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NATOPS FLIGHT MANUAL

NAVY MODEL

CT-39E/G

AIRCRAFT

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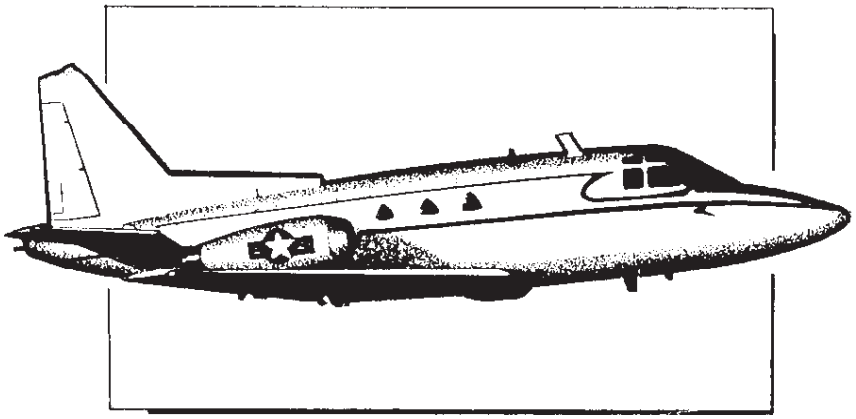

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NATEC ELECTRONIC MANUAL

Mach One Manuals

DEPARTMENT OF THE NAVY
OFFICE OF THE CHIEF OF NAVAL OPERATIONS
WASHINGTON, D.C. 20350

1 February 1994

LETTER OF PROMULGATION

1. The Naval Air Training and Operating Procedures Standardization (NATOPS) Program is a positive approach toward improving combat readiness and achieving a substantial reduction in the aircraft mishap rate. Standardization, based on professional knowledge and experience, provides the basis for development of an efficient and sound operational procedure. The standardization program is not planned to stifle individual initiative, but rather to aid the commanding officer in increasing the unit's combat potential without reducing command prestige or responsibility.

2. This manual standardizes ground and flight procedures but does not include tactical doctrine. Compliance with the stipulated manual procedure is mandatory except as authorized herein. In order to remain effective, NATOPS must be dynamic and stimulate rather than suppress individual thinking. Since aviation is a continuing, progressive profession, it is both desirable and necessary that new ideas and new techniques be expeditiously evaluated and incorporated if proven to be sound. To this end, commanding officers of aviation units are authorized to modify procedures contained herein, in accordance with the waiver provisions established by OPNAVINST 3710.7, for the purpose of assessing new ideas prior to initiating recommendations for permanent changes. This manual is prepared and kept current by the users in order to achieve maximum readiness and safety in the most efficient and economical manner. Should conflict exist between the training and operating procedures found in this manual and those found in other publications, this manual will govern.

3. Checklists and other pertinent extracts from this publication necessary to normal operations and training should be made and carried for use in naval aircraft.

BRENT M. BENNITT
Rear Admiral, U.S. Navy
Director, Air Warfare Division

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RECORD OF CHANGES

Change No. and Date of Change	Date of Entry	Page Count Verified by (Signature)

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INTERIM CHANGE SUMMARY

The following Interim Changes have been canceled or previously incorporated in this manual:

INTERIM CHANGE NUMBER(S)	REMARKS/PURPOSE
1 thru 10	Previously incorporated

The following Interim Changes have been incorporated in this Change/Revision:

INTERIM CHANGE NUMBER(S)	REMARKS/PURPOSE
11	Updated the Trim Arming Switch Section, the Emergency Procedures, and the Ice and Rain Section.
12	Updated the Battery Temperature Indicating and Isolation System and the Emergency Procedures

Interim Changes Outstanding – To be maintained by the custodian of this manual:

INTERIM CHANGE NUMBER(S)	ORIGINATOR/DATE (or DATE/TIME GROUP)	PAGES AFFECTED	REMARKS/PURPOSE

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PTAUZYUW RUENAAA4603 2911706-UUUU--RULSTGP.
ZNR UUUUU
P R 181532Z OCT 94 ZYB PSN 756890M35
FM CNO WASHINGTON DC//N889//
TO ALL SABERLINER AIRCRAFT ACTIVITIES//
INFO RULSTGP/NAVTACSUPPACT WASHINGTON DC//60//
RUCTPOH/NAVAEROPMEDINST PENSACOLA FL//00MM//
BT
UNCLAS //N03711//
MSGID/GENADMIN/N889//
SUBJ/INTERIM CHANGE TO CT-39EG AIRCRAFT NATOPS FLIGHT MANUAL//
REF/A/DOC/NAVAIR/01FEB94//
AMPN/REF A IS NAVAIR 01-60GBE-1 (CT-39E/G NATOPS FLIGHT MANUAL) DTD 01FEB94//
RMKS/1. THIS IS INTERIM CHANGE NUMBER 13 TO REF A (CT-39E/G NFM). THIS MSG MODIFIES INFORMATION ON USE OF ANTISKID SYSTEM WHEN ANTI SKID BRAKE FAIL CAUTION LIGHT IS ILLUMINATED.
2. CHANGE REF A (CT-39E/G NFM) AS FOLLOWS:
A. CHAPTER 2, PARAGRAPH 2.29.3 ANTISKID SYSTEM, PAGE 2-153 -
(1) DELETE SECOND SUBPARAGRAPH, WHICH BEGINS "IF A MALFUNCTION DEVELOPS...."
(2) ADD REPLACEMENT SECOND SUBPARAGRAPH AS FOLLOWS:
PAGE 02 RUENAAA4603 UNCLAS
"IF THE ANTISKID CONTROL VALVE IS OPENED CONTINUOUSLY FOR MORE THAN APPROXIMATELY THREE SECONDS, THE ENTIRE SYSTEM WILL BE DEACTIVATED ELECTRICALLY. THIS WILL RESULT IN ILLUMINATION OF THE ANTISKID BRAKE FAIL CAUTION LIGHT. COMPLETE CONTROL OF THE NORMAL BRAKING SYSTEM IS THEN RETURNED TO THE PILOT AS A RESULT OF A FAIL-SAFE SYSTEM DESIGN."
B. CHAPTER 2, PARAGRAPH 2.29.3.2 ANTISKID BRAKE FAILURE CAUTION LIGHT, PAGE 2-153 -
(1) DELETE FIRST SENTENCE, WHICH BEGINS "THE ANTISKID BRAKE FAIL CAUTION LIGHT ON THE ...."
(2) ADD REPLACEMENT FIRST SENTENCE, AS FOLLOWS:
"THE ANTI SKID BRAKE FAIL CAUTION LIGHT ON THE CAUTION-WARNING LIGHT PANEL (12, FIGURE 2-33), ILLUMINATES WHENEVER A HYDRAULIC BYPASS CONDITION EXISTS IN THE SYSTEM.
C. CHAPTER 2, PARAGRAPH 2.29.3.2 ANTISKID BRAKE FAILURE CAUTION LIGHT, PAGE 2-153 -
(1) DELETE: NA
(2) ADD WARNING AT END OF PARAGRAPH AS FOLLOWS:
PAGE 03 RUENAAA4603 UNCLAS

WARNING

LOSS OF ELECTRICAL SIGNAL FROM THE WHEEL TACHOMETER TO THE ANTISKID CONTROL BOX WILL RENDER THE ANTISKID SYSTEM INOPERATIVE ON THE AFFECTED WHEEL(S) WITHOUT ILLUMINATING THE ANTI SKID BRAKE FAIL CAUTION LIGHT. THIS CONDITION COULD ALLOW THE AFFECTED WHEEL(S) TO SKID, CAUSING TIRE FAILURE AND POSSIBLE LOSS OF DIRECTIONAL CONTROL.

D. CHAPTER 21, PARAGRAPH 21.11 PILOT QUESTION BANK, PAGE

SUMMARY OF APPLICABLE TECHNICAL DIRECTIVES

Information relating to the following recent technical directives has been incorporated in this manual

CHANGE NUMBER	DESCRIPTION	PLANNED METHOD OF INCORPORATION
68-12 69-23	(Sabreliner Service Change) Provide increased gross weight for Bunos 158380 - 158383	Depot/Organizational Field Team
AFC-74	(Supplemental Type Certificate STC SA834CE) Altitude Alerting and Reporting System	Depot Field Team
AD-72-19-4	(Air Worthiness Directive) NICAD Battery Temperature Sensing and Isolation System	Depot Field Team
AFC-74	Rosemount Probe Pitot Static System	Depot Field Team
73-6 73-10	Thrust Reverser Changes	Depot Field Team
SSL Rev 1	Audible Fire Warning System	Depot Field Team
*	* Information on five new CT-39G aircraft including WXR-80 Weather Radar, DF-206 ADF, Air-Conditioning, Pressurization, and Oxygen System Changes.	Depot Field Team

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Information relating to the following recent technical directives will be incorporated in a future change

CHANGE NUMBER	DESCRIPTION	PLANNED METHOD OF INCORPORATION

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LIST OF ABBREVIATIONS/ACRONYMS

A

A/A. Air-to-air.

AC. Aircraft commander.

ac. Alternating current.

ACT. Activation.

ADB. Aircraft discrepancy book.

ADF. Automatic direction finder.

ADIZ. Air defense identification zone.

AGL. Above ground level.

AM. Amplitude modulation.

APPR. Approach.

ATC. Air traffic control.

B

BFO. Beat frequency oscillator.

C

CAS. Calibrated airspeed.

cg. Center of gravity.

CH. Channel.

cw. Continuous wave.

D

D/G. Directional gyro.

dc. Direct current.

DH. Decision height.

DME. Distance measuring equipment.

E

EAS. Equivalent airspeed.

EGT. Exhaust gas temperature.

EPR. Engine pressure ratio.

ETA. Estimated time of arrival.

F

FAA. Federal Aviation Administration.

FCF. Functional checkflights.

FLIPs. Flight information publications.

fpm. Feet per minute.

G

g. Gravity.

GA. Go-around.

GPU. Ground power unit.

GS. Glideslope.

H

HF SSB. High frequency single sideband.

HF. High frequency.

HRD. High-rate-of-discharge.

Hz. Hertz.

I

IAS. Indicated airspeed.

ICAO. International Civil Aviation Organization.

IFF. Identification friend or foe.

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IFR. Instrument flight rule.

IFS. Integrated flight system.

IFS LTS. Integrated flight system lights.

ILS. Instrument landing system.

IMN. Indicated Mach number.

K

KHz. KiloHertz.

KIAS. Knots indicated airspeed.

L

L. Left.

LO SEN. Low sensitivity.

LSB. Lower sideband.

M

MAFS. Maintenance action forms.

MDA. Minimum decision altitude.

MEM. Memory.

MHz. MegaHertz.

N

NAVAID. Navigational aid.

nm. Nautical miles.

NOTAMs. Notices to airmen.

P

PA. Public address.

PAC. Pilot at the controls.

PIC. Pilot in command.

PNAC. Pilot not at the controls.

pph. Pounds per hour.

psi. Pounds per square inch.

R

R. Right; range.

R/T. Rate-of-turn.

RAM. Radar absorption material.

REC. Receive.

RMI. Radio magnetic indicator.

RMT. Remote.

rpm. Revolutions per minute.

S

SAR. Search and rescue.

SDLM. Standard depot level maintenance.

SOP. Standard operating procedures.

SQL. Squelch.

SSB HF. Single sideband high frequency.

STDY. Standby.

T

T+2. Total air temperature.

T/R. Transmit/receive.

T2P. Transport second pilot.

T3P. Transport third pilot.

TAC. Transport aircraft commander.

tacan. Tactical air navigation.

TAMB. Ambient air temperature.

TAS. True airspeed.

TFR. Transfer.

TTS. Time to station.

TX. Transmit.

U

UHF. Ultrahigh frequency.

USB. Upper sideband.

V

V. Voice.

V_1. Critical engine failure speed.

V_2. Takeoff safety speed.

vac. Volts alternating current.

vdc. Volts direct current.

VF. Navy fighter squadron.

VFR. Visual flight rules.

VHF. Very high frequency.

VMC. Visual meteorological conditions.

V_{MCA}. Minimum control speed (air).

V_{MCG}. Minimum control speed (ground).

VMO. Marine observation squadron.

VOR. VHF omnidirectional range.

V_R. Rotation speed.

V_S. Stall speed.

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PREFACE

SCOPE

The NATOPS flight manual is issued by the authority of the Chief of Naval Operations and under the direction of Commander, Naval Air Systems Command in conjunction with the naval air training and operating procedures standardization (NATOPS) program. This manual contains information on all aircraft systems, performance data, and operating procedures required for safe and effective operations. However, it is not a substitute for sound judgment. Compound emergencies, available facilities, adverse weather or terrain, or considerations affecting the lives and property of others may require modification of the procedures contained herein. Read this manual from cover to cover. It's your responsibility to have a complete knowledge of its contents.

APPLICABLE PUBLICATIONS

The following applicable publications complement this manual:

NAVAIR 01-60GBE-1B, Pilot's Pocket Checklist

NAVAIR 01-60GBE-1C, Card Checklist

NAVAIR 01-60GBE-1.1E, Emergency Procedures (Passengers)

NAVAIR 01-60GBE-1F, Functional Checkflight Checklist

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Note

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UPDATING THE MANUAL

To ensure that the manual contains the latest procedures and information, NATOPS review conferences are held in accordance with the current OPNAVINST 3710.7.

CHANGE RECOMMENDATIONS

Recommended changes to this manual or other NATOPS publications may be submitted by anyone in accordance with the current OPNAVINST 3710.7.

Routine change recommendations are submitted directly to the model manager on OPNAV 3710/6 (4-90) shown on the next page. The address of the model manager of this aircraft is:

Commander
Fleet Logistics Support Wing
8100 W. Jefferson
Naval Air Station Bldg 1239
Dallas, TX 75211-9513

NATOPS/TACTICAL CHANGE RECOMMENDATION
OPNAV 3710/6 (4-90) S/N 0107-LF-009-7900

DATE _____

TO BE FILLED IN BY ORIGINATOR AND FORWARDED TO MODEL MANAGER

FROM (Originator)	Unit
TO (Model Manager)	Unit

Complete Name of Manual/Checklist	Revision Date	Change Date	Section/Chapter	Page	Paragraph

Recommendation (be specific)

☐ CHECK IF CONTINUED ON BACK

Justification

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Signature	Rank	Title

Address of Unit or Command

TO BE FILLED IN BY MODEL MANAGER *(Return to Originator)*

FROM	DATE

TO

REFERENCE

(a) Your Change Recommendation Dated _____

☐ Your change recommendation dated _____ is acknowledged. It will be held for action of the review conference planned for _____ to be held at _____

☐ Your change recommendation is reclassified URGENT and forwarded for approval to _____ by my DTG _____ .

/S/_____ MODEL MANAGER	_____ AIRCRAFT

Change recommendations of an urgent nature (safety of flight, etc.) should be submitted directly to the NATOPS Advisory Group Member in the chain of command by priority message.

YOUR RESPONSIBILITY

NATOPS flight manuals are kept current through an active manual change program. Any corrections, additions, or constructive suggestions for improvement of its content should be submitted by routine or urgent change recommendation, as appropriate, at once.

NATOPS FLIGHT MANUAL INTERIM CHANGES

Flight manual interim changes are changes or corrections to the NATOPS flight manuals promulgated by CNO or NAVAIRSYSCOM. Interim changes are issued either as printed pages or as a naval message. The interim change summary page is provided as a record of all interim changes. Upon receipt of a change or revision, the custodian of the manual should check the updated interim change summary to ascertain that all outstanding interim changes have been either incorporated or canceled; those not incorporated shall be recorded as outstanding in the section provided.

CHANGE SYMBOLS

Revised text is indicated by a black vertical line in either margin of the page, adjacent to the affected text, like the one printed next to this paragraph. The change symbol identifies the addition of either new information, a changed procedure, the correction of an error, or a rephrasing of the previous material.

WARNINGS, CAUTIONS, AND NOTES

The following definitions apply to "WARNINGs," "CAUTIONs," and "Notes" found throughout the manual.

WARNING

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

CAUTION

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

Note

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

WORDING

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

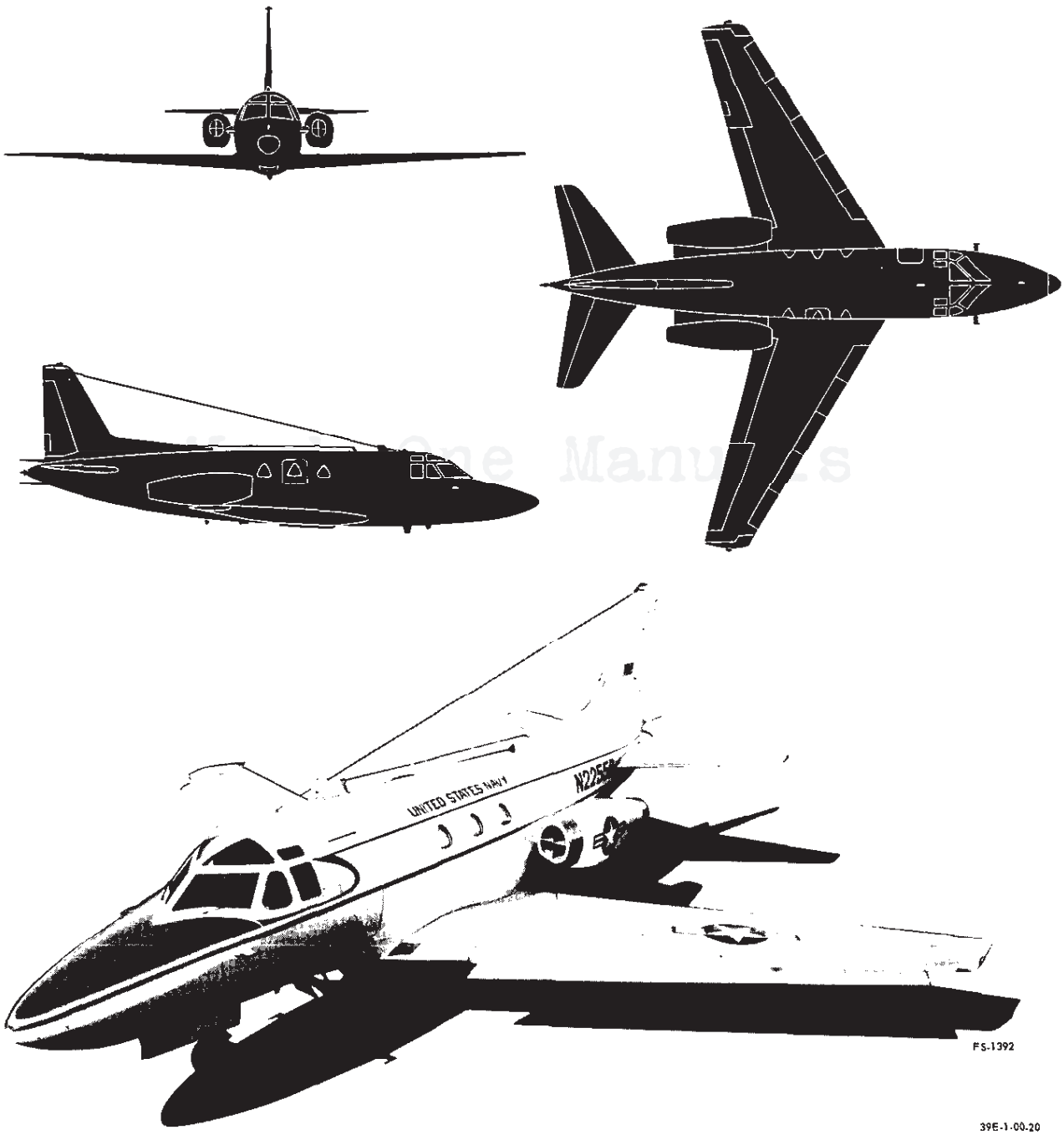
"Shall" has been used only when application of a procedure is mandatory.

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

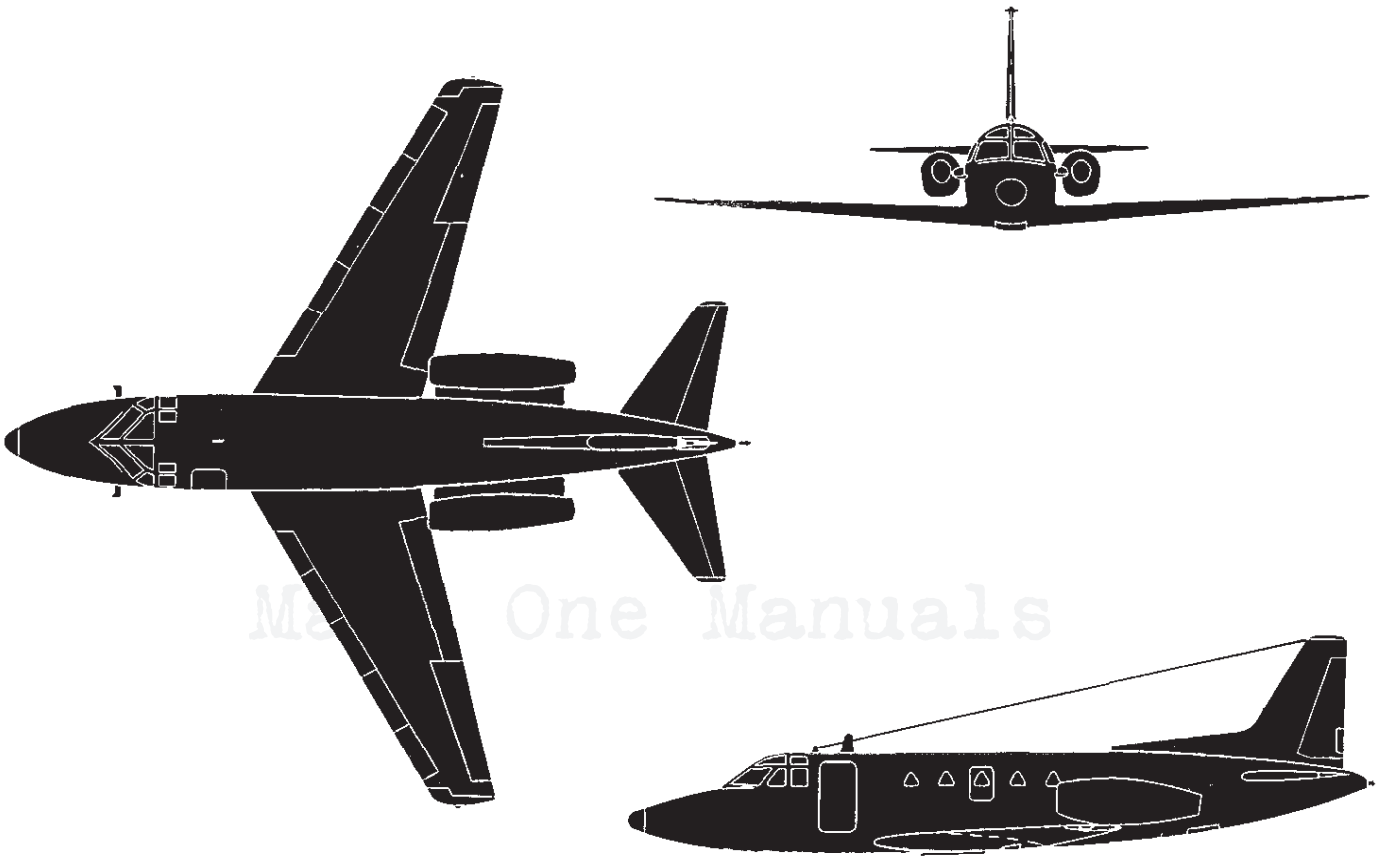
"May" and "need not" have been used only when application of a procedure is optional.

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

CT-39E



CT-39G Sabreliner



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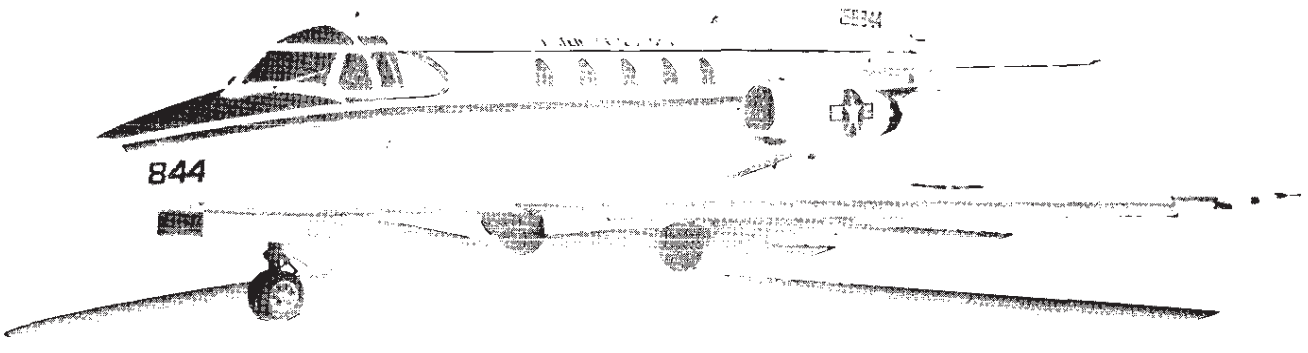

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PART I

Aircraft

Chapter 1 — General Description

Chapter 2 — Systems

Chapter 3 — Servicing

Chapter 4 — Operating Limitations

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CHAPTER 1

General Description

1.1 AIRCRAFT

The CT-39E and CT-39G aircraft (Figures 1-1 and 1-2) built by Rockwell International Corporation, is a low-wing, twin-jet monoplane with an axial-flow, pod-mounted engine on each side of the aft fuselage.

The cockpit and cabin compartments are pressurized and soundproofed for high-altitude flight. A plug-type entry door on the left side of the fuselage forward of the wing rotates down, forming steps for entry. A ground escape hatch is on the right side of the fuselage on CT-39E aircraft and on both sides of the fuselage on CT-39G aircraft. The 28° sweptback wing has aerodynamically operated slats on the leading edge and electrically operated flaps on the inboard trailing edge. An electrically actuated, hydraulically operated speedbrake that can be used at any airspeed, is on the lower surface of the fuselage.

The fuel supply is carried in the integral wing tanks and fuselage tank. Fuel servicing is performed through a single-point pressure refueling receptacle on the inboard lower surface of the right wing. Three gravity-flow fuel filler receptacles are provided: one on the upper surface of each wing panel at the tip and one on the aft fuselage right side upper surface.

1.2 AIRCRAFT BUREAU NUMBERS

Each aircraft is assigned a serial number (bureau number) painted on the aft fuselage. Bureau numbers are as follows:

CONTRACTOR NUMBER	AIRCRAFT BUREAU NUMBER
CT-39E	
282-84	157353
282-85	157354
282-95	158380
282-93	158381
282-92	158382
282-96	158383
CT-39G	
306-52	158843
306-55	158844
306-65	159361
306-66	159362
306-67	159363
306-69	159364
306-70	159365
306-104	160053
306-105	160054
306-106	160055
306-107	160056
306-108	160057

1.2.1 Aircraft Dimensions. Overall dimensions of the aircraft are as follows:

1. Wing span — 44.64 feet.
2. Length (CT-39E) — 43.74 feet.

3. Length (CT-39G) — 46.92 feet.

4. Height — 15.99 feet.

1.3 AIRCRAFT WEIGHT

1. Maximum start of takeoff gross weight — 19,612 (CT-39E); 20,172 (CT-39G).

2. Maximum gross ramp and taxi weight — 19,922 (CT-39E); 20,372 (CT-39G).

The approximate basic weight is as follows:

3. Aircraft plus engine oil and trapped fuel (pounds) — 10,500 (CT-39E); 11,400 (CT-39G).

For more detailed weight information, refer to paragraph 4.14, WEIGHT LIMITATIONS.

1.4 GENERAL ARRANGEMENT

For general arrangement, see Figure 1-2.

1.4.1 Interior Arrangement. The aircraft cabin compartment is divided into two sections: an intermediate cabin area and an aft cabin area. The intermediate cabin area contains the baggage compartment, a coat compartment, and the entry door. The aft cabin area contains the passenger seats. A center-facing, plane captain/passenger seat with a chemical toilet underneath is opposite the baggage compartment. On CT-39G aircraft 158843 and 158844, a center facing seat is also located on the left side of the aft end of the cabin with a chemical toilet under the seat. On all the other CT-39E/G aircraft, a forward-facing three-place couch is at the rear of the aircraft.

1.4.2 Flightcrew. Accommodations are provided for a pilot, a copilot, and a flightcrew plane captain.

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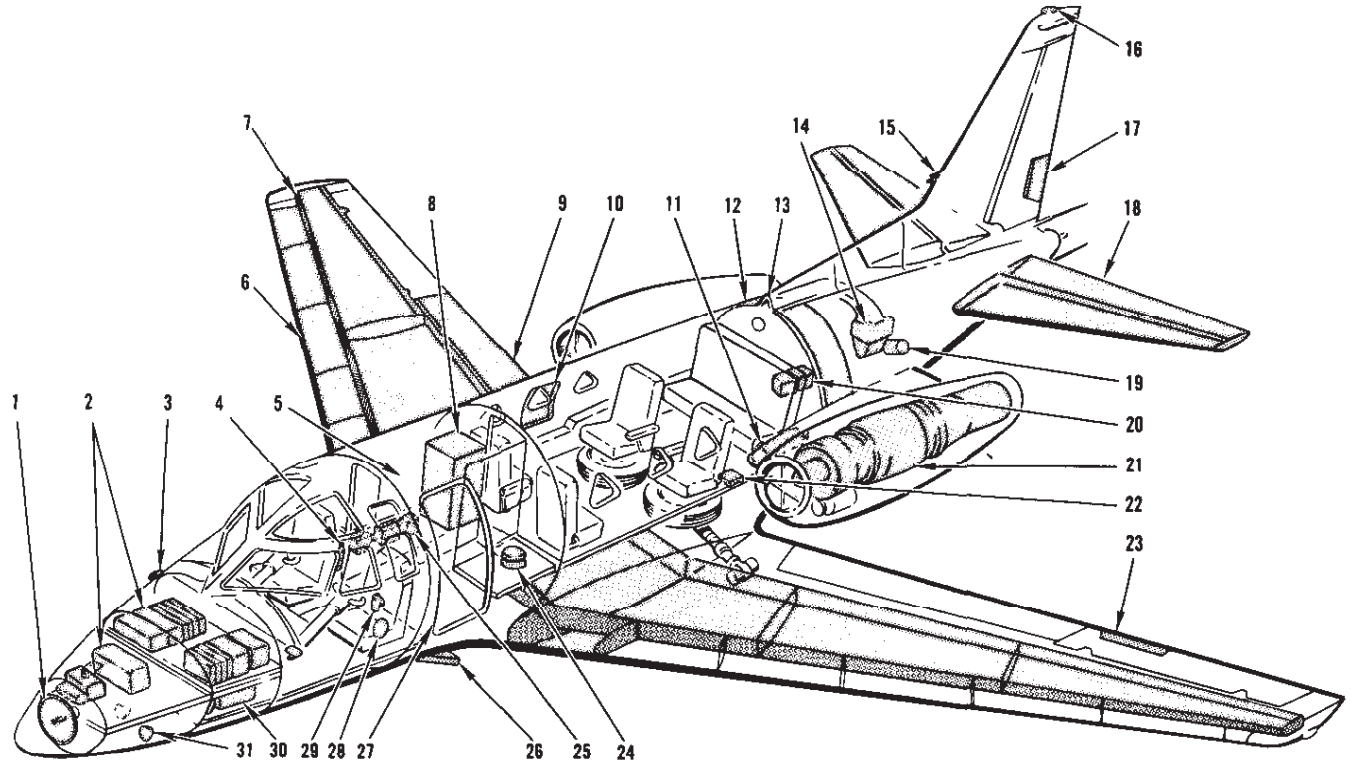

1. WEATHER RADAR ANTENNA
2 COMMUNICATION, NAVIGATION, AND RADAR EQUIPMENT
3 PITOT HEAD (TYPICAL BOTH SIDES)
4 WALKAROUND OXYGEN BOTTLE
5 BAGGAGE COMPARTMENT
6 WING SLATS
7 INTEGRAL WING FUEL TANKS
8 10-MAN LIFE RAFT
9 WING FLAP
10 GROUND ESCAPE HATCH
11 OXYGEN BOTTLE
12. FUSELAGE FUEL CELL
13. RAM-AIR INLET
14 HEAT EXCHANGER
15 ENGINE PRESSURE RATIO PITOT HEADS
16 ANTICOLLISION LIGHT
17 RUDDER TRIM TAB
18 TRIMMABLE HORIZONTAL STABILIZER
19 HYDRAULIC RESERVOIR
20 BATTERIES
21 JT12A TURBOJET ENGINE (TYPICAL BOTH SIDES)
22 DC EXTERNAL POWER RECEPTACLES
23. AILERON TRIM TAB
24 PLANE CAPTAIN SEAT WITH CHEMICAL TOILET
25. GYROS
26. SPEED BRAKE
27 ENTRY DOOR
28 WING ICE CHECK LIGHT (TYPICAL BOTH SIDES)
29 HAND FIRE EXTINGUISHER
30. INVERTERS
31 RETRACTABLE LANDING AND TAXI LIGHTS

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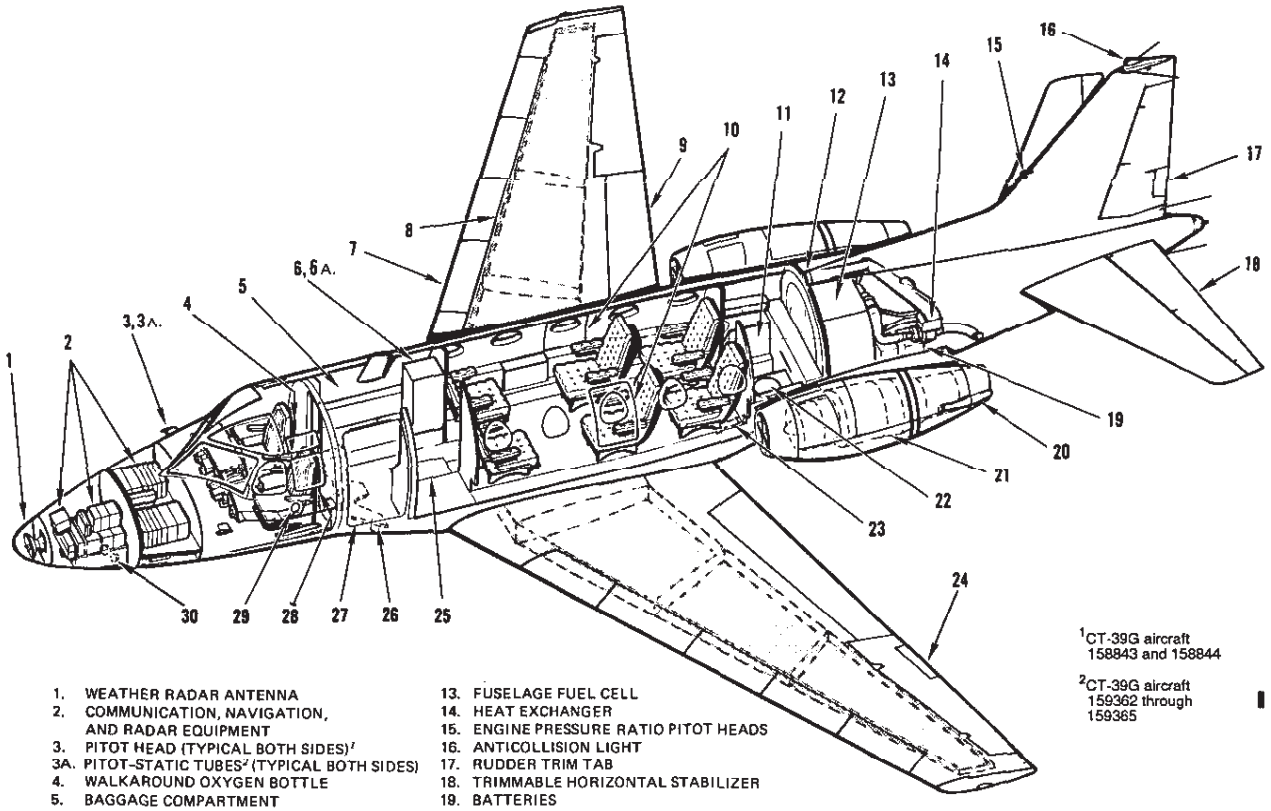
Figure 1-1. CT-39E General Arrangement



1. WEATHER RADAR ANTENNA
2. COMMUNICATION, NAVIGATION, AND RADAR EQUIPMENT
3. PITOT HEAD (TYPICAL BOTH SIDES)[1]
3A. PITOT-STATIC TUBES[2] (TYPICAL BOTH SIDES)
4. WALKAROUND OXYGEN BOTTLE
5. BAGGAGE COMPARTMENT
6. 10-MAN LIFE RAFT[1]
6A. TWO 7-MAN LIFE RAFTS[2]
7. WING SLATS
8. INTEGRAL WING FUEL TANKS
9. WING FLAP
10. GROUND ESCAPE HATCH (BOTH SIDES)
11. LAVATORY[1]
12. RAM-AIR INLET
13. FUSELAGE FUEL CELL
14. HEAT EXCHANGER
15. ENGINE PRESSURE RATIO PITOT HEADS
16. ANTICOLLISION LIGHT
17. RUDDER TRIM TAB
18. TRIMMABLE HORIZONTAL STABILIZER
19. BATTERIES
20. THRUST REVERSER (TYPICAL BOTH SIDES)
21. JT12A TURBOJET ENGINE (TYPICAL BOTH SIDES)
22. CHEMICAL TOILET
23. OXYGEN BOTTLE
24. AILERON TRIM TAB (LEFT SIDE ONLY)
25. PLANE CAPTAIN SEAT
26. SPEED BRAKE
27. ENTRY DOOR
28. HAND FIRE EXTINGUISHER
29. WING ICE CHECK LIGHT (TYPICAL BOTH SIDES)
30. RETRACTABLE LANDING AND TAXILIGHTS

[1]CT-39G aircraft 158843 and 158844

[2]CT-39G aircraft 159362 through 159365

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Figure 1-2. CT-39E General Arrangement — Typical

ITEM	CT-39E AIRCRAFT	CT-39G AIRCRAFT – 158843 AND 158844	CT-39G AIRCRAFT — 159362 THRU 159365	CT-39G AIRCRAFT – 160053 THRU 160057
GROSS WEIGHT (RAMP AND TAXI)	19,922	20,372	20,372	20,372
GROSS WEIGHT (START OF TAKE-OFF)	19,612	20,172	20,172	20,172
ENGINES	P&W JT12A-8	P&W JT-12A-8	P&W JT12A-8	P&W JT12A-8
THRUST REVERSERS	NO	YES	YES	YES
THRUST REVERSER AUTO STOW	NO	YES	YES	YES
BATTERY TEMPERATURE INDICATING AND ISOLATION SYSTEM	YES (MANUAL)	YES (MANUAL)	YES (AUTOMATIC)	YES (AUTOMATIC)
PITOT-STATIC COMPENSATED SYSTEM	NO	NO (843) YES (844)	YES	YES
AIR CONDI-TIONING REFRIGERATION UNIT	TWO-WHEEL UNIT	TWO-WHEEL UNIT	THREE-WHEEL UNIT	THREE-WHEEL UNIT
FUEL JETTISON	YES	YES	YES	NO
HYDRAULIC PRESSURE AUDIO WARNING SYSTEM	YES (SILENCER CAPABILITY)	YES (SILENCER CAPABILITY)	YES	YES
AUTOPILOT	COLLINS AP-103F	COLLINS AP-103F (843) AP-105 (844)	COLLINS AP-103F	COLLINS AP-105
EMERGENCY TRIM DISCONNECT	FOR AIRCRAFT WITH SABRE-LINER SERVICE BULLETIN 74-6	FOR AIRCRAFT WITH SABRE-LINER SERVICE BULLETIN 74-6	YES	YES
AUDIBLE TRIM WARNING	FOR AIRCRAFT WITH SABRE-LINER SERVICE BULLETIN 74-7	NO	FOR AIRCRAFT WITH SABRE-LINER SERVICE BULLETIN 74-7	YES
FLIGHT DIRECTOR SYSTEM	COLLINS FD-108	COLLINS FD-109D (843) FD-109Y (844)	COLLINS FD-109G	COLLINS FD-109Y

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Figure 1-3. Table of Differences (Sheet 1 of 2)

ITEM	CT-39E AIRCRAFT	CT-39G AIRCRAFT – 158843	CT-39G AIRCRAFT — 159362 THRU 159365	CT-39G AIRCRAFT – 160053 THRU 160057 AND 158844
UHF RADIO	AN/ARC-51AX AN/ARC-159	AN/ARC-51AX	AN/ARC-51AX AN/ARC-159	AN/ARC-159
VHF RADIO	COLLINS 618M-1A 618M-2D	COLLINS 618M-2D	COLLINS 618M-2D	COLLINS VHF-20B
HF AM RADIO	SUNAIR ASB-100 COLLINS 718U-5	SUNAIR ASB-100	COLLINS 718U-5	COLLINS 718U-5
AUDIO SYSTEM	COLLINS 346-B3	COLLINS 346-B3	COLLINS 346-B3	COLLINS 346-B3
TACAN NAVIGATION	AN/ARN-91 SIERRA 7000	AN/ARN-91	AN/ARN-91 SIERRA 7000	SIERRA 7000
VOR/ILS VHF NAVIGATION	COLLINS 51RV-1	COLLINS 51RV-1	COLLINS 51RV-1	COLLINS VIR-30A
LF ADF NAVIGATION	COLLINS DF-203	COLLINS DF-203	COLLINS DF-206	COLLINS DF-206 ADF-60A
WEATHER RADAR	PRIMUS-400	PRIMUS-400	PRIMUS-400	PRIMUS-400
DIRECTIONAL INDICATING SYSTEM	COLLINS MC-102	COLLINS MC-102	COLLINS MC-102	COLLINS MC-103
ATC TRANSPONDER	COLLINS 621A-3	COLLINS 621A-3	COLLINS 621A-3	COLLINS TDR-90
MARKER BEACON	COLLINS 51Z-4	COLLINS 51Z-4	COLLINS 51Z-4	COLLINS PART OF VIR-30A
RADIO ALTIMETER	COLLINS AL-101	COLLINS AL-101	COLLINS AL-101	COLLINS ALT-50

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Figure 1-3. Table of Differences (Sheet 2 of 2)

CHAPTER 2

Systems

2.1 AIR-CONDITIONING AND PRESSURIZATION SYSTEM

Hot compressed air, bled from the final stage of the engine compressor, is used to maintain the desired cabin temperature and pressurization. Controls for the system are on the center pedestal and are accessible to either the pilot or the copilot. An external pressure source may be connected to the system for testing or for air-conditioning during ground operation. The air-conditioning and pressurization system is shown schematically in FO-7, FO-8, or FO-9 (as applicable).

2.1.1 Air-Conditioning. The temperature of the air supplied to the cabin is regulated by an automatic control system. Hot, compressed engine air is extracted from the final engine compressor stage through four ports, two on each engine. These ports are manifolded at each engine to form a single duct that is routed into the aft fuselage. Air is bled from this duct for pressurization of the entry door seal. In the aft fuselage, the duct from each engine is joined into a common duct that routes the hot compressor air through or around a heat exchanger and refrigeration unit as required. In this manner, either engine is capable of furnishing air to the complete system.

Each engine has a bleed-air shutoff valve that protects the surrounding structure and equipment in case of downstream duct rupture or air contamination. Reverse-flow check valves prevent reverse flow during single-engine operation.

Air from the common duct of the two engines passes through a flow-limiting valve and divides one branch to be cooled, the other to remain hot. The air to be cooled passes through the heat exchanger where it is partially cooled by a crossflow of ram air before being directed to an expansion turbine for further cooling. This cold air then passes through a water separator and divides. One branch joins the hot-air line; the other continues into the pressurized section to supply cold air to adjustable "eyeball" outlets. A cabin flood valve permits air that has been tempered in the heat exchanger to bypass the cooling turbine and water separator and enter the supply line downstream of a hot-air mixing valve. In bypassing these two high-pressure drop items, the flood valve automatically provides an adequate amount of air to the cabin under conditions where the supply bleed air pressure is only slightly higher than cabin pressure, such as during a descent from high altitude. Downstream of the hot-air mixing valve, a ram-air line enters the system to provide an emergency air supply. The hot-air branch from the flow-limiting valve passes through the hot-air mixing valve and joins the cold air from the refrigeration unit and heat exchanger. The hot-air mixing valve operates automatically to admit the proper amount of hot air into the lines in response to cockpit temperature controls and sensors. The air, mixed to the proper temperature by the mixing valve, enters the cabin through flush outlets in the ceiling to provide thermostatically controlled air at low velocity. The ducting continues through the cabin to cockpit outlets behind the pilot and copilot seats. These cockpit outlets will emit the same temperature air as in the cabin. All conditioned air entering the cabin is ducted through mufflers to reduce cabin noise. For emergency operation, ram air is used to ventilate the aircraft.

On CT-39G aircraft 159362 through 159365 and 160053 through 160057, the air-conditioning system incorporates the installation of a three-wheel refrigeration unit. (See FO-9.) This refrigeration unit includes a boost compressor, an expansion turbine, and a cooling air fan on a common shaft. After passing through the flow-limiting valve, as previously explained, the bleed air to be cooled passes through the primary section of the heat exchanger in the refrigeration unit where it is partially cooled. This partially cooled air is pressure regulated and then routed to the compressor section. The air is further compressed, then cooled, as it passes through the secondary heat exchanger. Final cooling and expansion takes place as the air is released to drive the turbine wheel which, in turn, drives the compressor and fan. The fan draws ambient ram air over the heat exchanger, and the heat-laden air is directed overboard. The cooled and expanded air out of the turbine is routed through a water separator. A sensor, installed downstream of the water separator, senses air

temperature at that location. The sensor controls the opening and closing of a temperature control valve. The control valve controls the flow of compressor inlet air that is mixed with turbine discharge air to maintain a minimum of 35 °F into the water separator. This serves to protect against the formation of ice in the water separator. After the cooled air leaves the water separator, it divides, as previously explained. Also, at this point, a cabin air blower is installed to boost the volume of air into the cabin. The blower is powered by the dc secondary bus and controlled by the CABIN BLOWER switch on the air-conditioning system control panel. The switch is effective only if the CABIN AIR switch is positioned at RH ENG, LH ENG, BOTH ENG, or RAM and the essential bus is energized.

2.1.2 Cockpit Floor and Side Console Heating

2.1.2.1 Floor Heaters (CT-39E). Electric heating elements in rubber pads under the carpet warm the floor in the pilot and copilot foot area. The heating elements cycle on when the floor skin temperature drops to 28 °F and cycle off when the floor skin temperature rises to 43 °F. The system is entirely automatic (no pilot control). The heating elements are powered by the dc essential bus and are protected by a circuit-breaker on the copilot overhead aft circuit-breaker panel.

2.1.2.2 Floor Heaters and Side Console Heaters (CT-39G). Electric heating elements under the carpet warm the floor in the pilot and copilot foot area. Heater pads are also installed on each side console in the cockpit. The floor and console heaters function automatically (no pilot control) and cycle on when the temperature at the thermal switch drops to 45 °F and cycle off when the temperature increases to 60 °F. The heaters are powered from the dc essential bus and are protected by a circuit breaker on the right-hand console circuit-breaker panel. On 160053 through 160057, five heater pads are installed under the carpet in the cabin. These heater pads are powered by the same system that powers the cockpit heater pads.

2.1.3 Pressurization. The interior is pressurized by the flow of air from the cockpit and cabin air outlets. Pressurization is controlled by a pressure controller and is maintained at a selected pressure differential, not to exceed 8.2 psi to 40,000 feet and an 8,000-foot cabin altitude from 40,000 to 45,000 feet, by a regulator that controls the outflow of pressurized air from the interior. The pressurization schedule is shown in Figure 2-1. The pressure in the cockpit and cabin is maintained by the cabin pressure regulator (outflow valve) that in turn, is controlled automatically by the pressure controller. A limit control regulates the outflow valve to a 15,000-foot isobaric schedule in case the outflow valve should fail in the open position at altitudes in excess of 13,500 ±1,500 feet. A pneumatic relay is installed between the controller and outflow valve and increases the sensitivity to altitude and/or pressure changes, thereby reducing cabin fluctuations. A vacuum pump (CT-39G, 158844, 159362 through 159365, and 160053 through 160057) or jet pump (CT-39E, 158843) supplies the negative pressure required for operation of the outflow valve and the cabin pressure dump valve at low altitude (below 14,300 feet). An altitude switch is connected electrically to the vacuum pump and controls pump operation by sensing ambient pressure. If the outflow valve or limit control were to fail, the cabin pressure dump valve would prevent cabin pressure from exceeding 9.1 ±0.1 psig above outside atmospheric pressure. The FASTEN SEAT BELT sign and CABIN PRESSURE FAIL light will illuminate when cabin altitude exceeds 10,000 feet. If cabin altitude exceeds 13,500 ±500 feet, the oxygen warning horn will sound and cabin lights, passenger oxygen light, NO SMOKING, and WEAR OXYGEN MASK signs will illuminate.

In case of a pressurization duct failure, cabin pressurization can be maintained by use of an emergency pressurization duct that is opened by a switch in the cockpit. The emergency pressurization air is supplied only by the right-hand engine. In case of emergencies, pressurization can be dumped by a cabin pressure dump valve.

2.1.3.1 Door Seals. Pressurization for the seal of the entry door is provided by air from the engine compressors (FO-7 and FO-8) and is automatically controlled by a pressure regulator. Check valves in the system are designed to prevent reverse flow if an engine should fail and to maintain pressure even if both engines should fail. A reservoir stores higher engine pressure for use if the engine pressure should drop below the required 25 psi. The pressure regulator incorporates a solenoid shutoff valve that is energized open when the weight of the aircraft is off the landing gear to permit pressurization of the seal. When the weight of the aircraft is on the landing gear, the solenoid shutoff valve is deenergized closed, shutting off pressurization air and dumping the seal. A door seal pressure dump valve is installed on the aft side of the bulkhead aft of the entrance door. This dump valve provides a means of deflating the entrance door seal in case the solenoid valve fails to return to the closed position when the shutoff valve is deenergized. The seals for the pilot sliding window are not included in the door seal system. The entry door secondary seal and pilot sliding window are of a dynamic type and subject only to pressure from within.

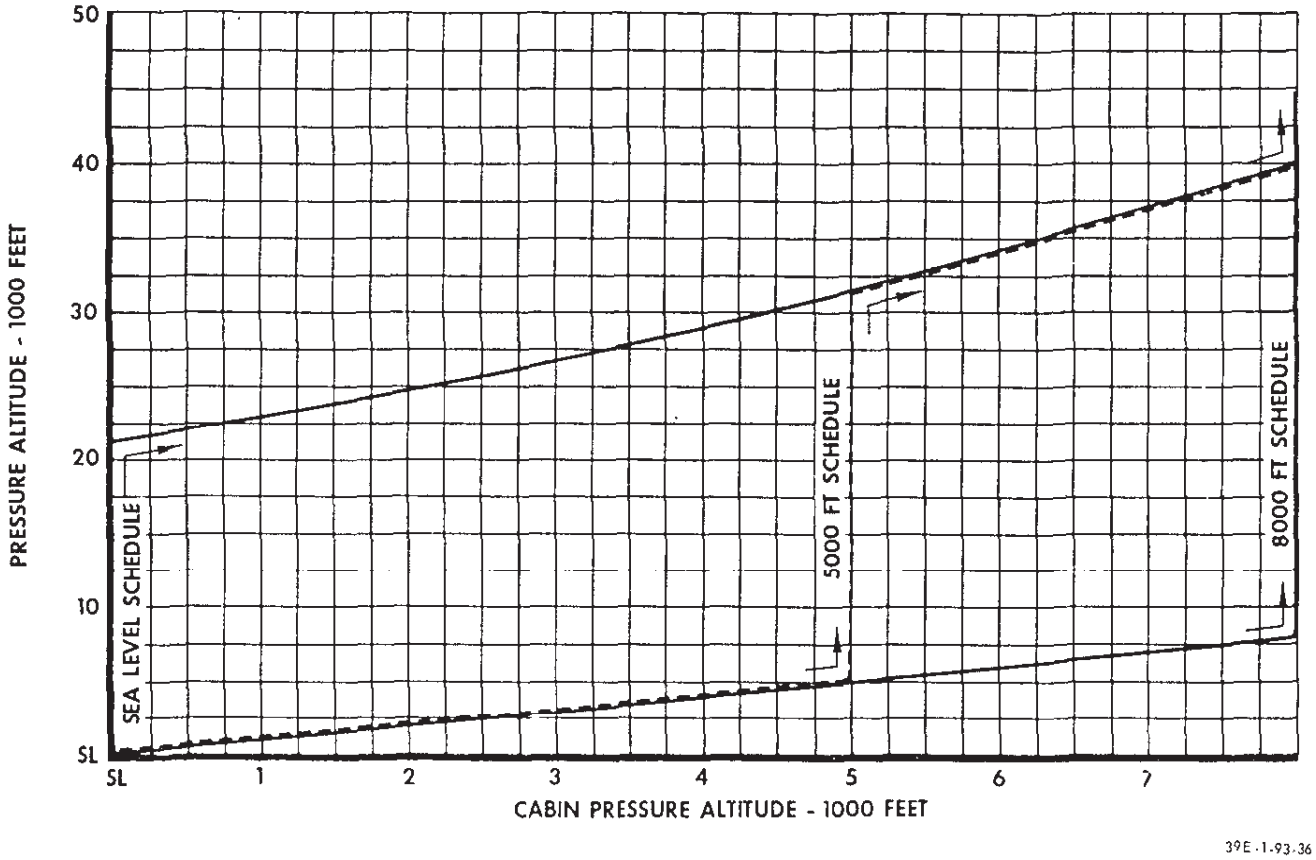

Figure 2-1. Pressurization Schedule

2.1.3.2 Nose Pressurization. On CT-39G aircraft 159362 through 159365 and 160053 through 160057, the nose section of the aircraft is not pressurized. However, a line to supply pressurization for the weather radar system wave guide is provided. On CT-39E aircraft and CT-39G aircraft 158843 and 158844, to reduce the arcing and corona effects of the electronic equipment when the aircraft altitude is in excess of 29,000 feet, the nose section of the aircraft is pressurized to maintain a nominal altitude of 29,000 feet. Outflow air from the cabin pressure regulator is used to pressurize the nose, and the equipment bay pressure regulator controls pressure to a maximum differential of 2.4 psi. If the pressure in the nose exceeds 2.4 psi differential, a pressure relief valve (reset on the ground only) dumps nose pressurization at 3.05 ±0.2 psi differential. Loss of pressurization to the nose is indicated by illumination of the nose pressure failure caution light on the caution-warning light panel.

2.1.3.3 Cabin Emergency Pressurization. An emergency pressurization duct from the bleed-air system of the right-hand engine can be manually selected to supply cabin emergency pressurization air in case of a failure of the normal pressurization duct. The emergency pressurization air to the cabin is picked up between the bleed-air shutoff valve and the right-hand engine. This airflow after passing through an electrically controlled shutoff valve and through a check valve goes into the distribution ducts to the cabin and cockpit air outlets. The shutoff valve is opened by selecting EMER PRESS on the cabin air selector switch. The valve is closed, however, regardless of the cabin air selector switch if the weight of the aircraft is on either main landing gear, if the bleed air exceeds 60 psi, or if the fire-extinguisher handle for the right-hand engine is pulled.

2.1.4 Air-Conditioning and Pressurization System Controls

2.1.4.1 Cabin Air Selector Switch. The CABIN AIR switch on the air-conditioning system control panel (Figure 2-2) is powered by the dc essential bus and the ac secondary bus. The switch has seven positions: EMER PRESS, BOTH ENG, LH ENG, RH ENG, OFF, RAM, and RAM & DUMP. The switch controls the two-engine bleed-air shutoff valves: one in each engine compressor hot-air line. Turning the switch to BOTH ENG opens both normally closed engine bleed-air shutoff valves simultaneously to allow full flow compressor air in the air-conditioning and pressurization system.

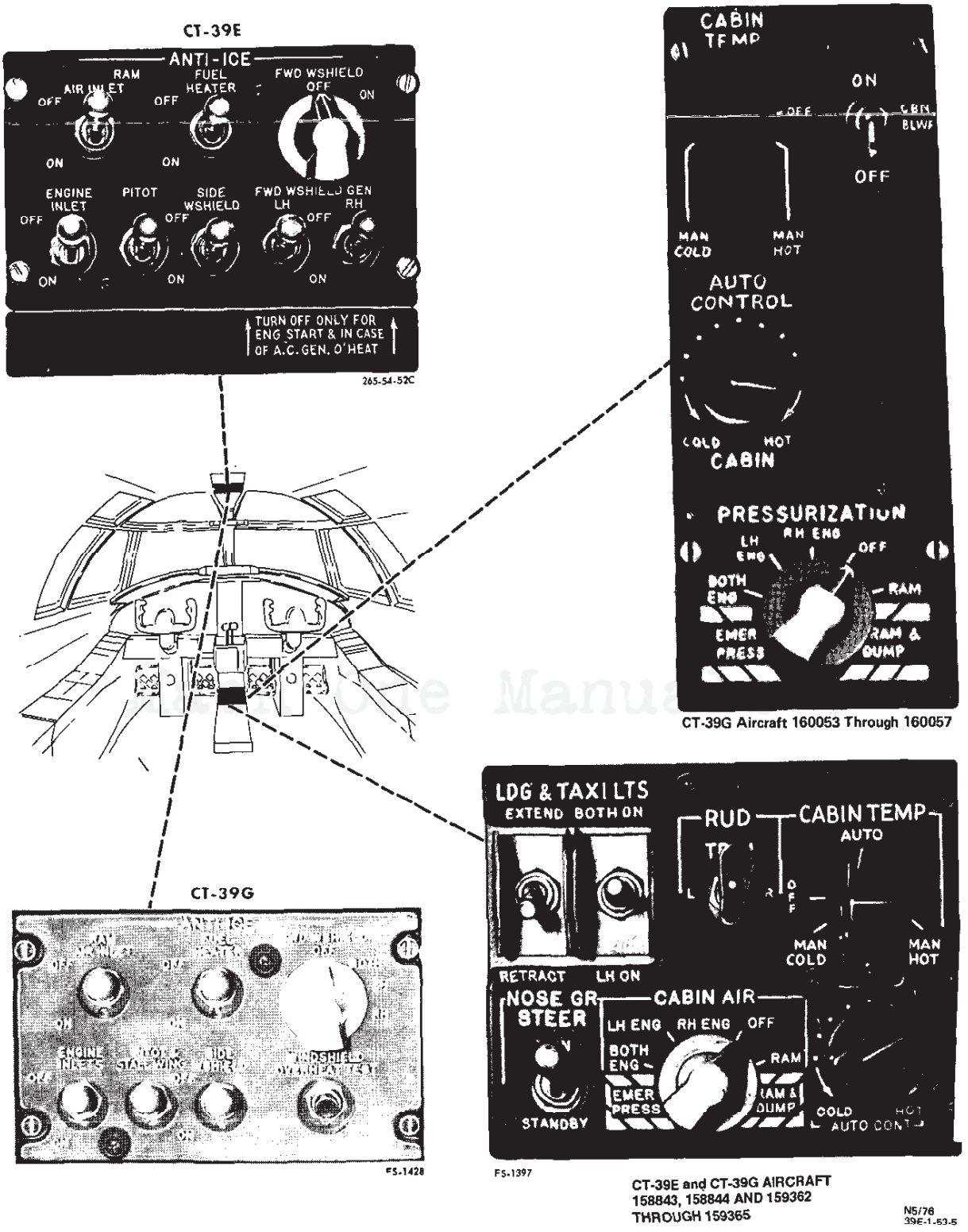

Figure 2-2. Air-Conditioning, Pressurization, and Anti-Icing System Controls

Turning the switch to either LH ENG or RH ENG shuts off the nonselected engine compressor air in case of contamination.

Note

- The shutoff valves are also pneumatically operated and will close anytime the compressor discharge pressure drops below 4 psi.
- Single-engine compressor air is sufficient to maintain normal air-conditioning and pressurization.
- During single-engine operation, the cabin air selector switch must be positioned to the engine in operation.

Turning the switch to OFF shuts off the compressor air from each engine. Turning the switch to RAM closes the compressor bleed-air shutoff valve for each engine and opens the emergency ram-air supply valve. Turning the switch to RAM & DUMP opens the cabin air dump valve and emergency ram-air supply valve in addition to closing the compressor bleed-air shutoff valves to dump the cabin pressure. Turning the switch to EMER PRESS closes the compressor bleed-air shutoff valve for each engine and opens the emergency pressurization valve so that bleed air from the right-hand engine supplies emergency pressurization to the cabin and cockpit air outlets.

2.1.4.2 Cabin Air Temperature Selector Switch. The CABIN TEMP switch on the air-conditioning system control panel (Figure 2-2) is powered by the ac secondary bus and provides normal control of cabin and cockpit air temperature. When the switch is at the guarded AUTO position, a hot-air mixing valve is modulated to maintain the setting selected by a cabin air temperature rheostat through a temperature control box. With the switch at OFF, the automatic control system is inoperative and the valve remains in the position prevailing at the time the switch was turned to OFF. If the automatic temperature control system fails or if manual control is desired, moving the switch to the spring-loaded MAN HOT or MAN COLD position opens or closes the hot-air mixing valve to produce a corresponding change in temperature. When using manual control, the switch should be toggled momentarily and released to OFF. Allow sufficient time between applications to minimize temperature overshoot.

2.1.4.3 Cabin-Cockpit Temperature Control Switch (CT-39G 158843, 158844, and 159362 Through 159365 Aircraft). The TEMP CONTROL switch on the air-conditioning system control panel (9A, FO-3) is used to select either the cockpit or the cabin temperature sensor for temperature control. The switch positions are marked CABIN and COCKPIT.

2.1.4.4 Cabin and Cockpit Air Temperature Rheostat Knob. The AUTO CONT knob on the air-conditioning system control panel (Figure 2-2) is powered by the ac primary bus and sweeps from COLD to HOT. The knob may be positioned at any place within this range and will automatically maintain a selected temperature of between 60 and 80 °F when the cabin air temperature selector switch is at AUTO. The cabin and cockpit air temperature rheostat is inoperative when the cabin air selector switch is at OFF, RAM, RAM & DUMP, or EMER PRESS or when the cabin air temperature selector switch is at any position other than AUTO.

2.1.4.5 Cabin and Cockpit Cooling Air Outlets. A cold-air outlet is on each side of the cockpit above the consoles and also in the cabin at each station. These "eyeball" outlets are adjustable spherical units that can be positioned to direct cold air from the air-conditioning system in any direction desired. A knurled ring around the opening can be rotated to regulate the volume of air. The ring is turned clockwise to decrease or counterclockwise to increase the outflow of air.

2.1.4.6 Cabin Pressure Controller. The cabin pressure controller (9, FO-2) is on the right console. A dial on the face of the controller displays altitude settings in thousands of feet. Cabin altitude may be selected by the CABIN ALT knob to the right of the dial. Rotating the knob moves a pointer on the dial and turns a plate within a window just below the dial. The pointer shows the selected cabin altitude, and the windowed plate shows maximum aircraft altitude, in thousands of feet, at which the cabin altitude can be maintained. When this aircraft altitude is exceeded, the selected cabin altitude will no longer be maintained, but will climb with aircraft altitude at 8.2 psi differential pressure to an aircraft altitude of 40,000 feet. From 40,000 to 45,000 feet, the differential will change to maintain the cabin altitude at 8,000 feet. The RATE knob, to the left of the dial, controls the rate of change of cabin altitude within a selectable pressure range of approximately 50 to 2,000 fpm. When the arrow on the knob is halfway between MIN and MAX (straight up), a range of approximately 500 fpm is obtained. Two methods may be used for pressurization control during climb.

1. Method No. 1 (nonconstant rate of change) — If the CABIN ALT knob is set at 1,000 feet, the altitude in the window will be 23,000 feet. After takeoff, the cabin altitude will climb to 1,000 feet at a rate of climb determined by the setting of the RATE knob. The differential pressure will

increase as the aircraft climbs, until at 23,000 feet the differential pressure is 8.2 psi. As the aircraft continues to climb above 23,000 feet, the cabin altitude will climb to maintain the 8.2 psi differential pressure. The RATE knob will not be effective. At 40,000 feet, the cabin altitude will be 8,000 feet and the pressure differential will be 8.2 psi. From 40,000 to 45,000 feet, the pressure differential will change to maintain the 8,000-foot cabin altitude.

2. Method No. 2 (constant rate of change) — The maximum aircraft altitude to be flown and the minimum time to reach that altitude should be determined. The CABIN ALT knob should be turned until the desired aircraft altitude appears in the window. The cabin altitude that will be maintained at the selected aircraft altitude is then indicated by the pointer. The rate of cabin altitude change should be determined and the RATE knob adjusted to that setting.

Note

Aircraft altitudes over 40,000 feet or cabin altitudes over 8,000 feet should not be selected. The cabin altitude is automatically maintained at 8,000 feet at aircraft altitudes in excess of 40,000 feet.

Example:

1. Takeoff elevation — Sea Level.
2. Maximum altitude to be flown — 40,000 Feet.
3. Time to climb — 20 Minutes.
4. Adjust the CABIN ALT knob so that 40 (40,000 feet) appears in the window.
5. The cabin altitude pointer is at 8 (8,000 feet).

$$\frac{\text{Cabin altitude to be flown (feet)}}{\text{Time to climb (minutes)}} = \text{Rate of cabin altitude change.}$$

or

$$\frac{8{,}000}{20} = 400.$$ Adjust rate of change 400 fpm.

With takeoff at sea level, the cabin altitude would increase at 400 fpm until at an aircraft altitude of 40,000 feet it would be 8,000 feet. The differential between cabin and atmospheric pressure would gradually increase from zero at sea level to 8.2 psi at 40,000 feet.

Note

Check cabin altitude rate of change with the cabin rate-of-change indicator. If the indicator is not showing the rate selected, the RATE knob should be reset.

Pressurization control during a typical descent is as follows: On the letdown, the CABIN ALT knob is set as required. The rate of cabin altitude change is determined and the RATE knob adjusted to that setting.

Example:

1. Assume an altitude of 40,000 feet (cabin altitude 8,000 feet) at start of letdown.
2. Desired cabin altitude — 1,000 Feet.
3. Time to descend — 20 Minutes.
4. Adjust the CABIN ALT knob so that the pointer is at 1 (1,000 feet).

$$\frac{\text{Cabin altitude minus field elevation}}{\text{Time to descend}} = \text{Rate of cabin altitude change.}$$

or

$$\frac{8{,}000 - 1{,}000}{20} = 350.$$ Adjust rate of change to 350 fpm.

As the aircraft descends, the cabin altitude will decrease at the selected rate-of-cabin-altitude change and will be at 1,000 feet by the time the aircraft reaches 1,000 feet.

Note

If a more rapid descent is desired, the RATE knob should be readjusted so that the cabin altitude will decrease at the new rate of descent.

2.1.5 Air-Conditioning and Pressurization System Indicators

2.1.5.1 Cabin Altimeter and Differential-Pressure Indicator. This indicator (64, FO-1) is vented both to outside air pressure and to pressure within the cabin and has two pointers, each traveling over a separate arc of calibrations. The OUTSIDE pointer, fixed to a rotating plate, shows flight pressure altitude. The CABIN pointer, fixed to a rotating band, shows both cabin altitude and differential pressure (between cabin and outside air) in pounds per square inch. The differential scale is visible through a cutout in the flight pressure altitude plate. During flight, the OUTSIDE pointer rotates with increasing altitude, exposing the differential scale. The highest figure exposed indicates the existing pressure differential.

2.1.5.2 Cabin Pressure Rate-of-Change Indicator. The rate of change in pressure altitude of the cockpit and cabin is indicated by the cabin rate-of change indicator (28, FO-1). This indicator functions similarly to the flight vertical velocity indicator, but is vented only to pressure within the cockpit.

2.1.5.3 Cabin Pressure Failure Warning Light. This placard type red warning light on the caution-warning light panel (12, Figure 2-33) is powered by the dc essential bus. The light comes on to read CABIN PRESS FAIL when the cabin altitude rises to 10,000 feet. If the cabin pressure altitude drops below 10,000 feet, the warning light goes out. Cabin altitude should be verified with the cabin altimeter and differential pressure indicator.

2.1.5.4 Cabin Air Hot Caution Light. This placard-type amber caution light on the caution-warning light panel (12, Figure 2-33) is powered by the dc essential bus. The light comes on to read CABIN AIR HOT when the temperature of the air in the distribution ducting upstream of the cabin and cockpit air outlets exceeds 265 °F. If light comes on during normal operation of the pressurization system, move cabin air temperature selector switch to MAN COLD. If the light goes out, this indicates a failure of the cabin air temperature control and the flight can continue, using manual temperature control. If the light stays on, this indicates a failure of the hot-air mixing valve. Place cabin air selector switch at EMER PRESS and continue flight using emergency pressurization system. When the EMER PRESS position is selected, compressor bleed air from the right-hand engine is supplied directly to the air outlet distribution ducts. Therefore, the cabin air hot caution light probably will come on soon after the emergency pressurization system is engaged. A reduction of the power setting of the right-hand engine during emergency system operation will lower the temperature of the incoming air so that the caution light will go out.

Note

With the cabin air selector switch at OFF or RAM, the air-conditioning and pressurization system is inoperative and cabin pressure will bleed off.

2.1.5.5 Aft Fuselage Hot Caution Light. This placard-type amber caution light on the caution-warning light panel (12, Figure 2-35) is powered by the dc essential bus. The light comes on to read AFT FUS HOT when there is an overtemperature condition (above 275 °F) in the vicinity of the bleed-air pressurization ducts in the aft fuselage.

2.1.5.6 Door-Open Caution Light. Refer to paragraph 2.11, ENTRY DOOR.

2.1.6 Recommended Operating Procedures for Pressurization System. The following procedure is useful in reducing cabin pressurization surges to a minimum:

1. Open sliding window.
2. Position CABIN AIR selector switch to OFF.
3. Set cabin pressure controller to approximately 250 feet above field elevation and rate-of-change knob as desired.
4. With engines at IDLE for at least 15 seconds, turn CABIN AIR switch to LH ENG or RH ENG.
5. Hold CABIN TEMP selector switch to MAN COLD, allowing sufficient time for hot air mixing to close, and release to OFF.

Note

At idle power, this provides the minimum airflow to the cabin. Supply temperatures normally are approximately 50 °F on a cool day. At high engine powers, the airflow increases and the temperature will decrease. If the temperature is felt to be cold, the manual control may be modulated so that the supply airflow is warmed.

6. Close sliding window (throttles at IDLE).

7. Move throttles at reasonable rate when taxiing and for takeoff.

8. After takeoff, place CABIN AIR selector switch to BOTH ENG and position cabin temperature selector switch to AUTO.

9. Set cabin pressure controller for desired cabin altitude and rate of change (the preferred rate is 300 fpm).

10. When making throttle changes, it may become necessary to adjust cabin altitude selection and rate selection.

Note

Decreasing engine power will tend to increase the cabin altitude. When decreasing power, lower cabin altitude and increase the rate to 1,000 fpm or more. When increasing power, the action is converse.

11. Program the aircraft descent so that the cabin can descend to the field altitude (the preferred rate is 300 fpm).

2.2 ANTI-ICING SYSTEMS

Note

Anti-icing systems are not to be used as deicing systems. An anti-icing system shall be turned on before areas of known icing conditions (visible moisture) are entered.

Anti-icing is accomplished either electrically or with final-stage engine compressor air. (See Figure 2-3.) Engine compressor air (about 260 °C) (500 °F)) is used for anti-icing the engine air inlet duct, engine guide vanes, engine air inlet duct flow-straightener vanes, engine nose fairing, and heat and vent water separator downstream of the cooling turbine. Electrical heaters are used to prevent ice formation on the pitot heads, ram-air inlet, and engine pressure ratio heads on the dorsal fin and the stall warning vanes. The windshield is electrically heated for defogging, birdproofing, and anti-icing. Switches to control these systems are on an overhead panel in the cockpit.

2.2.1 Engine Inlet Duct and Guide Vane Anti-Icing System. The inlet duct, each engine guide vane, and the engine nose fairing are protected from ice by hot engine compressor air that is routed internally. The compressor air passes through a pressure-limiting shutoff valve that is controlled by a switch in the cockpit.

2.2.1.1 Engine Inlet Anti-Icing Switch. The two-position ENGINE INLET switch on the anti-icing control panel (24, FO-3) is powered by the dc essential bus. Placing the switch at ON energizes the normally closed inlet anti-icing shutoff valve (one on each engine) open and deenergized the normally open inlet guide vane anti-icing valve (one on each engine). Compressor air then flows to the leading edge of the engine inlet duct, splitter vanes, bullet nose, and guide vanes of both engines. Placing the switch at OFF shuts off all engine anti-icing.

CAUTION

For ground operational checks, do not operate the engine inlet anti-icing system for extended periods. The hot compressor bleed air used for engine inlet anti-icing may cause structural damage to the inlet duct while the aircraft is on the ground.

2.2.1.2 Engine Anti-Icing Failure Lights. Two placard-type engine anti-icing failure amber caution lights (12, Figure 2-33), one for each engine, are powered by the dc essential bus. The lights come on to read LH ENG DE-ICE FAIL or RH ENG DE-ICE FAIL whenever either of the respective engine anti-icing valves does not correspond to the position selected by the engine inlet anti-icing switch. If a light comes on with the engine inlet anti-icing switch at ON, this indicates that either or both of the respective engine anti-icing valves are closed. There is no corrective action for this condition; however, the aircraft altitude and/or course should be changed to avoid further icing.

Note

Insufficient bleed air at low-power setting may also cause a light to come on. Maintain at least 1.5 EPR to ensure engine anti-icing.

If a light comes on with the switch at OFF, this indicates that either or both of the respective engine anti-ice valves are open. Normal engine anti-icing is still available. If an engine anti-icing failure light stays on after the switch has been placed at OFF, one of the valves has not completely closed and heat is still available to the engine. As soon as the valve closes, the light will go out.

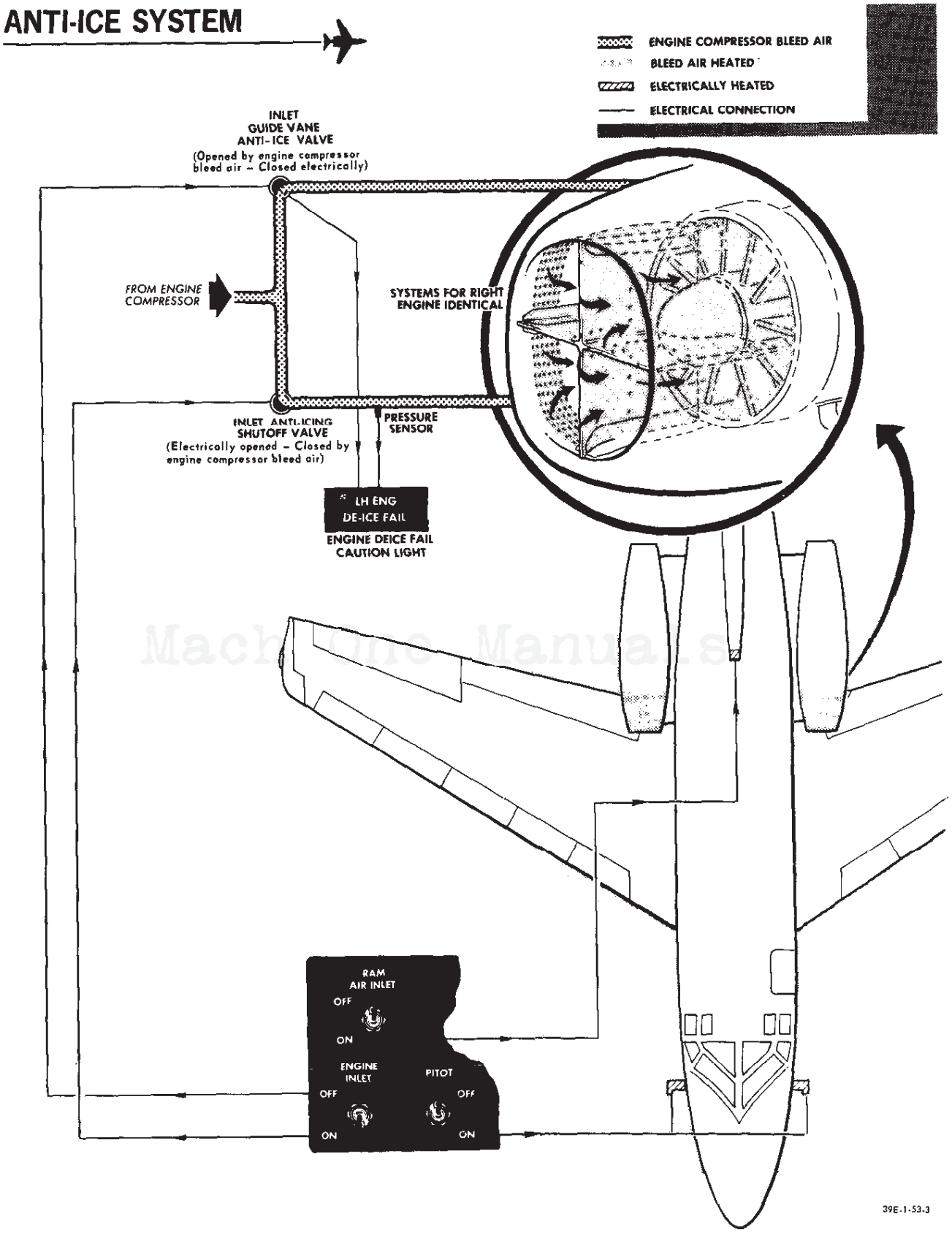

Figure 2-3. Anti-Icing System

CAUTION

Do not operate the engine inlet anti-icing system on the ground for extended periods of time. The hot compressor bleed-air used for engine inlet anti-icing may cause structural damage to the inlet duct.

2.2.2 Ram-Air Inlet Anti-Icing System. The ram-air inlet in the dorsal fin is protected from icing by electrical heaters that are controlled by a switch in the cockpit. The system is designed as an anti-icing system, not as a deice system, and must be turned on before areas of known icing conditions (visible moisture) are entered. It should not be turned on unless the ambient air temperature is between 41 and 23 °F (+5 and -5 °C). The ram-air inlet heater is powered by the dc secondary bus. The dorsal fin ram-air inlet heater will operate on the ground if turned on, but only at a fraction of full heat.

2.2.2.1 Ram-Air Inlet Anti-Icing Switch. The two-position RAM AIR INLET switch (FO-3) is powered by the dc essential bus. Moving the switch to ON electrically heats a spraymat element on the ram-air inlet on the dorsal fin. The ram-air inlet heater will operate at a reduced rate on the ground if the switch is at ON. When airborne, full heat will be available for the ram-air inle heater. Moving the switch to OFF shuts off the ram-air inlet heater.

2.2.3 Windshield Anti-Icing System (CT-39E). The windshields are heated by power provided from the two engine-driven, variable-frequency windshield ac generators and controlled by the dc essential bus through the windshield temperature control box. (See Figure 2-4.) Automatic heating of both windshields is controlled by a switch on the anti-icing control panel. Caution lights warn of generator failure and generator and windshield overheat conditions. A windshield generator switch is provided for each generator and can be used to take the affected generator off the line in case of a system malfunction.

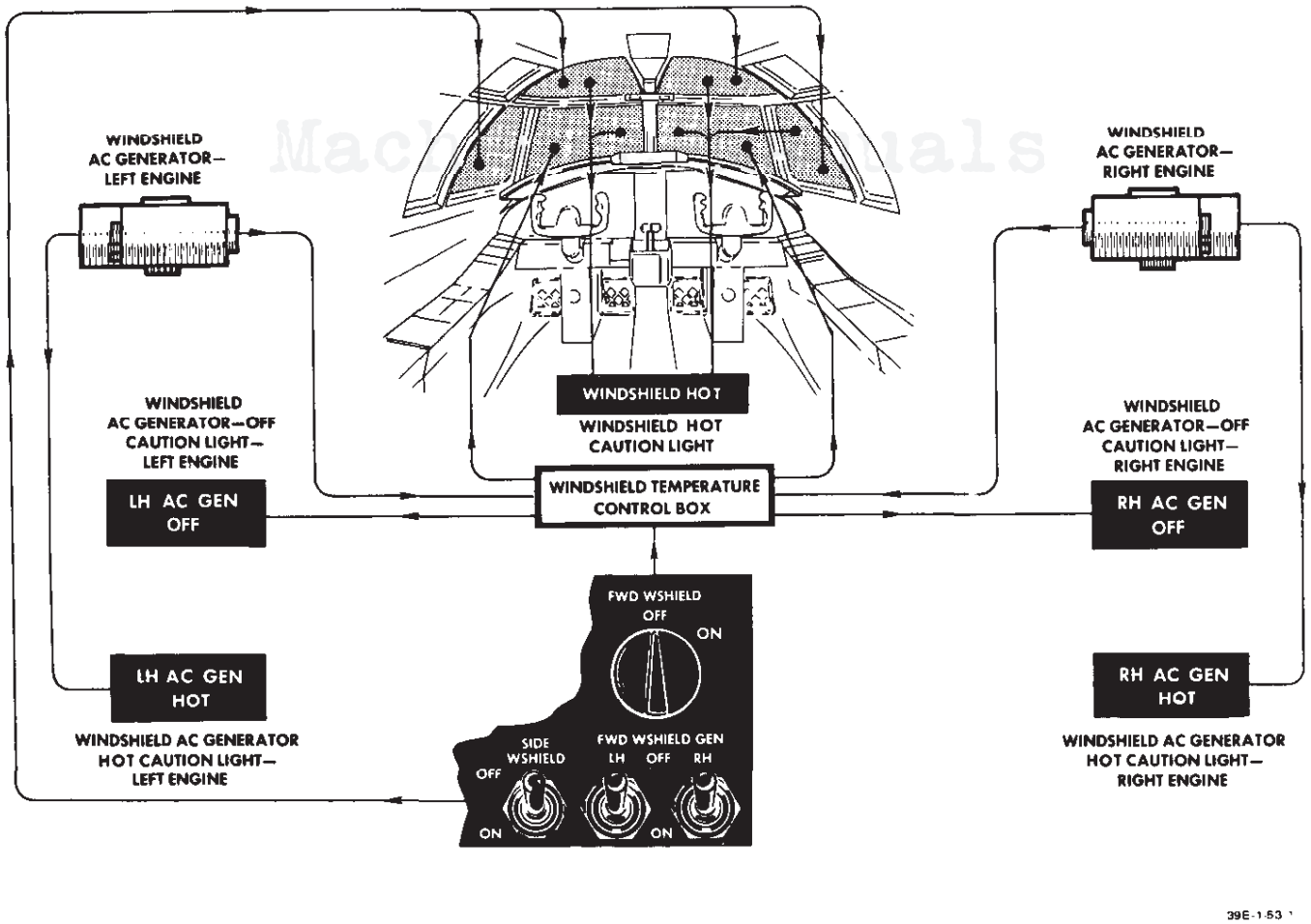

Figure 2-4. Windshield Anti-Icing System (CT-39E)

Note

If either engine-driven dc generator fails or is turned off, the respective windshield ac generator will be inoperative.

The pilot sliding window, copilot side window, and both overhead windows are heated by dc secondary bus power. System operation is controlled by a switch on the anti-icing control panel. The windshield hot caution light will also come on if these glass areas (except the pilot sliding window) are overheated. An overheat switch opens in case of an overheated condition.

2.2.3.1 Windshield AC Generator Switches. The two FWD WSHIELD GEN switches on the anti-icing control panel (FO-3) have two positions: ON AND OFF. Placing the switches at ON permits ac power from both generators to energize the windshield anti-icing system. Both switches should be at ON after engine start. Should a malfunction occur in either generator circuit, the affected generator can be disabled by moving the respective generator switch to OFF. When there is no output from one ac generator, the respective windshield will be heated by the other ac generator at a reduced rate if the aircraft has been modified by Service Bulletin 68-6. Otherwise, it will be unheated. Both switches should be placed at OFF before engine shutdown.

2.2.3.2 Forward Windshield Anti-ice Switch. The rotary-type, two-position FWD WSHIELD anti-icing switch on the anti-icing control panel (FO-3) has two positions: ON and OFF. Placing the switch at ON permits the windshield temperature control unit to sense and control the temperature of both windshields. The switch should remain in this position during flight. When there is no output from one ac generator, the respective windshield will be unheated. Placing the switch at OFF shuts off all power to the windshield heaters.

CAUTION

Turn the windshield anti-icing system on before takeoff to ensure birdproofing of the windshields.

2.2.3.3 Side Windshield Anti-Icing Switch. The two-position SIDE WSHIELD anti-icing switch on the anti-icing control panel (FO-3) is powered by the dc essential bus. Placing the switch at ON heats the pilot sliding window, copilot side window, and both overhead windows with dc secondary bus power. The windshield hot caution light comes on if these glass areas (except the pilot sliding window) are overheated.

2.2.3.4 Windshield Hot Caution Light. This placard-type amber caution light on the caution-warning light panel (12, Figure 2-33) is powered by the dc essential bus. The light comes on to read WINDSHIELD HOT whenever an overheat condition exists in the windshields, copilot side window, or either of the overhead windows.

Note

If the forward windshields have been exposed to solar heating, the windshield hot caution light may come on when the dc essential bus is energized regardless of the position of the windshield anti-ice switch. If the copilot side window or the overhead windows have been exposed to solar heating, the windshield hot caution light may come on immediately after the side and overhead windshield anti-ice switch is placed at ON. If either condition occurs, corrective action is not necessary and the windshield anti-ice system may be operated as required.

2.2.3.5 Ac Generator Hot Caution Lights. Two placard-type amber caution lights, one for each windshield ac generator, on the caution-warning light panel (12, Figure 2-33) are powered by the dc essential bus. The lights come on to read RH AC GEN HOT or LH AC GEN HOT when the respective windshield ac generator is overheated.

2.2.3.6 Ac Generator-Off Caution Lights. Two placard-type amber caution lights, one for each windshield ac generator, on the caution-warning light panel (12, Figure 2-33) are powered by the dc essential bus. The lights come on to read RH AC GEN OFF or LH AC GEN OFF. After the out-of-temperature range trigger circuit has been armed, these lights come on whenever the temperature of the respective windshield has gone outside the temperature control band. The lights will also come on if the respective ac generator switch is at OFF, if the respective ac generator has failed, or when the respective dc generator is off the line. When the ac generator is turned off in flight, the light will not come on until the windshield temperature has dropped below the temperature control band. Except for one condition, when a light comes on, the respective forward windshield ac generator switch should be placed at OFF. The one exception is during flight operations conducted under heavy precipitation at low ambient temperatures.

This condition imposes the greatest thermal demand on the windshield anti-icing system and ac generator output may not be able to maintain windshield temperatures within the control band. As the temperature in any one windshield goes below the low-temperature trigger circuit, the ac generator for that windshield is automatically taken off the line and the respective ac generator-off caution light comes on. The windshield anti-icing system should be reactivated by momentarily, placing the rotary forward windshield anti-icing switch at OFF, and then at ON. However, if either windshield is unheated or suspected of being cold, the rotary switch should be left at OFF for 3 minutes to allow the controller to cool and provide power-limiting protection. Windshield heat will be below the control band until ambient conditions allow temperature rise; at that time, the out-of-temperature trigger circuit will become armed.

2.2.3.7 Windshield Anti-Icing System Operation. Figure 2-5 is a guide for operation of the windshield anti-icing system.

2.2.4 Windshield Anti-Icing System (CT-39G). The single-element forward windshields, heated by ac power from the No. 2 inverter or the instrument inverter, are controlled by dc essential bus power through the left and right windshield controllers. Normal system power is supplied by the No. 2 inverter; emergency power is supplied by the instrument inverter. Selective heating of both forward windshields is controlled by a switch on the anti-ice control panel. A switch for testing the overheat sensing circuit is also on the anti-ice control panel. Caution lights warn when either forward windshield is overheated or when the power to either forward windshield is off.

The pilot sliding window, copilot side window, and both overhead windows are heated by dc secondary bus power. System operation is controlled by a switch on the anti-ice control panel. A caution light will come on if these glass areas (except the pilot sliding window) are overheated. An overheat switch opens in case of an overheated condition. The windshield anti-icing system is shown schematically in Figure 2-6.

2.2.4.1 Forward Windshield Anti-Ice Switch. Selective heating of the forward windshields is controlled by the rotary-type FWD WSHIELD anti-ice switch (Figure 2-6) on the anti-ice control panel. The switch is powered by the dc essential bus and has four positions: OFF, BOTH, LH, and RH. With the INVERTER switch at No. 1 and both dc generators operating, placing the FWD WSHIELD switch at BOTH, LH, or RH energizes the No. 2 inverter power relay to connect dc essential bus power to the No. 2 inverter and energizes the No. 2 inverter delta-wye relay to connect the No. 2 inverter to the windshield heating system. During a time delay of 75 to 120 seconds to prevent thermal shock to the windshields, the windshield controllers gradually increase No. 2 inverter power to the windshield heating elements of the selected windshield.

If the No. 1 inverter fails or if the inverter switch is placed at No. 2, the No. 2 inverter delta-wye relay will be deenergized to remove the No. 2 inverter from the windshield heating system. Also, the instrument inverter power relay and instrument inverter delta-wye relay will be energized to connect instrument inverter power to the windshield heating system. When the instrument inverter is powering the windshield heating system, electrical power is distributed to one windshield only. A stepdown transformer reduces the power to the windshield to one-half that normally provided. With the windshield anti-ice switch at BOTH, windshield heating is normally provided for both windshields, but if the instrument inverter is powering the windshield heating system, only the left windshield is heated. Regardless of which inverter is powering the windshield heating system, with the windshield anti-ice switch positioned at LH or RH windshield heating is provided only for the selected windshield. Placing the switch at OFF shuts off all power to the windshield heating system.

CAUTION

Turn the windshield anti-icing system on before takeoff to ensure birdproofing of the windshields.

During normal operating conditions (No. 1 inverter and windshield heat selected) in the event of a dc generator failure, the No. 2 inverter will automatically be shut down and the instrument inverter will then provide one-half of the normal power to the pilot windshield or to the copilot windshield if selected. During this condition, if the No. 1 inverter should fail, the No. 2 inverter will automatically power the 115-vac primary bus.

2.2.4.2 Windshield Overheat Test Switch. The pushbutton-type windshield overheat test switch on the anti-ice control panel (Figure 2-6) is used to test the windshield overheat system. Pressing the test switch will illuminate the WINDSHIELD HOT - LH and

	CONDITION	CAUSE	CORRECTIVE ACTION
1.	WINDSHIELD HOT caution light on immediately when FWD WSHIELD switch is placed in ON or SIDE WSHIELD switch is placed ON.	Solar heating.	No corrective action required.
2.	WINDSHIELD HOT caution light comes on in flight.	Windshield overheating.	a. Turn FWD WSHIELD switch to OFF. If light stays on, turn FWD WSHIELD switch ON and SIDE WSHIELD switch to OFF. b. If light goes out with FWD WSHIELD switch at OFF, turn both AC generator switches off and FWD WSHIELD to ON. After a cooling period the light will go out. After the light is out, turn FWD WSHIELD switch to OFF, turn either ac generator switch to ON and FWD WSHIELD switch to ON; if light stays out, leave switches as is; if light comes on, turn off the two switches, turn on opposite AC generator switch, and again place the FWD WSHIELD switch at ON.[1]
3.	After the engines are started and the FWD WSHIELD switch is placed at ON, the LH or RH AC GEN OFF caution light illuminates immediately.	System failure.	Turn windshield heating off and investigate.[1] Turn respective ac generator switch OFF.
4.	LH or RH AC GEN OFF caution light on during taxi.	AC generators limited in output below 58% engine rpm.	Turn FWD WSHIELD switch to OFF[1] for 2 minutes, then back to ON.
5.	LH and/or RH AC GEN OFF caution light comes on in cruising flight.	Generator, controller, or windshield malfunction or excessive heating demands induced by precipitation.	Check heat by touch. Leave FWD WSHIELD switch at ON. Turn respective ac generator switch to OFF.[1] Investigate after landing.
6.	LH or RH AC GEN OFF caution light comes on in low rpm descent.	Generators are limited in output below 58% engine rpm.	Reset by turning FWD WSHIELD switch to OFF then back to ON.[1]
7.	Distortion or bubbling of either windshield.	Windshield heat control or windshield has failed.	Turn off respective ac generator switch.[1]

[1] If the windshield heat is inoperative for either or both forward windshields, do not exceed 250 knots IAS below 10,000 feet altitude.

Figure 2-5. Windshield Anti-Icing Operation (CT-39E)

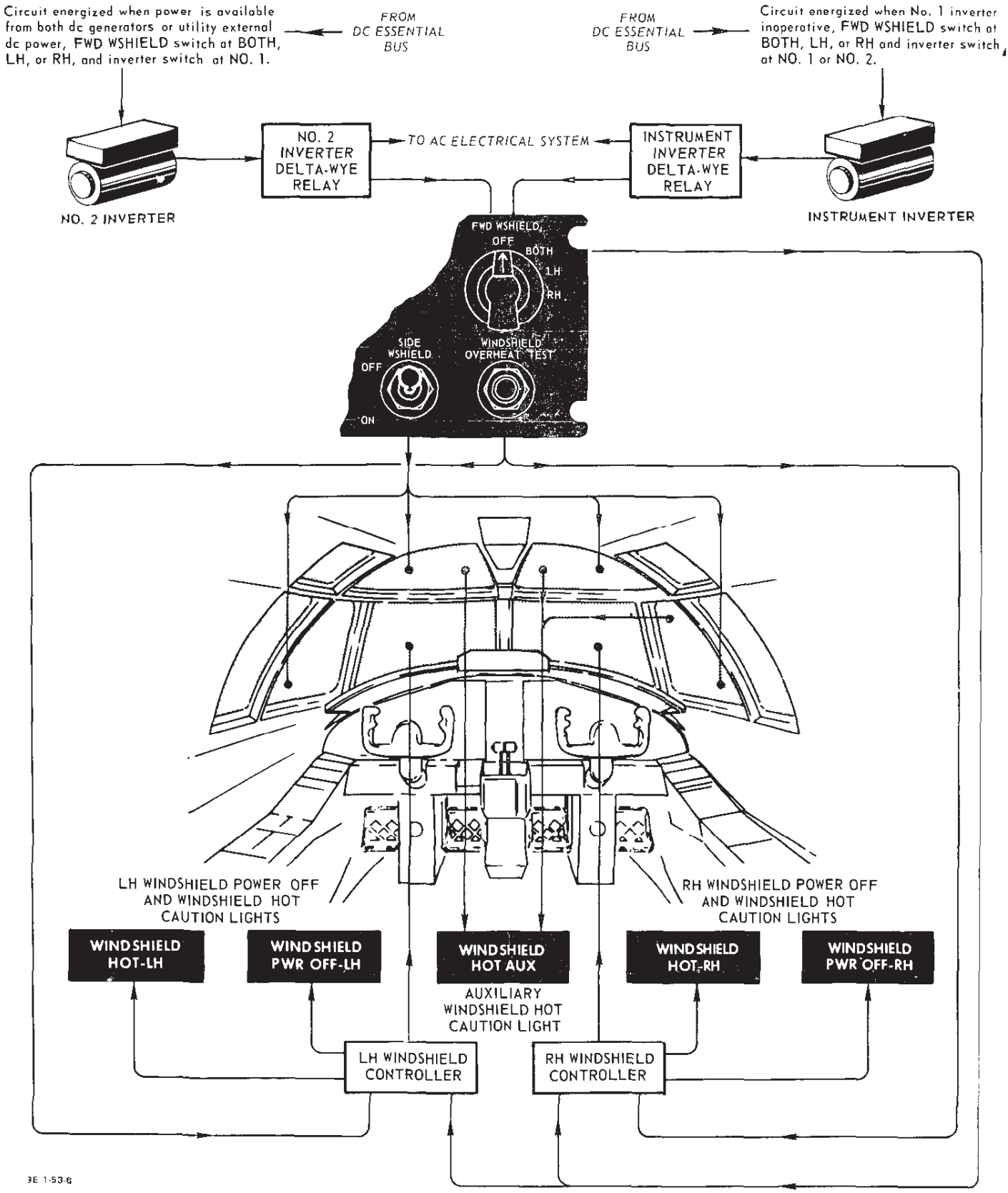

Figure 2-6. Windshield Anti-Icing System (CT-39G)

WINDSHIELD HOT - RH caution lights if the overheat sensing circuits are in proper operating condition. The overheat test switch is powered by the dc essential bus.

2.2.4.3 Side Windshield Anti-Ice Switch. The two-position SIDE WSHIELD anti-ice switch on the anti-ice control panel (Figure 2-6) is powered by the dc essential bus. Placing the switch at ON heats the pilot sliding window, copilot side window, and both overhead windows with dc secondary bus power.

2.2.4.4 Windshield Power-Off Caution Lights. Two placard-type amber caution lights, one for each windshield, on the caution warning light panel (12, Figure 2-33) are powered by the dc essential bus. The lights come on to read WINDSHIELD PWR OFF - LH or WINDSHIELD PWR OFF - RH when a failure of the respective ac power input occurs.

2.2.4.5 Windshield Hot Caution Lights. Two placard-type amber caution lights, one for each windshield, on the caution warning light panel (12, Figure 2-33) are powered by the dc essential bus. The lights come on to read WINDSHIELD HOT - LH or WINDSHIELD HOT - RH when an overheat condition exists in the respective windshield.

2.2.4.6 Auxiliary Windshield Hot Caution Light. This placard-type amber caution light on the caution-warning light panel (12, Figure 2-33) is powered by the dc essential bus. The light comes on to read WINDSHIELD HOT - AUX whenever an overheat condition exists in the copilot side window, or either of the overhead windows.

2.2.4.7 Windshield Anti-Icing System Operation. Figure 2-7 is a guide for operation of the system.

2.2.5 Heaters. Electrical heaters are provided for the airspeed pitot heads, engine pressure ratio pitot heads, and stall warning vanes.

2.2.5.1 Pitot Heaters. The right pitot heater is powered by the dc secondary bus; the left pitot heater is powered by the dc essential bus. Ground operation is controlled by a switch on the anti-icing control panel.

2.2.5.2 Engine Pressure Ratio Pitot Heaters. The engine pressure ratio pitot heaters are powered by the dc paralleling bus. Ground operation is controlled by a switch on the anti-icing control panel.

2.2.5.3 Stall Warning Vane Heaters. The stall warning vane heaters are powered by the dc essential bus. Ground operation is controlled by a switch on the anti-icing control panel.

2.2.5.4 Pitot and Stall Warning Anti-Icing Switch. This two-position PITOT & STALL warning switch is on the anti-icing control panel (FO-3). Placing this switch at ON when the weight of the aircraft is on the landing gear turns on the pitot heaters, engine pressure ratio pitot heaters, and partial heat to stall warning vane heaters. Placing the switch at OFF shuts off all power to the heaters. When the weight of the aircraft is removed from the landing gear, the heaters operate continuously, regardless of the switch position as long as one generator is on line. If both dc generators fail or are off the line and the pitot heat switch is off, power is not available to the left and right pitot heaters and the EPR probes. Placing the pitot heat switch to ON restores power to the left pitot heater. The stall warning vanes are always heated in flight as long as the essential dc bus is powered.

2.2.6 Pitot Heat Monitoring System. For aircraft with AFC 627 installed, an indicator caution light on the copilot instrument panel reads L Pitot heat on, R Pitot heat on. These lights will illuminate when the pitot switch is on and there is power to the appropriate pitot tube. The system is powered by the secondary dc bus. R Pitot heat is inoperative on deck with only one generator.

2.2.7 Wing Ice-Check Light Switch. The wing ice-check lights may be used to check for the presence of ice on the wings. Refer to paragraph 2.22, LIGHTING EQUIPMENT.

2.2.8 Fuel Filter Deicing System. Refer to paragraph 2.13, FUEL HEATER SYSTEM.

2.3 COMMUNICATION AND ASSOCIATED ELECTRONIC EQUIPMENT

2.3.1 Table of Communication and Associated Electronic Equipment. See Figure 2-8. See Figure 2-9 for antenna locations.

2.3.2 Audio System. The pilot and copilot each have a 346B-3 audio system that combines interphone communications and audio facilities of the airborne communication and navigation equipment. Each system includes audio amplifier, control panel, control wheel interphone-microphone switch, and speaker. Two additional speakers are in the cabin. Three separate jacks — one for a hand microphone, one for oxygen mask microphone, and one for the headphones — are above each side console. The system amplifies audio signals to speaker and headphone level and permits selective reception on all available frequencies and transmission

	CONDITION	CAUSE	CORRECTION ACTION
1.	WINDSHIELD HOT - LH or WINDSHIELD HOT - RH caution light on immediately when FWD WSHIELD switch is placed at BOTH, LH, or RH.	Solar heating.	No corrective action required.
2.	WINDSHIELD HOT - AUX caution light on immediately when SIDE WSHIELD switch is placed at ON.	Solar heating.	No corrective action required.
3.	WINDSHIELD HOT - LH or WINDSHIELD HOT - RH caution light cycles on and off intermittently.	Left or right forward windshield normal control system has failed. (Regulating approximately 5° F above normal.)	No corrective action required in flight. Investigate after landing.
4.	WINDSHIELD HOT - LH or WINDSHIELD HOT - RH caution light comes on steady in flight.	Left or right forward windshield overheating.	If LH light comes on steady, turn FWD WSHIELD switch to RH;[1] if RH light comes on steady, turn FWD WSHIELD switch to LH;[1] if both lights come on steady, turn FWD WSHIELD switch to OFF.[1]
5.	WINDSHIELD PWR OFF - LH or WINDSHIELD PWR OFF - RH caution light comes on.	Failure of respective ac power input.	If LH or RH light comes on and it is desirable that heat be restored to that particular windshield, place FWD WSHIELD switch to position that corresponds to light.[1] If light does not go out, select other windshield.[1]
6.	WINDSHIELD HOT - AUX caution light comes on in flight.	Left or right overhead windshield or copilot's side windshield overheating.	Turn SIDE WSHIELD switch OFF.

[1] If the windshield heat is inoperative for either or both windshields, do not exceed 250 knots IAS below 10,000 feet altitude.

Figure 2-7. Windshield Anti-Icing Operation (CT-39G)

on HF, VHF, and UHF frequencies. The audio system is also connected electronically to the landing gear and Mach-airspeed warning systems. The audio system is powered by the dc essential bus.

2.3.2.1 Audio System Control Panel. Identical audio system control panels (Figure 2-10) are on the pilot and copilot consoles. Mixer switches labeled VOR, TAC, MKR, VHF, UHF, HF, ADF-R, and ADF-V control the mixed signal facility of the audio amplifier. Any or all of these switches may be moved from the center OFF position, forward to SPEAKER, or aft to PHONE. Thus, the selected signal or any combination of eight receiver signals may be received through the speaker or in the headphones. There are two volume knobs on the panel: one labeled PHONE for controlling volume in the headphones and the other labeled SPEAKER for controlling volume through the speaker. Any of three microphone selector buttons labeled VHF, UHF, and HF can be pressed to connect the desired channel for transmission. A fourth microphone selector button labeled PA can be pressed to connect the cabin speakers into the interphone system. The microphone selector buttons are interconnected so that only one can be pressed at a time. Moving the AUTO AUDIO switch from OFF to SPEAKER or PHONE permits reception (by speaker or headphones) on the frequency selected by a microphone

TYPE	DESIGNATION	USE	OPERATOR	RANGE	LOCATION OF CONTROLS
UHF RADIO	AN/ARC-51AX AN/ARC-159	TWO-WAY VOICE COMMUNICATION	PILOT OR COPILOT	LINE OF SIGHT	CENTER CONSOLE
HF AM RADIO	SUNAIR ASB-100 COLLINS 718U-5	TWO-WAY VOICE COMMUNICATION	PILOT OR COPILOT	UP TO 2500 MILES	CENTER INSTRUMENT PANEL CENTER PEDESTAL
VHF RADIO	COLLINS 618M-1A 618M-2D VHF-20B	TWO-WAY VOICE COMMUNICATION	PILOT OR COPILOT	LINE OF SIGHT	INSTRUMENT PANEL SHROUD
AUDIO SYSTEM	COLLINS 346B-3	INTERSTATION COMMUNICATION AND RADIO SIGNAL AMPLIFICATION	PILOT OR COPILOT	WITHIN THE AIRCRAFT	PILOT'S AND COPILOT'S SIDE CONSOLE
TACAN NAVIGATION	AN/ARN-91 SIERRA SAN-7000	TACAN STATION BEARING AND DISTANCE	PILOT OR COPILOT	LINE OF SIGHT UP TO 300 NAUTICAL MILES	CENTER CONSOLE INSTRUMENT PANEL SHROUD
VOR/ILS/DME (VHF NAVIGATION)	COLLINS 51RV-1 VIR-30A DME-40	VOR, LOCALIZER, GLIDE SLOPE AND RANGE INFORMATION	PILOT OR COPILOT	LOCALIZER 45 MILES, OMNI 300 MILES, DEPENDING ON ALTITUDE	INSTRUMENT PANEL SHROUD
LF ADF NAVIGATION	COLLINS DF-203 DF-206 ADF-60A	AUTOMATIC AND MANUAL BEARING TO ANY RADIO STATION AND AURAL CODE AND VOICE RECEPTION	PILOT OR COPILOT	20-200 MILES DEPENDING ON FREQUENCY AND TIME OF DAY	CENTER INSTRUMENT PANEL
DIRECTIONAL INDICATING SYSTEM	COLLINS MC-102 MC-103	HEADING INFORMATION	PILOT OR COPILOT	--	PILOT'S AND COPILOT'S OUTBOARD INSTRUMENT PANELS
ATC TRANSPONDER	COLLINS 621A-3 TDR-90	RESPONDS TO ATC GROUND RADAR INTERROGATIONS FOR AIRCRAFT IDENTIFICATION AND AUTOMATIC ALTITUDE REPORTING	PILOT OR COPILOT	LINE OF SIGHT UP TO 200 NAUTICAL MILES	INSTRUMENT PANEL SHROUD
MARKER BEACON	COLLINS 51Z-4 PART OF VIR-30A	RECEIVES MARKER SIGNALS	AUTOMATIC	VERTICAL DISTANCE BETWEEN AIRCRAFT AND GROUND	INSTRUMENT PANEL SHROUD
WEATHER RADAR	PRI-400	WEATHER AND GROUND MAPPING	PILOT OR COPILOT	300 MILES	CENTER PEDESTAL

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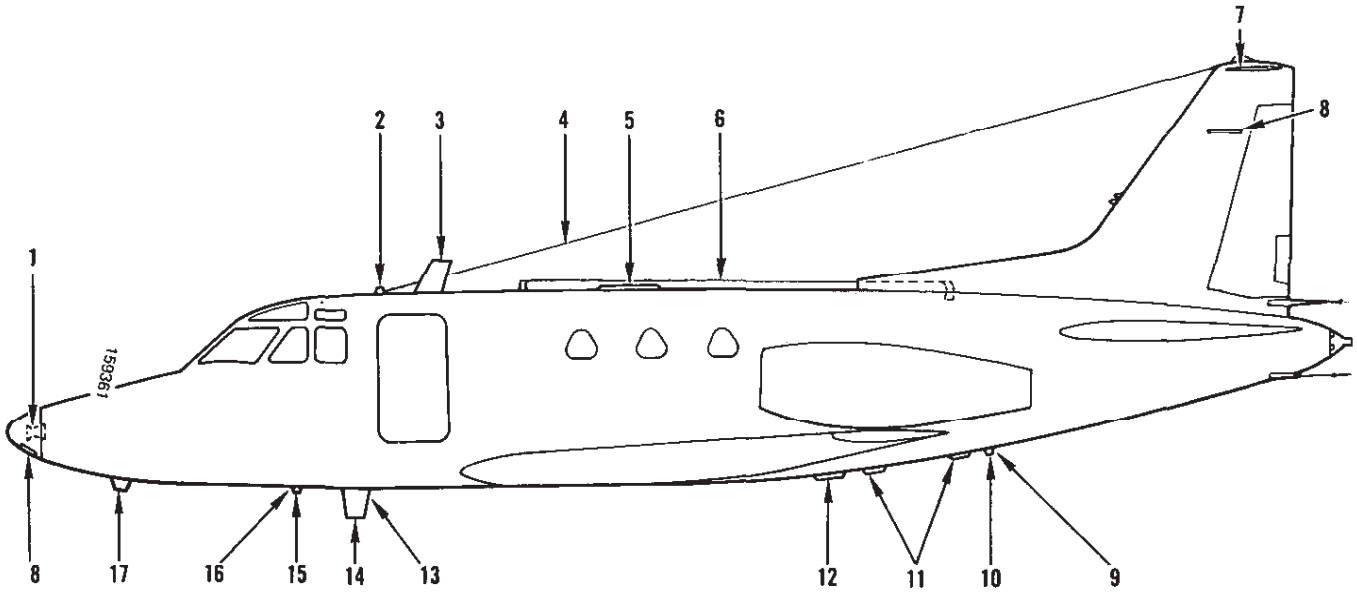
Figure 2-8. Communication, Navigation, and Associated Electronic Equipment



1. Weather Radar
2. TACAN
3. VHF Comm
4. HF Comm
5. ADF Loop
6. ADF Sense
7. VOR Localizer 2,3
8. VOR Localizer 1
9. ATC Transponder 2, 3
10. TACAN 1
11. Radio Altimeter
12. Marker Beacon
13. VHF and UHF Comm 2
14. TACAN 3
15. TACAN 2
16. DME 1
17. ATC Transponder 1
18. Glide Slope

1 CT-39G, BUNO 160053 thru 160057
2 CT-39E, BUNO 157352 thru 157354 & 158380 thru 158383 & CT-39G, BUNO 158843 and 158844.
3 CT-39G, BUNO 159362 thru 159365.

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Figure 2-9. Radio and Radar Antenna Locations

selector button regardless of the positions of the mixer switches.

2.3.2.2 Interphone-Microphone Switch. This rocker-type switch (Figure 2-29) on the pilot and copilot control wheels is used in conjunction with an oxygen mask microphone. Interphone communication or radio is initiated by moving the switch to INTER or MIC, respectively.

2.3.2.3 Operation of Audio System. For reception, transmission, and interphone communication, proceed as follows:

1. Mixer switches — Desired Receiver at SPEAKER or PHONE.

2. SPEAKER or PHONE volume knob — Comfortable Level.

3. Microphone selector button — VHF, UHF, or HF for Broadcast. PA for Cabin Speakers.

4. AUTO AUDIO switch — As Desired.

5. Interphone-microphone switch — INTER or MIC.

   or

6. Hand microphone button — Press.

   Release button when transmission completed to permit reception.

2.3.3 High-Frequency Single-Sideband Communication System (Aircraft With SUNAIR ASB-100 Installed). The HF SSB ASB-100 communication system provides long-range, two-way voice communication on 10 preset, single- or double-frequency simplex channels between 2.00 and 18.00 MHz. The audio signal is directed through the audio system, and system power is received from the dc essential bus.

2.3.3.1 HF SSB Communication System Control Panel. All controls and indicators for operation of the HF SSB communication system are on a control panel (Figure 2-11) on the center instrument panel.

2.3.3.2 Volume Control Knob. The VOL control knob (6, Figure 2-11) controls power to the system and regulates the audio level. Clockwise rotation of the knob applies power to the system and increases the audio level.

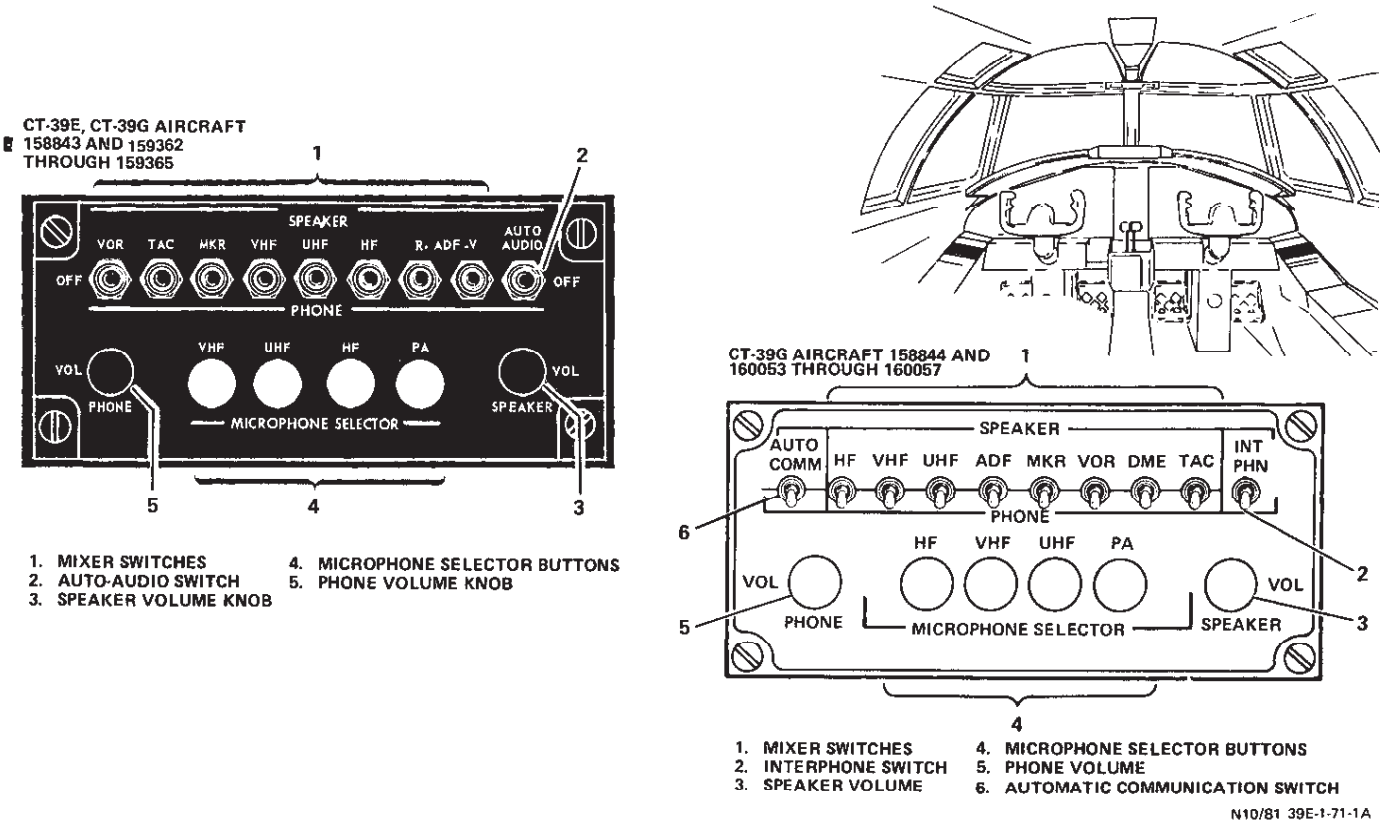

Figure 2-10. Audio System Control Panel

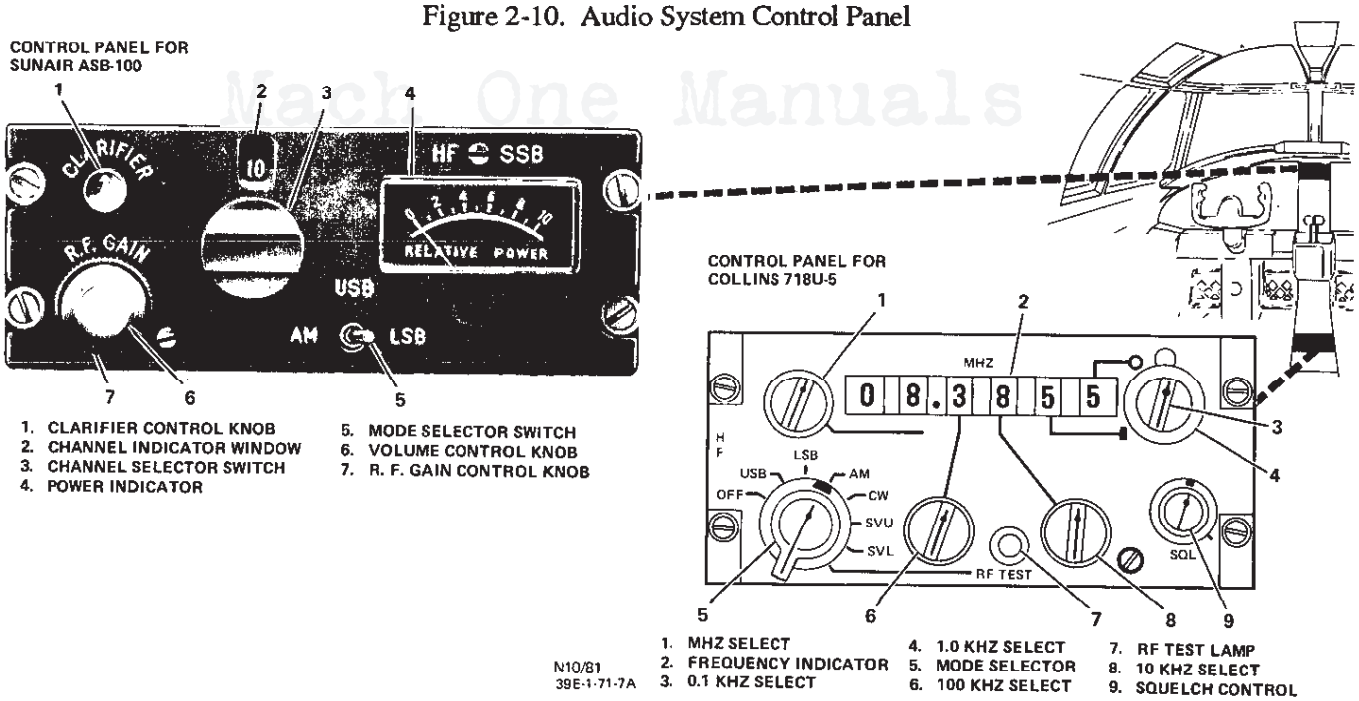

Figure 2-11. HF SSB Communication System Control Panel

2.3.3.3 R.F. GAIN Control Knob. The R. F. GAIN control knob (7, Figure 2-11) provides a means of adjusting the receiver sensitivity to the desired level.

2.3.3.4 CLARIFIER Control Knob. The CLARIFIER control knob (1, Figure 2-11) provides a means of obtaining natural-voice-quality audio.

2.3.3.5 Channel Selector Switch. The channel selector switch (3, Figure 2-11) is used to select the desired preset channel. The selected channel will appear in the channel indicator window (2, Figure 2-11). Each channel with its related frequency is listed on an HF SSB frequency indicator card in the cockpit.

2.3.3.6 Mode Selector Switch. The three-position mode selector switch (5, Figure 2-11) is used to select the desired mode of operation. Available modes are USB for upper sideband operation, LSB for lower sideband operation, and AM for compatible-amplitude modulation.

2.3.3.7 Power Indicator. The power indicator (4, Figure 2-11) indicates power output and the correct tuning of the antenna system. For sideband operation, the meter deflects only when the microphone is spoken into. For AM operation, the meter indicates carrier level when the microphone switch is keyed.

2.3.3.8 HF SSB Communication System Operation

1. Volume control knob — On.
2. Mode selector switch — As Desired.
3. Channel selector switch — Select Desired Channel.
4. HF mixer switch — As Desired.
5. Speaker or phone volume control knob — As Desired.
6. R.F. GAIN control knob — Adjust as Necessary.
7. CLARIFIER control knob — Adjust To Obtain Best Audio.

To transmit:

8. HF microphone selector button — Depress.
9. Microphone switch — Depress.

2.3.4 High-Frequency Single-Sideband Communication System (Aircraft With Collins 718U-5 Installed). The HF SSB communication system provides long-range two-way voice communication at frequencies between 2.0000 to 29.9999 MHz spaced at 100 Hz. The transceiver operates in the USB, LSB, and AM (carrier plus USB for AM transmission) as selected by the mode selector on the control panel.

2.3.4.1 HF SSB Communication System Control Panel. All controls (except volume) and indicators for operation of the transceiver are on a control panel (Figure 2-11) on the center pedestal. The audio output of the transceiver is controlled by the volume control on the audio control panel. A phone is also installed in the cabin to permit a passenger to communicate from the cabin to a station as selected on the control panel on the center pedestal. The phone also incorporates a volume control to vary the audio level.

2.3.4.2 Frequency Selector Knobs. The five frequency knobs are used to select the desired operating channel. The selected channel appears in the channel indicator window.

2.3.4.3 SQUELCH Control. The SQUELCH control provides a selection of eight squelch threshold level settings for the receiver circuits in the transceiver. Rotating the SQUELCH control counterclockwise adjusts the receiver so that a stronger receiver signal is required to overcome the squelch threshold.

2.3.4.4 Mode Selector Switch. The five-position mode selector switch is used to select the desired mode of operation. Mode selections are USB, LSB, AM, and RF TEST. Moving the selector switch from the OFF position applies 28-vdc essential bus power to transceiver.

2.3.4.5 RF TEST Indicator Light. This indicator light provides an operational status of the system. With the mode selector switch in the RF TEST position and the mic button keyed, if the light is on (a steady glow or flashing), it indicates a fault in the system.

2.3.4.6 HF SSB Communication System Operation

1. Electrical power — ON.
2. Audio control panel mixer switches (select HF COMM) — SPKR.
3. Frequency selector knobs — Set to Desired Frequency.

4. Mode selector switch — As Desired.

5. SQUELCH control — Adjust To Eliminate Background Noise.

6. HF microphone selector button on audio control button — Depress.

7. Microphone switch — Depress to Transmit.

2.3.5 VHF Communication System (VHF-21B). The communication system provides two-way air-to-ground and ground-to-ground voice communications in the frequency range of 118 to 151.975 MHz in increments of 25 kHz. The system is controlled by the Comm Control located on the instrument panel shroud.

2.3.5.1 VHF-21B Receiver. The receiver is a dual-conversion super heterodyne receiver. The receiver has two automatic squelch circuits (carrier and noise). The receiver is located in the nose compartment radio rack and is powered by 28 vdc from the dc essential bus.

2.3.5.2 CTL-22 VHF Control. The CTL-22 VHF control (Figure 2-12) provides tuning control through 1,360 channels from 118.000 to 151.975 MHz. The control contains the following displays and controls.

1. Active frequency display — The active frequency (frequency to which the receiver is tuned) and diagnostic messages are displayed in the upper window.

2. Preset frequency display — The preset (inactive) frequency and diagnostic messages (Figure 2-13) are displayed in the lower window.

3. Compare annunciator — ACT momentarily illuminates when frequencies are being changed. If ACT continues to flash, the actual radio frequency is not identical to the frequency shown in the active window.

4. Annunciators — The COM control contains MEM, RMT, and TX annunciators. The MEM annunciator illuminates whenever a stored frequency is being displayed in the lower window. The RMT annunciator indicates that the receiver is being remotely tuned by another system and is not implemented on the CT-39E. The TX annunciator illuminates whenever the receiver is transmitting.

5. Volume control — The volume control is concentric with the power and mode switch.

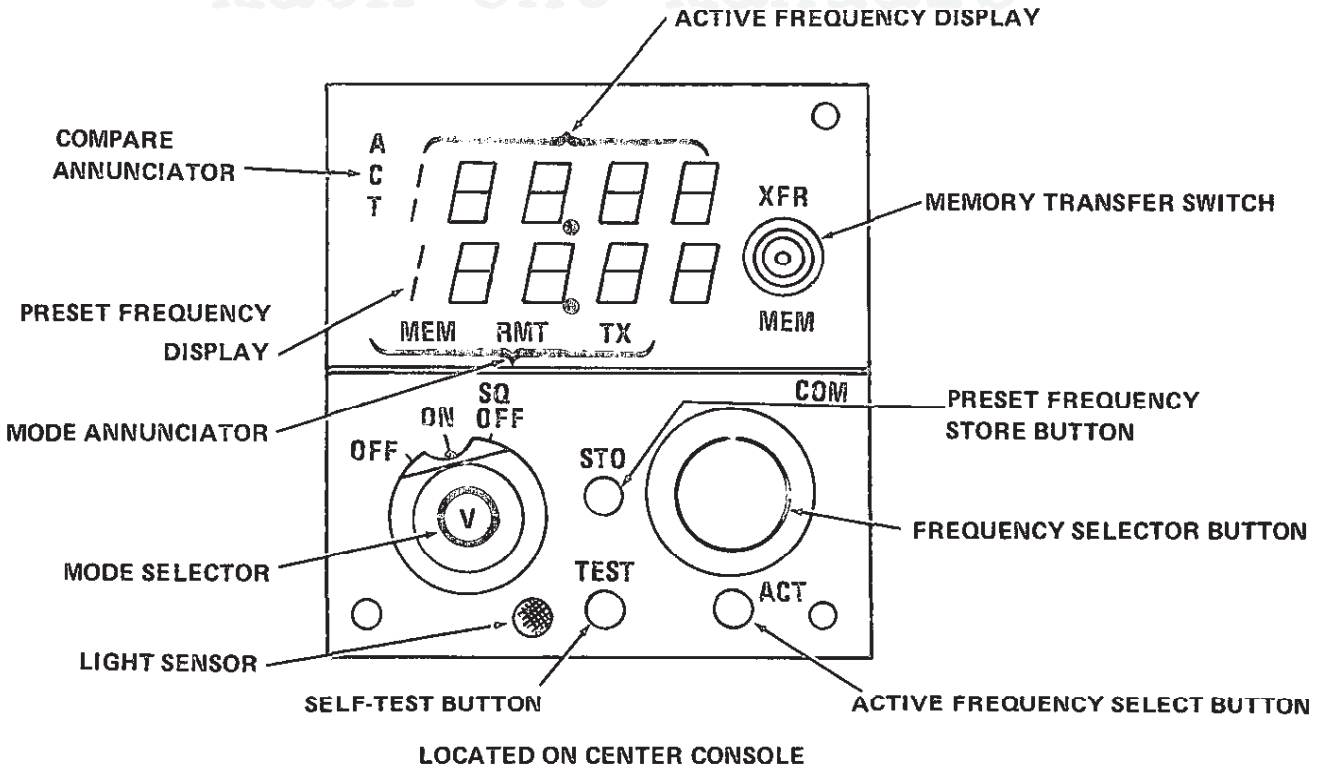

Figure 2-12. CTL-22 VHF Control

FAULT CODE	DESCRIPTION
00	NO FAULT FOUND
01	5 VDC BELOW LIMIT
02	5 VDC ABOVE LIMIT
03	12 VDC BELOW LIMIT
04	12 VDC ABOVE LIMIT
05	SYNTHESIZER NOT LOCKED
07	NOISE SQUELCH OPEN WITHOUT SIGNAL
08	NOISE SQUELCH NOT OPEN WITH SIGNAL
12	BCD FREQUENCY CODE INVALID
13	2/5 FREQUENCY CODE INVALID
14	SERIAL MESSAGE INVALID
15	FREQUENCY OUT OF RANGE
16	FORWARD POWER BELOW LIMIT
17	TRANSMITTER TEMPERATURE EXCESSIVE
21	TUNING VOLTAGE OUT OF LIMIT AT HIGHEST RECEIVE FREQUENCY
22	TUNING VOLTAGE OUT OF LIMIT AT 118 MHZ
23	LOCAL OSCILLATOR OUTPUT BELOW LIMIT
24	NO-SIGNAL AGC VOLTAGE TOO HIGH
25	INADEQUATE AGC VOLTAGE INCREASE WITH RF SIGNAL
27	TRANSMITTER TIMED OUT

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Figure 2-13. CTL-22 Diagnostic Codes

6. Power and mode switch — The power and mode switch contains three detected positions.

   a. ON/OFF — The ON and OFF positions switch system power.

   b. SQ OFF — The SQ OFF position disables the receiver squelch circuits, so noise should be heard. Use this position to set the volume control or, if necessary, to try to receive a very weak signal that cannot break squelch.

7. Light sensor — The built-in light sensor automatically controls the display brightness.

8. XFR/MEM switch — This switch is a three-position, spring-loaded, toggle switch. When held to the XFR position, the preset frequency is transferred up to the active display and the receiver retunes. The previously active frequency becomes the new preset frequency and is displayed in the lower window. When this switch is held to the MEM position, one of the six memory frequencies through the display (...2,3,4,5,6,1,2,3...).

9. Frequency select knobs — Two concentric knobs control the preset or active frequency display. The larger knob changes the three digits to the left of the decimal point in 1-MHz steps. The smaller knob changes the two digits to the right of the decimal point in 50-kHz steps (or in 25-kHz steps for the first two steps after the direction of rotation has been reversed). Numbers roll over at the upper and lower frequency limits.

10. ACT button — Push the ACT button for approximately 2 seconds to enable the frequency select knobs to directly retune the receiver. The bottom window will display dashes and the upper window will continue to display the active frequency. Push the ACT button a second time to return the control to the normal two-display mode.

11. STO button — The STO button allows up to six preset frequencies to be selected and entered into the control's nonvolatile memory. After presetting the frequency to be stored, push the STO button. The upper window displays the channel

number of available memory (CH 1 through CH 6); the lower window continues to display the frequency to be stored. For approximately 5 seconds, the MEM switch may be used to advance through the channel numbers without changing preset frequency to memory in the selected location. After approximately 5 seconds, the control will return to normal operation.

12. TEST button — Press the TEST button to initiate the radio self-test diagnostic routine. The transceiver performs a complete self-test routine requiring about 5 seconds. In the test mode the active and preset intensity will vary from minimum to maximum. Several audio tones are heard in the audio system. Active display indicators display dashes and the preset indicators will display 00. If a fault is found, then a fault annunciator will appear below the preset display (see Figure 2-13 for fault codes).

2.3.5.3 Operating Procedures. The frequencies displayed on the CTL-22 COMM control show only five of the six digits. The sixth digit is always a 0 (when the fifth digit is a 0 or 5) or a 5 (when the fifth digit is a 2 or a 7). Therefore, the sixth digit does not need to be displayed.

The transceiver and the CTL-22 COMM control are turned on by rotating the power and mode switch on the COMM control to the ON position. When the transceiver if first turned on, it sounds a brief tone while the microprocessor checks its own memory. If there is a memory defect, the tone continues to indicate the transceiver can neither receive nor transmit.

After the memory check, the CTL-22 displays the same active and preset frequencies that were present when the equipment was last turned off.

Note

If two short 800-Hz tones are heard, the transceiver has detected an internal fault. Push the TEST button on the CTL-22 to initiate self-test and display the fault code.

Adjust the volume level and perform a quick squelch test by setting the power and mode switch on the CTL-22 to SQ OFF and adjusting the volume level with background noise. After a comfortable listening level has been established, return the power and mode switch to the ON position. All background noise should disappear unless a station or aircraft is transmitting on the active frequency.

Frequency selection is made using either the frequency select knobs or the XFR/MEM (transfer/memory recall switch).

Rotation of either frequency select knob increases or decreases the frequency in the preset frequency display. The larger outer knob changes the frequency in 1-MHz increments (number to the left of decimal point). The smaller inner knob changes the frequency in 50-kHz increments when first rotated or in 25-kHz increments for the first two increments after the direction of rotation is reversed.

After the desired frequency is set in the preset frequency display, it can be transferred to the active frequency display by momentarily setting the XFR/MEM switch to XFR. At the same time that the preset frequency is transferred to the active display, the previously active frequency is transferred to the active display. A short audio tone is applied to the audio system to indicate the active frequency has been changed, and the ACT annunciator on the control flashes while the transceiver is tuning to the new frequency.

Note

If the ACT annunciator continues flashing, it indicates that the transceiver is not tuned to the frequency displayed in the active display.

The CTL-22 has a nonvolatile memory that permits storing up to six preset frequencies. Once stored, these frequencies can be recalled to the preset display by positioning the XFR/MEM switch to the MEM position. The storage location (CH 1 through CH 6) for the recalled frequency is displayed in the active frequency display while the XFR/MEM switch is held in the MEM position.

All six stored frequencies can be displayed one at a time in the preset display by repeatedly positioning the XFR/MEM switch to the MEM position. After the desired stored frequency has been recalled to the preset display, it can be transferred to the active display by momentarily positioning the XFR/MEM switch to the XFR position.

During normal operation, all frequency selections and revisions are done in the preset frequency display.

The active frequency can be selected directly with the frequency select knobs by pushing the ACT button for about 2 seconds. The active frequency selection mode is indicated by dashes appearing in the preset

display. Also, the ACT annunciator will flash as the frequency select knobs are turned to indicate transceiver is being returned.

Note

If the ACT annunciator continues flashing after the frequency has been selected, it indicates that the transceiver is not tuned to the frequency displayed in the active display.

To return to the preset frequency selection made, merely push the ACT button again for about 2 seconds.

As a safety feature, the CTL-22 automatically switches to the active frequency selection mode when a frequency select knob is operated while the STO, TEST, or XFR/MEM switches are actuated or while the remote transfer or memory recall inputs to the control are grounded.

Up to six preset frequencies can be stored in the nonvolatile memory in the CTL-22 for future recall. To program the memory, select the frequency in the preset frequency display using the frequency select knobs and push the STO button once. One of the channel numbers (CH 1 through CH 6) will appear in the active display for approximately 5 seconds. During this time, the channel number can be changed without changing the preset frequency by momentarily positioning the XFR/MEM switch to the MEM position. After the desired channel number has been selected, push the STO button again to store the frequency.

Note

When storing a frequency, the second actuation of the STO button must be done within 5 seconds after selecting the channel number or the first actuation of the STOP button. If more than 5 seconds elapse, the control will revert to the normal modes of operation and the second store command will be interpreted as the first store command.

After a frequency has been stored in memory, it will remain there until changed by using the STO button. Memory is retained even when the unit is turned off for an extended period of time.

Each time the push-to-talk button is pressed, the microprocessor in the transceiver starts a 2-minute timer. (The TX annunciator on the CTL-22 is illuminated whenever the transmitter in on.) If the transmitter is still on at the end of 2 minutes, the microprocessor turns it off. Most intentional transmissions last much less than 1 minute; a 2-minute transmission is most likely the result of a stuck microphone button. This timing feature protects the ATC channel from long-term interference.

When the microprocessor turns off the transmitter it switches the receiver to receive operation. The stuck mic button prevents received signals or even the two warning beeps from being heard. The microprocessor then waits until the push-to-talk button opens to sound the two beeps.

When it is necessary to transmit for more than 2 minutes, just release the mic button briefly and then press it again when pausing for a breath. The 2-minute timer resets and starts a new count each time the push-to-talk button is pressed.

To operate VHF COMM, proceed as follows:

1. Ensure power is supplied to all associated equipment.
2. Select CTL-22 control mode switch to ON.
3. Select desired frequency on CTL-22 either by setting the frequency known or transferring for internal memory.
4. Select VHF on ICS panel.
5. Press appropriate microphone button to make transmission.

2.3.6 VHF Communication System (CT-39G Aircraft 158844 and 160053 Through 160057). A VHF-20B VHF communication system provides two-way voice communication between 116.00 and 151.950 MHz. The audio signal is directed to the audio system, and system power is received from the dc essential bus. On this system, the SQL adjustment is an internal part of the transceiver. The SQL is disabled when the TEST button is depressed.

2.3.6.1 VHF Communication System Control Panel. Except for the volume and microphone selector control, all controls and indicators for operations of the VHF communication system are on a control panel (Figure 2-14) on the center pedestal. Volume is controlled by the phone or speaker volume control on the audio system control panel.

2.3.6.2 Power Control Knob. The power control knob (Figure 2-14) is on the top knob of three stacked

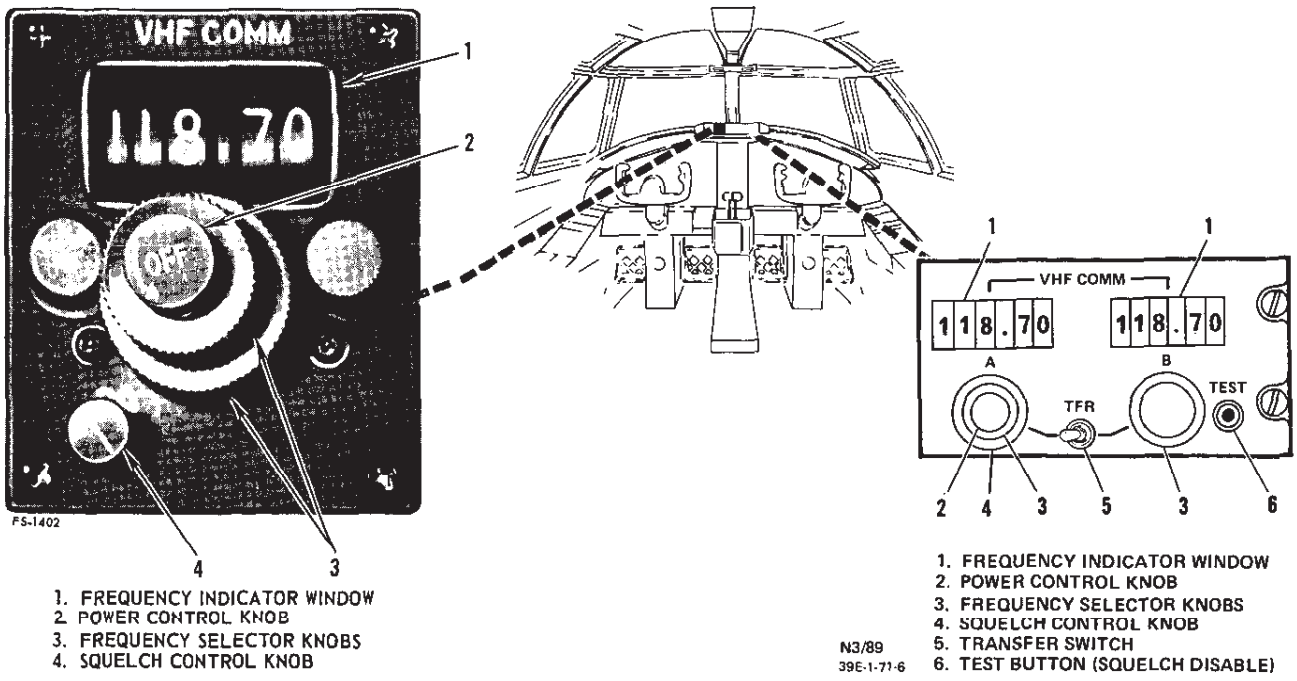


Figure 2-14. VHF Communication System Control Panel

knobs on the face of the control panel. Clockwise rotation of the knob applies power to the system.

2.3.6.3 Frequency Selector Knobs. Two sets of frequency knobs are provided. The middle and bottom of the left knob control the frequency selection in the left frequency window. The right two knobs control the frequency selection in the right frequency window.

2.3.6.4 Transfer Switch. The TFR switch is used to select either frequency as dialed by the frequency selector knob. Two different frequencies may be set in the frequency windows and, by use of the TFR switch, either frequency may be used.

2.3.6.5 VHF Communication System Operation

1. Power control knob — ON.
2. Frequency selector knobs — Set to Desired Frequency.
3. TFR switch — Position to Selected Frequency.
4. VHF mixer switch — As Desired.
5. Speaker or phone volume control knob — As Desired.

To transmit:

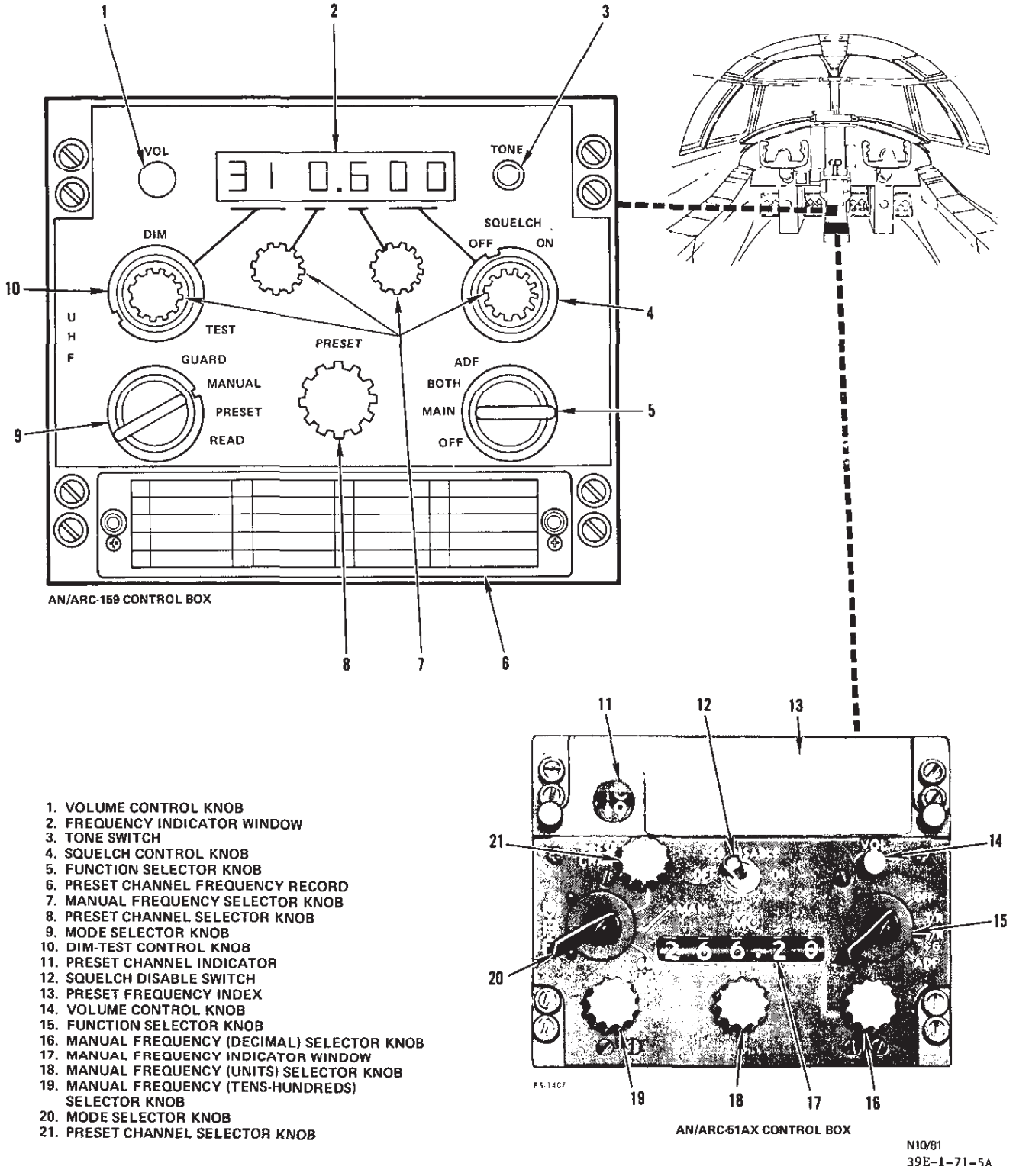
6. VHF microphone selector button — Depress.
7. Microphone switch — Depress.

2.3.7 UHF Communication System (CT-39E, CT-39G Aircraft With AN/ARC-51AX Installed). The UHF AN/ARC-51AX communication system provides communication in the UHF band from 225.00 to 399.95 MHz on 20 preset channels or 3,500 manually set frequencies. In addition, the system has the capability of monitoring the military guard frequency (243.00 MHz) while operating on any other frequency. The audio signal is directed through the audio system, and system power is required from the dc essential bus.

2.3.7.1 UHF Communication System Control Panel. All controls and indicators for operation of the UHF communication system are located on a control panel (Figure 2-15) on the center pedestal.

2.3.7.2 Function Selector Knob. The function selector knob (15, Figure 2-15) has four positions marked OFF, T/R, T/R + G, and ADF. Rotating the knob to the T/R position activates the RT unit for normal transmission and reception on the channel selected; the guard receiver is inoperative. Rotating the knob to the T/R + G position also activates the RT unit for normal use and, in addition, activates the guard receiver to allow monitoring guard frequency. The ADF position is inoperative.

2.3.7.3 Mode Selector Knob. The three-position mode selector knob (20, Figure 2-15) permits selection of the desired operating mode. The knob is marked PRESET CHAN, MAN, and GD XMIT. The PRESET CHAN position is used when selecting one of the 20



1. VOLUME CONTROL KNOB
2. FREQUENCY INDICATOR WINDOW
3. TONE SWITCH
4. SQUELCH CONTROL KNOB
5. FUNCTION SELECTOR KNOB
6. PRESET CHANNEL FREQUENCY RECORD
7. MANUAL FREQUENCY SELECTOR KNOB
8. PRESET CHANNEL SELECTOR KNOB
9. MODE SELECTOR KNOB
10. DIM-TEST CONTROL KNOB
11. PRESET CHANNEL INDICATOR
12. SQUELCH DISABLE SWITCH
13. PRESET FREQUENCY INDEX
14. VOLUME CONTROL KNOB
15. FUNCTION SELECTOR KNOB
16. MANUAL FREQUENCY (DECIMAL) SELECTOR KNOB
17. MANUAL FREQUENCY INDICATOR WINDOW
18. MANUAL FREQUENCY (UNITS) SELECTOR KNOB
19. MANUAL FREQUENCY (TENS-HUNDREDS) SELECTOR KNOB
20. MODE SELECTOR KNOB
21. PRESET CHANNEL SELECTOR KNOB

Figure 2-15. UHF Communication System Control Panel

preset frequencies. The MAN position is used when utilizing frequencies that are selected by the manual frequency selector knobs. The GD XMIT position tunes the main RT unit to the guard frequency of 243.0 megacycles.

2.3.7.4 Preset Channel Selector Knob. The preset channel selector knob (11, Figure 2-15) permits selection of 1 of 20 preset frequencies. With the mode selector switch at PRESET, movement of the preset channel selector knob changes the frequency to that of the channel selected. There are 20 channels that may be

individually selected. The selected channel number will appear in the preset channel indicator window (11, Figure 2-15). Frequencies for each channel are written on a preset frequency index (13, Figure 2-15) on the face of the control panel. Frequencies of the preset channels can be changed during flight.

2.3.7.5 Manual Frequency Selector Knobs. Three manual frequency selector knobs (16, 18, and 19, Figure 2-15) are provided for manually selected frequencies. Manual frequency selection can be made in steps of 50 kHz from 225 through 399.95 MHz. The first selector knob will select the first two digits of the desired frequency with a range of 22 through 39 in multiples of 10. The second knob selects the third digit and has a range of 0 through 9. The third knob selects the last two digits in multiples of 0.05 and has a range of 0.00 through 0.95. The selected frequency is displayed in a window on the face of the control panel.

2.3.7.6 Volume Control Knob. The volume control knob (14, Figure 2-15) increases the volume of the receiver when turned clockwise and decreases it when turned counterclockwise.

2.3.7.7 Squelch Disable Switch. The squelch disable switch (12, Figure 2-15) is a two-position switch marked ON and OFF. The switch is provided so that the squelch can be selected for compatibility with the strength of the signal being received. Placing the switch at ON disables (turns off) the squelch. Placing the switch to OFF turns the squelch on.

2.3.7.8 Frequency Indicator Window. The frequency indicator window (17, Figure 2-15) shows the frequency selected for transmission or receiving. The window has five digits; the first two digits and last two digits are set by frequency selector knobs to the left and right of the window. The third or center digit is set by a knob directly below the window.

2.3.7.9 UHF Communication System Operation. To operate the UHF communication system, proceed as follows:

1. Function selector knob — T/R + G.
2. Mode selector knob — PRESET CHAN.
3. Preset channel selector knob — Turn to Desired Channel.
4. Squelch disable switch — OFF.
5. UHF mixer switch — As Desired.

Note

Allow 3 minutes for warmup.

6. Volume control knob — As Desired.

To cut out guard frequency:

7. Function selector knob — T/R.

To transmit:

8. UHF microphone selector button — Depress.
9. Microphone switch — Depress.

To set up a manual frequency:

10. Mode selector knob — MAN.
11. Manual frequency selector knobs — As Desired.

Note

To augment loudness of signals that are weak because of distance, move squelch disable switch to ON.

2.3.8 UHF Communication System (CT-39E, CT-39G Aircraft With AN/ARC-159 Installed). The UHF AN/ARC-159 communication system provides communication in the UHF band from 225.00 to 399.975 MHz on 20 preset channels or 7,000 manually set frequencies. In addition, the system has the capability of monitoring the military guard frequency (243.00 MHz). The audio signal is directed through the audio system, and system power is required from the dc essential bus.

2.3.8.1 UHF Communication System Control Panel. All controls and indicators for operation of the UHF system are located on the control panel on the center pedestal (Figure 2-15).

2.3.8.2 Volume Control Knob. The volume control knob (Figure 2-15) increases or decreases the audio output level to the audio system. Clockwise rotation increases the audio output level.

2.3.8.3 Frequency Indicator Window. The frequency indicator window (Figure 2-15) shows the frequency selected for transmission or receiving. The window has six digits that are set by four frequency selector knobs. The first two digits are set by the left

knob, the third and fourth digits are set by the two middle frequency knobs, and the right two digits are set by the right-hand frequency knob.

2.3.8.4 Tone Switch. The tone switch is a momentary pushbutton switch. With the switch depressed and held, a 1020-Hz tone is heard in the headset or speaker. This provides a quick check of the receiver.

2.3.8.5 Squelch Control Knob. The squelch control knob (Figure 2-15) is a two-position switch marked ON and OFF. This control is provided so that the squelch can be selected for compatibility with the strength of the signal being received. Placing the switch at ON turns on the squelch. Placing the switch to OFF turns the squelch off.

2.3.8.6 Function Selector Switch. The function selector switch is a four-position rotary switch used to select the desired system function. With the switch at OFF, the system is inoperative. With the switch at MAIN, the system is in the normal receive transmit condition. At BOTH, the guard receiver is enabled along with the normal transmit and receive functions. The ADF position is inoperative. A separate ADF system is provided.

2.3.8.7 Preset Channel Frequency Record. The preset channel frequency record is provided to list the frequencies of the 20 preset channels.

2.3.8.8 Manual Frequency Selector Knobs. The four (Figure 2-15) manual frequency selector knobs are provided for manually selecting the desired frequency. Manual frequency selection can be made in steps of 25 kHz from 225 through 399.975 MHz. The selected frequency is displayed in a window on the face of the control panel.

2.3.8.9 Manual Preset Channel Selector Knob. The manual preset channel selector selects any 1 of the 20 preset channels. As a channel is selected and the mode selector switch is positioned to READ, the selected channel frequency appears in the frequency window. The channel frequencies are written on the frequency index (Figure 2-15) on the face of the control panel.

2.3.8.10 Mode Selector Knob. The four-position mode selector knob (Figure 2-15) is used to select the desired operating mode. When positioned at GUARD, the receiver-transmitter is tuned to the guard frequency. When at MANUAL, the receiver-transmitter operates normally using only the manually selected frequency. When positioned to PRESET, the receiver-transmitter operates on the channel as selected by the preset channel selector knob (Figure 2-15). At READ, the receiver-transmitter operates on the preset selected channel and the frequency readout will appear in the frequency window.

2.3.8.11 Dim-Test Control Knob. The dim-test control knob adjusts the light intensity of the frequency readouts. When positioned at TEST, the frequency readouts will go to a bright level.

2.3.8.12 UHF Communication System Operation. To operate the UHF communication system, proceed as follows:

1. Aircraft electrical power — ON.
2. Function selector knob — MAIN.
3. Manual frequency selector knob — Set to Operating Frequency.
4. Mode selector knob — MANUAL.
5. Squelch control knob — As Required.
6. Audio control panel — Select UHF and Speaker.
7. Monitor communications.

Guard frequency:

8. Mode selector switch — GUARD (243.0 MHz should be displayed in frequency window).
9. Function selector knob — BOTH.
10. Monitor guard frequency.

Preset frequency:

11. Mode selector knob — PRESET.
12. Function selector knob — MAIN.
13. Preset channel selector knob — To Operating Frequency.
14. Mode selector knob — READ (preset channel frequency should appear in frequency window).
15. Monitor communications.

To transmit:

16. Function selector knob — MAIN.

17. Mode selector knob — MANUAL.

18. Manual frequency selector knob — Set to Operating Frequency.

19. Audio control panel — Depress UHF Microphone Button.

20. Microphone switch — Depress, Request a Radio Check.

21. Microphone switch — Release, Monitor Radio Check.

2.3.9 Air Traffic Control Transponder System (CT-39E, CT-39G Aircraft 158843 and 159362 Through 159365). The ATC transponder system provides identification of aircraft on the plan position indicator of the ground controller. The system is integrated by radar pulses received from a ground station and replies automatically with a series of pulses. Reply pulses are coded to supply identification reporting. The interrogation frequency is 1030 MHz and the reply frequency is 1090 MHz. System power is received from the ac secondary bus. Automatic altitude reporting capability is provided by the installation of an encoding altimeter and a static defect correction module that operate in conjunction with the ATC transponder system. (Refer to paragraph 2.18, INSTRUMENTS.) Mode C operation provides altitude information only to the controllers. Modes B and D are not in use at this time.

2.3.9.1 ATC Transponder Control Panel. The ATC transponder control panel (Figure 2-16) contains all controls for operation of the ATC transponder system.

2.3.9.2 Function Selector Switch. The three-position rotary function selector switch (Figure 2-16) is used to select the desired system function. With the switch in the STBY position, a 60-second time delay is introduced and the system is held in a standby condition. With the switch in the ON or LO SENS position, the system is operational after the 60-second time delay is completed or immediately if the system was warmed up in the STBY position.

2.3.9.3 Code Selector Knobs. The four code selector knobs (Figure 2-16) are used to select the code that will be displayed in the code window. The left inner knob is used to select the hundreds digit, the left outer knob is used to select the thousands digit, the right inner knob is used to select the tens digit, and the right outer knob is used to select the units digit. Clockwise rotation of each knob increases the respective digit.

2.3.9.4 Mode Selector Switch. The mode selector switch (6, Figure 2-16) is used to select the requested mode of operation. Normally the selector is set to A, but modes B, C, or D may be requested at the option of the controller.

2.3.9.5 Identification Switch. The pushbutton-type IDENT switch (1, Figure 2-16) is used for additional identification when requested by the ground controller. When the switch is depressed, an additional pulse is added to the normal reply and a special identification is displayed on the ground controller ATC radar screen. The identification pulse is transmitted for approximately 20 seconds after the switch is depressed and released.

2.3.9.6 Test Switch. The pushbutton TEST switch (7, Figure 2-16) provides a means of testing the ATC transponder system. When the switch is depressed, a simulated mode A or mode B interrogation, determined by the position of the mode selector switch, is supplied to the system. If the system is operating normally, the reply light will illuminate.

2.3.9.7 Reply Light. The green REPLY light (8, Figure 2-16) comes on when the system is interrogated by the ATC ground station or when the test button is depressed and the system is operating normally.

2.3.9.8 Altitude Reporting Switch. The altitude reporting switch (5, Figure 2-16) has two positions: ALT RPTG and OFF. When in the ALT RPTG position, the ATC transponder will provide replies to mode C interrogations, regardless of the mode selector switch position, as well as providing replies to interrogations in the selected mode. When the switch is at OFF, the mode selector switch must be at C to provide automatic altitude reporting.

2.3.9.9 Operation of ATC Transponder System

1. Function selector switch — ON. Allow approximately 80 seconds for system warmup.

Note

The function selector switch should be set to SBY or LO SENS only when requested by the ground controller.

2. Mode selector switch — As Required.

3. Code selector knobs — As Required.

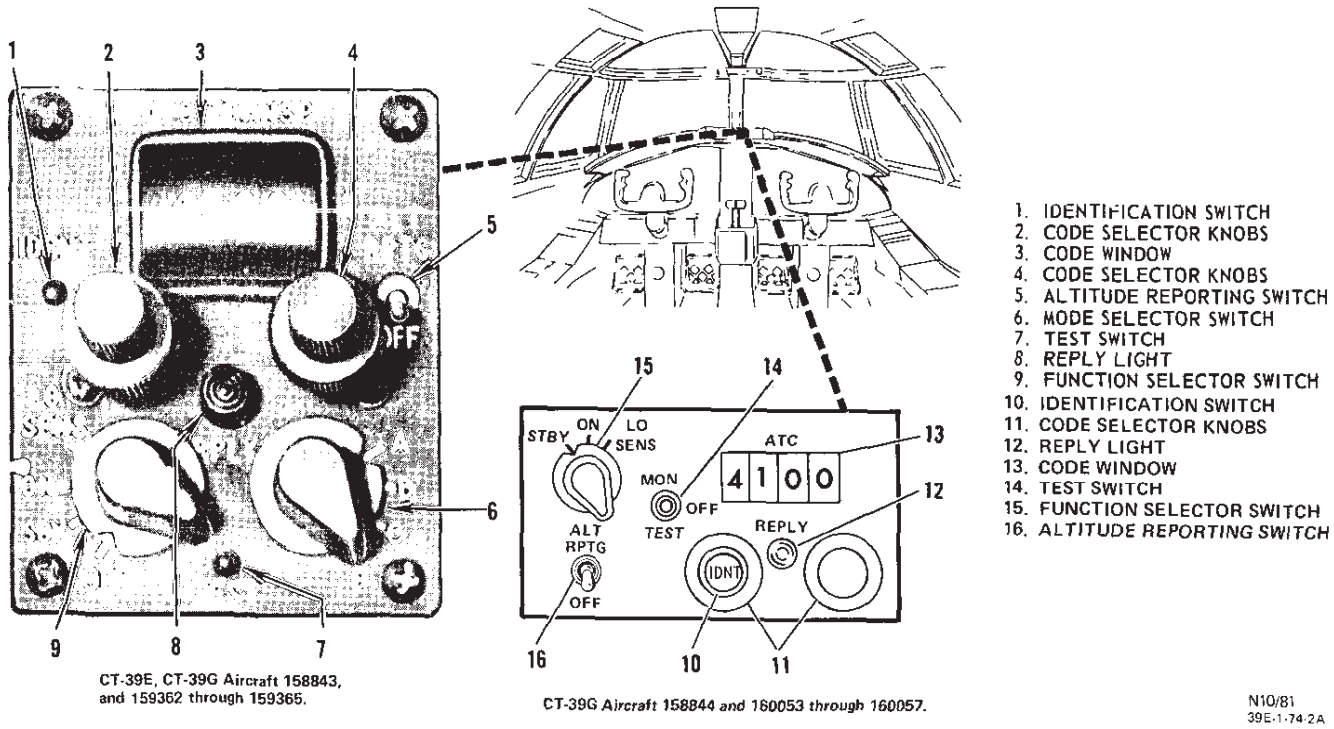


Figure 2-16. ATC Transponder Control Panel

4. Altitude reporting switch — As Required.

5. Test switch — Depress. Verify that reply light comes on to indicate system is operating normally.

2.3.10 ATC Transponder System (CT-39G Aircraft 158844 and 160053 Through 160057). The ATC transponder system provides identification of aircraft on the plan position indicator of the ground controller. The system is integrated by radar pulses received from a ground station and replies automatically with a series of pulses. Reply pulses (mode A operation) are coded to supply identification reporting. The interrogation frequency is 1030 MHz; the reply frequency is 1090 MHz. System power is received from the 28-vdc essential bus. Automatic altitude reporting capability (mode C operation) is provided by the installation of an encoding altimeter that operates in conjunction with the ATC transponder and static systems. (Refer to paragraph 2.17, INSTRUMENTS.)

2.3.10.1 ATC Transponder Control Panel. The ATC transponder control panel (Figure 2-16) contains all controls for operation of the ATC transponder system.

2.3.10.2 Function Selector Switch. The three-position rotary function selector switch (Figure 2-16) is used to select the desired system function. With the switch in the STBY position, a 60-second time delay is introduced and the system is held in a standby condition. With the switch in the ON or LO SENS (low sensitivity) position, the system is operational after the 60-second time delay is completed or immediately if the system was warmed up in the STBY position.

2.3.10.3 Code Selector Knobs. The four code selector knobs (Figure 2-16) are used to select the code that will be displayed in the code window. The left inner knob is used to select the hundreds digit, the left outer knob is used to select the thousands digit, the right inner knob is used to select the tens digit, and the right outer knob is used to select the units digit. Clockwise rotation of each knob increases the respective digit.

2.3.10.4 Identification Switch. The pushbutton-type IDENT switch (Figure 2-16) is used for additional identification when requested by the ground controller. When the switch is depressed, an additional pulse is added to the normal reply and the two parallel bars on the controller radarscope are filled in. The identification pulse is transmitted for approximately 20 seconds after the switch is depressed and released.

2.3.10.5 MON/TEST Switch. This switch (Figure 2-16) provides a means of testing the ATC transponder system (in both mode A and mode C) operation. With the function selector switch at ON and the ALT RPTG/OFF switch at OFF, when the MON/TEST is at TEST, a simulated mode A interrogation is supplied to the system. If the system is operating normally, the reply light will illuminate. With ALT RPTG/OFF switch at ALT RPTG, mode C operation is tested.

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2.3.10.6 Reply Light. The green REPLY light (Figure 2-16) comes on when the system is interrogated by the ATC ground station or when the MON/TEST switch is at TEST position if the system is operating normally.

2.3.10.7 Altitude Reporting Switch. The altitude reporting switch (Figure 2-16) has two positions: ALT RPTG and OFF. When in the ALT RPTG position, the ATC transponder will provide replies to mode "C" interrogations regardless of the function selector switch position. When the switch is at OFF, altitude reporting is terminated.

2.3.10.8 Operation of ATC Transponder System

1. Function selector switch — ON.
2. MON/TEST switch — OFF.

Allow approximately 60 seconds for system warmup.

Note

The function selector switch should be set to STBY or LO SENS only when requested by the ground controller.

3. Code selector knobs — Set to Any Frequency Other Than 7777, 7700, or 7600.
4. ALT RPTG switch — OFF.
5. MON/OFF switch — TEST Position. Verify That Reply Light Comes On To Indicate System Is Operating Normally (mode A operation).
6. ALT RPTG switch — ALT RPTG Position.
7. MON/TEST switch — TEST Position. Verify That Reply Light Comes On To Indicate System Is Operating Normally (mode C operation).

2.4 ELECTRICAL SYSTEM — CT-39E AIRCRAFT

The 28-vdc system is powered by two 30-volt, 400-ampere engine-driven starter-generators, one on each engine. Two 25.2-volt, nickel-cadmium batteries furnish standby power for the dc system and can be used for engine start when an external power source is not available. Both batteries are connected in parallel to the dc system; but during a battery-powered start, the batteries are series-connected with the combined voltage of both batteries applied to the starters. Ac is supplied by a dc-powered, main three-phase, 2,500-volt ampere inverter. A standby inverter, with the same power rating, automatically provides alternating current if the main inverter fails. If both of these inverters fail, a three-phase instrument inverter provides instrument power. The instrument inverter also provides instrument power during engine starts.

For ground operations, external power can be supplied to the dc system and to the starter circuit.

2.5 ELECTRICAL SYSTEMS — CT-39G AIRCRAFT

The 28-volt, dc system is powered by two 30-volt, 400-ampere engine-driven starter-generators, one on each engine. Two 25.2-volt, nickel-cadmium batteries furnish standby power for the dc system and can be used for engine start when an external power source is not available. Both batteries are connected in parallel to the dc system; but during a battery-powered start, the batteries are series-connected with the combined voltage of both batteries applied to the starters. Ac is supplied by a dc-powered No. 1, three-phase, 2,500-volt ampere inverter. A No. 2 inverter with the same power rating automatically provides alternating current if the No. 1 inverter fails. If both of these inverters fail, a three-phase instrument inverter provides instrument power. The No. 2 inverter provides power for windshield anti-icing. For ground operations, external power can be supplied to the dc system and to the starter circuit.

During normal operating conditions (No. 1 inverter and windshield heat selected) in the event of a dc generator failure, the No. 2 inverter will automatically be shut down and the instrument inverter will then provide one-half of the normal power to the pilot windshield or copilot windshield if selected. During this condition, if the No. 1 inverter should fail, the No. 2 inverter will automatically power the 115-vac primary bus.

2.5.1 Dc Electrical Power Distribution. Dc power is distributed from the following electrical buses: the battery bus, paralleling bus, dc essential bus, and dc secondary bus. In addition, there is a starter bus to supply power to the starters only. The dc electrical system is shown schematically in FO-10 or FO-11 (as applicable).

2.5.1.1 Battery Bus. The battery bus is connected directly to the batteries. During normal operation, the battery bus is powered by both batteries connected in parallel. However, when a battery-powered engine start is made, battery No. 2 is disconnected automatically from this bus and the batteries are then series-connected to supply the higher voltage demands of the starters. During this period, battery No. 1 continues to supply the battery bus. The battery bus can be connected to and power the paralleling bus and the dc essential bus.

2.5.1.2 Paralleling Bus (CT-39E, CT-39G Aircraft 158843 and 158844). The paralleling bus can be energized by the battery bus, dc generators, or utility external power. Normally, the paralleling bus powers all other dc buses except the starter bus. The paralleling bus is energized if power is available from either dc generator. On the ground, the paralleling bus is energized directly by utility external power or by the battery bus if the electrical master and battery switches are ON.

2.5.1.3 Paralleling Bus (CT-39G Aircraft 159362 Through 159365 and 160053 Through 160057). The paralleling bus can be energized by the battery bus, dc generators, or utility external power. The paralleling bus is connected to the battery and essential buses when the electrical master and either battery switch is at ON (provided a battery overtemperature condition does not exist). During normal operation (one or both dc generators operating), this bus supplies electrical power to the essential and battery buses through the dc bus tie in relays. If both dc generators are turned off or fail in flight, the battery bus will power the paralleling and essential buses. During this condition, the pilot should turn off all nonessential equipment (including the hydraulic pump) in order to conserve battery power. On the ground, the paralleling bus is energized directly by utility external power, or by the battery bus if the electrical master and battery switches are ON. Refer to applicable battery temperature system operation for a complete description of a battery isolation.

2.5.1.4 Dc Essential Bus (CT-39E, CT-39G Aircraft 158843 and 158844). The dc essential bus is normally energized by the paralleling bus if either dc generator is on the line. Also, on the ground, the essential bus is energized through the paralleling bus by battery bus power if the electrical master and battery switches are on or by utility external power.

In flight, if both dc generators fail or are off the line, the essential bus automatically becomes powered directly from the battery bus. If this automatic power transfer fails, the dc essential bus can be connected to the battery bus by setting the battery switch at the ESS (EMERG) position.

2.5.1.5 Dc Essential Bus (CT-39G Aircraft 159361 Through 159365 and 160053 Through 160057). The dc essential bus is powered by the paralleling bus when the electrical master and either battery switch is at ON. On the ground, the essential bus may be powered by the paralleling bus when external electrical power is applied to the aircraft or by the battery bus when the electrical master and either battery switch is at ON.

In flight, if both dc generators are turned off or fail and a battery hot condition does not exist, the essential bus is powered by the battery bus. If a battery hot condition exists during a failure of both dc generators, the battery switches must be placed in the EMER position to provide power to the essential bus from the battery bus. During both conditions as stated above, the pilot should turn off all nonessential equipment (including the hydraulic pump) in order to conserve battery power. Refer to applicable battery temperature system operation for a complete description of a battery isolation.

Note

The sealed batteries in the interior emergency lighting system power units are kept charged by the dc essential bus. If the essential bus fails, these batteries illuminate the emergency lights.

2.5.1.6 Dc Secondary Bus. The dc secondary bus is energized by the paralleling bus, while the aircraft is on the ground, only if both dc generators are on the line or if utility external power is applied. In flight, the secondary bus will be energized when one dc generator is on the line.

2.5.1.7 Starter Bus. The starter bus is powered by both batteries connected in series during a battery-powered engine start or by external power through the starter external power receptacle. During an external power start, the batteries are not connected to the starter bus.

2.5.2 Essential Bus Circuit Breaker Out Caution Light. This placard-type ESS BUS C/B OUT caution light and both MASTER CAUTION lights illuminate when one or more of the five essential bus circuit breakers located in the aft cabin trip or disengage.

A test button is installed for system integrity checks and, when pushed, illuminates the MASTER CAUTION and ESS BUS C/B OUT caution lights. The system should be tested prior to each flight.

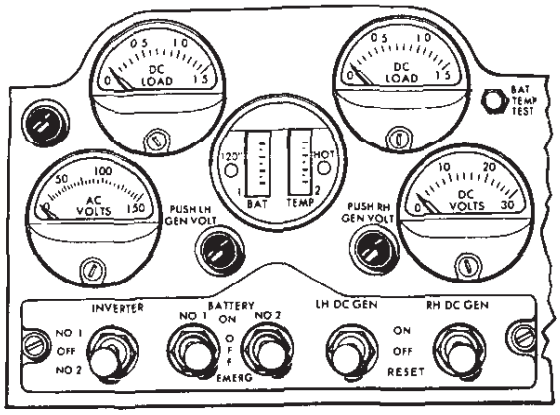
2.5.3 Battery Temperature Indicating and Isolation System (CT-39G Aircraft 159362 Through 159365 and 160053 Through 160057). A battery temperature indicating and isolation system allows battery temperatures to be monitored and provides for the isolating of one or both batteries from the aircraft electrical system. The system includes a battery switch for each battery, a temperature sensor for each battery, a battery temperature indicator (dual), a test switch, and the system caution and warning lights. Each battery is controlled by its respective three-position battery switch. During normal operation, both battery switches will be in the ON position. If either battery temperature reaches 120 °F, the amber caution light on the battery temperature indicator will illuminate. Generator output voltage is reduced by approximately 2 volts and the charging rate to both batteries is reduced (except aircraft with solid state voltage regulators installed). If either battery temperature reaches 150 °F, the red warning light on the battery temperature indicator will illuminate, a BATTERY HOT caution light on the caution-warning light panel will illuminate, both master caution lights will illuminate, and both batteries will be automatically disconnected (as long as one dc generator or external power is on the line). This automatic disconnect feature may be overridden for one or both batteries by placing the respective battery switch in the EMERG position. The automatic disconnect feature will only disconnect the batteries if the dc electrical buses are being powered by the aircraft dc generator (s) or external power.

2.5.3.1 Battery Temperature Indicator. A dual battery temperature indicator is located on the electrical indicator and control panel (Figure 2-17). The indicator provides vertical strip temperature indicators for both No. 1 (left-hand) and No. 2 (right-hand) batteries. The temperature scale is calibrated in 5° increments from 105 to 185 °F. Installed in the face of the indicator are the amber 120 °F caution light and the red 150 °F warning light. When either battery temperature reaches 120 °F, the amber caution light illuminates. If either battery temperature reaches 150 °F, the red warning light illuminates.

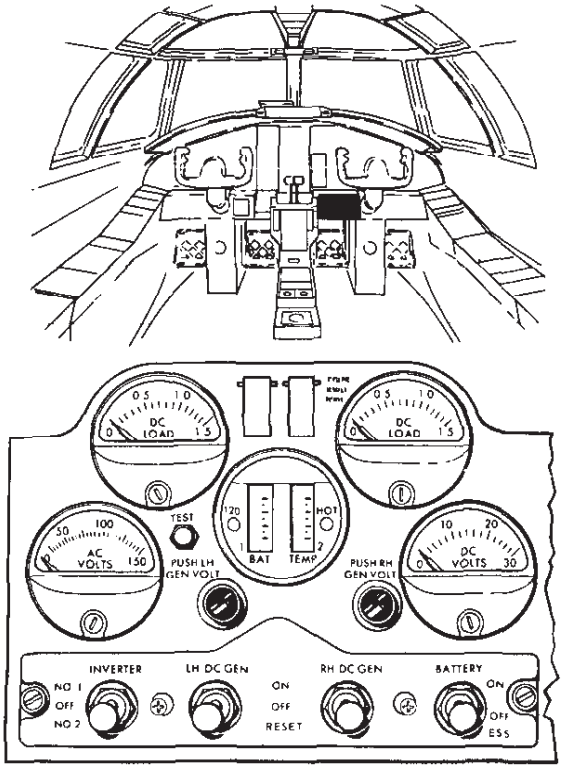
2.5.3.2 Battery Temperature System Test Switch. A battery temperature system test switch is located on the electrical indicator and control panel (Figure 2-17). The switch is labeled BATT TEMP TEST and, when actuated, allows both needles of the indicator to rise to the top of the scale. As the needles pass through 120 °F, the amber caution light on the indicator will illuminate and, as the needles pass through 150 °F, the red warning light on the indicator will illuminate. The BATTERY HOT caution light on the caution-warning light panel will illuminate and the batteries will be disconnected (as long as one dc generator or external power is on the line).

2.5.4 Battery Temperature Indicating and Isolation System (CT-39E, CT-39G Aircraft 158843 and 158844). A battery temperature indicating and isolation system allows battery temperatures to be monitored and provides two switches for use in isolating of one or both batteries from the aircraft electrical system (see Figure 2-17). The system consists of a temperature sensor for each battery, a dual battery temperature indicator (with an amber (120 °F) and red (150 °F) indicator light), two isolation switches, a test switch, a circuit breaker, and a BATT HOT light that is installed on the caution-warning panel. During normal operation (battery switch at ON and isolation switches at NORM) in the event of a battery overtemperature condition (150 °F or higher), the affected battery(ies) should be isolated from the paralleling bus by positioning the respective isolation switch at ISOL. This removes the respective battery from the charging source (dc generator). During the 150 °F temperature condition, a BATT HOT warning light on the caution-warning panel (Figure 2-33) illuminates along with the master caution lights and the red warning light on the battery temperature indicator.

With the isolation switches positioned at O'RIDE (aircraft with three-position switches), the isolation relays are grounded via their respective isolation switch and the main battery switch can be turned off, thereby isolating the batteries from the parallel and essential buses. The batteries are still connected to the battery bus; however, when the main battery switch is turned off, the battery relay deenergizes and the battery bus is disconnected from the parallel bus. Subsequent to starting the first engine and operating at 65-percent rpm, the MAIN BATTERY SWITCH may be positioned at OFF and both isolation switches positioned at O'RIDE which actually overrides the main battery switch and allows the isolation relays to be deenergized. When the starter button is depressed for the second engine start, the batteries will be placed in series on the starter bus. The operating generator will be powering the essential bus via the parallel bus during the ISOLATED second engine start. This method of starting the second engine eliminates an additional charging cycle on the batteries after the first engine start, prevents the cockpit lights from dimming, and prevents a possible loss of ac power that is the result of a high current draw during the second engine start when utilizing the normal engine starting procedures. After starting the second engine using the O'RIDE



CT-39G 159362 thru 159365
and 160053 thru 160057



CT-39E, CT-39G 158843 and 158844

ELECTRICAL BUS AVAILABILITY

POWER SOURCE	DC BUSSES					AC BUSSES	
	BATTERY	STARTER	PARALLEL	ESSENTIAL	SECONDARY	SECONDARY	PRIMARY
UTILITY EXTERNAL DC POWER							
ENGINE START EXTERNAL POWER							
BOTH DC GENERATORS							
ONE DC GENERATOR							
BATTERY POWER ONLY							
MAIN INVERTER							
STAND-BY INVERTER							
INSTRUMENT INVERTER							

ENERGIZED DE-ENERGIZED IN-FLIGHT ONLY GROUND ONLY

N11/84
39E-1-54-5B

Note

On CT-39G aircraft 159362 through 159365 and 160053 through 160057, the paralleling bus is powered anytime the dc essential bus is powered. Battery power airborne or on the ground can power the dc essential/parallel bus.

Figure 2-17. Electrical Indicator and Control Panel

switches, the battery and isolation switches should be positioned to the normal operating condition provided a battery overtemperature condition does not exist.

In flight, the battery temperatures should be monitored and if either battery reaches 120 °F (as noted by illumination of the amber light on the indicator), the temperatures should be monitored closely for any further increase. If the temperature increases to 150 °F, the battery(ies) should be isolated from the paralleling bus positioning the isolation switch(es) to ISOL. If the temperature continues to rise, the aircraft should be landed as soon as possible.

2.5.4.1 Battery Temperature Indicator. A dual battery temperature indicator is located on the electrical indicator and control panel (Figure 2-17). The indicator provides vertical strip temperature indicators for both Nos. 1 and 2 batteries. The temperature scale is calibrated in 5-percent increments from 105 to 185 °F. Installed in the face of the indicator are the amber 120 °F caution light and the red 150 °F warning light. When either battery temperature reaches 120 °F the amber light illuminates. If either battery temperature reaches 150 °F, the red warning light illuminates.

2.5.4.2 Battery Temperature System Test Switch. A battery temperature system test switch is located on the electrical indicator and control panel (Figure 2-17). The switch is labeled BATT TEMP TEST and, when actuated, allows both needles of the indicator to rise to the top of the scale. As the needles pass through 120 °F, the amber caution light on the indicator will illuminate and, as the needles pass through 150 °F, the red warning light on the indicator will illuminate. The BATT HOT caution light on the caution-warning light panel and the master caution light will illuminate.

2.5.4.3 Battery Isolation Switches. The battery isolation switches are located on the electrical indicator and control panel (Figure 2-17). The three-position switch is labeled NORM, ISOL, and O'RIDE. The switch is at NORM during normal operation. When positioned at ISOL, the batteries are disconnected from the paralleling bus. The O'RIDE position allows a second engine start utilizing the batteries only. Aircraft with two-position switches are identical in operation except the O'RIDE position is not provided.

2.5.4.4 Battery Hot Caution Light. The BATT HOT caution light illuminates when the temperature of either battery reaches 150 °F. The caution light is located on the caution-warning light panel panel (Figure 2-33). Illumination of the BATT HOT caution light will also trigger the master caution lights.

2.5.5 Electrical Master Switch. The two-position guarded ELEC MASTER switch (27, Figure FO-3) on the overhead panel controls all dc power. When the switch is at ON, battery bus power is supplied to the battery switch to make it effective and the contacts of the reverse-current relays can close when dc generator output is available to put the generators on the line. Moving the electrical master switch to OFF takes the dc generators off the line by opening the contacts of the reverse-current relays and power is removed from the battery switch. The battery bus is the only dc bus that remains energized when the electrical master switch is off.

2.5.6 BATTERY Switch (CT-39E Aircraft and CT-39G Aircraft 158843 and 158844). The BATTERY switch, on the electrical indicator and control panel (Figure 2-17), controls battery bus power to the dc system, but is effective only if the electrical master switch is at ON. When the battery switch is at ON, a bus tie-in relay connects the battery bus to the paralleling bus. In flight, this relay closes only if either dc generator is on the line. Therefore, if both dc generators fail in flight or are turned off, the paralleling bus is disconnected automatically from the battery bus and the paralleling bus can no longer power the dc essential bus. When this occurs, an emergency relay is actuated if the electrical master and battery switches are on to ensure that the dc essential bus remains energized by connecting it directly to the battery bus. Should this automatic power transfer system fail, moving the battery switch to ESS energizes the essential bus emergency relay to connect the dc essential bus to the battery bus. On the ground, if the electrical master and battery switches are ON, the parallel bus energizes the essential bus.

CAUTION

The contacts of the essential bus emergency relay are used in flight only when the relay is energized by either of these two emergency transfer methods. To prevent unnecessary use of the relay, the ESS position of the battery switch should be used in flight only as required.

Whenever the battery switch is at OFF, isolation relays disconnect the batteries from the battery bus. If

the battery switch is turned OFF and both dc generators are on the line, all dc buses except the battery and starter buses remain energized.

2.5.7 BATTERY Switches (CT-39G Aircraft 159362 Through 159365 and 160053 Through 160057). The dual battery switches on the electrical indicator and control panel (Figure 2-17) control battery bus power to the dc bus system, but are effective only if the electrical master switch is at ON. The three-position battery switches are labeled ON, OFF, and EMERG. When either battery switch or both battery switches are placed at ON, the battery isolation relay(s) are energized and the battery(ies) are connected to the battery bus. With the battery bus energized, the essential bus relay and the battery bus relay are energized, connecting the battery bus to the dc essential bus and the paralleling bus.

In flight, if either dc generator should fail, the battery bus will continue to energize the dc essential bus and the paralleling bus. If both dc generators fail, the essential bus relay will open, the essential bus emergency relay will automatically be energized closed, and the battery bus will continue to power the dc essential bus and the paralleling bus. If the essential bus emergency relay should fail, as evidenced by power loss to the essential bus, moving either battery switch to EMERG will ensure that battery bus power is connected to the dc essential bus and the paralleling bus.

With the battery switches at OFF, both batteries are disconnected from the aircraft electrical system. With the switches at EMERG, the battery isolation system is bypassed and the battery bus is connected to the dc essential bus and the paralleling bus.

2.5.8 Dc Generator Switches. The left and right dc generator (LH DC GEN and RH DC GEN) switches on the electrical indicator and control panel (Figure 2-17) control the dc generators, provided the electrical master switch is at ON. When the engine speed is above approximately 40-percent rpm and the dc generator switch is at ON, the respective dc generator powers the system. If dc generator voltage becomes excessive (31 to 33 volts), an overvoltage relay cuts the affected generator out of the circuit. Then the switch may be held at RESET momentarily and released to ON in an attempt to restore generator operation. When a dc generator switch is OFF, the corresponding generator is off the line. The dc generator switches are spring loaded from RESET to OFF.

Note

During engine start (ground or air) or when the engine master switch is at OFF, or when the engine fire pull handle is actuated, the corresponding dc generator is taken off the line regardless of the setting of the dc generator switch.

2.5.9 Dc Voltmeter. The dc voltmeter on the electrical indicator and control panel (Figure 2-17) normally shows the voltage available at the dc essential bus. When the left or right pushbutton switch (PUSH LH GEN VOLT or PUSH RH GEN VOLT) below the voltmeter is pressed, the voltmeter shows the voltage of the selected left or right dc generator. The voltmeter should show about 28 volts when engine speeds are above idle revolutions per minute.

2.5.10 Dc Loadmeters. The left and right dc loadmeters on the electrical indicator and control panels (Figure 2-17) show the electrical load being drawn from the corresponding dc generators in terms of percentage of total generator output. Because of the balance control that equalizes the load of both generators, the loadmeter readings should be within approximately 10 percent of the rated load and within one division of each other.

2.5.11 Dc Generator-Off Caution Light. This placard-type DC GEN OFF caution light on the caution-warning light panel (12, Figure 2-33) is powered by the dc essential bus and comes on when either dc generator has failed or is taken off the line because of overvoltage or undervoltage. Because one light is used for both dc generators, it is necessary to refer to the dc loadmeters to determine which generator is not on the line. (The failed generator will have a very low or zero loadmeter reading, and the operational generator will have a high loadmeter reading.) The dc generator-off caution light comes on during engine start until engine speed is at idle and both generators are operating normally. The master caution lights come on when the dc generator-off caution light comes on.

2.5.12 Dc Generator Overheat Caution Lights. Two placard-type caution lights (LH DC GEN HOT and RH DC GEN HOT) are on the caution-warning light panel (12, Figure 2-33). These lights are powered by the dc essential bus and come on when the respective dc generator is overheated. The master caution lights also come on when either over-temperature light is illuminated. During prolonged engine cranking, a generator overheat light may come on, but it should go out when the start is obtained. If an overheat light comes on in flight, the affected generator should be turned off to relieve the generator from all electrical loads.

2.5.13 External Power Receptacles. The two dc external power receptacles are within a single access door on the left side of the fuselage behind the wing

trailing edge (Figure 3-12). The oval dc utility receptacle permits all dc buses except the starter bus to be powered from an external source for system operation and battery charging. The rectangular dc starter receptacle permits an external source to power only the starter bus. When the external power source plug is connected to the starter receptacle, it actuates a switch which prevents the aircraft batteries from being connected to the starter bus. There is no ac external power receptacle.

2.5.14 Circuit Breakers. The dc and ac electrical distribution circuits are protected by push-to-reset type, trip-free circuit breakers. After being tripped by a circuit overload, a trip-free circuit breaker cannot be reset immediately because it needs a cooling period. The dc essential bus is protected by a three-conductor circuit. Circuit breakers accessible to the flightcrew are shown in Figure 2-18.

2.5.15 Ac Electrical Power Distribution. Ac power from the main (or No. 1) or standby (or No. 2) inverter is distributed from the 115-vac secondary bus and the 115-vac primary bus through transformers, two 26-vac buses, and four 5-volt indirect light buses. On CT-39E aircraft, an engine-driven ac generator on each engine provides power for windshield anti-icing only. On CT-39G aircraft, the No. 2 inverter provides power for windshield anti-icing. The ac electrical system is shown schematically in FO-12 and FO-13.

2.5.15.1 Ac Primary Bus. The 115-volt ac primary bus is normally powered off the ac secondary bus. However, during engine starts or in the event of main (or No. 1) and standby (or No. 2) inverter failure, the ac primary bus is automatically powered by a three-phase, instrument inverter. Ac primary bus power is applied to a stepdown transformer which, in turn, powers a 26-vac bus.

2.5.15.2 Ac Secondary Bus. The 115-vac secondary bus is normally powered by the main (or No. 1) inverter. If the standby (or No. 2) inverter is turned on or is operating because of a main (or No. 1) inverter failure, the ac secondary bus becomes powered by the standby (or No. 2) inverter. Ac secondary bus power is applied to stepdown transformers that in turn, power one 26-volt and four 5-vac buses.

2.5.15.3 Automatic Inverter Changeover Control (CT-39E). The automatic inverter changeover control during normal operation (inverter switch at MAIN) energizes the main inverter power relay and connects the dc essential bus to the main inverter. When the voltage output of the main inverter drops below 90 volts, the automatic inverter changeover control deenergizes the main inverter power relay, energizes and locks in the standby inverter power relay, and energizes the ac power transfer relay to connect standby inverter power to the 115-vac secondary bus. The automatic inverter changeover control is bypassed and the ac transfer relay is energized when the inverter switch is manually positioned at STBY.

The inverter control circuitry allows ensured manual selection of standby ac power, bypassing the automatic transfer system if the instrument inverter control relay malfunctions. If an ac electrical power loss to the engine and left-hand flight instruments occurs when the inverter selector switch is at MAIN, the inverter selector switch is placed at STBY.

2.5.15.4 Automatic Inverter Changeover Control (CT-39G). The automatic inverter changeover control during normal operation (inverter switch at No. 1) energizes the No. 1 inverter power relay and connects the dc essential bus to the No. 1 inverter. When the voltage output of the No. 1 inverter drops below 90 volts, the automatic inverter changeover control deenergizes the No. 1 inverter power relay, deenergizes the No. 2 inverter delta-wye relay to disconnect the No. 2 inverter from the windshield heating system, and energizes the ac power transfer relay to reconnect No. 2 inverter power to the 115-vac secondary bus. In addition, the instrument inverter power and instrument inverter delta-wye relays are energized to connect the instrument inverter to the windshield heating system. Placing the inverter switch at No. 2 will bypass the automatic changeover control and accomplish this same ac and windshield heating system power transfer.

2.5.16 INVERTER Switch (CT-39E). The three-position INVERTER switch on the electrical indicator and control panel (Figure 2-17) uses dc essential bus power to control the main and standby inverters. For normal operation, the inverter switch should be at MAIN. With the switch in this position, the main inverter is selected and supplies power to all ac buses, and the standby inverter is held inoperative through a transfer relay as long as the main inverter is operating. If the main inverter fails during normal operation, the inverter changeover control automatically engages the standby inverter. The standby inverter may also be selected manually by the STBY position of the inverter switch.

Moving the inverter switch to OFF disconnects dc power from the inverter control circuits and shuts down the inverter.

2.5.17 INVERTER Switch (CT-39G). The three-position INVERTER switch on the electrical indicator and control panel (Figure 2-17) uses dc essential bus power

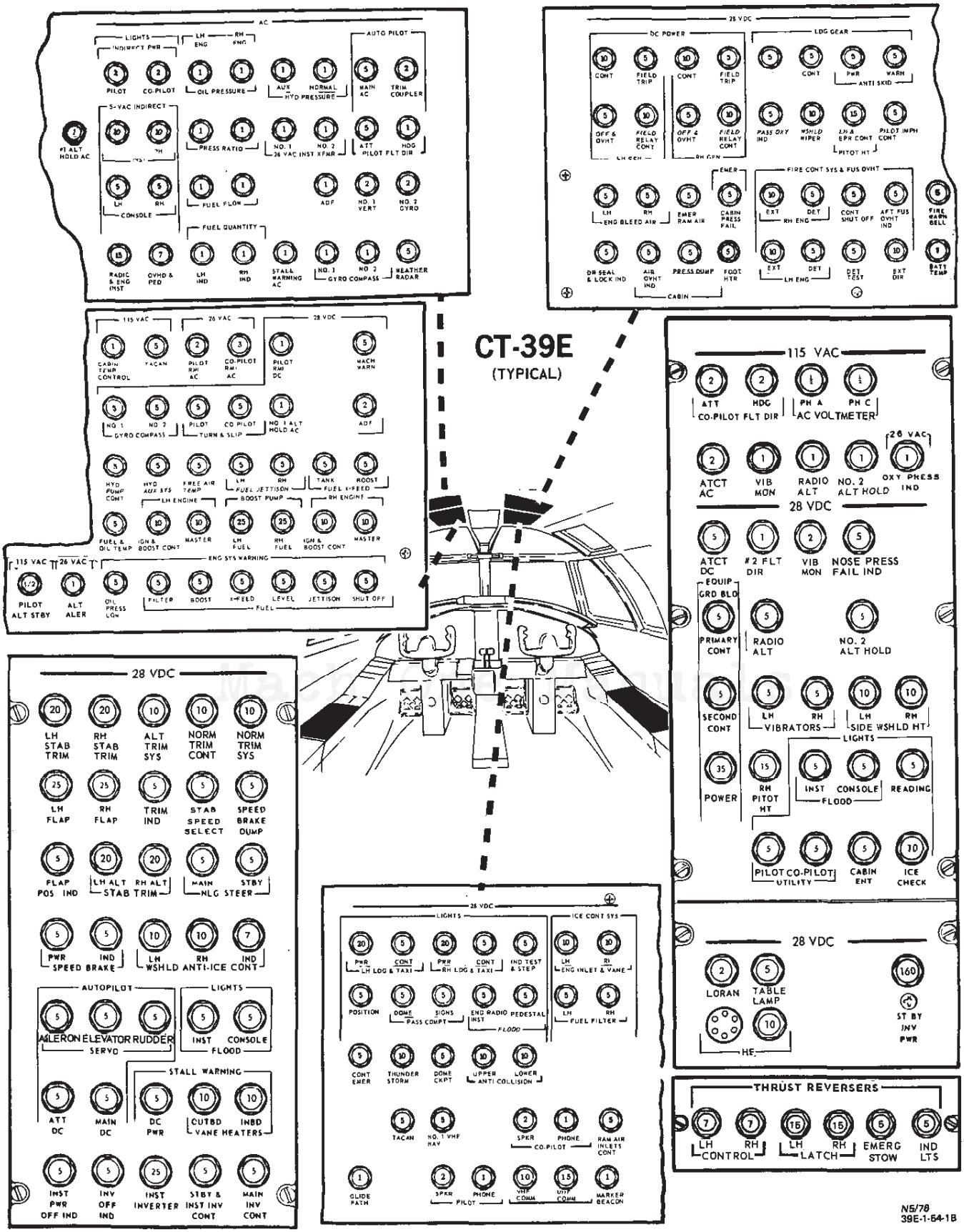


Figure 2-18. Circuit Breaker Panels (Sheet 1 of 5)

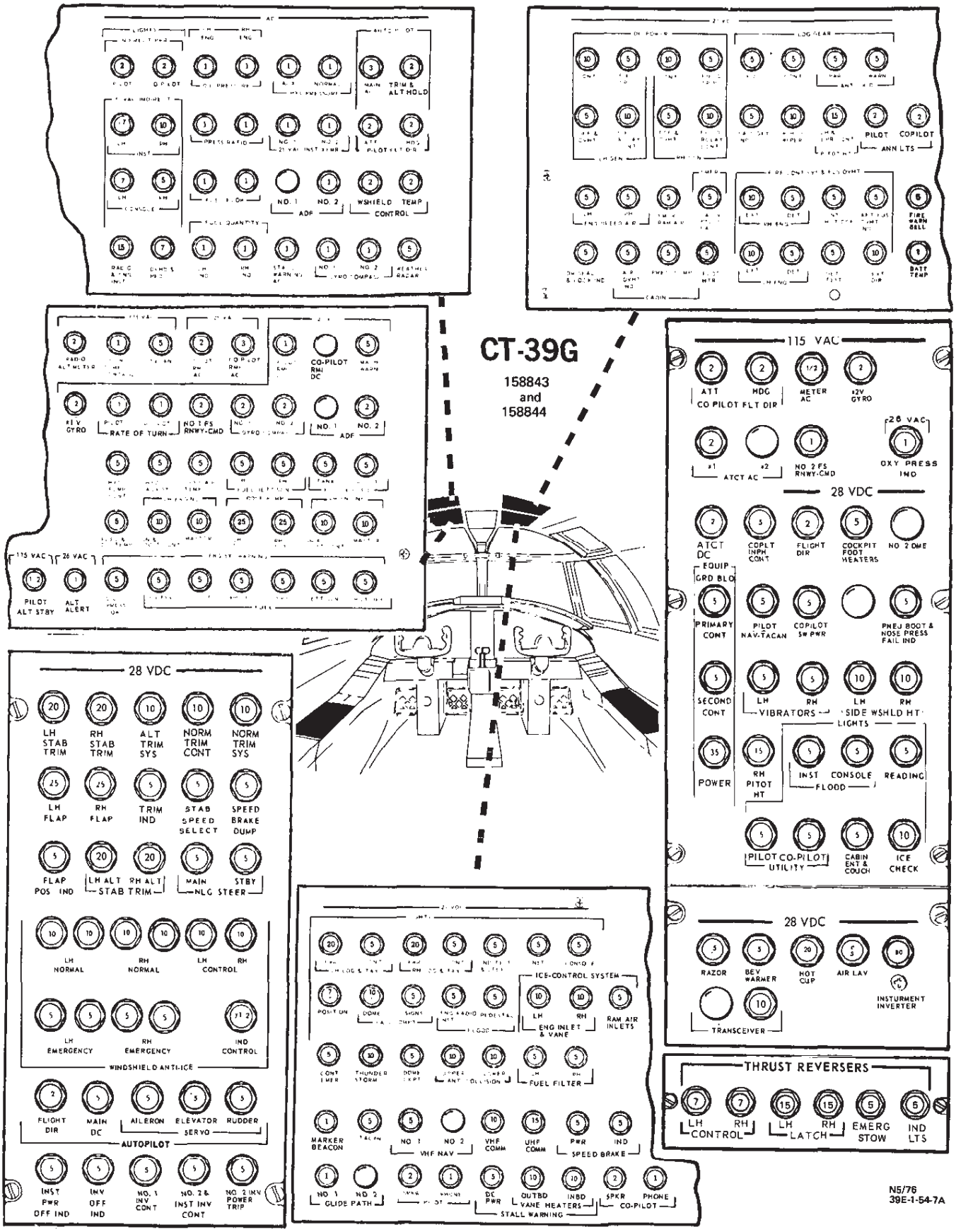


Figure 2-18. Circuit Breaker Panels (Sheet 2 of 5)

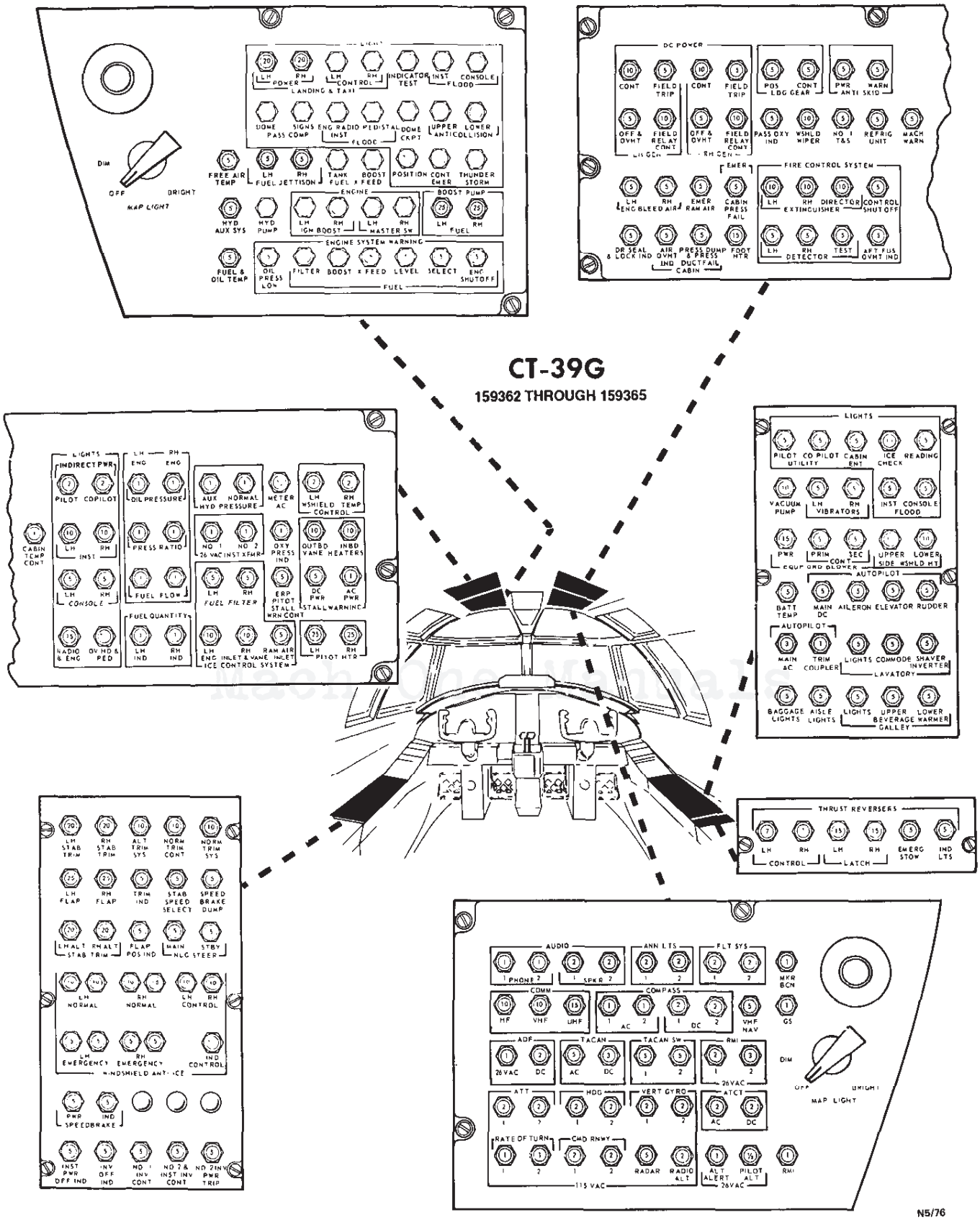


Figure 2-18. Circuit Breaker Panels (Sheet 3 of 5)

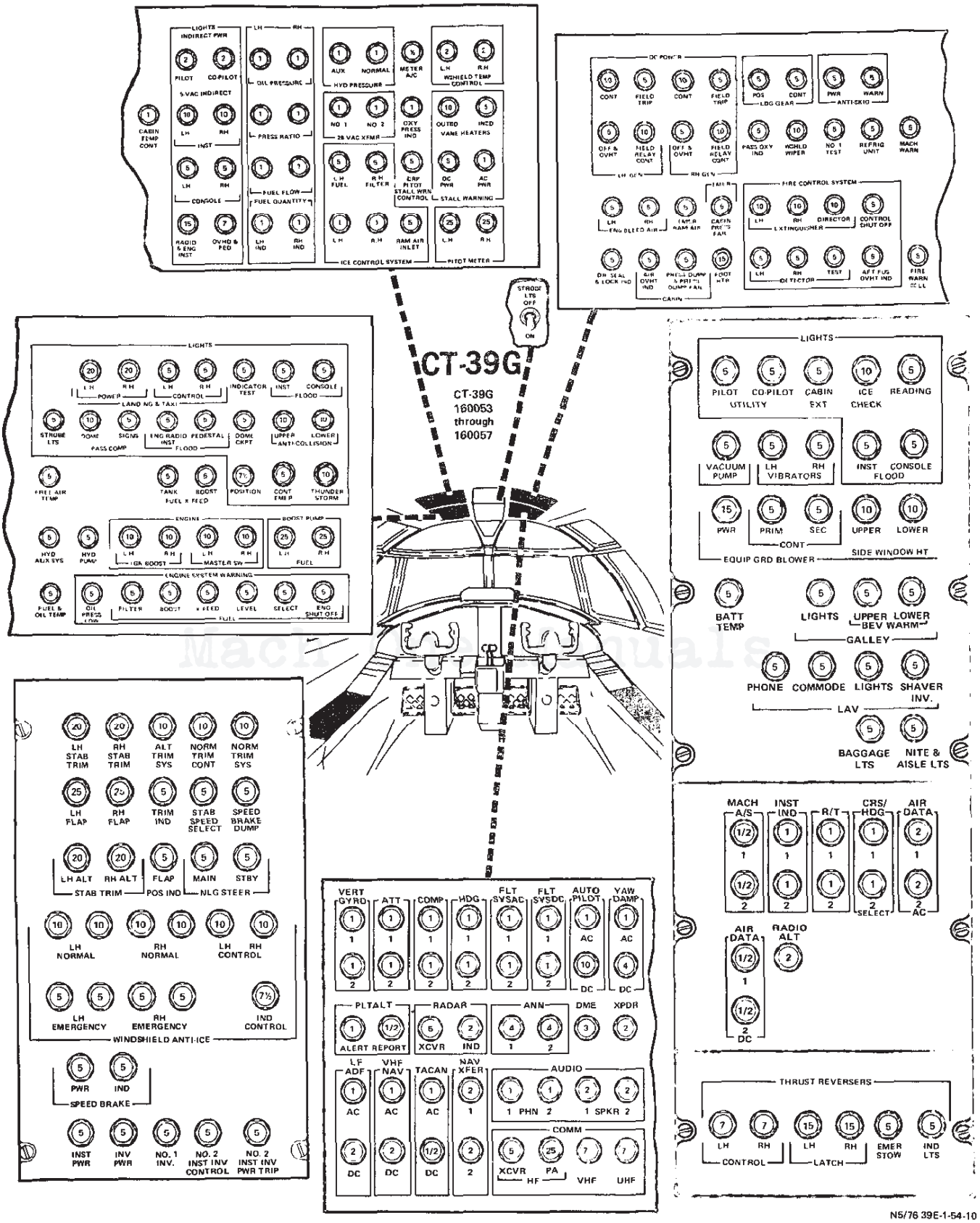


Figure 2-18. Circuit Breaker Panels (Sheet 4 of 5)

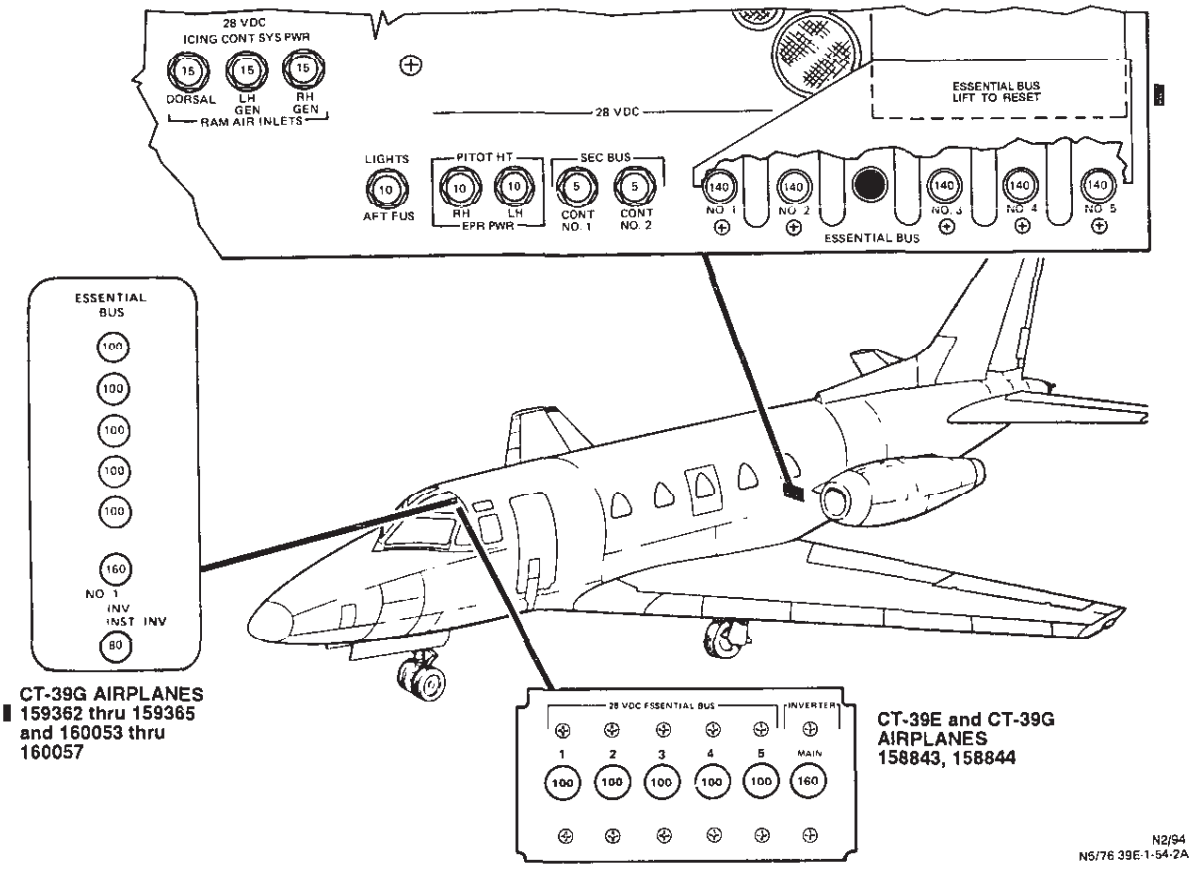


Figure 2-18. Circuit Breaker Panels (Sheet 5 of 5)

to control the Nos. 1 and 2 inverters. For normal operation, the inverter switch should be at NO. 1. With the switch in this position, the No. 1 inverter is selected and supplies power to all ac buses. The No. 2 inverter supplies power to the windshield anti-icing system if the FWD WSHIELD switch is at BOTH, LH, or RH as long as the No. 1 inverter is operating. If the No. 1 inverter fails during normal operation, the No. 2 inverter will transfer to the ac bus system and the instrument inverter will come on to power the windshield heating system. This transfer may also be selected by manually placing the inverter switch at NO. 2. Moving the inverter switch to OFF disconnects dc power from the inverter control circuits and shuts down the inverters. During normal operating conditions (No. 1 inverter and windshield heat selected) in the event of a dc generator failure, the No. 2 inverter will automatically be shut down and the instrument inverter will then provide one-half of the normal power to the pilot windshield. During this condition, if the No. 1 inverter should fail, the No. 2 inverter will automatically power the 115-vac primary bus.

2.5.18 Inverter Protection (CT-39G). If the No. 1 inverter fails, it will be locked out through the automatic changeover relay. This will also occur if the generators are turned off. To reset, place inverter switch to OFF and then return switch to the No. 1 position. If the No. 2 inverter power supply increases to or exceeds 200 amps, a thermal trip will take place, locking out the inverter. To reset (only once), pull out and push in the No. 2 inverter power trip circuit breaker. Failure of the instrument inverter control relay could result in two inverters powering the ac primary bus. If this is suspected, pull the instrument inverter power circuit breaker.

2.5.19 Inverter Voltage Test (INV VOLT TEST) Switch (CT-39E). This three-position INV VOLT TEST switch on the electrical indicator and control panel (Figure 2-17) is used to select the phase of ac voltage indicated on the ac voltmeter.

2.5.20 Ac Voltmeter (CT-39E). The ac voltmeter on the electrical indicator and control panel (Figure 2-17) shows the voltage of the particular phase of inverter output according to the selected position of the inverter voltage test switch. The voltmeter indicates the voltage supplied by the main, the standby, or the instrument inverter, whichever is operating.

2.5.21 Ac Voltmeter (CT-39G). The ac voltmeter on the electrical indicator and control panel (Figure 2-17) shows the voltage supplied by the No. 1, the No. 2, or the instrument inverter, whichever is powering the primary bus.

2.5.22 Ac Frequency Meter. The ac frequency meter shows the frequency of the No. 1, the No. 2, or the instrument inverter, whichever is powering the primary bus. On CT-39G aircraft 158843 and 158844, the frequency meter is on the electrical indicator and control panel (Figure 2-17). On CT-39G aircraft 159361 through 159365 and 160053 through 160057, the frequency meter is on the right-hand instrument subpanel.

2.5.23 Inverter Failure Caution Lights (CT-39E). Two placard-type caution lights (MAIN INV FAIL and STANDBY INV FAIL) are on the caution-warning light panel (12, Figure 2-33). The lights are powered by the dc essential bus and come on to indicate a failure of the respective inverter.

2.5.24 Inverter-Off Caution Lights (CT-39G). Two placard-type caution lights (NO. 1 INV OFF and NO. 2 INV OFF) are on the caution-warning light panel (12, Figure 2-33). The lights are powered by the dc essential bus and come on to indicate a failure of the respective inverter.

2.5.25 Ac Instrument Power-Off Caution Light (CT-39E). This placard-type AC INST PWR OFF caution light is on the caution-warning light panel (12, Figure 2-36). This light is powered by the dc essential bus and comes on when the ac primary bus is not energized. With the inverter switch at MAIN, illumination of the ac instrument power-off caution light is followed by the light going out when the standby inverter is engaged automatically. If the standby inverter is on because of main inverter failure, illumination of the ac instrument power-off caution light is followed by the light going out when the instrument inverter is automatically engaged. Illumination of the ac instrument power-off caution light under this condition indicates the failure of the instrument inverter, resulting in a complete loss of all ac electrical power.

2.5.26 Ac Instrument Power-Off Caution Light (CT-39G). This placard-type AC INST PWR OFF caution light is on the caution-warning light panel (12, Figure 2-33). This light is powered by the dc essential bus and comes on when the ac primary bus is not energized. With the inverter switch at No. 1, illumination of the ac instrument power-off caution light is followed by the light going out when the No. 2 inverter is engaged automatically. If the No. 2 inverter has been turned on or is on because of No. 1 inverter failure, illumination of the ac instrument power-off caution light is followed by the light going out when the instrument inverter is automatically engaged. Illumination of the ac instrument power-off caution light under this condition indicates the failure of the instrument inverter, resulting in a complete loss of all ac electrical power.

2.6 ELECTRONIC EQUIPMENT COOLING

Cooling of the electronic equipment on the ground is accomplished by a ground air blower in the nose compartment. (See FO-7 and FO-8.) Air is drawn into the nose compartment and circulated into the electronics equipment compartment through the open equipment bay pressure regulator. When both dc generators are operating or utility external dc power is applied, the blower will operate continuously on the ground. However, when only battery power is available and/or only one dc generator is operating, a thermal switch allows only intermittent operation of the blower. This is to prevent unnecessary power drain on the batteries and overheating of the generator.

2.7 EMERGENCY EQUIPMENT

2.7.1 Engine Fire and Overheat Detection Systems. A continuous-element-type alarm circuit is installed in each engine pod to detect any engine fire or overheat condition. A sealed firewall in each engine pod separates the forward fire section from the aft overheat section. The fire and overheat detection system is controlled by a hermetically sealed control unit powered by 28 vdc through the essential bus. When a fire or overheat condition exists, a red warning light in the respective fire pull handle comes on and the fire warning bell rings.

2.7.1.1 Fire Warning Lights. Red fire warning lights in the two engine fire pull handles (Figure 2-19) are provided to warn of an overheat or fire condition in either engine pod. When a fire or overheat condition is detected, the warning light in the fire pull handle for the respective pod illuminates. The warning light circuit may be tested by pressing the fire and overheat detection circuit test button.

2.7.1.2 Fire Warning Bell. A fire warning bell is mounted behind the copilot on the forward right side of the station bulkhead. When a fire or overheat condition exists, the respective engine fire and overheat detection system will ring the bell. A red warning light in the respective fire pull handle will indicate the warning. Pulling the fire pull handle will arm that engine fire extinguishing system and shut off the fire warning bell. The fire warning bell circuit may be tested by pressing the fire and overheat detection circuit test button.

2.7.2 Fire and Overheat Detection Circuit Test Button. This push-to-test button on the fire emergency control panel (Figure 2-19) is used to test the

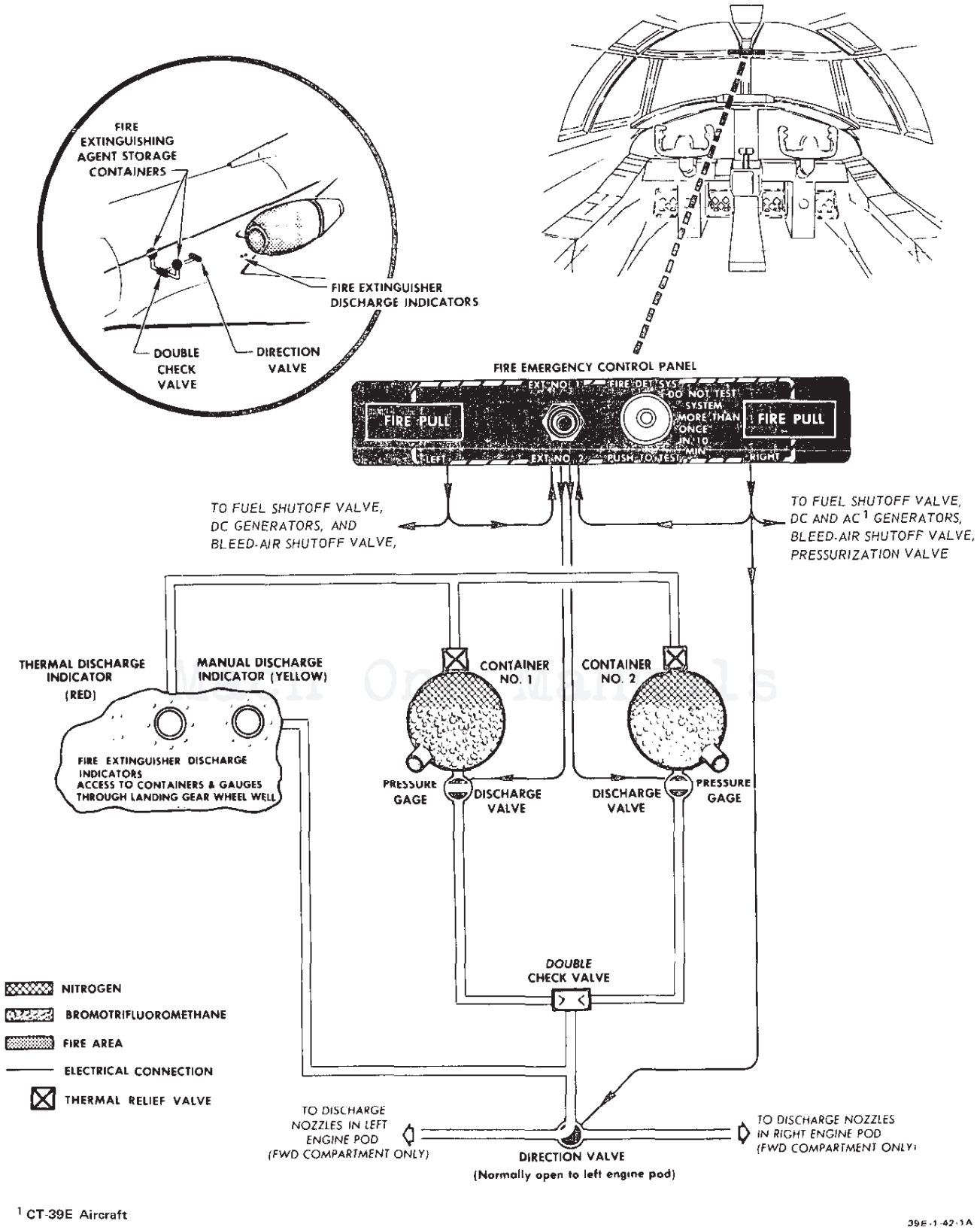


Figure 2-19. Engine Fire Extinguishing System

engine fire and overheat detection circuits and the aft fuselage overheat detector circuit. Pressing the button illuminates both engine fire pull handles and the aft fuselage hot caution light if the circuits are continuous and the control units are in proper operating condition. Pressing the push-to-test button will also ring the fire-warning bell. Each engine fire pull handle and caution light is equipped with two light bulbs. Both bulbs in each handle and caution light should come on when the fire and overheat detection circuit test button is pressed. The button is powered by the dc essential bus.

CAUTION

Do not test the fire detection system more than once in any 10-minute period as damage may result if relays are not allowed to cool sufficiently.

2.7.3 Aft Fuselage Overheat Detection System. A series of thermal switches surrounding the bleed-air pressurization ducts in the aft fuselage section detect abnormal temperature conditions (above 275 °F (135 °C)) that may result from bleed air escaping from a ruptured duct. The system warns of an overheat condition by automatically turning on the amber aft fuselage hot caution light on the caution-warning light panel. When the detector circuit is tested by pressing the fire detection system test button, the aft fuselage hot caution light comes on. The aft fuselage overheat detection system is independent of the engine fire and overheat detection system.

2.7.3.1 Aft Fuselage Hot Caution Light. This amber caution light (12, Figure 2-33) warns of an overheat condition around the bleed-air pressurization ducts in the aft fuselage and reads AFT FUS HOT when illuminated.

2.7.4 Engine Fire-Extinguishing System. An HRD (high rate-of-discharge), fixed fire-extinguishing system is provided in the forward compartment of each engine. Bromotrifluoromethane (CB_rF_3), used as the agent, is stored under pressure in two spherical containers. A pressure gauge on each container is visible from the main wheelwell. On the bottom of each container is a discharge valve with a cartridge that is ignited by 28-vdc power when the fire extinguisher selector switch is actuated (engine fire pull handle previously pulled). When a cartridge is fired, the contents of one container are discharged by nitrogen pressure and forced through delivery lines to a direction valve from where it is routed into the forward compartment of the affected engine. The sealed firewall between the forward and aft engine compartments prevents the extinguishing agent from entering the aft compartment. Two disk-type indicators are flush mounted in the fuselage below the left engine. If the contents of either or both containers have been discharged into the engine compartments, the yellow disk will be ruptured. If the contents of either or both containers have been discharged overboard automatically as a result of an overheat condition causing excessive pressure within the containers, the red disk will be ruptured. If both the disks are intact, the system is not discharged. The indicators are readily available for visual inspection and shall be checked for condition before each flight.

2.7.4.1 Engine FIRE PULL Handles. Two mechanically interconnected FIRE PULL handles, one for each engine, are provided on the fire emergency control panel for the purpose of isolating either engine pod. (See Figure 2-19.) When the handle is pulled, the respective fuel shutoff valve is closed, the dc and ac (CT-39E aircraft) generators are taken off the line, the engine air-bleed valve is shut off, the fire extinguisher selector switch is armed, the discharge valve on each container is armed, the thrust reverser hydraulic control valves are deactivated if the thrust reverser switch is in the ON position, the fire bell is silenced, and the fire control panel is illuminated. Pulling the right-hand fire pull handle positions the direction valve to direct the agent to the right-hand engine. The left fire pull handle has no effect on the direction valve because the valve is normally open to the left engine pod. When the right-hand engine fire pull handle is actuated, the emergency bleed-air shutoff valve is energized closed. If the second engine fire pull handle is pulled, the first handle automatically retracts to its original position and, except for the generator, those items previously turned off are reinstated unless the engine master switch to the first engine had been placed in the OFF position.

2.7.4.2 Fire Extinguisher Selector Switch. The fire extinguisher selector switch on the fire emergency control panel (Figure 2-19) is powered by the dc essential bus. The switch has three positions: EXT NO. 1, EXT NO. 2, and an unmarked off (center) position. When the selector switch is positioned to either EXT NO. 1 or EXT NO. 2 after having been armed by an engine fire pull handle, the extinguishing agent from one container is discharged to the engine pod selected by the handle. After the initial container has been expended, the fire extinguisher selector switch may, if necessary, be positioned in the opposite direction to discharge the second container.

2.7.5 Emergency Lighting. Emergency lighting is provided in the interior of the aircraft to be used in case of a complete electrical failure. Four emergency

lights are installed: one light is directed at the emergency escape hatch; one light is directed on the cabin floor; one light is directed at the main entry door; and one light in the cockpit on the bulkhead, aft and to the left of the copilot head, is directed on the pilot controls and panels. The four lights are powered in pairs by two separate power units, each containing a sealed 7.2-volt nickel-cadmium battery and a power-off sensing unit, connected to the dc electrical system through the essential bus. Under normal operating conditions, dc essential bus power keeps the emergency power unit batteries charged. In case of essential bus failure, the voltage drop is sensed and the emergency lights automatically illuminate. Each battery will sustain its pair of lights approximately 30 minutes.

2.7.5.1 Emergency Light Switch. The EMER light switch on the lights control panel (23, FO-3) is a three-position, toggle-type switch with TEST, ARM, and DISARM positions. To check the emergency lights, batteries, and circuitry, position the switch at TEST and all four lights should illuminate. With the engines operating and the electrical master switch on, positioning the emergency light switch at ARM puts the circuit into a standby condition until there is a voltage drop in the dc essential bus. The emergency lights are automatically illuminated when this voltage drop energizes the batteries in the power units. When power is applied to the dc essential bus and the emergency light switch is positioned at ARM, the emergency batteries are recharged. To prevent battery discharge, the emergency light switch should always be placed in the DISARM position before the battery and electrical master switches are placed at OFF before leaving the aircraft. When the batteries are fully discharged, it takes approximately 16 hours to fully recharge them.

2.7.6 Ground Escape Hatches. The plug-type emergency escape hatch (Figure 13-1) on the right side of the cabin of CT-39E aircraft and on both sides of the cabin of CT-39G aircraft is provided for ground escape and ditching. The hatches may be unlocked by handles from the inside or outside of the aircraft and lifted into the cabin compartment.

2.7.7 Hand Fire Extinguisher. A portable fire extinguisher is mounted on the cockpit bulkhead aft of the pilot seat to be used by the crew for cockpit or cabin fires. The extinguishing agent, CO_2, can be used most effectively on normal combustion, hydrocarbon, or electrical fires. If the extinguisher has been used, the lockwire on the trigger will be broken. On CT-39G aircraft 160053 through 160057, a water-type fire extinguisher is also installed in the cabin. This extinguisher is used for fabric fires.

2.7.8 Walkaround Oxygen Bottle. Refer to paragraph 2.25, OXYGEN SYSTEM.

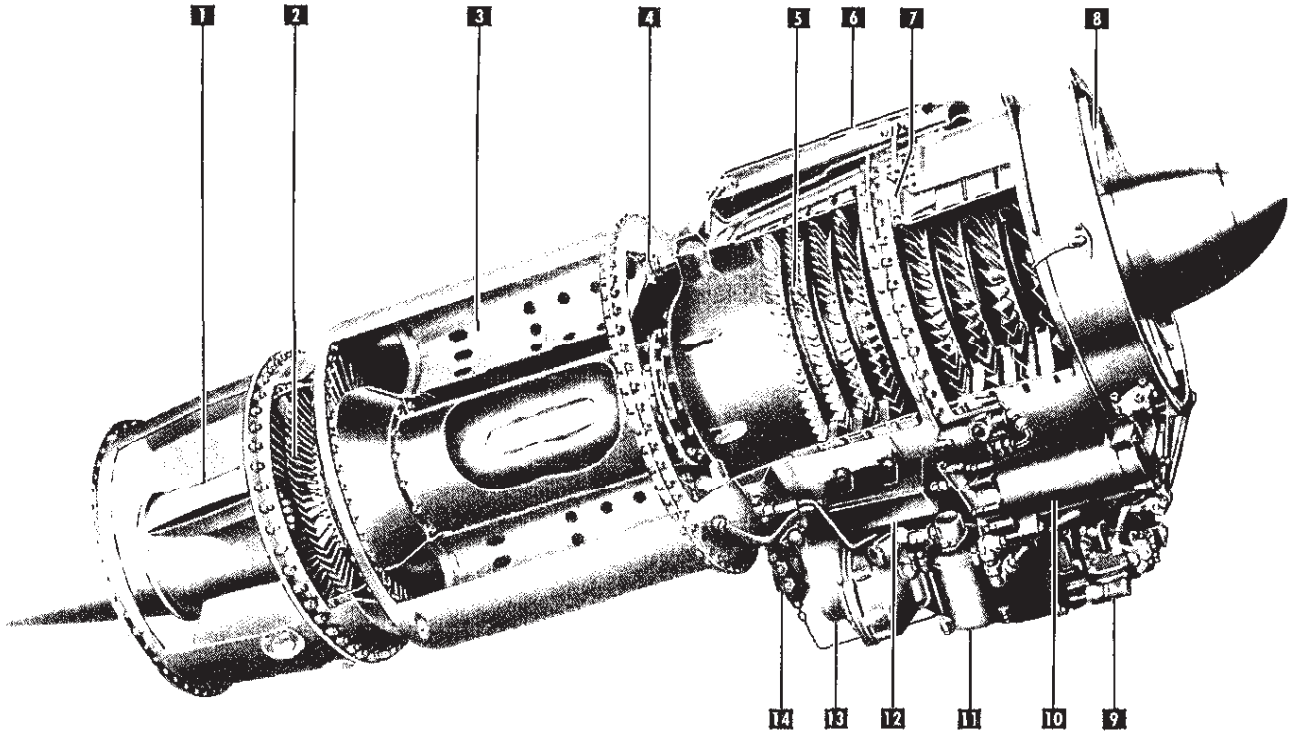
2.7.9 Liferaft. On all CT-39 aircraft, there is stowage space for a 10-man or two 4- to 7-man liferafts in the baggage compartments.

2.8 ENGINES

Power is supplied by two Pratt & Whitney JT12A-8A gas turbine engines. (See Figure 2-20.) The rated sea-level static thrust of each is 3,300 pounds at takeoff thrust. The engines are shock mounted in separate pods attached to the aft fuselage section by pylons. The engine has a single-rotor, nine-stage, axial-flow compressor driven by a two-stage reaction turbine. The combustion chamber contains eight burner cans into which fuel is sprayed through single, dual-orifice nozzles mounted at the inlet of each can. High-energy ignition units and spark igniters are used to start combustion and provide continuous ignition. A hydromechanical fuel control schedules the fuel flow to provide the thrust called for by the throttle setting in the cockpit with automatic compensation for conditions at the compressor air inlet and compressor discharge. An air-bleed valve opens automatically to bleed interstage compressor air overboard to facilitate engine starting and operation at low thrust. An anti-ice system protects the engine guide vanes from ice formation. The engine power takeoff gearbox, used to drive engine accessories, is waistmounted and is driven by the compressor rotor through a bevel gear and shaft system that also serves as the input system during starting.

2.8.1 Engine Fuel Control System. Fuel flow to the engine is mechanically controlled by throttle movement and is delivered and regulated by the engine fuel control system. This system includes the engine-driven fuel pump unit, fuel heater, fuel bypass valve, hydromechanical fuel control unit, fuel oil cooler, and pressurization and dump valve. The engine fuel control system is shown schematically in Figure 2-21.

2.8.1.1 Engine Fuel Control Unit. An engine-driven hydromechanical fuel control unit regulates fuel flow to the engine. During operation, the fuel flow is determined by a variable-orifice main metering valve in the control that is governed by control signals from a mechanical computer. The metering system meters the fuel flow to be sent to the engine burners in accordance with the amount of thrust required. The computing system monitors four operating parameters: compressor inlet pressure, engine rpm, throttle position, and compressor discharge pressure. The computing system then adjusts the metering system accordingly, thus providing for optimum engine performance.



FS-744

Mach One Manuals

1. EXHAUST CONE
2. TWO-STAGE REACTION TURBINE
3. CAN-ANNULAR COMBUSTION CHAMBER
4. FUEL NOZZLE
5. COMPRESSOR SECTION
6. OIL TANK
7. COMPRESSOR BLEED STRAP
8. AIR INTAKE
9. FUEL CONTROL
10. FUEL HEATER
11. FUEL FILTER
12. IGNITION UNIT
13. ACCESSORY DRIVE GEAR BOX
14. PRESSURIZATION AND DUMP VALVE

39E-1-40-1

Figure 2-20. JT12A Engine

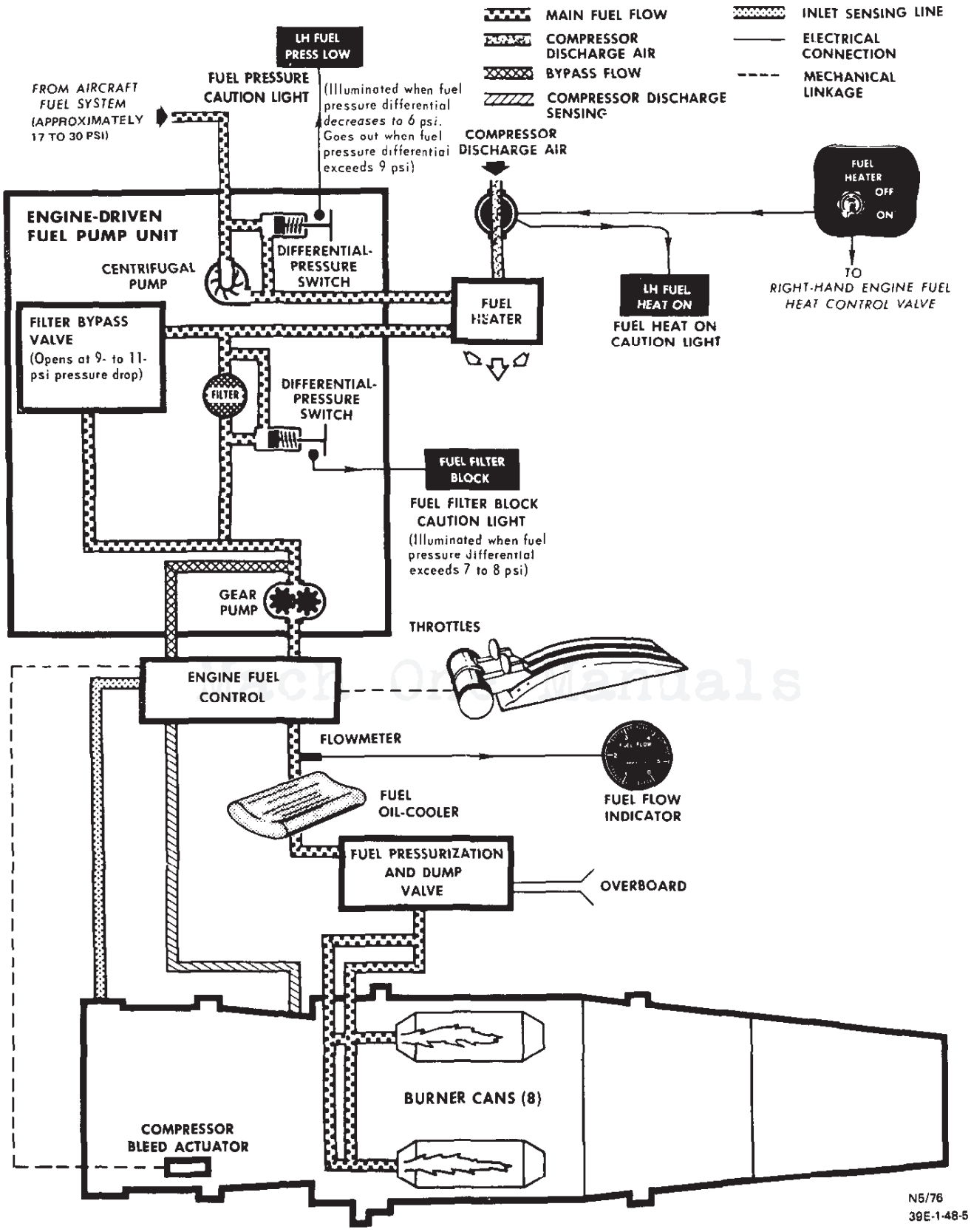


Figure 2-21. Engine Fuel Control System

Note

Condensation of water droplets in the burner pressure sensing line may cause a hung engine deceleration or high engine idle condition at altitudes above 20,000 feet because of ice blockage of the line or ice restriction of burner pressure bellows movement. The resulting burner pressure bellows false position can usually be corrected by fuel heater operation for 1 minute. If heater operation appears to help but does not completely correct the condition in 1 minute, heater operation for 1-minute periods may be repeated until the condition is corrected or no further improvement results. Do not operate the fuel heater continuously as excessive continuous operation of the fuel heater may cause overheating of the engine oil.

2.8.1.2 Fuel Pump Unit. The engine-driven pump unit supplies the high fuel pressure required by the engine. The unit has a centrifugal element, a filter, a filter bypass valve, and gear element. The centrifugal element provides an interstage fuel pressure and also has suction-feed capabilities in case of boost pump failure. FAA requirements dictate that the engine have a suction-feed altitude capability of at least 6,000 feet above the approved operational takeoff pressure altitude of the aircraft. It has been demonstrated that the aircraft has suction-feed capabilities that exceed the FAA requirements. It has also been determined that, at lower fuel temperatures, suction-feed operation can be accomplished at much higher altitudes. See Figure 2-22 for a suction-feed capability envelope. It is noted that suction-feed operation would normally be required only if the fuel crossfeed is not selected or if both tank-mounted fuel boost pumps become inoperative.

A differential-pressure switch is installed across the centrifugal element to sense fuel pressure on each side of the pump. When the pressure differential across the pump decreases to 6 psi, the respective low fuel pressure caution light comes on. When the pressure differential increases to 9 psi or more, the caution light goes out. A filter is between the centrifugal boost stage and the gear stage to ensure protection of engine

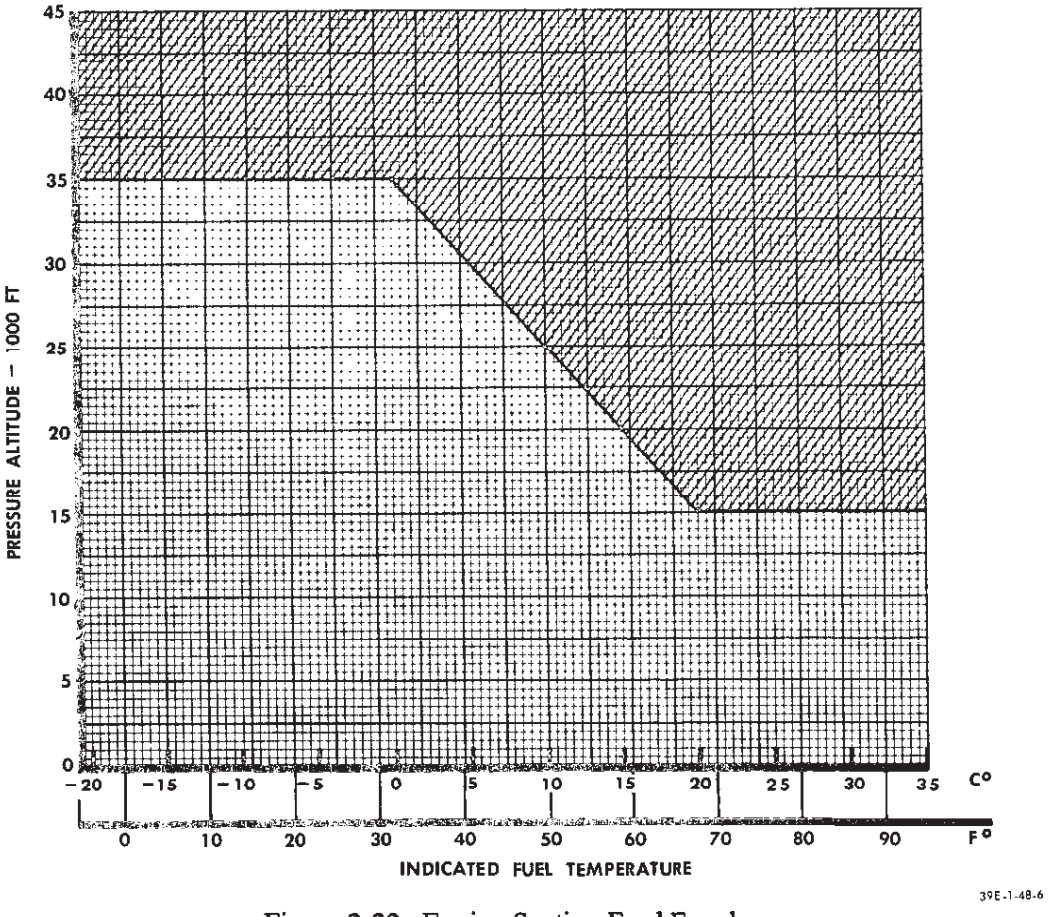


Figure 2-22. Engine Suction Feed Envelope

fuel components. In case the differential fuel pressure across the filter increases to 7 to 8 psi, a differential-pressure switch is actuated and the fuel-filter-blocked caution light comes on. If the differential fuel pressure increases to 9 to 11 psi, the filter bypass valve will open to provide unfiltered fuel flow to the gear stage of the pump.

2.8.1.3 Fuel-Filter-Blocked Caution Light. The placard-type, amber FUEL FILTER BLOCK caution light is on the caution-warning light panel (12, Figure 2-33). This light may be energized from one of two differential-pressure switches, one on each engine. These pressure switches sense fuel pressure drop across the fuel filters. In case either the left- or right-engine fuel filter becomes contaminated with ice or foreign matter and the pressure drop across the filter increases to 7 to 8 psi, the differential-pressure switch closes and the caution light will come on. The differential-pressure switch opens when the pressure drop decreases to 6 to 7 psi, and the caution light will go out. In case the fuel pressure drop increases to 9 to 11 psi, the filter bypass valve automatically opens, allowing full fuel flow to the engine. However, the caution light will remain on until normal fuel flow occurs. By using the fuel temperature in conjunction with the FUEL FILTER BLOCK caution light, it is possible to determine whether filter contamination is caused by ice or by foreign particles. This light is powered by the dc essential bus and is protected by a circuit breaker on the overhead circuit breaker panel.

2.8.1.4 Fuel Heater. A fuel heater, installed on each engine upstream of the filter, is used to prevent the fuel filter from becoming blocked with ice with subsequent operation of the filter bypass. The fuel passes through the heater and is heated when compressor discharge air is passed through the heater. The compressor discharge bleed air to each engine is controlled by a motor-driven valve when the fuel heater switch is placed in the ON or OFF position. When the valves are not completely closed, the respective fuel heater caution light will come on.

2.8.1.5 Fuel Heater Caution Light. Refer to paragraph 2.13, FUEL HEATER SYSTEM.

2.8.1.6 Fuel Pressurization and Dump Valve. The engine fuel pressurization and dump valve automatically directs fuel to one or both fuel manifolds, depending on engine operating conditions. At low flow, fuel passes through the primary manifold only. At high flows, both manifolds are used with fuel flow through the secondary manifold being scheduled by the pressurizing valve as a function of the pressure drop across the primary manifold and nozzles. The dump portion of the valve is a two-position valve that is hydraulically actuated by a pressure signal from the fuel control. The pressure signal maintains the dump valve in a closed position during engine operation. On shutdown, fuel from the control is shut off immediately, the pressure signal drops rapidly, and the dump valve opens, permitting fuel in the manifold to drain overboard. On starting, the dump valve is again closed by the high-pressure signal.

2.8.1.7 Fuel Oil Cooler. A fuel oil cooler is installed on the engine to cool the engine lubricating oil. All engine fuel passes through the fuel oil cooler which is installed downstream of the hydromechanical fuel control. As the engine lubricating oil passes through the cooler, the heat is transmitted to the inlet fuel.

No controls are provided for the fuel oil cooler since its function is automatic.

2.8.2 Engine Air-Bleed System. The engine has two separate air-bleed systems: high-pressure air bleed and overboard air bleed. High-pressure air bleed is available for airframe use such as supply of the cabin pressurization and engine inlet duct and guide vane anti-icing systems. The overboard air-bleed system is required to prevent compressor instability. High-pressure air is bled from the engine through two ports on the diffuser-case bleed manifold. Interstage air from the fourth stage is bled overboard to preclude compressor instability at certain thrust conditions. A compressor bleed valve within the compressor case is actuated at predetermined conditions, as scheduled by the fuel control.

2.8.3 Compressor Interstage Air-Bleed System. Engine interstage air is bled overboard from the fourth stage of each engine to preclude compressor instability at low engine thrust conditions. Operation of the compressor interstage air-bleed system is completely automatic. The interstage bleed valves are opened or closed by the engine fuel control at engine speeds ranging from 79- to 81-percent rpm.

Note

Engine speed fluctuations may occur in the 79- to 81-percent rpm range if the interstage bleed valve cycles. If the fluctuations do occur, advance or retard the throttles as necessary to obtain stable operation outside this range.

Because of a malfunction or failure, the bleed valve may not close as scheduled, resulting in the possibility of the valve remaining open at high thrust levels. Operating with the valve open for extended periods at thrust levels where it is normally closed imposes undesirable stress in the compressor section of the engine. It, therefore, is desirable to recognize this condition and reduce the

throttle to the bleed-valve-open point (approximately 81-percent rpm).

An engine with a bleed valve stuck in the open position can be readily identified by comparing revolutions per minute, fuel flow, and EGT between engines at a constant EPR. Because of the increased output of the engine with the malfunctioning valve, the revolutions per minute, fuel flow, and EGT will all read higher. At takeoff thrust, these differences become quite substantial, with EGT approaching the operating limits and precomputed EPR may not be obtainable for takeoff thrust setting.

2.8.4 Engine Pods. The engine pods (Figure 1-2), one on each side of the aft fuselage section, are attached to the fuselage by pylon structures. Each engine pod houses the engine, inlet duct, engine tailpipe, and engine accessories. The engine inlet duct assembly is attached to and installed with the engine. Each pod is equipped with engine inlet duct and guide vane anti-icing, overheat and fire detection warning system, and fire extinguishing lines. Engine air inlet duct flow-straightener vanes are installed in each nacelle nose assembly. The four vanes of the flow straightened are of hollow construction.

The pod is divided into two sections by a fire seal at the aft end of the compressor section that separates the compressor section from the combustion and exhaust section. In-flight cooling is provided by ram air to each compartment. On the ground, cooling is accomplished by convection circulation in the compressor compartment and by aspiration through the aft compartment. The aft compartment cooling air exits through an annular space around the exhaust nozzle opening. A fan in the starter-generator provides cooling-air circulation during ground operation of the unit and receives ram-air cooling during flight. Access to the engine is gained through four large, removable engine access doors. The lower half of the engine pod and three access doors on the upper half are removable. The lower door opens on hinges for line maintenance and is easily removable for engine removal.

CAUTION

Lower and aft engine cowling should be removed prior to prolonged high power turns to prevent scorching and/or buckling of engine cowlings.

2.8.5 Throttles and Friction Lock. Engine thrust is controlled by the throttles (Figure 2-23). The throttles are mounted in quadrant on top of the cockpit center pedestal where they are accessible to the pilot or copilot. The two throttles are connected to their respective engine fuel controls by mechanical linkage. A thumb-operated speedbrake switch is mounted on the left engine throttle. Each throttle is provided with a separate safety latch on the forward side that must be depressed before the throttle can pass from CLOSE to IDLE back into the CLOSE position. Both throttles are interlocked with the control surface gustlock to prevent reaching takeoff power when the gustlock is on. A friction control knob is provided on the pilot side of the throttle quadrant. There are throttle-actuated switches in the quadrant that control the sequence of events in the fuel boost pump, engine ignition, landing gear warning circuits, speedbrake open caution light, and thrust reverser system.

2.8.6 Engine Master Switches. A positive-position engine master switch (1 and 15, FO-3) is provided for each engine. These switches are on the forward end of the center pedestal, one on each side of the throttle quadrant. Moving the engine master switch to ON opens the respective engine fuel shutoff valve and electrical power is made available for subsequent starter operation. With the selected engine master switch in the ON position and the respective starter button depressed, switches in the throttle quadrant are actuated when the respective throttle is moved out of the CLOSE position to energize the fuel boost pump and engine ignition circuits. The engine master switches also control dc essential bus power to the thrust reverser system. The engine master switches are powered from the dc essential bus and each is protected by a circuit breaker on the overhead circuit breaker panel.

Note

- After engine shutdown, the engine master switch should be turned to OFF before the electrical master and battery switches are positioned at OFF. This allows the fuel shutoff valve to close. Then the electrical master and battery switches should be positioned at OFF.

- On aircraft with AFC 648 installed, when moving the engine master switch to on, the respective electrical fuel pump is powered.

2.8.7 Starter System. Engine starts may be made by use of the aircraft batteries or external power or by air starting. A combination starter-generator operates as a starter until engine speed reaches approximately 31- to 35-percent rpm and the starter current has decreased to approximately 200 amperes. At this time, the starter ceases to torque the engine and the solenoid-held push-pull starter button is automatically released, indicating

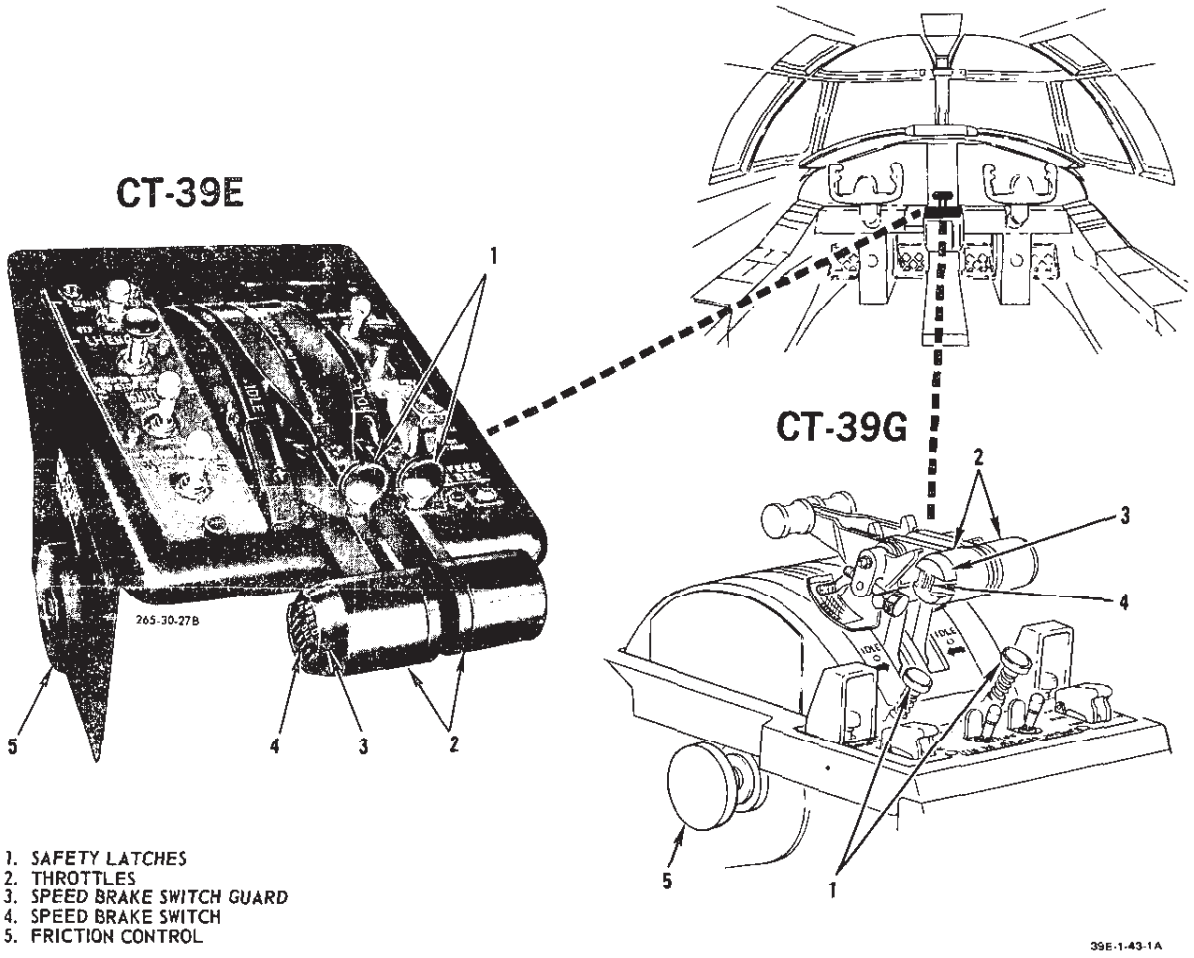


Figure 2-23. Throttles

the cranking cycle is ended. As engine revolutions per minute increases, the starter-generator begins to function as a generator. The generator output is automatically connected to the dc bus system at approximately 40-percent rpm. During a battery start, the two 25.2-volt batteries are automatically connected in series. The 800-ampere starting current reduces the voltage across the starter terminals to about 20 volts. As engine revolutions per minute increases, the starter current drops and terminal voltage increases. One battery supplies voltage and load indications and the other battery supplies voltage to the battery bus for engine ignition. During an external power start, the batteries are disconnected (by insertion of the external power plug) from the starting bus. Starter operation is entirely automatic after the starter button has been pushed in. Should the engine fail to start or if it is desired to stop the cranking cycle, the starter button may be manually pulled up. The starting system is designed so that either engine may be started first.

2.8.7.1 Starter Buttons. A solenoid-held push-pull starter button (2 and 16, FO-3) is provided for each engine. These PUSH TO START PULL TO STOP STARTER buttons are on the engine switch panels, one on each side of the throttle quadrant, and are used when normal battery or external power starts are made. When the starter button is actuated, a solenoid holds the button down during the cranking portion of the starting cycle. A relay in the starter controller automatically deenergizes the ignition and releases the starter button when the engine has accelerated to approximately 31- to 35-percent rpm and the starter load has decreased to approximately 200 amperes when using external electrical power for starting. During battery starts, the revolutions per minute may be somewhat higher. There is a red starting cycle light in each starter button that comes on during ground or air engine starting. This light goes out when the starting cycle ends and the starter-generator begins to function as a generator. The electrically powered starting cycle can be interrupted at anytime by pulling the starter button up.

2.8.7.2 AIR START Switch. The AIR START switch (4, FO-3) is on the left engine switch panel. This

switch has three positions marked LEFT HAND ON, OFF, and RIGHT HAND ON and is to be used for air starting of either engine. When the switch is actuated for either engine, dc power is supplied directly from the dc essential bus to the respective engine ignition unit when the respective throttle is advanced from the CLOSE position. The starting cycle light in the engine starter button comes on in the same manner as during a normal engine start; however, the light will remain on until the AIR START switch is placed in the OFF position. After an engine airstart, as indicated by engine EGT and revolutions per minute, the AIR START switch shall be replaced to the OFF position.

Note

The affected dc generator cannot be reset until the AIR START switch is placed in the center (OFF) position.

2.8.8 Ignition System. Initial ignition is provided through spark igniters in the Nos. 3 and 6 burners. The remaining burners are ignited through interconnecting flame tubes that also sustain continuous combustion. Controls for both spark igniters are provided by the IGN & BOOST CONT circuit breaker, ENG MASTER switch, BATTERY switch, starter button or AIR START switch, and throttle boost and ignition switches. In addition, the "continuous" spark igniter for the No. 3 burner may be energized selectively using the IGNITION switch or both spark igniters energized in conjunction with the stall-warning system. The "intermittent" spark igniter for the No. 6 burner because of its hotter operation is subject to an ignition duty cycle. This duty cycle applies when both igniters are operating as follows:

2 minutes ON, 3 minutes OFF.
2 minutes ON, 23 minutes OFF.

Igniter operation is indicated by illumination of the red light in the respective starter button. The ignition system operation is available in three modes. In each mode in the following paragraphs, normal switch positions are assumed for all switches except the IGNITION switch. The IGNITION switch position is as given.

1. Engine starting mode, IGNITION switch — NORMAL.

   External dc or aircraft battery power (essential bus for airstart) is applied to both spark igniters until start is completed or terminated. Ignition duty cycle limit applies.

2. Selective mode, IGNITION switch — CONTINUOUS.

   Essential bus power is applied to "continuous" igniter until IGNITION switch is placed at NORMAL or ignition is terminated by other ignition controls. Operation in this mode is recommended for flight in turbulent air and precipitation. No ignition limit.

3. Automatic mode, IGNITION switch — NORMAL or CONTINUOUS.

   Essential bus power is applied to both spark igniters automatically in conjunction with activation of stall-warning system. Ignition continues for duration of stall warning or termination by other control means. Ignition duty cycle applies.

2.8.8.1 Hot Start. Several factors can cause a hot start (EGT greater than 525 °C (977 °F)), and the pilot should be able to recognize the early signs to enable him to take corrective action (retard throttle to CLOSED).

A low battery state or initiating the start at too low a revolutions per minute may cause a hot start.

After initial ignition has been indicated by EGT rise, monitor the EGT, fuel flow, and revolutions per minute. A fuel flow of over 600 pounds during initial starting will result in an improper fuel-air ratio that will invariably cause a very rapid rise in EGT. The maximum allowable starting EGT will probably be exceeded if the throttle is not retarded by the time the EGT indicator reaches 475 °C (887 °F).

A hot start may result from engine revolutions per minute hanging up and failing to accelerate to idle revolutions per minute in a normal manner after ignition. The pilot should be alert to recognize a steady increase in revolutions per minute after ignition as well as stable temperature indications and anticipate preventive action for a hot start as soon as the engine fails to increase in a smooth and constant manner.

2.8.8.2 Engine Idle Hangup. Engine idle hangup ("hung idle") is characterized by normal or lower than normal idle rpm (37 to 41 percent) and failure to accelerate with advancing throttle. The possibility of hung idle occurring is much greater during static ground operation than in flight. Since the acceleration margin is decreased by increasing inlet air temperature and decreasing barometric pressure, the probability of hung idle occurring is greater on hot days and at high-altitude fields. Engine/fuel control combinations prone to hanging at idle are consistently slow in accelerating from idle to

takeoff and may exhibit unstable idle. Extraction of aircraft accessory power (cabin air, anti-icing, generator loads, etc.) effectively reduces the acceleration fuel flow operation. An engine/fuel control combination that hangs at idle or accelerates slowly may require engine/engine fuel control maintenance and should be reported.

Should idle hangup occur in flight during a waveoff or touch-and-go landing, the hangup may be alleviated by placing the AIR START switch at LEFT HAND ON or RIGHT HAND ON.

CAUTION

The ignition duty cycle is limited to 2 minutes on, 3 minutes off, 2 minutes on, and 23 minutes off.

The respective dc generator may have to be reset to restore operation after the AIR START switch is placed back at OFF. Corrective maintenance should be performed before the next flight.

2.8.8.3 Acceleration Check. Engine acceleration time may be checked by first operating the engines at takeoff thrust and noting the stabilized tachometer readings and the throttle positions. Retard the throttles to IDLE and allow the engines to stabilize. Advance the throttles to the previously noted takeoff thrust position in 1 second or less. The engines should accelerate to takeoff thrust revolutions per minute within 8 seconds and within a 4-second time limit differential between engines, and the EGT shall not exceed the red-line limit.

2.9 ENGINE OIL SYSTEM (Figure 2-24)

Each engine has a self-contained engine oil lubricating system consisting of an oil tank, a pump, an oil filter, engine scavenge pumps, and a fuel oil cooler. The oil serves for both lubricating and cooling. Operation of the oil system is completely automatic, requiring no control action by the flightcrew. The oil tank total capacity is 2.03 U.S. gallons; however, the oil capacity is 1.6 U.S. gallons. The oil tank is mounted on the engine over the compressor section. An oil servicing access door is provided on the upper forward surface of each engine pod. The engine oil pump is on the engine accessory gearbox. An oil pressure of 45 ±5 psi, relative to internal breather pressure (tank pressure), is maintained by an oil pressure relief valve downstream of the oil filter. Oil scavenge pumps are provided to pick up oil from the bearing compartments and gearbox and return it through the fuel oil cooler back to the oil tank. All return oil passes through the fuel oil cooler except during cold-weather starting. Refer to servicing chart (Figure 3-1) for engine oil specification and grade.

2.9.1 Oil Pressure Gauges. The two oil pressure gauges (67, FO-1) are mounted at the bottom of the center instrument panel and are calibrated in increments of 5 psi from 0 to 100 psi. These gauges show the pressure of oil within the engines and are powered from the 26-volt No. 1 ac bus. Each oil pressure indicating system is protected by a circuit breaker on the overhead circuit breaker panel.

2.9.2 Oil Temperature Gauge. Oil temperature indication for both engines is included in the combination fuel and oil temperature gauge (37, FO-1) on the copilot outboard instrument panel. Each oil temperature gauge is calibrated in 10° increments from -60 to 150 °C (-76 to 302 °F). An oil temperature probe is installed in the accessory section of each engine and is connected to its respective oil temperature indicator. There is no pilot-operated control for oil temperature except to advance the throttle and increase power on the engine. When the power is increased (fuel flow is increased) and/or altitude is reduced, fuel flow and cooling effect are increased.

CAUTION

If the temperature does not return to the operating range within a reasonable time after the throttle is advanced, the oil pressure gauge should be closely monitored because an engine oil system failure and subsequent engine failure could occur.

The gauge is powered by the dc essential bus and is protected by circuit breakers on the overhead circuit breaker panel.

2.9.3 Low Oil Pressure Caution Light. A placard-type amber OIL PRESS LOW caution light is on the caution-warning light panel (12, Figure 2-33). This light is actuated by a pressure switch in either engine oil system. When engine oil pressure drops to 28 ±1 psi, the pressure switch closes and the low oil pressure light comes on. When pressure is greater than 32 psi, the switch opens and the low oil pressure light goes out. This light is powered by the dc essential bus and is protected by a single push-pull circuit breaker on the overhead circuit breaker panel.

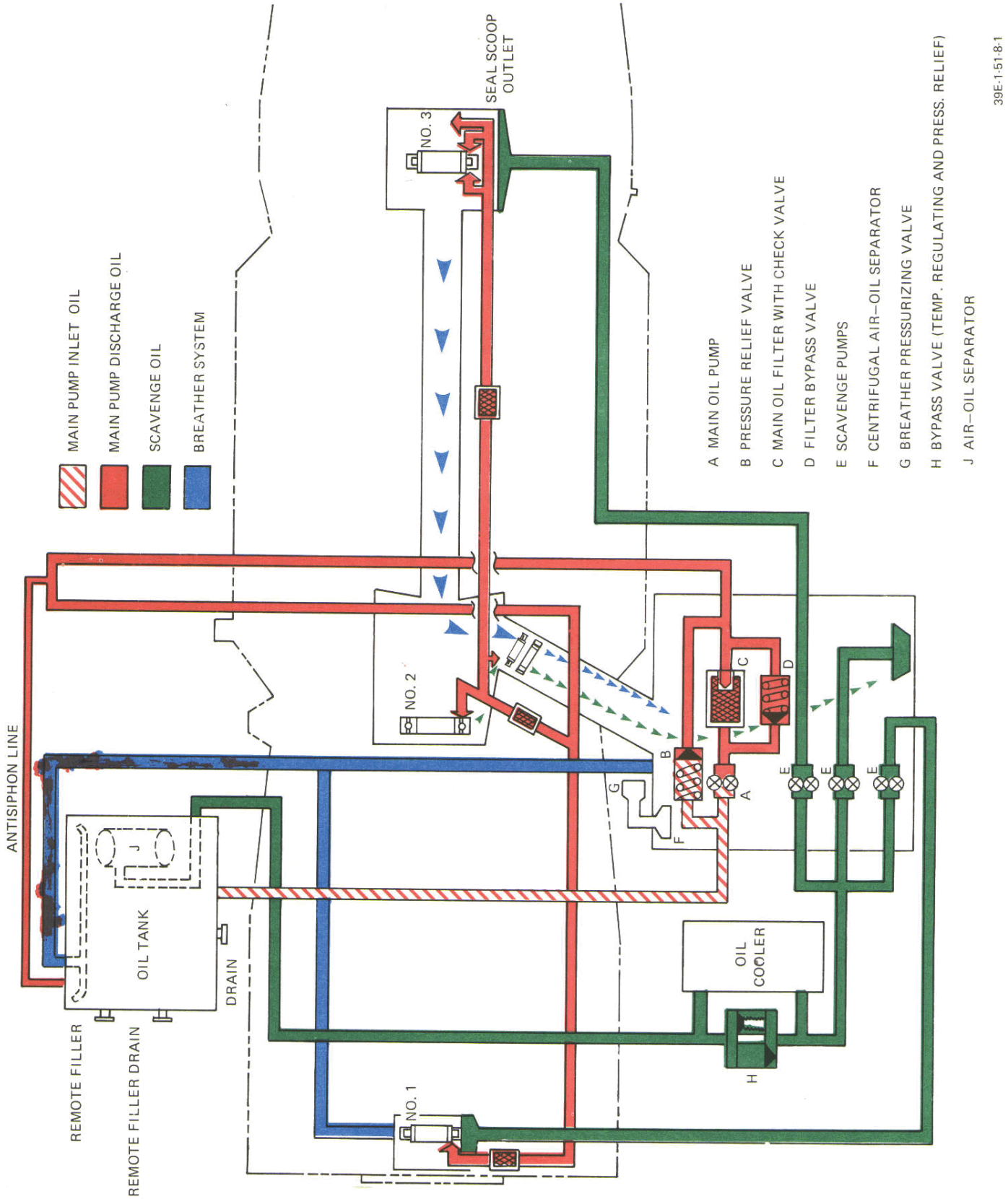


Figure 2-24. JT12 Engine Oil System

2.9.4 Engine Indicators

2.9.4.1 Engine Pressure Ratio Gauges. Two indicators (Figure 2-25), one for each engine, are mounted on the pilot inboard instrument panel. These gauges indicate the engine pressure ratio as a measure of thrust being developed by the engine. This is the ratio of turbine discharge total pressure to compressor inlet total pressure. Turbine discharge total pressure is an average pressure taken by four probes manifolded together immediately downstream of the turbine. Compressor inlet total pressure is measured by two electrically heated pitot-pressure heads, one for each instrument, mounted on the leading edge of the vertical stabilizer. The pressure ratio gauges are powered by the 115-vac secondary bus.

2.9.4.2 Exhaust Gas Temperature Gauges. An exhaust gas temperature gauge (23, FO-1) is provided for each engine. These two gauges are mounted on the engine instrument panel where they may be monitored by the pilot or copilot. The gauges are calibrated in 20° increments from 0 to 1,000 °C. The indicators provide the pilot with a visual indication of engine EGT. The gauge system employs self-generating electrical units and does not require electrical power from the aircraft for operation.

2.9.4.3 Tachometers. Two tachometers (21, FO-1) are mounted on the pilot inboard instrument panel. Each tachometer is calibrated in 2-percent increments from 0- to 100-percent rpm and registers engine speed in percentage of 16,030, which equals 100-percent revolutions per minute. The small dial within the larger dial is provided so that the pilot may more easily read the exact revolutions per minute percent (i.e., when the large pointer is between 90 and 100 and the pointer on the small dial is at 3, the rpm percent is 93). Each tachometer receives power from its respective tachometer generator that is geared to the engine accessory section driven by the compressor rotor. The tachometer is, therefore, independent of the aircraft electrical system.

2.9.4.4 Fuel Flow Indicators. Refer to paragraph 2.14, FUEL SYSTEM — AIRCRAFT.

2.9.4.5 Engine Synchronization Indicator. An engine synchronization indicator (Figure 2-26) is installed on the copilot inboard instrument panel. It has a double-end pointer that will rotate either clockwise or counterclockwise through 360°. On the face of the indicator are the words SLOW and FAST, each associated with a curved arrow. When the pointer rotates counterclockwise (SLOW), this means the right engine is running slower than the left engine. Pointer rotation clockwise (FAST) indicates the right engine is running faster than the left engine. The pointer rotates when the speed differential between the two engines is 4 percent or less. When the engines are in synchronization, the pointer stands still.

The indicator is operated by electrical signals generated by the left and right engine-driven tachometer generators. The signal from the left engine tachometer generator is used as the master signal. The signal from the right engine tachometer generator when out of synchronization with that from the left engine tachometer generator causes the pointer to rotate in the direction corresponding to faster or slower right engine speed.

2.10 ENGINE THRUST REVERSERS (CT-39G)

A means of decelerating the aircraft during the landing roll is provided by an independently operated thrust reverser installed on each engine. The system is electrically controlled by dc essential bus power and hydraulically actuated by the normal hydraulic system. If a failure of the normal hydraulic system should occur, the thrust reverser hydraulic accumulator will

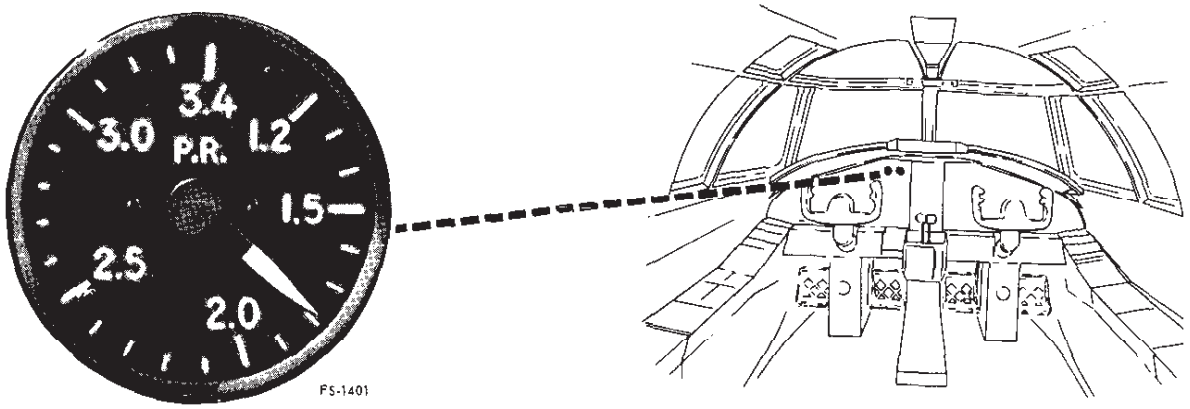


Figure 2-25. Engine Pressure Ratio Gauges

Figure 2-26. Engine Synchronization Indicator

provide sufficient hydraulic pressure to operate the thrust reversers through one complete cycle (deploy and stow). The system is designed to be used on the ground only. The system is wired through the main landing gear downlock switches. The thrust reversers cannot be deployed in flight as long as the thrust reverser control switch is at OFF.

The thrust reverser system for each engine consists of two doors, a hydraulic control valve, a hydraulic actuator, a latch unlock solenoid, an adjustable thrust reverser lever stop assembly, a thrust reverser control lever unlock solenoid, several control and indicating limit switches, and three advisory lights. When deployed, the doors direct engine exhaust gases forward, over, and under the engine pod. In stowed position, these doors form the aft section of the engine pod fairing. The doors are deployed and stowed by a hydraulic actuator on the outboard side of each engine. The actuator has an internal mechanical lock that provides the primary means for retaining the reverser doors in the stowed position. Secondary locks on the inboard side of each engine are provided in the event the actuator internal lock should fail.

Reverser door deployment and stowing and reverse thrust power application are controlled by two thrust reverser levers, one of each engine. As a secondary means of stowing the thrust reverser doors, a thrust reverser control switch is provided which, when placed at EMERG STOW, maintains hydraulic pressure in the stow direction (in flight or on the ground). Three advisory lights for each thrust reverser system indicate the status of the respective system.

If an indication of engine fire should occur, pulling the FIRE PULL T-handle of the affected engine will deactivate the respective thrust reverser control valve. The control valve will return to neutral, isolating the pod from the hydraulic fluid pressure source, provided the thrust reverser control switch is at ON. The fire supporting potential of the thrust reverser system is thus reduced to a very small quantity of unpressurized fluid in the engine pod lines and actuator. As a design safety feature, the doors will remain deployed should a failure occur to the system when the doors are deployed. The thrust reverser system is shown schematically in Figure 2-27.

2.10.1 Thrust Reverser Levers. Engine reverse thrust is controlled by two independently operated thrust reverser levers (Figure 2-28) immediately forward of the throttles. Before the thrust reverser levers can be moved up to deploy the thrust reverser doors, the throttles must be at IDLE. When the main landing gear is down and locked and the thrust reverser control switch is placed at ON, initial upward movement of the thrust reverser levers energizes the secondary latch unlock solenoid which disengages the secondary locks and initiates thrust reverser door deployment. A thrust reverser control lever interlock solenoid prevents aft movement of the thrust reverser levers beyond idle reverse thrust until the doors are fully deployed. Once the doors are fully deployed, the thrust reverser levers can be moved aft to increase reverse thrust. Returning the thrust reverser levers to the full forward position decreases reverse thrust to idle and stows the thrust reverser doors. There is no reverse idle interlock or detect when moving the levers forward.

2.10.2 Thrust Reverser Lever Stop Assemblies. An adjustable thrust reverser lever stop assembly (Figure 2-28) is installed on each thrust reverser lever to provide a reference for setting maximum allowable reverse thrust. Each stop assembly consists of a fixed field temperature scale calibrated in °F, a field elevation scale calibrated in thousands of feet on the end of an adjustable bellcrank, and a stop lug on the thrust reverser lever. The stop assembly is used when a landing is made at any field up to 2,500 feet elevation. Before landing, the applicable field elevation and temperature index marking should be aligned by turning the knurled adjustment knob as necessary. When these scales are properly aligned, thrust reverser lever travel is limited by the stop assembly so that the maximum allowable thrust is not exceeded. Above 2,500 feet elevation, the thrust reverser levers should be actuated only to the precomputed EPR setting.

2.10.3 Thrust Reverser Control Switch. A three-position thrust reverser control switch (Figure

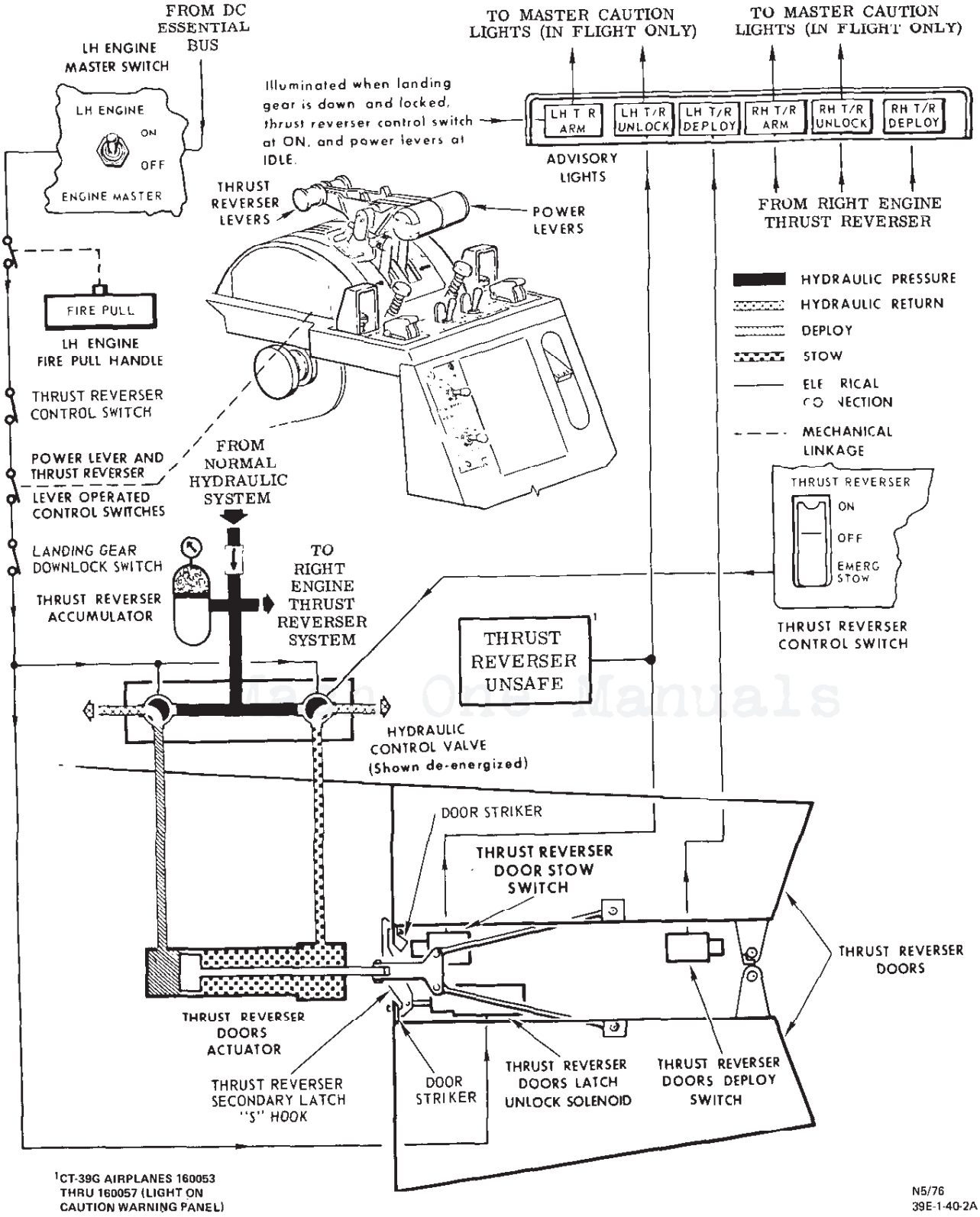


Figure 2-27. Thrust Reverser System

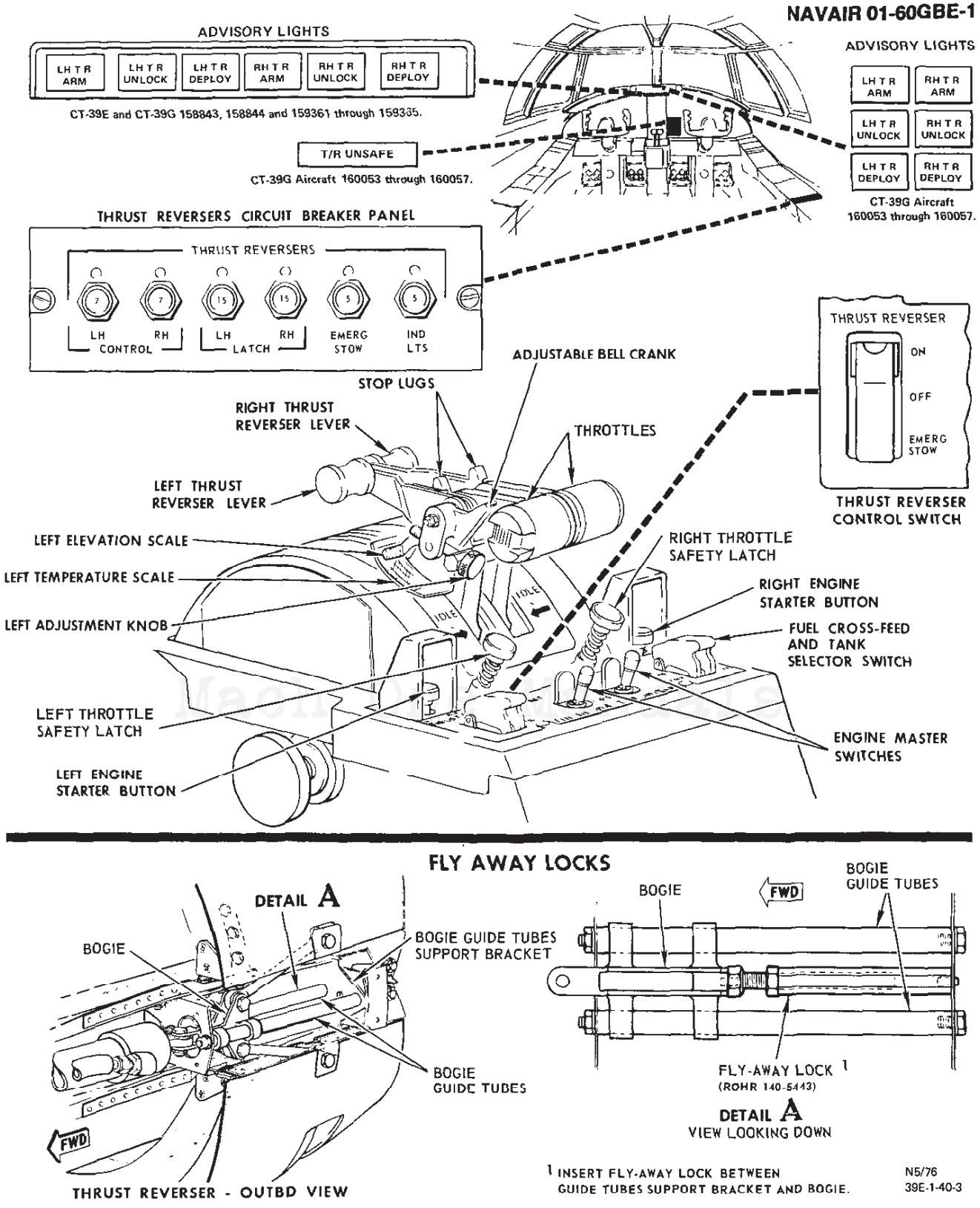


Figure 2-28. Thrust Reverser Controls and Indicators

2-28) on the engine switch panel controls dc essential bus power to the thrust reverser system. When the switch is at ON, power is available for normal operation of the thrust reverser system. When the switch is at EMERG STOW, the gear and throttle safety switches are bypassed, the stow solenoids of both the left and the right thrust reverser hydraulic control valves are energized, and hydraulic pressure is directed and maintained to the actuator stow ports. When the switch is in the OFF position, all electrical power is removed from the thrust reverser system.

2.10.4 Thrust Reverser Arm Lights (CT-39G Aircraft 158843, 158844, and 159362 Through 159365). The amber thrust reverser LH T/R ARM and RH T/R ARM lights (Figure 2-28), powered by the dc essential bus, are on the edge of the instrument panel shroud visor in front of the pilot. These lights come on and stay on as long as the thrust reverser control switch is at ON and the throttles are at IDLE and indicate that electrical power is available for thrust reverser deployment. The arm lights can be tested through the caution light test switch circuit and dimmed with the pilot flight instrument light rheostat. In addition, in flight only, if either arm light comes on, the master caution light will also illuminate.

2.10.5 Thrust Reverser Unlock Lights (CT-39G Aircraft 158843, 158844, and 159362 Through 159365). The red thrust reverser LH T/R UNLOCK and RH T/R UNLOCK lights (Figure 2-28), powered by the dc essential bus, are on the edge of the instrument panel shroud visor in front of the pilot. These lights will come on anytime the respective thrust reverser doors are not completely stowed or the secondary lock is not in the engaged position. The unlock lights can be tested through the caution light test switch circuit. In addition, in flight only, if either unlock light comes on, the master caution light will also illuminate. For aircraft changed by Service Bulletin 76-3, the UNLOCK light will not dim by the dimming control when the thrust reversers are armed.

2.10.6 Thrust Reverser Deploy Lights (CT-39G Aircraft 158843, 158844, and 159362 Through 159365). The blue thrust reverser LH T/R DEPLOY and RH T/R DEPLOY lights (Figure 2-28), powered by the dc essential bus, are on the edge of the instrument panel shroud visor in front of the pilot. These lights come on anytime the thrust reverser doors are fully deployed. The deploy lights can be tested through the caution light test switch circuit.

2.10.7 Thrust Reverser Arm, Unlock, and Deploy Lights (CT-39G Aircraft 160053 Through 160057). The thrust reverser ARM, UNLOCK & DEPLOY lights (Figure 2-28) are provided with a mask in which the background illuminates and the lettering is silhouetted. The thrust reverser lights may be dimmed using the light dimming rheostat; however, if an UNLOCK light should illuminate with the control switch at OFF, the light will illuminate bright, bypassing the dimming circuit. At this time, the UNLOCK light serves to indicate an abnormal condition since the thrust reverser system and lights should not operate with the control switch at OFF. In addition, in flight only, if either the UNLOCK or ARM light comes on, the master caution light will also illuminate. The ARM light illuminates when the reverser control switch is at ON and the throttles are at IDLE. The DEPLOY light illuminates anytime the thrust reversers are fully deployed. The lights may be tested through actuation of the caution light test switch and are located in the center of the shroud over the instrument panel.

2.10.8 Thrust Reverser Automatic Stow System (CT-39G Aircraft). A thrust reverser automatic stow system is installed to provide additional protection against an inadvertent thrust reverser deployment. The automatic stow system consists of a THRUST REV UNSAFE caution light and, for each reverser, an auto stow relay. The system is automatically energized through actuation of the stow switch. The automatic stow protection is available only after the landing gear is released from its downlocks.

When the automatic stow system is initially energized, the thrust reverser UNLOCK light may be momentarily illuminated. This momentary illumination occurs during the time between the actuation of the stow switch and the time the reverser is hydraulically actuated into the stow position. When the auto stow circuit is energized, the THRUST REV UNSAFE and the MASTER CAUTION lights are illuminated.

Note

The automatic stow system will be energized only on the left or right thrust reverser system that has experienced a malfunction.

2.10.8.1 Thrust Reverser Unsafe Caution Light. An amber THRUST REV UNSAFE caution light (Figure 2-33) is installed on the indicator, caution, and warning light panel. This light when illuminated indicates that one thrust reverser auto stow circuit is energized; the affected reverser is hydraulically pressurized in the stow position. The unsafe light will remain illuminated and the reverser will remain stowed until power is removed from the dc essential bus and will

illuminate again when power is restored if the malfunction was not corrected.

Note

With the THRUST REV UNSAFE caution light illuminated, the affected thrust reverser cannot be deployed.

2.10.9 Thrust Reverser Flyaway Locks. Two flyaway locks (Part No. 140-5443) are supplied as part of loose equipment with each aircraft. (See Figure 2-28.) If it is desired that the thrust reverser be mechanically locked out, the locks may be installed. The flyaway locks provide a positive mechanical block against thrust reverser deployment; and, if used, they should be installed on both left and right thrust reversers. The locks may be installed as follows:

1. With thrust reverser doors stowed, position battery and electrical master switches at OFF.
2. Remove outboard stang fairing from each engine.
3. Adjust flyaway lock to a short dimension and install between guide tube support bracket and bogie.
4. With lock in position, extend threaded end until lock is tight.
5. Tighten jam nut and safety with lockwire.
6. Reinstall the two stang fairings.

2.10.10 Thrust Reverser Operation. The thrust reverser may be actuated as soon as practical after touchdown. It is recommended that the thrust reversers be actuated on each landing. This will standardize the landing procedure and will ensure that the reversers are ready for higher levels of reverse thrust if needed. Before landing, precompute the maximum allowable reverse thrust EPR value. When reverse thrust is applied, the EPR gauge should be monitored to ensure that the maximum allowable EPR values are not exceeded. If the reverse thrust EPR values are exceeded, the engine thrust reverser structural limit is exceeded. For landings at airports with a pressure altitude of less than 2,500 feet, the thrust reverser lever stop assemblies should be adjusted before landing. For landings at airports with a pressure altitude in excess of 2,500 feet, it will be necessary to fully retract the thrust reverser lever stop assemblies and monitor the EPR gauges, pulling the thrust reverser levers back only to the precomputed EPR value.

To obtain maximum benefit from reverse thrust, the reversers should be operated as follows:

2.10.10.1 Before Landing

1. Reverse thrust EPR setting — Compute.
2. Reverser lever stops — Set (for field elevation below 2,500 feet).
3. Thrust reverser control switch — EMERG STOW.
4. Throttles — Retard to IDLE at or Before Touchdown.

2.10.10.2 After Touchdown

1. Thrust reverser control switch — ON. With the throttles at IDLE, the two arm lights will be on.
2. Thrust reverser levers — Pull Up to Interlock.

Maintain a slight back pressure on levers.

3. Unlock and deploy lights — Check On.
4. Reverser levers — Pull AFT to Predetermined EPR Setting or Preset Mechanical Stops.
5. Brakes — As Required.
6. Nosewheel steering — Engage.
7. Nose — Hold Down With Elevator or Nosedown Trim.

2.10.11 Thrust Reverser Stow Procedure. Stow the thrust reverser as follows:

Before decelerating to 60 KIAS:

1. Move thrust reverser levers forward to reverse idle.

Note

The thrust reverser levers should be moved forward slowly. It will take approximately 10 seconds for the engines to decelerate from a maximum reverse thrust revolutions per minute to an idle revolutions per minute. Do not stow from a high revolution per minute to avoid aircraft acceleration.

At taxi speed with engines at IDLE:

2. Stow thrust reverser levers.

Mach One Manuals

After deploy and unlock lights go out:

3. Thrust reverser control switch — OFF.

2.10.12 Thrust Reverser Emergency Stow. Refer to paragraph 15.24, THRUST REVERSER UNSAFE.

2.11 ENTRY DOOR

A plug-type entry door is on the left side of the fuselage forward of the wing. When this door and all other cabin openings are closed, an engine is operating, and the weight of the aircraft is off the landing gear, an inflatable pressure seal automatically inflates and seals the door. A flush-type rotary latch is provided on each side of the door to be used as a handgrip while opening or closing the door. When the door is open, steps within the door and a stowable handrail provide easy access into the aircraft. Helper springs within the structure are installed to facilitate raising or lowering the door-open caution light in the cockpit comes on when the door is not closed and locked. A bungee assembly (Figure 2-43) may be installed on the entry door to prevent wind gust damage to the door when the aircraft is on the ground. The assembly, which has a red streamer, must be removed before flight. (See Figure 8-1 for entry door operation.)

2.11.1 DOOR OPEN Caution Light. A DOOR OPEN caution light, on the caution-warning light panel (12, Figure 2-33) indicates when the entry door is not closed and locked. The light is powered from the dc essential bus. A switch is installed at the latches of the entry door. If the switch is not closed, the door-open caution and master caution lights come on. The lights shall be out before takeoff.

2.12 FLIGHT CONTROL SYSTEM

The flight control surfaces (ailerons, rudder, and elevators) are mechanically operated from a conventional set of dual controls. A spring-type bias bungee is incorporated to establish a no-load center position for the elevator controls by interconnecting the elevators and horizontal stabilizers. This no-load position varies automatically with stabilizer position and improves the apparent longitudinal stability. Servos are incorporated for use with the autopilot system.

Trim is accomplished by electromechanical actuators that position the trim tabs on the left aileron and rudder and reposition the horizontal stabilizer. Two mechanically interconnected actuators are used to actuate the horizontal stabilizer. When the landing gear is retracted, horizontal trimming is accomplished at a low-speed rate; with the gear extended, the trimming rate is high. Two complete and independent trim control electrical circuits, a normal and an alternate circuit, are provided. A separate trim position indicator is provided for each trim system.

A bobweight is installed in the horizontal control system for obtaining a positive stick force of 10 pounds per g of normal acceleration. A bobweight balance bungee is used to statically balance the system as a positive 1g.

A mechanical flight control gustlock is provided which interconnects with the throttle mechanism. When the gustlock is engaged, the throttles cannot be advanced above the IDLE position.

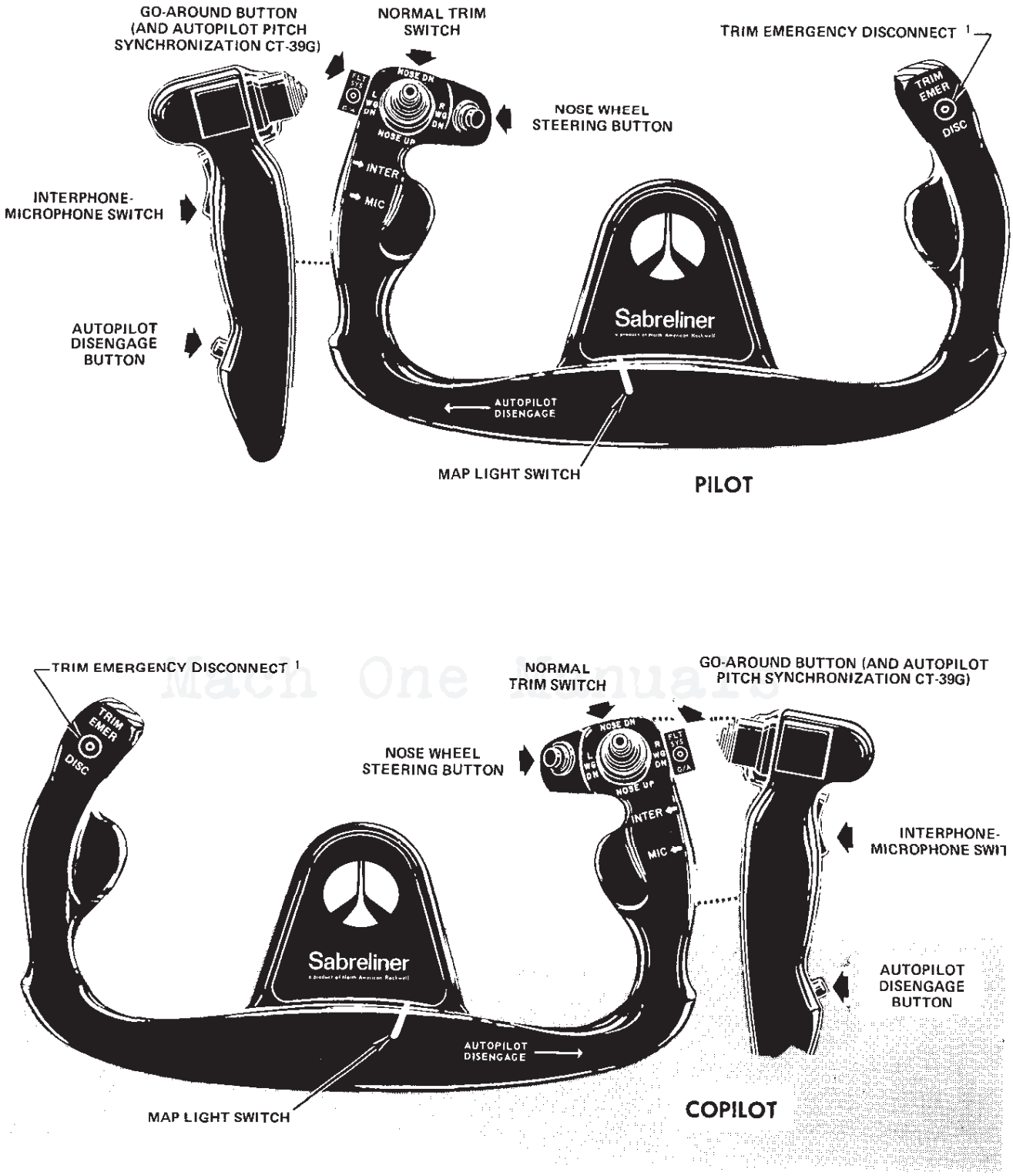
A stall warning system, interconnected with the flight control system, shakes the control columns to warn of approaching aircraft stall conditions.

2.12.1 Control Wheels. The pilot and copilot control wheels (Figure 2-29) are identical except that incorporated switches are in reverse locations. The nosewheel steering, interphone-microphone, trim, go-around, emergency trim disconnect (CT-39E, CT-39G 158843, 158844, 159362 through 159365 changed by Service Bulletin 74-6, and 160053 through 160057) and autopilot disengage switches are on the outboard grip of each wheel. The interphone-microphone and autopilot disengage switches are on the forward face of the wheel.

2.12.2 Rudder Pedals. A set of rudder pedals is provided for the pilot and copilot, which control the rudder action through direct mechanical linkage to the rudder. The wheelbrakes are actuated by pressure on the top of the rudder pedals.

2.12.2.1 Rudder Pedal Adjustment Knob. Comfortable leg-length adjustment of the rudder pedals may be accomplished by a rudder pedal adjustment knob (57 and 78, FO-1) on the control pedestal directly below the control wheels. Turning the knob counterclockwise adjusts the pedals aft (toward the pilot); clockwise turning adjusts the pedals forward.

2.12.3 Gustlock Handle. The push-pull gustlock handle (80, FO-1) is below the control wheel on the pilot side only. To apply the gustlock, the handle is moved out of the detent and aft of the engaged position while the throttles are at IDLE or below. Movement of the control systems through neutral engages the control lock. The throttles are also blocked so that only idle and limited taxi power are available. To release the gustlock, the handle is moved out of the detent and pushed to full-forward stowed position.



1 CT-39G AIRPLANES 160053 THRU 160057 AND AIRPLANES CHANGED BY SERVICE BULLETIN 74-6

Figure 2-29. Control Wheels

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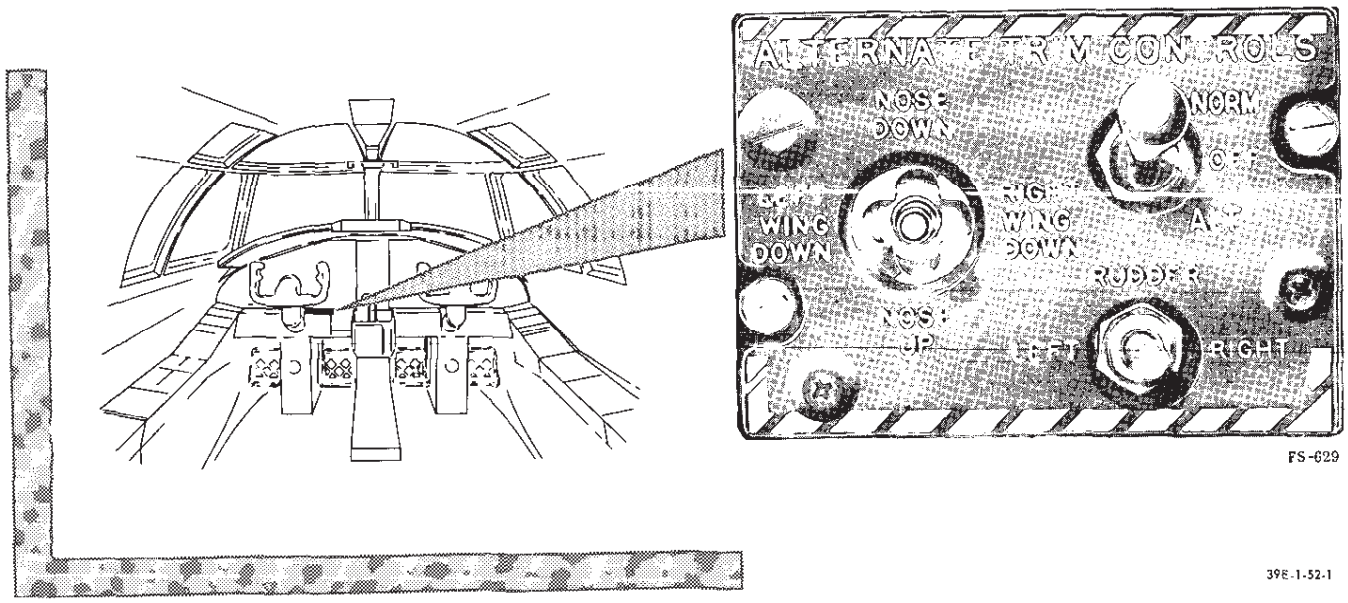


Figure 2-30. Alternate Trim Control Panel

Note

Engine starting may be performed without releasing the gustlock. This may be desirable during windy or gusty weather operations; however, nosewheel steering is not operative.

2.12.4 Trim System Selector Switch. A positive-lock trim system selector switch on the alternate trim control panel (Figure 2-30) is powered by the dc essential bus. The switch has three positions: NORMAL, ALTERNATE, and OFF. When the switch is in the NORMAL position, longitudinal and lateral trim are controlled by the normal trim switch on the control wheel and directional trim is controlled by the normal rudder trim switch on the center pedestal. With the trim control selector switch in the ALTERNATE position, all trim is controlled by the alternate longitudinal, lateral, and directional trim switches on the alternate trim control panel. With the switch in the OFF position, both trim systems are inoperative.

2.12.5 Trim Arming Switch. The trim arming switch (Figure 2-29) is in the center of the normal trim switch so that pressing the normal trim switch also depresses the trim arming switch which supplies power to the normal trim switch circuitry. The arming switch returns to the up (off) position when pressure is removed.

2.12.6 Normal Trim Switch. Normal trim of the horizontal stabilizer or the aileron tab is through a five-position normal trim switch (Figure 2-29) on each control wheel outboard grip. Trim is effected by holding the wheel for the desired flight attitude and operating the normal trim switch after depressing the arming switch to remove the load (control forces) from the wheel. Trim switch positions are L WG DN, NOSE DN, R WG DN, and NOSE UP. When the switch is released, it automatically returns to the OFF (center) position and trim action stops.

2.12.7 Emergency Trim Disconnect Buttons (CT-39E, CT-39G 158843, 158844, 159362 Through 159365 Aircraft Changed by Service Bulletin 74-6, and 160053 Through 160057). A TRIM EMER DIS button is located on the inboard grip of each control wheel. The emergency trim disconnect buttons are operable only with the trim system selector switch at NORM or ALT.

Note

During autopilot operation, check to ensure the flight control trim selector switch is placed at the NORM or ALT position. Autopilot trimming will occur with the switch in the center OFF position; however, the emergency trim disconnect button will not function.

If stabilizer, rudder, or aileron trimming does not stop when the normal trim switch is released (normal trimming is accomplished only when (1) the trim arming switch is depressed to energize the arming circuit and (2) the normal trim switch is displaced to energize the actuation circuit), then actuation of the

trim disconnect button will stop the trim action. The disconnect button will also stop trim action induced by the autopilot or through the alternate trim system.

A trim disconnect amber caution light is installed adjacent to the indicator, caution-warning light panel. When horizontal stabilizer, rudder, and aileron trimming action is stopped by use of the trim disconnect switch, the TRIM DIS caution light and master caution lights are illuminated. The light may be extinguished and normal trim restored (providing a malfunction does not exist in the system) by placing the trim system selector switch at OFF and then back to NORM.

2.12.8 Rudder Trim Switch. Normal trim of the rudder is provided through the three-position RUD TRIM switch (28, FO-3) on the center pedestal below the throttles. Holding the rudder trim switch at R trims the nose of the aircraft to the right; holding it at L trims the nose left. When the rudder trim switch is released, it automatically returns to the OFF (center) position and trim action stops. The normal trim switches are powered by the dc essential bus.

2.12.9 Alternate Trim Switches. An alternate trim system is provided to be used in case of a normal trim system failure. When the trim control selector switch (Figure 2-30) is positioned at ALTERNATE, the ailerons and horizontal stabilizer are trimmed though the five-position alternate trim switch on the alternate trim control panel. Holding the alternate trim switch to either LEFT WING DOWN or RIGHT WING DOWN causes the corresponding wing to be trimmed down. Holding the alternate trim switch forward to NOSE DOWN trims the aircraft nose down; holding it aft to NOSE UP trims the nose up. When the switch is released, it automatically returns to the OFF (center) position and trim action stops. The alternate system trims the horizontal stabilizer at the low-speed rate. Alternate trim of the rudder is provided through a three-position switch on the alternate switch control panel. Holding the trim switch at RIGHT trims the nose of the aircraft to the right; holding it at LEFT trims the nose left. When the switch is released, it automatically returns to the OFF (center) position and trim action stops. The alternate trim switches are powered by the dc essential bus.

2.12.10 Audio Trim Signal (CT-39E, CT-39G Aircraft 158843, 158844, 159362 Through 159365 Changed by Service Bulletin 74-7, and 160053 Through 160057). An audio signal is provided as an aid to the pilot when trimming of the horizontal stabilizer is taking place. Anytime the normal trim switch on either the pilot or copilot control wheel is actuated to the NOSE DN or NOSE UP position, the signal will sound and continue as long as the switch is held in position. The signal will occur during manual trimming and autopilot trimming of the horizontal stabilizer. The signal device is located on the overhead control panel near the dome light.

2.12.11 Trim Position Indicators. There are three trim position indicators (34, 35, and 36, FO-1) on the copilot inboard instrument panel, powered by the dc essential bus. The horizontal stabilizer trim position indicator indicates the aircraft noseup trim condition in terms of horizontal stabilizer leading edge trim position. The rudder trim tab position indicator is calibrated 15° right or left in increments of 5°. The aileron trim tab position indicator is calibrated 12° wing up or wing down in increments of 4°.

2.12.12 Stall Warning (Stick-Shaker) System. A stall warning system warns the pilot by a shaking of the control columns when the aircraft is approaching stall speed or when the speed of the aircraft is approaching that which may cause an engine to flame out. The system consists of two similar transducers, mounted in the leading edge of the right wing; a flap monitor switch; nose and main gear load switches; a signal summing unit; and two control column shakers, one for each control column.

The flap monitor switch provides a signal to the summing unit at 20° of flap travel to compensate for a change in stall speeds resulting from flap movement. When a warning occurs, airspeed should be increased to a speed that causes the warning to cease. The system is inoperative when the weight of the aircraft is on the nosegear. Heat anti-icing of the transducer vanes is provided automatically when the weight of the aircraft is off the main landing gear. The transducer vanes may be heated at a reduced rate when the weight of the aircraft is on the main landing gear by placing the pitot and stall warning switch on the anti-icing control panel at ON.

2.12.12.1 STALL WARNING TEST Switch. A three-position STALL WARNING TEST switch (81, FO-1) enables the system to be checked when the aircraft is on the ground or in flight. When the weight of the aircraft is on the nosegear and the wing flaps are up, placing the switch at either INBD VANE or OUTBD VANE causes the nosegear load switch to be bypassed, and the control columns will shake. With the weight of the aircraft off the nose gear, placing the switch at either INBD VANE or OUTBD VANE causes the control columns to shake. The switch is spring loaded to the NORMAL (center) position.

2.13 FUEL HEATER SYSTEM

Ice is removed from the fuel filters on each engine by heating the fuel with hot engine compressor air from its respective engine before it gets to the fuel filter. The heated air supply controlled by a motor-actuated valve on each engine, that is powered by the dc essential bus. A switch in the cockpit operates the valve on each engine, simultaneously, and illuminates a caution light for each valve when it is open. A differential pressure of 7 to 8 psi across the fuel filter will cause a caution light to come on, indicating the fuel heater should be operated.

2.13.1 FUEL HEATER Switch. The two-position FUEL HEATER switch (Figure 2-2) is powered by the dc essential bus. When the switch is moved to ON, the fuel heat valve in the engine compressor bleed-air duct of each engine opens and directs hot air through the fuel heaters. Moving the switch to OFF closes the valves.

2.13.2 Fuel Temperature Gauge. Refer to paragraph 2.14, FUEL SYSTEM — AIRCRAFT.

2.13.3 Fuel Heater Caution Light. The two placard-type fuel heater amber caution lights (12, Figure 2-33) come on to read RH FUEL HEAT ON and LH FUEL HEAT ON when their respective fuel heat valve is open and will remain on until that valve is closed.

2.13.4 Fuel-Filter-Blocked Caution Light. Refer to paragraph 2.8.1, Engine Fuel Control System.

2.13.5 Operation of Fuel Heater System. When the fuel temperature is at 0 °C (32 °F) or below or when the fuel-filter-block caution light comes on, operation of the fuel heater should be as follows:

1. Fuel heater switch — ON.

   Check that both fuel heater caution lights come on.

   WARNING

   Do not take off with fuel heater on because aircraft performance is reduced because of bleed-air extraction. If fuel heater use is necessary before takeoff, its operation should be completed before takeoff.

2. Fuel heater switch — OFF.

   After heat has been used for 1 minute, move fuel heater switch to OFF and check that both fuel heater caution lights go out.

3. Fuel-filter-blocked caution light — Check out.

   If the fuel-filter-blocked caution light remains on after 1 minute of heater operation, the filters may be blocked by foreign material other than ice.

   CAUTION

   Operate the heater 1 minute only during each 30 minutes for the remainder of the flight because prolonged and unnecessary use of the fuel heater may result in high oil temperature and speeds up fuel nozzle coking.

2.14 FUEL SYSTEM — AIRCRAFT

The aircraft fuel system (FO-14) consists of two separate fuel tanks in the wing and a fuselage tank mounted in the aft fuselage compartment. Each wing has a fuel-tight compartment that serves as a fuel tank. The wing is divided by a center bulkhead into two separate fuel tanks. Each tank extends from the center bulkhead outboard to the wingtip rib, providing a separate fuel supply system for each engine. A tank crossfeed valve is installed in the center bulkhead to permit utilization of fuel from both wing tanks by one engine or one boost pump. Fuel lines attached to each side of the tank crossfeed valve extend forward through the booster pump compartment into the center wing section. This prevents fuel from flowing through the tank crossfeed valve from one booster pump compartment to the other. Each wing tank contains three baffle plates that have check valves that allow fuel to flow by gravity feed toward the center section. A fuel ejector and boost pump are in each wing tank near the center bulkhead. To ensure automatic after-shutdown protection from thermal expansion, thermal relief is provided in the fuel shutoff and upstream check valve. Fuel tank capacities are listed in Figure 2-31; fuel specifications are given in Figure 3-1. For gravity and single-point refueling procedures, refer to Chapter 3.

2.14.1 Fuselage Tank. A bladder-type fuselage tank, mounted in liquid-tight, sealed, metal compartment, in the aft fuselage equipment has a capacity of 163 U.S. gallons (1,059.5 pounds) of usable fuel and 28 U.S. gallons of expansion space. During pressure refueling, automatic shutoff will occur at 161 U.S. gallons (1,046.5

pounds). This fuel drains by gravity feed through separate lines into the center wing section of both wing tanks. The fuselage tank is provided with a dual integral level control shutoff valve with dual test diaphragms to admit fuel to the tank during single-point pressure refueling. This tank may also be serviced through a gravity-flow type fuel filler cap on the upper right side of the fuselage. When the aircraft is being fueled for total fuel capacity (wing and fuselage tank) by use of the gravity refueling method, the system should be fueled from the fuselage tank. By holding each wing gravity-flow fuel filler cap flapper valve depressed while filling from the fuselage filler receptacle, the wing tanks will fill at the same time. As fuel rises to the level of the flapper valves, they must be released and then the fuselage tank filled to the required level. Anytime expansion occurs in the wing, the expanding fuel is vented to the fuselage tank. The expansion space in the fuselage tank serves as a liquid separator, retaining any liquid fuel that may pass from the wing into the vent lines. This fuel flows back into the aircraft fuel system as fuel is used. The fuselage tank is vented overboard through a vent line that extends directly aft out of the fuselage. An antisiphon line and check valve are installed in the fuselage tank to prevent siphoning during thermal expansion.

2.14.2 Fuel Tank Venting. The fuel system is vented internally and to the atmosphere to maintain a satisfactory pressure condition during any static or flight condition. The wing tanks are vented, compartment-to-compartment, and to the fuselage tank. The fuselage tank is vented to the atmosphere through a vent line that extends aft out through the tail cone into the screened-in area around the fuel jettison pipe. The primary purpose of the vent is to provide a proper balance of pressure within each fuel tank to prevent collapse or expansion of the tank beyond limits and to ensure proper fuel transfer. Separate vent exhausts, incorporated in the single-point wing refueling level control pilot and vent valves, on, each wingtip prevent excessive pressure buildup during pressure refueling. A climb vent line between the wing tanks and fuselage tank improves fuel transfer during climb and expedites completion of transfer when level flight is attained.

2.14.3 Fuel Boost Pumps. The submerged centrifugal boost pump in the aft center section (hopper compartment) in each wing tank supplies fuel under pressure through the fuel shutoff valve to its respective engine fuel system. Each fuel boost pump is energized whenever its respective engine master switch is ON. Whenever the fuel crossfeed and tank selector switch is positioned at CROSSFEED, both boost pumps are energized, provided one engine master switch is ON and the corresponding throttle is advanced to IDLE or above. The aircraft is equipped with a fuel pump crossfeed valve that connects the two fuel systems, allowing either fuel boost pump to supply fuel to both engines. The boost pumps are powered by the dc essential bus.

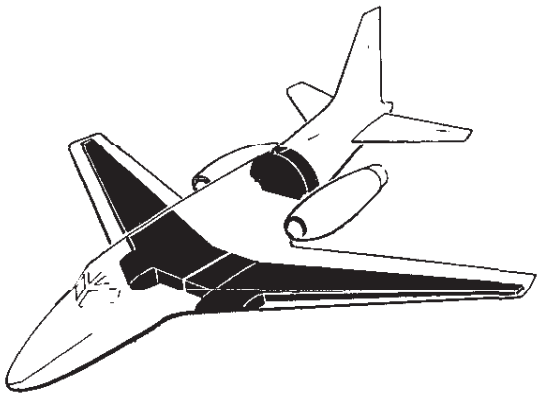
2.14.4 Fuel Transfer Ejector. A fuel transfer ejector in each wing fuel tank in the forward lower portion of the tank near the center bulkhead scavenges fuel from that area and directs it to the boost pump area. One-way check valves in the baffle surrounding the

DATA AS OF: 19 DEC 1961
BASED ON: TEST DATA

BASED ON JP-4 FUEL (6.5 POUND/GALLON—STANDARD DAY)

	USABLE FUEL IN LEVEL FLIGHT	FULLY SERVICED
LEFT WING TANK	451.3 GALLONS 2933.5 POUNDS	463.2 GALLONS 3010.5 POUNDS
RIGHT WING TANK	451.3 GALLONS 2933.5 POUNDS	463.2 GALLONS 3010.5 POUNDS
FUSELAGE TANK	160.5 GALLONS 1043 POUNDS	161 GALLONS 1046.5 POUNDS

TOTAL USABLE FUEL	
WITH FUSELAGE TANK FUEL	1063.1 GALLONS 6910.2 POUNDS
WITHOUT FUSELAGE TANK FUEL	902.6 GALLONS 5866.9 POUNDS



Note

To convert gallons of JP-5 fuel (MIL-T-5624), multiply gallons by 6.8 (Standard Day only).

Quantities shown are with pressure refueling. Total usable fuel after gravity refueling will be about 8 gallons (52 pounds) less than pressure refueling total.

39E-1-48-7
N12/78

boost pump permit fuel flow into the compartment to ensure availability of fuel to the boost pump. The ejectors are self-contained units that have no moving parts and operate by bleeding a portion of the boost pump flow through the submerged ejector that picks up fuel (by venturi action) and directs it to the boost pump compartment.

2.14.5 Fuel Heater. Refer to paragraph 2.13, FUEL HEATER SYSTEM.

2.14.6 Fuel Crossfeed and Tank Selector Switch. The FUEL-X-FEED & TANK SEL switch (17, FO-3) has four positions: NORM, -X-FEED, LH TANK, and RH TANK. For all normal operations, the switch is at the guarded NORM position. If low fuel pressure is indicated by the left or right low fuel pressure caution lights coming on, the switch should be placed at -X-FEED. Selecting the -X-FEED position opens the fuel tank and boost pump crossfeed valves and permits the operable boost pumps to supply the necessary fuel pressure. With the FUEL-X-FEED & TANK SEL switch in X-FEED position and in the event of a crossfeed relay failure or the FUEL X-FEED BOOST circuit breaker is pulled, the fuel tank crossfeed valve will close.

If the low fuel pressure caution light goes out, this indicates that a tank boost pump has failed. If the caution light stays on and engine operation is normal, this may indicate that the differential fuel pressure switch has failed. However, if engine operation is not normal, this indicates the engine driven centrifugal boost pump has failed. The LH TANK and RH TANK positions are for the purpose of wing fuel balancing. For the wing fuel balancing procedures, refer to paragraph 8.15, FUEL BALANCING. Selecting the position (LH TANK or RH TANK) that corresponds with the heavy tank deenergizes the boost pump in the opposite tank, maintains closed or closes the tank crossfeed valve, opens the pump crossfeed valve, and energizes the selected tank boost pump. This permits fuel from the heavy wing tank to be supplied to both engines. When the fuel quantity is equalized, place the switch at NORM. The switch is powered by the dc essential bus. The crossfeed system is protected by circuit breakers on the left forward overhead circuit breaker panel.

Note

During single-engine operation, place the switch at -X-FEED to provide the operating engine with all fuel available and use of both boost pumps. This also equalizes the fuel level in the wing tanks.

2.14.7 Fuel Quantity Switch. The FUEL QTY switch (73, FO-1) has three positions: TEST, TOTAL, and RH IND-RH WING LH IND-AUX. When the switch is moved to TEST, the indicator needles of both fuel quantity indicators should rotate counterclockwise. The function of this position is to test both fuel quantity indicating systems for proper operation by driving both needles away from their true indications. When the fuel quantity switch is placed back at TOTAL, the needles should return to their former positions and show total fuel. When the switch is placed at RH IND-RH WING LH IND-AUX, the right fuel quantity indicator shows usable fuel in the right wing and the left fuel quantity indicator shows usable fuel in the fuselage fuel tank.

2.14.8 Fuel Quantity Gauges. Two fuel quantity gauges (69, FO-1) indicate in pounds of fuel and are calibrated from 0 to 4,000 in increments of 100 pounds. These gauges indicate the total internal fuel supply. Each gauge reads the fuel quantity in one wing tank plus one half the total fuselage tank fuel regardless of the temperature or type of fuel. The total fuel quantity of the fuselage fuel tank may be read on the left fuel quantity indicator. (Refer to paragraph 2.14.7, Fuel Quantity Switch).

Note

- Fuel indications may vary in error as much as 200 pounds because of changes in aircraft attitude during a penetration approach. The same error indications will also apply with a prolonged noseup attitude; the error will be indicated at a much slower rate.

- Fuel quantity indications are sometimes affected when the cockpit instrument and console indirect lights are turned on. The amount of indication change is dependent upon the light intensity and the quantity of fuel remaining.

The fuel quantity indicating system is powered by the 115-vac primary bus. Circuit protection is provided by two circuit breakers on the left aft overhead circuit breaker panel.

2.14.9 Fuel Flow Indicators. The two fuel flow indicators (68, FO-1) are mounted on the pilot instrument panel. The flow indicators show the rate of fuel flow from the fuel control unit to the engine in pounds per hour. The gauges are calibrated in 100-pound increments from 0 to 4,000 pounds per hour. The indicators are powered by the ac primary bus and are protected by separate push-pull circuit breakers on the overhead circuit breaker panel.

2.14.10 Fuel Temperature Gauge. Fuel temperature indication is included in the combination fuel and oil temperature gauge (37, FO-1) on the copilot outboard instrument panel. The gauge is calibrated in 10° increments from -70 to 70 °C (-94° to 158 °F). The temperature probe is in the left wing fuel cell and indicates the temperature of the fuel before it enters the engine fuel control. By monitoring the temperature indicator, it is possible to apply fuel heat before a fuel ice condition exists. When using the fuel temperature indication in conjunction with the fuel-filter-blocked caution light, it is possible to determine whether filter contamination is caused by ice or foreign particles. The gauge is powered by the dc essential bus. Circuit protection is provided by a circuit breaker on the overhead circuit breaker panel.

2.14.11 FUEL SHUTOFF FAIL Caution Light. The placard-type, fuel shutoff fail amber caution light (12, Figure 2-33) comes on to read FUEL SHUTOFF FAIL whenever either or both fuel shutoff valves are not in a fully open or closed position as selected by the engine master switches or engine fire pull handles.

2.14.12 Fuel Tank Crossfeed Failure Caution Light. The placard-type, fuel tank crossfeed fail amber caution light (12, Figure 2-33) comes on to read FUEL TANK X-FD FAIL whenever the position of the tank crossfeed valve does not correspond with the position selected by the fuel crossfeed and tank selector switch. The light is protected by a circuit breaker on the left forward overhead circuit breaker panel.

2.14.13 Fuel Pump Crossfeed Failure Caution Light. The placard-type fuel pump crossfeed fail amber caution light (12, Figure 2-33) comes on to read FUEL PUMP X-FD FAIL whenever the position of the fuel pump crossfeed valve does not correspond with the position selected by the fuel crossfeed and tank selector switch.

Note

During fuel jettison (CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365) with the fuel crossfeed and tank selector switch at NORM, the fuel pump crossfeed fail caution light will not illuminate even though the fuel pump crossfeed valve position differs from that selected by the fuel crossfeed and tank selector switch.

The light is protected by a circuit breaker on the left forward overhead circuit-breaker panel.

2.14.14 Fuel Select Switch Not Normal Caution Light. The placard-type fuel selector switch not normal amber caution light (12, Figure 2-33) comes on to read FUEL SEL SW NOT NORMAL whenever the fuel crossfeed and tank selector switch guard is raised. On CT-39E aircraft modified by Service Bulletin 68-8 and CT-39G aircraft, the caution light is activated when the switch is moved from the NORMAL position and the entire sequence of positioning valves and activating/deactivating boost pumps, corresponding to the selection, has been satisfied. Failure of either crossfeed valve to properly function or a fuel boost pump to cease/begin operation, when required by that position of the fuel selector switch, will prohibit operation of the light.

2.14.15 Low Fuel Pressure Caution Lights. The placard-type low fuel pressure amber caution lights (12, Figure 2-33) come on to read LH FUEL PRESS and RH FUEL PRESS when actuated by either of two pressure switches. One switch, set to close when pressure drops to 5 ±1 psi, is downstream of the fuel shutoff valve and indicates tank boost pump output; the second pressure switch (differential pressure) indicates a differential pressure decrease across the engine-driven centrifugal boost pump. This switch closes at 6 ±1 psi and opens at 9 ±1 psi.

2.14.16 Low Fuel Level Caution Light. The placard-type low fuel level amber caution light (12, Figure 2-33) comes on to read FUEL LOW. The light normally receives its signal from either fuel quantity gauge and is energized when either fuel quantity gauge indicates approximately 300 pounds remaining. However, when the fuel quantity switch is at RH IND RH WING LH IND AUX, the light will illuminate only when there is less than approximately 300 pounds of fuel in the right wing. Circuit protection is provided by a circuit breaker on the left forward overhead circuit breaker panel.

2.14.17 Fuel Jettison System (CT-39E, CT-39G Aircraft 158843, 158844, and 159362 Through 159365). The aircraft is equipped with a fuel jettison system that may be used in any flight attitude or configuration. Normally, fuel is jettisoned at the rate of approximately 180 pounds per minute, depending on altitude, temperature, and thrust setting, and is automatically stopped by the action of the dual-float level switch in

either wing tank. A manual method of jettisoning fuel provides the flightcrew with an alternate means of fuel jettison in case of a malfunction of relays or float switches or will allow additional fuel to be jettisoned. Fuel jettisoning may be used when it is desirable to land with the lightest possible load.

Note

Fuel jettison is inoperative when the weight of the aircraft is on the landing gear.

2.14.17.1 Fuel Jettison Switch. This three-position, guarded switch (26, FO-3) is powered by the dc essential bus. The switch positions are AUTO, OFF, and MAN. Fuel jettisoning is protected by a circuit breaker on the left forward overhead circuit breaker panel. When the fuel jettison switch is placed at AUTO, the fuel boost pump crossfeed valve opens, the fuel tank crossfeed valve remains unmoved, the two fuel jettison valves open, and fuel jettisoning begins. Jettisoning is automatically stopped by the action of a dual-float level switch in each wing tank. When the fuel level in either wing tank decreases to 700 to 1,100 pounds remaining, the dual-float level switch opens the circuit, allowing the fuel boost pump crossfeed valve and the two fuel jettison valves to close. When the switch is held in the spring-loaded MAN position, the fuel boost pump crossfeed valve opens, the fuel tank crossfeed valve remains unmoved, the two fuel jettison valves open, and fuel is jettisoned until the switch is released back to OFF.

Note

When the fuel jettison switch is held at MAN, there is no automatic shutoff feature. Caution should be used because all remaining fuel could be jettisoned.

2.14.17.2 Fuel-Jettison-Open Caution Light. The placard-type fuel-jettison amber caution light (12, Figure 2-33) comes on to read FUEL JETT OPEN whenever either of the fuel jettison valves is in any position other than closed. If automatic jettisoning is being used, the fuel jettison switch should be placed at OFF when the caution light goes out. The light is protected by a circuit breaker on the left forward overhead circuit breaker panel.

2.14.18 Fuel Availability During Flight. If the aircraft is flown at a pitch angle in excess of about 8-1/2° noseup for a protracted period, fuel will be trapped in the outboard segment of each wing tank. Consequently, flameout could occur under this flight condition even though the fuel quantity gauges show a usable supply of fuel remaining in the tanks. During low-altitude climbs, the aircraft noseup pitch attitude exceeds 8-1/2° for only a short period of time, insufficient to induce the flameout condition. This is because low-altitude climb cannot normally be entered without the aircraft having been previously held in a level attitude (or nosedown attitude such as on the ground before takeoff, in level flight, or in a descent) that would permit gravity flow of the fuel. For all other normal flight conditions, the aircraft noseup pitch attitude will be less than that which would result in trapping the fuel in the outboard segments of the wing tanks. (The normal flight condition which most nearly approaches the critical 8-1/2° noseup pitch attitude is loitering flight, then, if recommended speeds are observed, the pitch attitude will not exceed about 6-1/2° noseup.)

2.15 HYDRAULIC SYSTEM

The hydraulic system (FO-15) is a 3,000-psi utility system powered by an electric-motor-driven pump. The hydraulic system supplies power for operation of the landing gear, main gear doors, speedbrake, wheelbrakes, thrust reversers (CT-39G), and nosewheel steering.

The hydraulic reservoir (fluid capacity, 2 U.S. gallons) supplies fluid to the hydraulic pump. The reservoir is pressurized by air to provide a constant head of pressure to the hydraulic pump, preventing pump cavitation, regardless of altitude. An air pressurization system supplies compressor bleed air from a storage tank through a pressure regulator and check valve to the reservoir. The air storage tank is pressurized from the engines to about 90 psi. A ground servicing connection permits the servicing of the air storage tank without running the engines.

A reservoir air pressure gauge and dump valve are installed adjacent to the reservoir. For normal operation, the air gauge will indicate a pressure of 35 to 37 psig (air pressurization system regulator setting). The hydraulic reservoir has a filler cap for gravity filling. An accumulator in the hydraulic system stores hydraulic pressure at 3,000 psi to supplement pump output during sudden high rates of demand. Two self-sealing couplings on the bottom of the fuselage below the left engine pod pylon permit groundtesting, filling, and flushing of the hydraulic power source. Pressure from the ground source is applied to the aircraft system pressure line through the ground pressure coupling and is prevented from flowing to the hydraulic pump by a check valve. Return flow to the ground return coupling passes through the hydraulic reservoir into the system supply lines and bypasses the hydraulic pump. (See Figure 3-1 for hydraulic fluid specification.)

A hydraulic pressure warning system is installed to provide the flightcrew with an audible warning of an abnormal hydraulic system condition. This system is an additional means of alerting the flightcrew prior to engine starting that hydraulic system pressure is low or may become inadequate for wheel-braking.

2.15.1 Hydraulic Power System. System pressure is provided by an electric-motor-driven hydraulic pump which operates according to the demands on the system. The pump motor is powered by the paralleling bus with pump operation controlled by dc essential bus power through a pressure switch. When system pressure is below about 2,700 psi, the pressure switch closes to energize the pump motor. The pump continues to operate until system pressure is about 3,000 psi. At this time, the switch opens, deenergizing the pump. A time-delay relay automatically shuts off electrical power to the pump motor if the pump runs continuously for over 1 minute. (This time limit exceeds the maximum operational demands on the pump.) A caution light shows when the time-delay relay interrupts electrical power to the pump, and a switch in the cockpit permits the pump power circuit to be reset. The 3,000-psi pressure maintained in an auxiliary hydraulic accumulator by the system pump can be selected to operate the essential hydraulic functions upon failure of the normal portion of the hydraulic power system. The auxiliary accumulator pressure is routed to the speedbrake, wheelbrake, nosewheel steering and auxiliary assist for gear retraction systems through separate lines, as shown in FO-15, to reduce the possibility of one hydraulic power system failure affecting the other. (The all-mechanical emergency operation of the landing gear makes it unnecessary to provide auxiliary pressure to the gear.) The auxiliary pressure should be selected and used only when required and provides sufficient fluid and pressure to complete a landing if the normal hydraulic system fails.

Note

A check of normal operation of the hydraulic pump can be made with both the normal and auxiliary accumulators fully charged, the auxiliary hydraulic power switch at OFF, and no hydraulic demand on the system. During such conditions, the pump should not operate more often than once every 5 minutes and should build up to a full 3,000 psi smoothly and promptly during each cycle. (An increase in the dc loadmeter reading shows when the pump is operating.) If pump operation exceeds the specified cycle, excessive internal leakage or bypass of the hydraulic system is indicated and the system should be checked out.

2.15.1.1 Hydraulic Pump Switch. The three-position hydraulic pump switch on the hydraulic control panel (28, FO-3) controls paralleling bus power to the hydraulic pump motor by means of dc essential bus power. When the switch is at AUTO, the pump motor is cycled on and off automatically by a pressure switch in response to the demands of the hydraulic system. The pump circuit is inoperative when the switch is at OFF & RESET. If the pump runs continuously for an excessive period (over 1 minute), a time-delay relay shuts off the pump motor by tripping a circuit breaker. To restore electrical power to the pump, the circuit breaker can be reset by moving the pump switch momentarily to OFF & RESET and then to AUTO.

CAUTION

Do not use the RESET & OFF position more than once to extinguish the HYD PRESS/PWR OFF and MASTER CAUTION lights. If the caution lights reappear, place the switch in the RESET & OFF position and leave off until the malfunction is corrected.

Selection of the MANUAL position permits hydraulic pump motor operation in case of failure of the automatic hydraulic pump control circuit. The hydraulic pump motor runs continuously when the switch is held in the MANUAL position. The switch should be held in the MANUAL position only long enough to restore hydraulic pressure to the normal operating range.

CAUTION

- If hydraulic pressure does not start to build up almost immediately when using the MANUAL position, release the switch. Place the switch at RESET & OFF and leave off until the malfunction is corrected. In any case, do not hold the switch in MANUAL for longer than 1 minute.
- When the switch is held in MANUAL position, monitor the normal system hydraulic gauge. Release the switch when the hydraulic pressure reaches the top of the normal operating range.

When the spring-loaded switch is released from the MANUAL position, it automatically returns to the center AUTO position.

2.15.1.2 Auxiliary Hydraulic Power Switch. Hydraulic power from the auxiliary accumulator is selected by a two-position switch on the hydraulic control panel (28, FO-3). When the switch is at its normally selected OFF position, the auxiliary shutoff valve is closed. If the normal hydraulic power system fails and the normal system is depleted, the switch should be moved to ON only when hydraulic pressure is required. This opens the auxiliary shutoff valve to direct auxiliary accumulator pressure to the speedbrake, wheelbrake, and nosewheel steering systems. The switch should be returned to OFF whenever possible to conserve auxiliary accumulator power. The auxiliary hydraulic switch is powered by the dc essential bus.

2.15.1.3 Hydraulic Pressure Gauges. The hydraulic pressure gauges (65, FO-1), on the center instrument panel, are powered by the No. 1 26-vac bus. In addition to the NOR (normal) system pressure gauge, an AUX (auxiliary) system gauge is installed that shows the auxiliary hydraulic pressure. The gauges are calibrated in 500 psi increments from 0 to 4,000 psi.

2.15.1.4 Hydraulic Pressure Power-Off Caution Light. The hydraulic pressure power-off caution light on the caution-warning light panel (12, Figure 2-33) is powered by the dc essential bus. This caution light comes on when any one of the following conditions exists:

1. Hydraulic system pressure is below 2,700 psi for 1 minute or more.
2. Hydraulic pump switch is in the OFF & RESET position.
3. Hydraulic pump motor draws excessive current.
4. Hydraulic pump motor develops excessive temperature.
5. Hydraulic pump motor runs in excess of 1 minute.
6. HYD PUMP CONT circuit breaker is out.
7. Hydraulic system pressure decreases to 1,900 ±100 psi. When pressure builds back up to 2,600 ±100 psi, the light will go off. The master caution light is not on under this condition.

Note

The hydraulic pump is protected by three circuits: overtime, overtemp, and overcurrent. The following may be performed to determine the fault:

1. Overtime will run in manual and will reset and is usually a result of a system leak.
2. Overtemp will run in manual, but will not reset; automatically resets when cool.
3. Overcurrent will not run in manual and will not normally reset.

2.15.1.5 Hydraulic System Low-Pressure Audible Warning. On CT-39G aircraft 159362 through 159365 and 160053 through 160057, an audible hydraulic pressure warning system is provided by the Mach-airspeed audible warning horn. This system is operable only with the weight of the aircraft on the landing gear. When either engine master switch is placed on the ON position with (1) the normal hydraulic system pressure below 2,000 ±100 psi, or (2) the hydraulic pump switch at OFF & RESET or MANUAL, or (3) either hydraulic pressure (AUX or NORMAL) circuit breaker disengaged (out), an interrupted tone will be heard from the Mach-airspeed warning unit and in the pilot headset. When pressure increases to 2,600 ±100 psi, the warning will stop. To prevent actuation of the warning horn, prior to engine starting, allow hydraulic system pressure to increase to 2,600 ±100 psi before placing either engine master switch at ON. This system has no warning horn silence capability.

2.15.2 Hydraulic System Fail Audible Warning (CT-39E, CT-39G Aircraft 158843 and 158844). The audible warning horn on the hydraulic control panel (Figure 2-32) is powered by the dc essential bus. To the right of the horn, a TEST-RESET switch and SILENCE button are provided. If the horn is silenced using the SILENCE button, the RESET position must be momentarily selected to rearm the warning system. The horn is armed whenever one or both generators is on and weight is on the main landing gear. The horn will sound if armed when:

1. Hydraulic pump switch is OFF and RESET.
2. HYD PUMP CONT circuit breaker is out.
3. Hydraulic pressure decreases to 1,900 ±100 psi.

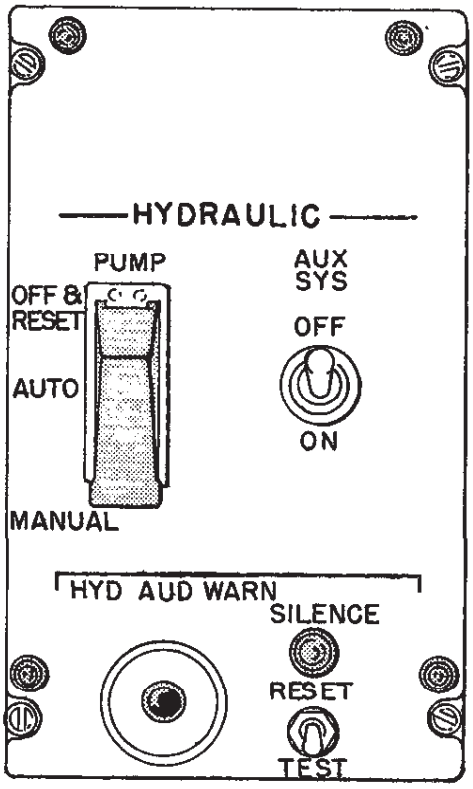


Figure 2-32. Hydraulic Control Panel

2.16 INDICATOR, CAUTION, AND WARNING LIGHT SYSTEM

A caution-warning light panel (12, Figure 2-33), mounted on the copilot instrument panel, contains 37 placard-type caution lights and one warning light. It is powered by the dc essential bus. All of the placard-type lights on the caution-warning light panel are amber caution lights with the exception of the cabin pressure failure and ac instrument and battery hot red warning lights. CT-39E and CT-39G (158843 and 158844) have a battery hot red warning light. Two amber master caution lights are installed below the instrument panel shroud, one in front of the pilot and one in front of the copilot. Annunciator panel lights will be accompanied by both master caution lights except one condition of low hydraulic pressure. This alerts the pilot or copilot to check the placard-type lights on the instrument panel to determine which system has not functioned properly. To extinguish the master caution lights so that they will indicate any additional failure, either master caution light must be depressed momentarily. This leaves only the placard-type caution light on until the fault is cleared. There are other caution and warning lights in the cockpit that do not illuminate the master caution lights when they come on. They are the two red starting cycle warning lights, the red landing gear unsafe warning light, the amber autopilot disengage light, the amber landing lights extended caution light, and the red warning lights in engine fire pull handles. Indicator lights in the cockpit are the white, blue, and amber marker beacon lights; the three green landing gear position indicator lights; the red passenger oxygen flow indicator light; the two green nosewheel steering on indicator lights; the two white minimum decision altitude lights, the two white go-around lights; and the white, green, and amber annunciator lights.

CAUTION

When removing or installing lamps, legend plates, or troubleshooting caution and warning panel, all electrical power must be off aircraft to avoid shorting out diodes and welding relay contacts.

2.16.1 Caution Light Test Switch. The CAUTION TEST switch, on the lighting control panel (Figure 2-46), is powered by the dc essential bus. The switch has three positions marked BRIGHT, DIM, and OFF (center) and is spring loaded to the OFF position. When the switch is actuated in either the BRIGHT or the DIM position, the respective circuit to all placard-type caution lights, master caution lights, and indicator and warning lights except the starting cycle warning lights and the autopilot disengage light is actuated to indicate any defective lamps.

2.16.2 Indicator, Caution, and Warning Lights. All indicator, caution and warning lights are illuminated by the dc essential bus and tested by the caution light test switch. The pilot flight instrument light rheostat provides the dimming control for all indicator, caution, and warning lights except the nosewheel steering-on indicator lights, the marker beacon lights, the minimum decision altitude lights, the go-around lights, the annunciator lights, the starting cycle warning lights, the autopilot disengage light, and the fire warning lights in the fire pull handles. For details concerning an individual light, refer to the respective system description.

2.17 INSTRUMENTS

The following paragraphs cover only instruments that are not part of a complete system such as the fuel system, engines, etc.

2.17.1 Pitot-Static System — CT-39E Aircraft and CT-39G Aircraft 158843. A complete independently operated pitot-static system (Figure 2-36) is provided for each set of pilot and copilot instruments.

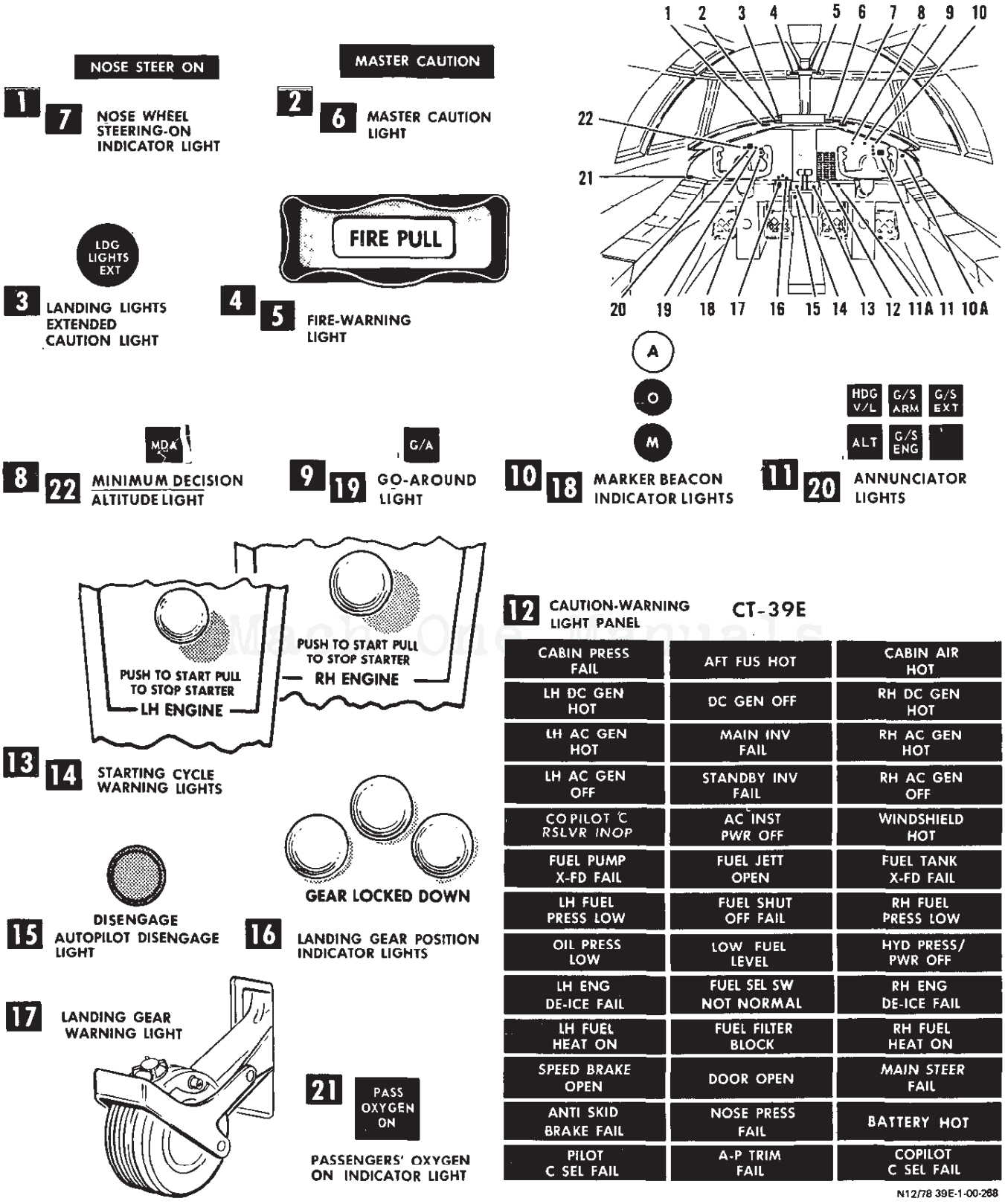


Figure 2-33. Indicator, Caution, and Warning Lights (Sheet 1 of 2)

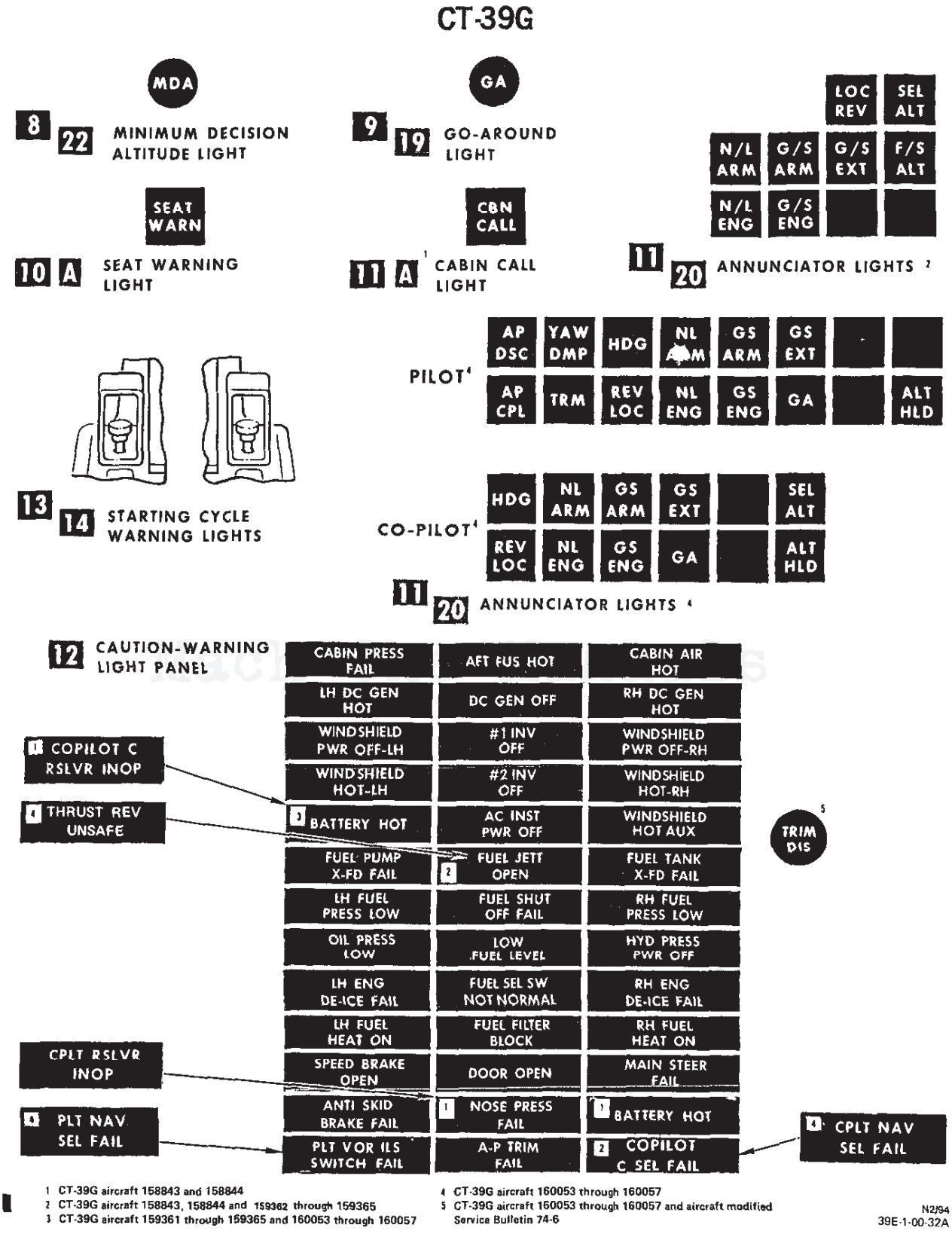


Figure 2-33. Indicator, Caution, and Warning Lights (Sheet 2 of 2)

INDICATION	PROBABLE CAUSE	RECOMMENDED ACTION
Cabin Pressure Fail (Activates) Fasten Seat Belts	1. *PRESSURE* Switch Failed. 2. Cabin Altitude at or over 10,000 feet. 3. Lo Engine power. 4. Leak in Cabin. 5. Leak in Aft Ducts.	1. Crew oxygen masks - ON; 100-Percent Oxygen. 2. Cabin air select switch - Emergency Pressure or Both Engines, as required. 3. Descend rapidly if necessary.
Aft Fuselage Hot	Temperature at or over 275° from Air Duct leakage or Fire.	1. Crew oxygen mask - ON; 100-Percent Oxygen. 2. Passenger oxygen normal and emergency switches open (if necessary). 3. *Cabin air select: below 15,000* feet - RAM; above 15,000 feet - EMERG PRESS.
Cabin Air Hot	1. Duct Temperature 265° due to a. Cabin Mixing Valve Failure, b. Auto Temperature Control Failure.	1. Select manual cold. If light goes out adjust temperature manually as required, if light remains on and temperature is high, select emergency pressurization.
Left or Right D.C. Generator Hot	Generator temperature 311° caused by 1. Loss of cooling air, 2. Bearing failure. 3. Commutator short. 4. Field Short. 5. Faulty probe.	1. Affected generator switch - OFF. If light does not go out, proceed as follows. 2. Power lever (affected eng) - OFF. 3. Fuel X-feed and tank SEL switch - X-FEED. 4. Land as soon as possible.
D.C. Generator Off	1. Generator Switch Off. 2. Control C/B Out. 3. Generator Shaft sheared. 4. Overvolts, undervolts or ground Fault.	1. W/s heat switch Off. 2. Check loadmeters. 3. Generator switch Off. 4. Check generator volts. 5. If zero reset, switch On If gen. fails to reset check C/B. If gen. fails to go ON, switch OFF and Pull control C/B. If gen. goes on W/S heat Switch. ON.
LH (or) AC GEN HOT	Overheating due to mechanical or electrical malfunction	Refer to windshield anti-icing system operation, Part I.
LH (or) RH AC GEN OFF	AC generators switch-off or refer to WINDSHIELD ANTI-ICING SYSTEM OPERATION, Section I.	*Refer to windshield anti-icing* system operation, Part I.
Main Inv Fail	Main inverter inoperative.	Check switch position or refer to inverter failure, Part V.
Standby Inv Fail	Must be accompanied by MAIN INV FAIL.	Refer to inverter failure, Part V.
Copilot C Rslvr Inop	Same NAVAID selected due to failure of switching relays.	Use pilots FD and course indicator for navigation.
AC Inst Prw Off	Must be accompanied by MAIN and STANDBY INV FAIL lights unless caused by indicating relay failure.	Refer to inverter failure, Part V.
Windshield Hot	Refer to WINDSHIELD ANTI-ICING SYSTEM OPERATION, Section I.	Refer to windshield anti-icing system operation, Section I.
Fuel Pump X-Feed Fail.	1. Valve did not reach selected position. 2. Boost pump X Feed C/B out with switch in X-Feed position.	1. If Selection was made due to low fuel pressure and low pressure lite goes out, valve is partially open. Use caution - check balance. Do not take-off with this condition. 2. C/B in. 3. To balance fuel, use steady-state constant-heading sideslip of one to one-half ball deflection with the heavy wing high.
Fuel Jettison Open	1. Switch in Manual or Auto. 2. Valves not closed with Switch Off.	Pull jettison C/B. If Light still on pull both boost pump C/B's and land. Power changes with caution
Fuel Tank X-Feed Fail.	Valve not in selected position.	1. Use LH tank/RH tank positions of the fuel X-feed and tank selector switch or differential engine power to maintain fuel balance. 2. Land as soon as practicable.
Left or Right Fuel Pressure Low	1. Line Pressure Switch Failed. 2. Diff. Press. Switch Failed. 3. Boost Pump Failed. 4. Line or Component Leak.	Select X-FEED. If light goes out - OK. If light remains on - move switch to NORMAL.
Fuel Shut Off Fail	Valve not in selected position.	If on Ground - Repair. If in flight - engine cannot be repaired.

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INDICATION	PROBABLE CAUSE	RECOMMENDED ACTION
Oil Pressure Low.	1. One Engine Shut Down. 2. If Both Engines running Check Gauges.	Check gauges and if below 35 psi shut down affected engine.
Low Fuel Level	1. Actual Low Level (297) in either or both tanks. 2. Faulty Gauge. 3. Tank Leak.	1. Select X-Feed and land immediately. 2. Gauge test and X-Check with A/C lateral trim. 3. Land immediately.
Hydraulic Pressure Power Off	1. Switch Off. 2. C/B Out. 3. Pump Cycling - No Master - Caution. 4. Pump Overheat. 5. Pump Overcurrent. 6. Pump Overtime. 7. Cycling Switch Failed.	1. Check switch position. 2. Check C/B position. 3. Reset pump. If light out OK - but monitor. 4. If light on select manual. If all fails, switch off and pull C/B.
Left or Right Engine De-Ice Fail	1. Low Engine Power. 2. Engine or Nacelle Valve Failed.	1. Increase engine power until light out. 2. Try X-Check through EPR gauges. 3. Increase air speed for friction heat, continuous ignition On and get out of Icing as soon as possible.
Fuel Selector Switch Not Normal	1. Switch Guard Raised. 2. Switch is moved from the normal position.	As required.
Left or Right Fuel Heat On	1. Filter Heat Switch On. 2. Switch Off and Valve Not in Closed Position. 3. C/B Out.	1. Position switch as required. 2. Safety circuit by pulling C/B and monitor engine instruments.
Fuel Filter Block	1. Ice on Filter. 2. Contamination.	1. Select filter heat and continuous ignition. 2. If light not out in few secs. land as soon as possible
Speed Brake Open	1. Brake Selected and Power over 95%. 2. Selector Valve Failed.	1. Select brake In. 2. Dump speed brake.
Door Open	1. Not Properly Locked. 2. Micro Switch Malfunction. 3. Structural Deformation.	1. Fasten seat belts sign - On. 2. Reduce airspeed and altitude.
Main Steer Fail	1. Switch in Standby. 2. Electrical Malfunction.	1. Stop aircraft. 2. Select STBY (if required). 3. Taxi to ramp. 4. Investigate cause.
Anti-Skid Brake Fail	1. Brakes applied after T/O. 2. Anti-Skid Switch Off. 3. Continuous Brake Release for 2.7 secs. 4. Electrical Failure in System.	1. Do not brake after T/O. 2. Select switch On. 3. Switch Off and On reset. 4. System inoperative.
Nose Pressure Fail	Differential Exceeded Relief Setting of 3.5.	1. Relief valve cannot be reset.
Pilots or Copilots "C" Selector Fail	1. Instrument Power Loss.	Use crossover system as required.
A-P Trim Fail.	Auto Pilots Servos Loaded.	Disconnect or turn Off auto pilot and fly manually.

NOTES: 1. Both master caution lights will illuminate with any annunciator light, EXCEPT HYDRAULIC PRESSURE POWER OFF may be on alone if demand of system exceeds pump supply (one min. max).
2. The master caution lights can be reset "Out" in all cases.
3. Where two legends are depicted for one location, refer to caution warning light annunciator panel for aircraft coding.

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Figure 2-34. Caution-Warning Light Panel (CT-39E) (Sheet 2 of 2)

INDICATION	PROBABLE CAUSE	RECOMMENDED ACTION
Cabin Pressure Fail (Activates) Fasten Seat Belts	1. Pressure Switch Failed. 2. Cabin Altitude at or over 10,000 feet. 3. Lo Engine power. 4. Leak in Cabin. 5. Leak in Aft Ducts.	1. Crew oxygen mask - On; 100-Percent Oxygen. 2. Cabin air select switch - Emergency Pressure or Both Engines, as required. 3. Descend rapidly if necessary.
Aft Fuselage Hot	Temperature at or over 275° from Air Duct leakage or Fire.	1. Crew oxygen masks - On; 100-Percent Oxygen. 2. Passenger oxygen normal and emergency switches open (if necessary). 3. Cabin air select: below 15,000 feet - RAM; above 15,000 feet - EMERG PRESS.
Cabin Air Hot	1. Duct Temperature 265° due to a. Cabin Mixing Valve Failure. b. Auto Temperature Control Failure.	1. Select manual cold. If light goes out adjust temperature manually as required, if light remains on and temperature is high, select emergency pressurization.
Left or Right D.C. Generator Hot	Generator temperature 311° caused by 1. Loss of cooling air. 2. Bearing failure. 3. Commutator short. 4. Field Short. 5. Faulty probe.	1. Affected generator switch - OFF. If light does not go out, proceed as follows. 2. *Power lever (affected eng)* - OFF. 3. Fuel X-feed and tank SEL switch - X-FEED. 4. Land as soon as possible.
D.C. Generator Off	1. Generator Switch Off. 2. Control C/B Out. 3. Generator Shaft sheared. 4. Overvolts undervolts or, ground Fault.	1. W/S heat Switch Off. 2. Check loadmeters. 3. Generator Switch Off. 4. Check generator volts 5. If zero reset, switch On If generator fails to reset check C/B. If generator fails to go On, switch Off and Pull control C/B. If generator goes on W/S heat Switch On.
Windshield Pwr Off - LH and RH	*Inverter failure.*	*Reselect inverters as required.*
No. 1 Inverter Off	No. 1 Inverter Failed.	Select No. 2 inverter (if auto transfer fails).
Windshield Hot - LH and RH	Overheated Windshield.	Turn off heat (decrease altitude and airspeed).
No. 2 Inverter Off	No. 2 Inverter Failed.	Reselect inverters as required.
Battery Hot or Copilot C Rslvr Inop	Charge Rate to High. Same NAVAIR Selected Due to Failure of Switching Relays.	Isolate batteries. Use pilots FD and course indicator for navigation.
AC Inst Pwr Off	Must be accompanied by MAIN and standby inv fail lights unless caused by indicating relay failure.	Refer to inverter failure, Part V.
Windshield Hot Aux	Refer to windshield anti-icing system operation, Part I.	Refer to windshield anti-icing system operation, Part I.
Fuel Pump X-Feed Fail.	1. *Valve did not reach* selected position. 2. Boost pump X-Feed C/B out with switch in X-Feed position.	1. If selection was made due to low fuel pressure and low pressure lite goes out, valve is partially open. Use caution - Check balance. Do not takeoff with this condition. 2. C/B in. 3. To balance fuel, use steady-state constant-heading sideslip of one to one-half ball deflection with the heavy wing high.
Fuel Jettison Open or Thrust Rev Unsafe	1. Switch in Manual or Auto. 2. Valves not closed with Switch Off. Doors not Stowed.	Pull jettison C/B. If light still on pull both boost Pump C/B's and land. Power changes with caution. Select EMERG STOW.
Fuel Tank X-Feed Fail.	Valve not in selected position.	1. Use LH tank/RH tank positions of the fuel X-feed and tank selector switch or differential *engine power to maintain fuel* balance. 2. *Land as soon as practicable.*
Left or Right Fuel Pressure Low	1. Line Pressure Switch Failed. 2. Diff. Press. Switch Failed. 3. Boost Pump Failed. 4. Line or Component Leak.	Select X-FEED. If light goes out - OK. If light remains on - move switch to NORMAL.
Fuel Shut Off Fail	Valve not in selected position.	If on ground - repair. If in flight - engine cannot be repaired.

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Figure 2-35. Caution-Warning Light Panel (CT-39G) (Sheet 1 of 2)

INDICATION	PROBABLE CAUSE	RECOMMENDED ACTION
Oil Pressure Low.	1. One Engine Shut Down. 2. If Both Engines running Check Gauges.	Check gauges and if below 35 psi shut down affected engine.
Low Fuel Level	1. Actual Low Level (297) in either or both tanks. 2. Faulty Gauge. 3. Tank Leak.	1. Select X–FEED and land immediately. 2. Gauge test and X-check with A/C lateral trim 3. Land immediately.
Hydraulic Pressure Power Off	1. Switch Off. 2. C/B Out. 3. Pump Cycling - No Master - Caution. 4. Pump Overheat. 5. Pump Overcurrent. 6. Pump Overtime. 7. Cycling Switch Failed.	1. Check switch position. 2. Check C/B position. 3. Reset pump. If light out OK - but monitor. 4. If light On select manual. If all fails, switch off and pull C/B
Left or Right Engine De-Ice Fail	1. Low Engine Power. 2. Engine or Nacelle Valve Failed.	1. Increase engine power until Light Out. 2. Try X-check through EPR Gauges. 3. Increase air speed for friction heat, continuous ignition On and get out of Icing as soon as possible.
Fuel Selector Switch Not Normal	Fuel Selector Switch Not in NORM Position	Reposition switch if necessary
Left or Right Fuel Heat On	1. Filter Heat Switch On. 2. Switch Off and Valve Not in Closed Position. 3. C/B Out.	1. Position switch as required. 2. Safety circuit by pulling C/B and monitor engine instruments.
Fuel Filter Block	1. Ice on Filter. 2. Contamination.	1. Select filter heat and continuous ignition. 2. If light not Out in few secs., Land as soon as possible.
Speed Brake Open	1. Brake Selected and Power over 95%. 2. Selector Valve Failed.	1. Select brake In. 2. Dump speed brake.
Door Open	1. Not Properly Locked. 2. Micro Switch Malfunction. 3. Structural Deformation.	1. Fasten seat belts sign - On. 2. Reduce airspeed and altitude.
Main Steer Fail	1. Switch in Standby. 2. Electrical Malfunction.	1. Stop aircraft. 2. Select STBY (if required). 3. Taxi to ramp. 4. Investigate cause.
Anti-Skid Brake Fail	1. Brakes applied after T/O. 2. Anti-Skid Switch Off. 3. Continuous Brake Release for 2.7 secs. 4. Electrical Failure in System.	1. Do not brake after T/O. 2. Select switch On. 3. Switch Off and On Reset. 4. System inoperative.
Nose Pressure Fail	Differential Exceeded Relief Setting of 3.5.	Relief valve cannot be reset.
Pilots or Copilots "C" Selector Fail or Plt Nav Sel Fail.	Instrument Power Loss.	Use crossover system as required.
A-P Trim Fail.	Auto Pilots Servos Loaded.	Disconnect or turn Off auto pilot and fly manually.

NOTES: 1. Both master caution lights will illuminate with any annunciator light, EXCEPT HYDRAULIC PRESSURE POWER OFF may be on alone if demand of system exceeds pump supply (One Min. Max).
2. The master caution lights can be reset "Out" in all cases.
3. Where two legends are depicted for one location, refer to caution warning light annunciator panel for aircraft coding.

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Figure 2-35. Caution-Warning Light Panel (CT-39G) (Sheet 2 of 2)

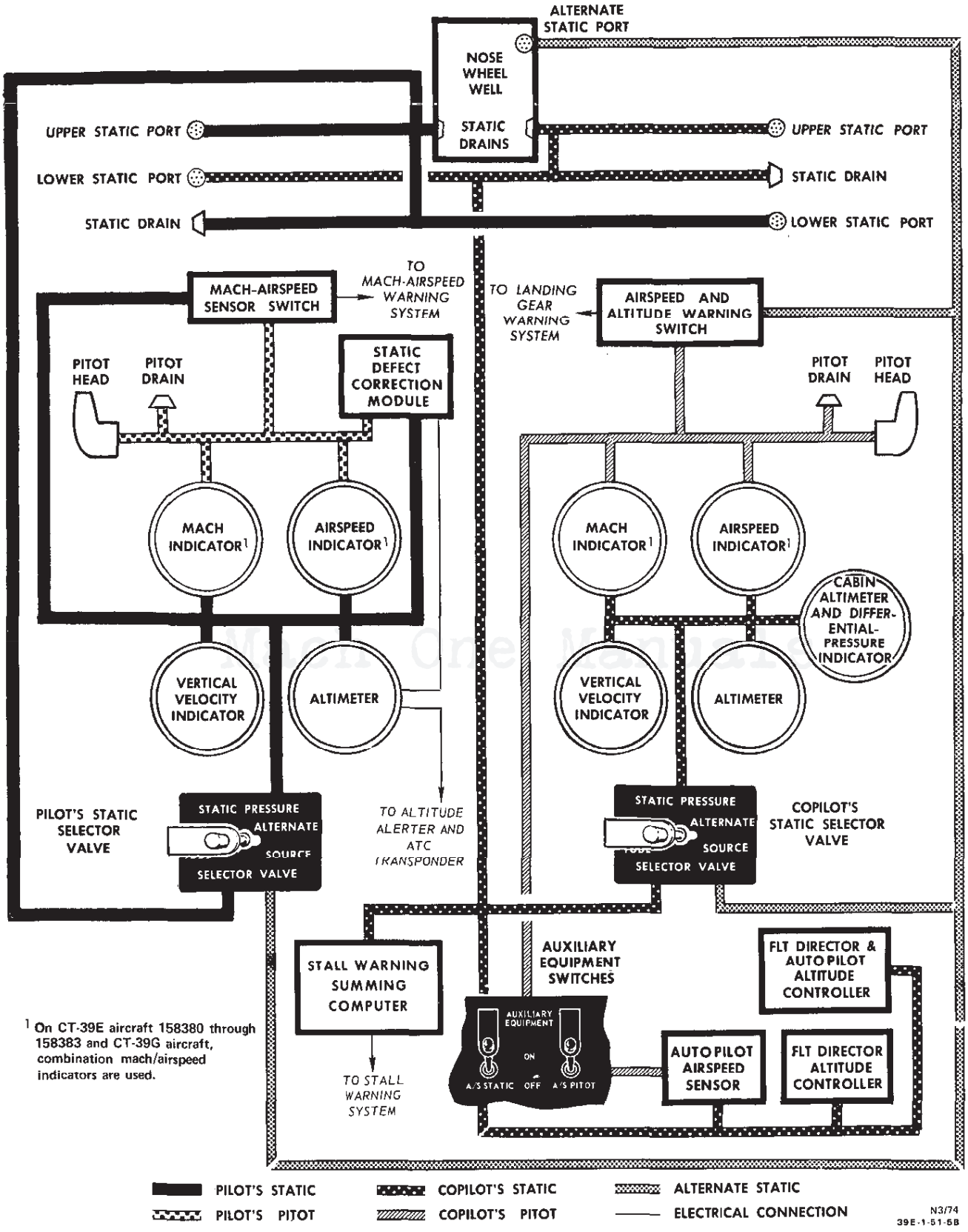


Figure 2-36. Pitot-Static System — CT-39E Aircraft 158843 and 158844

An alternate static-pressure source, common to both sets of instruments, is also provided. Guarded toggle-type selectors are on both the left and right instrument subpanels for transferring from the primary static-pressure source to the alternate static-pressure source. Dual, electrically heated pitot-pressure heads are installed, one on each side of the fuselage, below the windshield.

Fluctuating pitot-static instrument indications have occurred on some Sabreliners while flying in rain during the final approach for landing. Use of the alternate static-pressure source will stabilize the pitot-static instrument indications. However, because of the location of the alternate static-pressure port, airspeed and altimeter indications are not the same as when the normal system is used. Flight-test results indicate that at final approach airspeeds, the differences are small. Indicated airspeed on the alternate system is approximately 5 KIAS lower than on the normal system; the altimeter on the alternate system indicates approximately 60 feet lower than the normal system. Because these corrections are on the safe side, normal final approach airspeeds and indicated altitudes are recommended for use when operating on the alternate static-pressure system.

2.17.2 Pitot-Static System — CT-39G Aircraft Except 158843. On these aircraft, the pitot-static system consists of left- and right-hand pitot-static tubes, pilot and copilot static selector valves, and the necessary lines for interconnection to the pilot and copilot instruments. (See Figure 2-37.) The left-hand pitot-static tube provides pitot pressure for the pilot instruments; the right-hand tube provides pitot pressure for the copilot instruments as well as auxiliary equipment. Static ports in the left- and right-hand pitot-static tubes provide the static source. The static selector valves allow for selection of the source of static pressure.

The selector valves are normally positioned at BOTH. In this position, the static ports in both pitot-static tubes are interconnected to minimize sideslip effects. Dual electrically heated, aerodynamically compensated pitot-static tubes are installed, one on each side of the fuselage, below the windshield. The tubes are specifically designed for the Sabreliner aircraft and are aerodynamically compensated for static pressure position error at the mounting location. Aerodynamic compensation forms an integral part of the tube and is achieved by appropriate contour shaping of the tube and proper placement of the static ports.

2.17.3 Airspeed Indicators. The airspeed indicators (4 and 56, FO-1) are conventional airspeed indicators with the addition of an aneroid-operated maximum allowable airspeed pointer that indicates the maximum airspeed at any particular altitude[1].

2.17.4 Mach Indicators[1]. Mach indicators (60 and 87, FO-1) give a more usable speed indication at higher altitudes by indicating the ratio of aircraft speed to the speed of sound at the particular altitude and temperature existing during any period of flight. A single pointer indicates the Mach number on a dial face graduate in tenths[1].

2.17.5 Mach-Airspeed Indicators.[2] The Mach-airspeed indicators (4 and 56, FO-1) provide a simplified presentation of both indicated airspeed and Mach number. Both indications are read from the same pointer. The pointer indicates airspeed at low speeds and both indicated airspeed and Mach number at high speeds. The airspeed scale is stationary, but the Mach number scale is movable, controlled by an aneroid diaphragm that reacts to static-pressure changes as the altitude changes. The aneroid also automatically positions a maximum allowable airspeed pointer to indicate the maximum airspeed/Mach number at any particular altitude. The instrument incorporates an adjustable index (which can be used as a landing speed index). The index is set manually by a knob at the lower corner of the instrument.

2.17.6 Mach-Airspeed Audible Warning System. Audible warning that the aircraft has reached either the limit Mach or limit airspeed is provided by the Mach-airspeed audible warning system and amplified through the audio system. The system includes a pitot-static pressure-sensing switch, a relay, a warning horn, and related wiring. The horn is on the bulkhead behind the pilot. At the maximum allowable Mach-airspeed limit, an interrupted tone will be heard from the warning horn and in the pilot and copilot headsets. The warning tone is not reproduced through the loudspeakers. The system is powered by the dc essential bus and can be tested by the Mach-airspeed warning test button. On CT-39G aircraft 159361 through 159365 and 160053 through 160057, the Mach-airspeed audible warning system horn also provides for the audible warning of a hydraulic system low-pressure condition. This hydraulic pressure warning system is operable only with the weight of the aircraft on the landing gear.

[1] CT-39E aircraft 157352 through 157354

[2] CT-39E aircraft 158360 through 158383 and CT-39G indicators

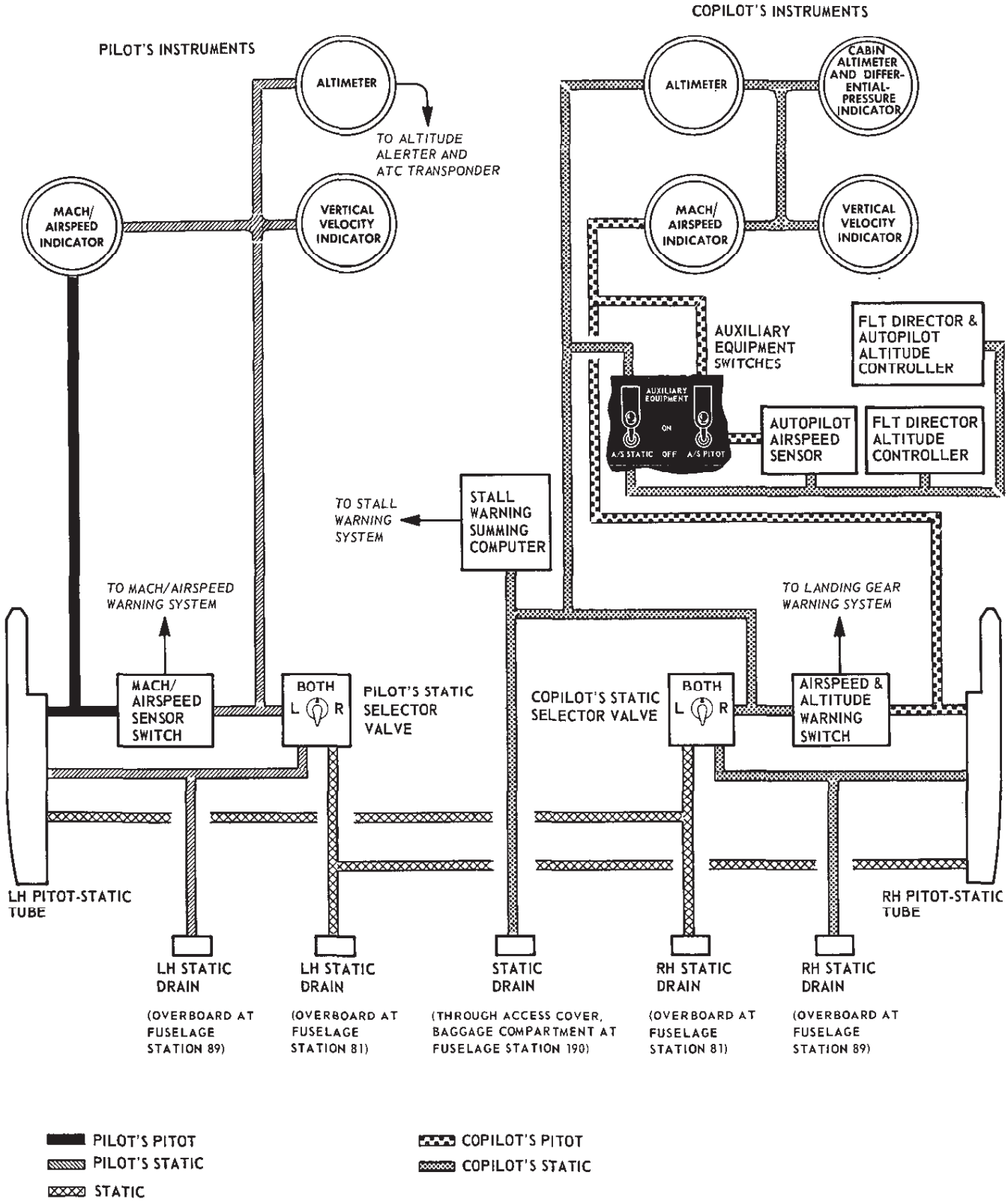


Figure 2-37. Pitot-Static System — CT-39G Aircraft 159362 Through 159365 and 160053 Through 160057

2.17.6.1 Mach-Airspeed Warning Test Button. This MACH-AS WARNING TEST pushbutton (82, FO-1) controls dc essential bus power for testing the Mach-airspeed audible warning system. Depressing the button energizes the warning system relay, causing the horn to sound and a steady tone to be heard in the headsets.

2.17.7 Vertical Velocity Indicators. The vertical velocity indicators (53 and 75, FO-1) indicate the rate of climb or descent in feet per minute or provide an indication of level flight. Actuation of the pointer is controlled by the rate of change of atmospheric pressure.

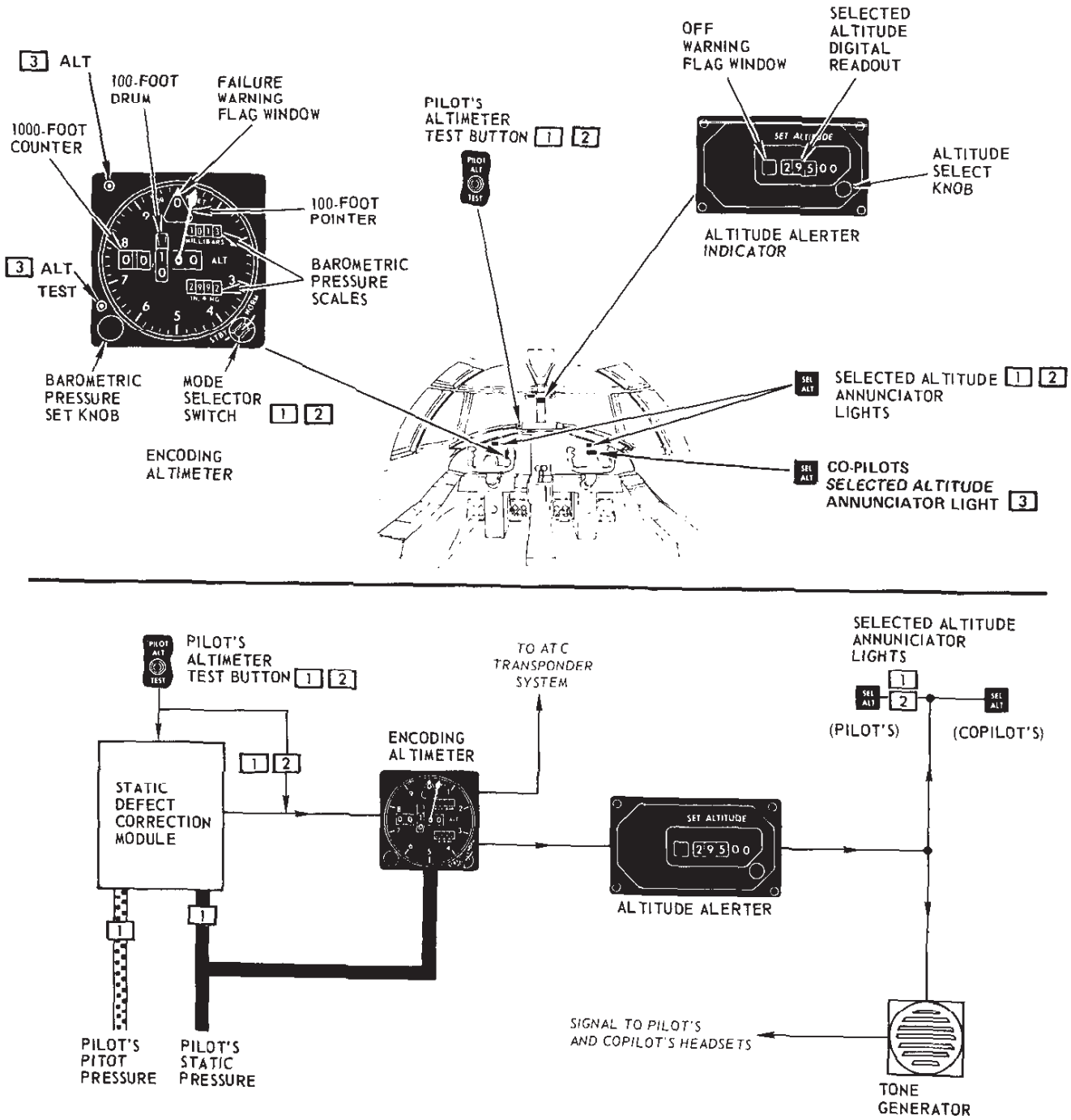
2.17.8 Copilot Altimeter. The copilot altimeter (72, FO-1) has 100- and 1,000-foot pointers plus a 10,000-foot pointer. The 10,000-foot pointer serves a second function as a warning indicator by exposing a striped section on the instrument face at altitudes below 16,000 feet. This type of altimeter offers improved readability and gives visual warning when an altitude of less than 16,000 feet is entered.

2.17.9 Encoding Altimeter. An encoding altimeter (Figure 2-38 and 72, FO-1), mounted on the pilot instrument panel, displays corrected pressure altitude and provides encoded altitude data (in increments of 100 feet) to the ATC transponder system. (Refer to paragraph 2.3, COMMUNICATION AND ASSOCIATED ELECTRONIC EQUIPMENT.) The altimeter also provides the altitude input to the altitude alerter indicator. The altimeter is a servo-type instrument. On CT-39E, CT-39G aircraft 158843, 158844, and 159361 through 159365, the altimeter operates in two modes of operation as selected by the STBY/NORM selector switch on the altimeter. On CT-39E aircraft and CT-39G aircraft 158843 and 158844 in the NORM position, a static defect correction module provides a correction input in the form of an electrical signal to the altimeter to compensate for static-pressure source position error at the mounting location. Additionally, internal trimming potentiometers are employed to remove altimeter scale errors determined prior to installation or during routine altimeter calibrations. On CT-39G aircraft 159361 through 159365 and 160053 through 160057, aerodynamically compensated pitot-static tubes have been installed and the static defect correction module is not required. On CT-39E, CT-39G aircraft 158843, 158844, and 159361 through 159365 in the NORM position, the pitot-static tubes are compensated for static pressure position error at the mounting location and in the STBY position, the altimeter provides uncorrected indications and output signals and when operating with the switch at the NORM position, the standard altimeter correction (Figure 22-1) is not used. The barometric pressure scales (inches of mercury and millibars) are set, simultaneously, with the knob on the lower left corner of the altimeter.

The altitude indication is presented on a counter, drum, and pointer display, covering a range of -1,000 to +50,000 feet. The first two digits (thousands of feet) are presented by a digital counter-type display. The third digit (hundreds of feet) is presented by a drum-type display. The last two digits (00) are fixed. Thus, the counterdrum display provides indications to the nearest hundred feet. The pointer also indicates altitude in hundreds of feet with a full revolution equivalent to 1,000 feet. The pointer display enables a more precise reading than the drum display since the scale is graduated in increments of 20 feet.

If, during operation, the static defect correction module (CT-39E aircraft and CT-39G aircraft 158843 and 158844) or altimeter fails or if a power failure occurs, an orange failure warning flag appears, in place of the "0," at the top of the altimeter dial. The altimeter indication should not be relied upon while the warning flag is in view: at such time, encoded altitude data are not fed to the ATC transponder or to the altitude alerter indicator. If the flag disappears when the mode selector switch is placed at STBY, the static defect correction module is defective. In this condition, the flag disappears (flag remains armed) and the altimeter provides uncorrected altitude data to the ATC transponder system and to the altitude alerter indicator. On CT-39E aircraft and CT-39G aircraft 158843 and 158844, the encoding altimeter is powered through the static defect correction module by the 115-vac secondary bus. The static defect correction module provides the 26-vac required for altimeter operation. On CT-39G aircraft 159361 through 159365, the altimeter is powered by the aircraft No. 2, 26-vac primary bus. On CT-39G aircraft 160053 through 160057, the altimeter is powered by the aircraft 26-vac primary bus.

2.17.9.1 Altimeter Test Button. The PILOT ALT TEST button (15A, FO-1) on the left side of the instrument panel shroud (CT-39E, CT-39G aircraft 158843, 158844, and 159361 through 159365) is a momentary pushbutton switch used to test the encoding altimeter. On CT-39G aircraft 160053 through 160057, the test switch is located on the encoding altimeter just above the barometric set knob. When the test button is depressed and held, an error signal is developed in the static defect correction module (CT-39E aircraft and CT-39G aircraft 158843 and 158844) and, in turn, the altimeter failure warning flag appears. The error signal drives the altimeter to 850 ±100 feet below the original indicated altitude. The failure warning flag may or may



1 CT-39E Aircraft and CT-39G Aircraft 158843 and 158844.
2 CT-39G Aircraft 159361 through 159365
3 CT-39G Aircraft 160053 through 160057

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39E-1-51-11A

Figure 2-38. Encoding Altimeter (Automatic Altitude Reporting) and Altitude Alerter

not disappear from view at this point, which is normal during self-test. When the test button is released, the altimeter returns to the original indication and, if in view, the failure warning flag disappears.

On aircraft 159361 through 159363, the button is located on the front left side. When depressed, a false signal will drive the altimeter to a minimum of 400 feet below the original indicated altitude. The following steps are recommended in order to check both the altimeter and the altitude alerter:

1. Set the altitude alerter to the 100-foot interval nearest to the altimeter reading and depress the press-to-test button.

2. The altimeter and alerter off flags appear.

3. The altimeter pointer descends a minimum of 400 feet, thereby activating the inner warning band circuit of the alerter. The altimeter alert lamp lights and the alerter audible signal sounds.

4. Release the press-to-test button. The altimeter returns to its correct reading, the alert light goes out, and the alerter and altimeter off flags drop out of view.

Performance of steps 1 through 4 above verifies operation and accuracy of the altimeter servo and the alerter circuits. It also demonstrates that the warning lights, audible signal, and failure flags of both units are functioning correctly.

2.17.10 Altitude Alerter Indicator. An altitude alerter indicator (Figure 2-38 and 32, FO-3) is mounted on the center windshield divider. The indicator receives input data from the encoding altimeter and controls a tone generator and annunciator lights that provide aural and visual alerting signals when approaching or deviating from a preselected altitude. When the tone generator operates, the tone is also audible in the headsets. A knob at the lower right corner of the indicator is used to preselect the desired altitude. The preselected altitude (0 to 50,000 feet) is presented on a digital counter-type display with the last two digits (00) fixed.

During climbout, when the aircraft is 1,000 ±50 feet below the selected altitude, the tone generator operates for 2 seconds and the pilot and copilot SEL ALT annunciator light comes on. When the aircraft is within 300 ±50 feet of the selected altitude, the SEL ALT lights go off. As long as the aircraft is maintained within a band 300 ±50 feet below or above the preselected altitude, the SEL ALT lights and tone generator remain off. Anytime the aircraft altitude deviates from this band, the tone generator operates for 2 seconds and the SEL ALT lights come on. The altitude alerter is powered by the No. 2 26-vac bus. If there is a loss of power, an OFF flag appears in the extreme left window of the indicator.

2.17.11 Turn-and-Slip Indicators (CT-39E Aircraft). The turn-and-slip indicators (13 and 52, FO-1) are electrically powered by the dc essential bus. A power indicator window on the instrument face indicates whether power is on or off. One standard needle-width deflection will accomplish a 360° turn in 4 minutes (1-1/2° per second rate of turn). Normally, the indicator is not used in banks exceeding 30°. On CT-39G aircraft, the turn-and-slip indicators are an integral part of the flight director indicators.

2.17.12 Accelerometer. The accelerometer (88, FO-1) has, in addition to the indicating pointer, two recording pointers (one for positive g loads, and one for negative g loads) that follow the indicating pointer to its maximum attained travel and remain at the maximum travel positions reached by the indicating pointer. Pressing the knob on the instrument ring returns the recording pointers to the normal (1g) position.

2.17.13 Magnetic Compass. A conventional magnetic compass (29, FO-3), suspended from the center windshield divider, is furnished for navigation in case of instrument or electrical system failure. A compass correction card is on the center windshield divider just below the compass.

Note

Windshield heat must be secured to receive reliable information from the magnetic compass.

2.17.14 Free Air Temperature Indicator. A free air temperature indicator (32, FO-1), mounted on the pilot instrument panel, is powered by the dc essential bus. The indicator needle is connected to the thermometer resistance bulb, so that changes in outside air temperature register on the indicator face by means of changes in electrical current between the bulb and indicator. The indicator is calibrated in °C.

Note

The free air temperature bulb is on the underside of the fuselage. Because of errors induced by reflection from the runway, the free air temperature indicator should not be used for EPR settings on the ground.

2.17.15 Flight Hour Meter (CT-39G Aircraft 159362 Through 159365 and 160053 Through 160057). An aircraft hour meter is mounted on the copilot vertical trim panel just above the console circuit breaker panel. The meter is powered by the dc essential bus and records aircraft flight time in hours and tenths of an hour. The meter is operative when either dc generator is functioning and the weight of the aircraft is removed from the landing gear.

2.18 INTEGRATED FLIGHT SYSTEM (CT-39E AIRCRAFT)

The integrated flight system consists of the FD-108 flight director system combined with the AP-103F autopilot. The system provides the pilot automatic control of heading, attitude, altitude, VOR/ILS/tacan radial interception, and tracking with automatic crosswind correction. The flight director indicator presents a three-dimensional, parallax-free presentation to ensure safe aircraft control when the autopilot is not used.

Two complete and independent sets of components are provided, one set each for the pilot and copilot, including the flight director indicator, course indicator, instrument amplifier, a flight steering computer, vertical and directional gyros, and inputs from the VOR/ILS, tacan, marker beacon receivers. A switching package in the system (course select switch) controls selection of navigational signals to the appropriate indicators. Depending on the operating mode selected, the flight computers supply steering and attitude commands to the flight director and course indicators. The pilot flight director system operating mode is controlled from the controller located on the center pedestal (27, Figure 2-39). The copilot flight director system operating mode is controlled by the mode selector switch on the lower right-hand edge of his flight director indicator. The mode selector switch on the pilot flight director indicator is inoperative.

The pilot has the option of coupling his flight director/course indicator presentation to the autopilot system or using the flight director system independently of the autopilot.

2.18.1 Integrated Flight System Controls and Indicators

2.18.1.1 Flight Director Indicator. The flight director indicators (Figure 2-40) on the pilot and copilot instrument panels provide a three-dimensional presentation of aircraft attitude combined with vertical and lateral steering commands from the command or V-bars. The indicators are identical and function the same except that the presentation on the pilot indicator can be coupled to the autopilot and the pilot mode selector switch is inoperative. If the autopilot is engaged, the pilot integrated flight system is in the same mode as the autopilot. Each flight director receives attitude information from its own vertical and directional gyros.

The fixed delta-shaped symbol represents the aircraft. Pitch and bank (roll) are displayed by the relationship of the aircraft symbol to the movable attitude tape background. The background is colored with a blue sky above and black ground beneath with a white horizon line. A 360° roll presentation is possible. A scale on the same background displays pitch attitude from zero to ±90°. A bank pointer moves in the direction in which the aircraft is rolled to show the amount of bank angle indicated on a semicircular scale, indexed at 10°, 20°, 30°, and 60° each side of center on the upper half of the instrument. Bank is also indicated by the relationship of the aircraft symbol to the horizon line.

Tapered V-command bars that move across the indicator are servodriven by signals supplied by the flight computer. The V-bars supply autopilot command inputs when the autopilot system is engaged. The bars roll clockwise or counterclockwise to command a left or right bank to maintain a selected heading or course, and the bars move up or down to indicate pitch correction for proper altitude and attitude and to intercept and maintain the correct glideslope position. Thus, the effects of airspeed, distance, crosswind, and approach angle are resolved into computed commands and the aircraft is maneuvered so that the aircraft symbol is "flown" into the V-bar until two are aligned to satisfy these commands.

Commands generated in the flight computer for input to the flight director indicator are determined by the position of the mode selector switch. The switch has four positions: GYRO, HEADING (pilot switch) or HDG (copilot switch), NAV/LOC (pilot switch) or N/L (copilot switch) and APPR.

Note

The autopilot power switch must be on, the pilot mode selector switch must be at GYRO or HEADING, and the turn knob must be in the center detent before the autopilot can be engaged.

The GYRO position can be used in approaching a VOR or tacan station or when maneuvering is required while over the station and before tracking the outbound radial. It is also used in operations where frequent heading changes are required, such as orbiting a station, to

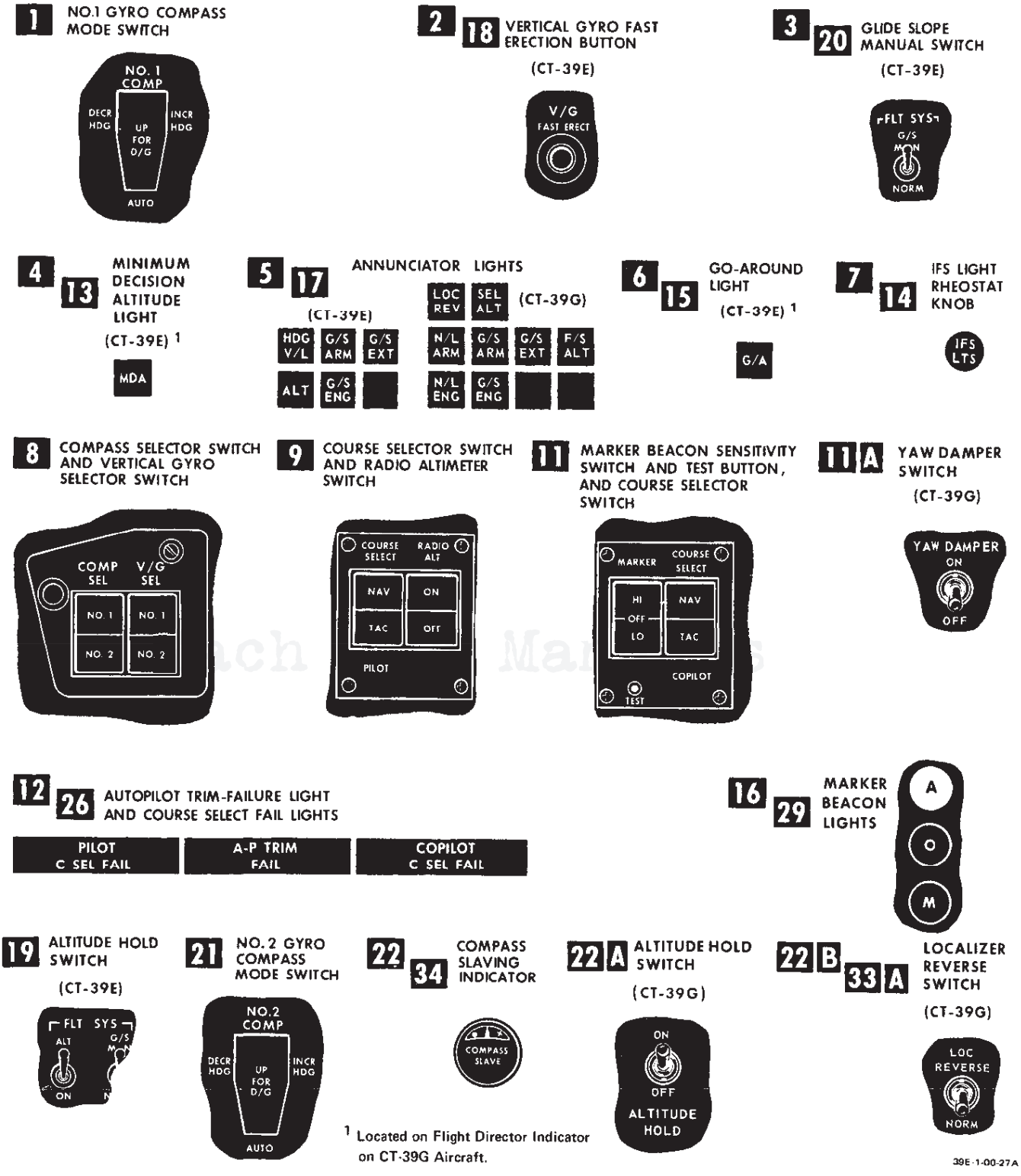
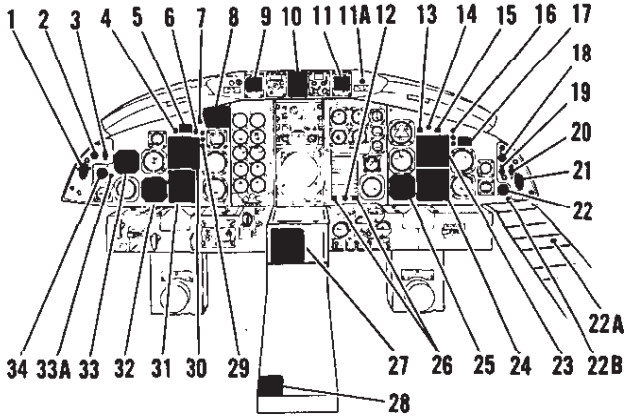
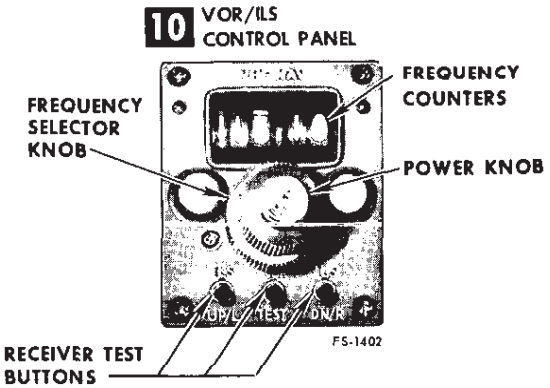
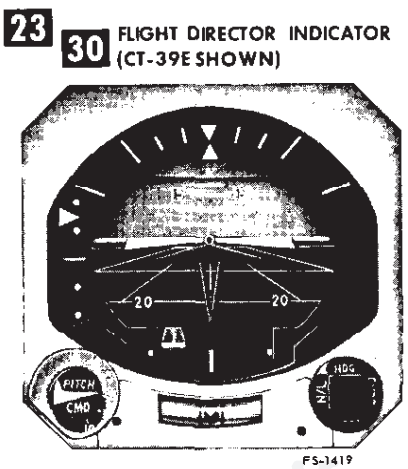
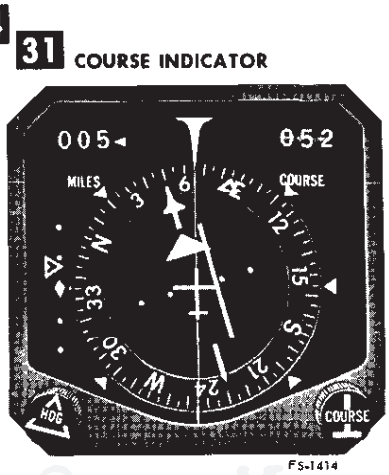


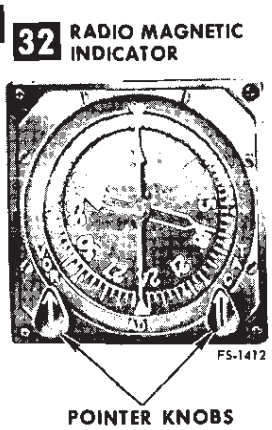
Figure 2-39. Integrated Flight and Autopilot System Controls and Indicators (Sheet 1 of 2)











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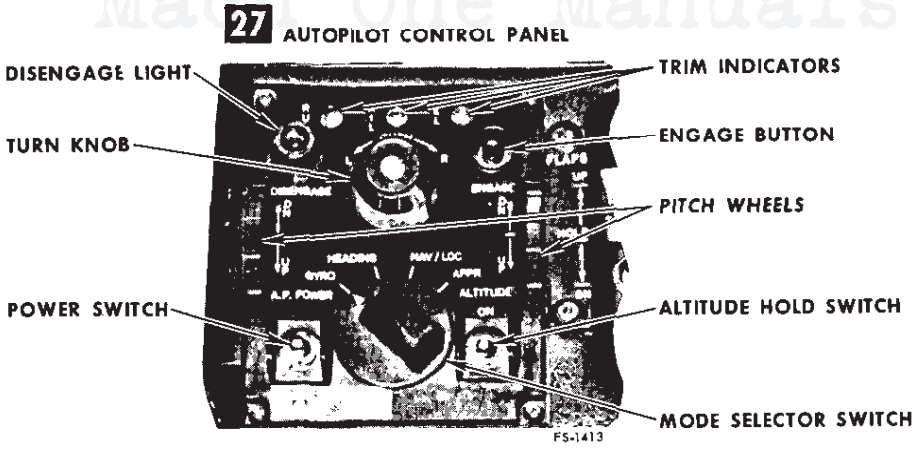

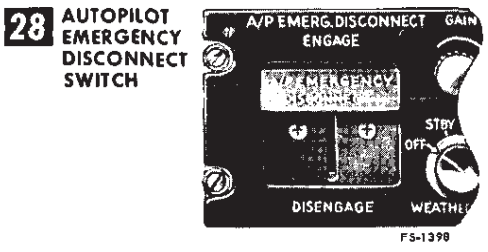

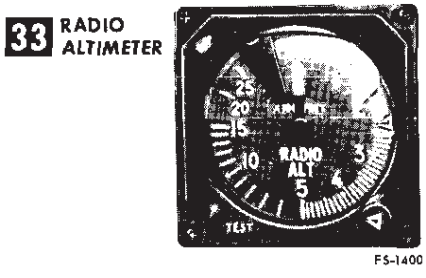

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Figure 2-39. Integrated Flight and Autopilot System Controls and Indicators (Sheet 2 of 2)

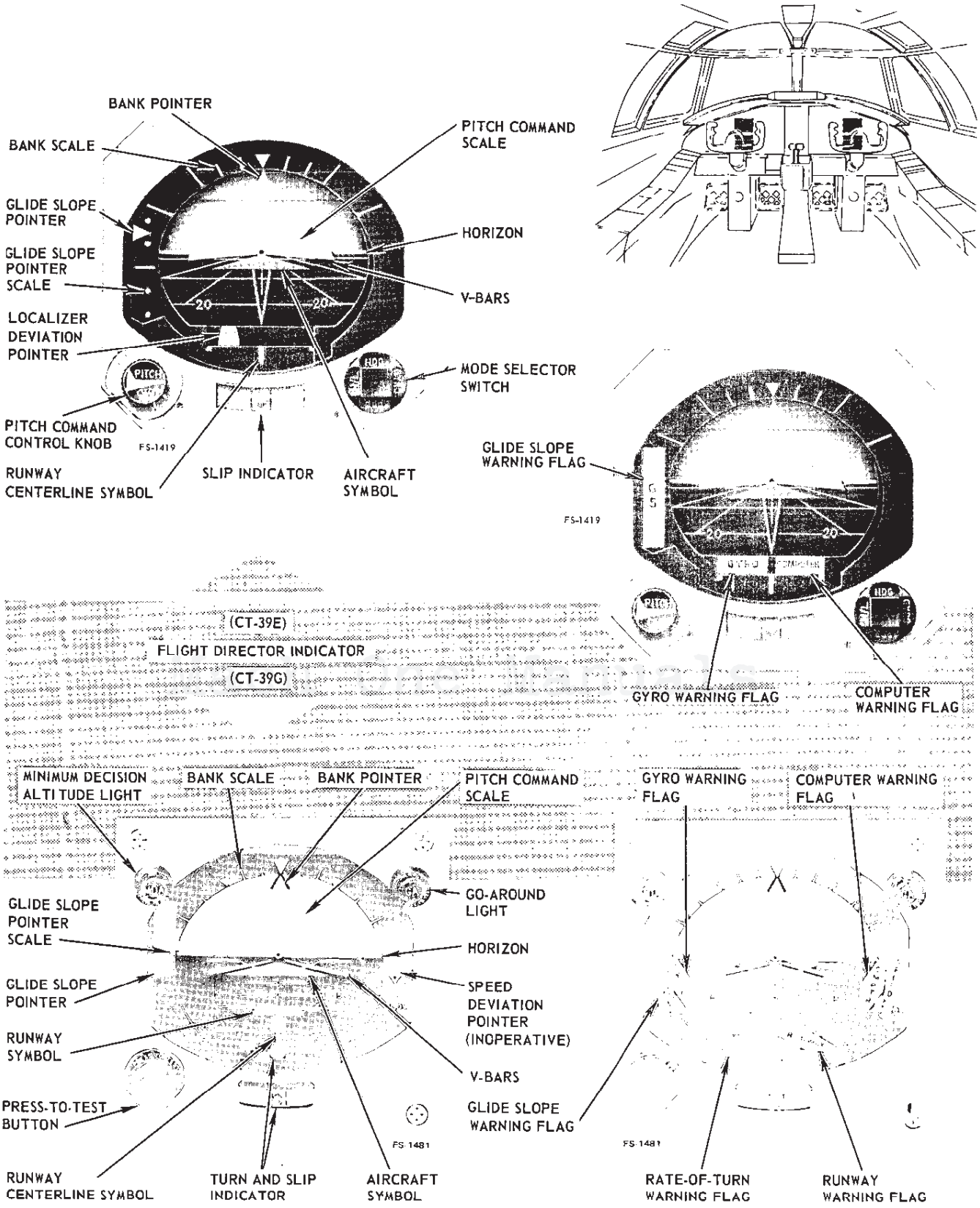

Figure 2-40. Flight Director and Course Indicators (Sheet 1 of 2)

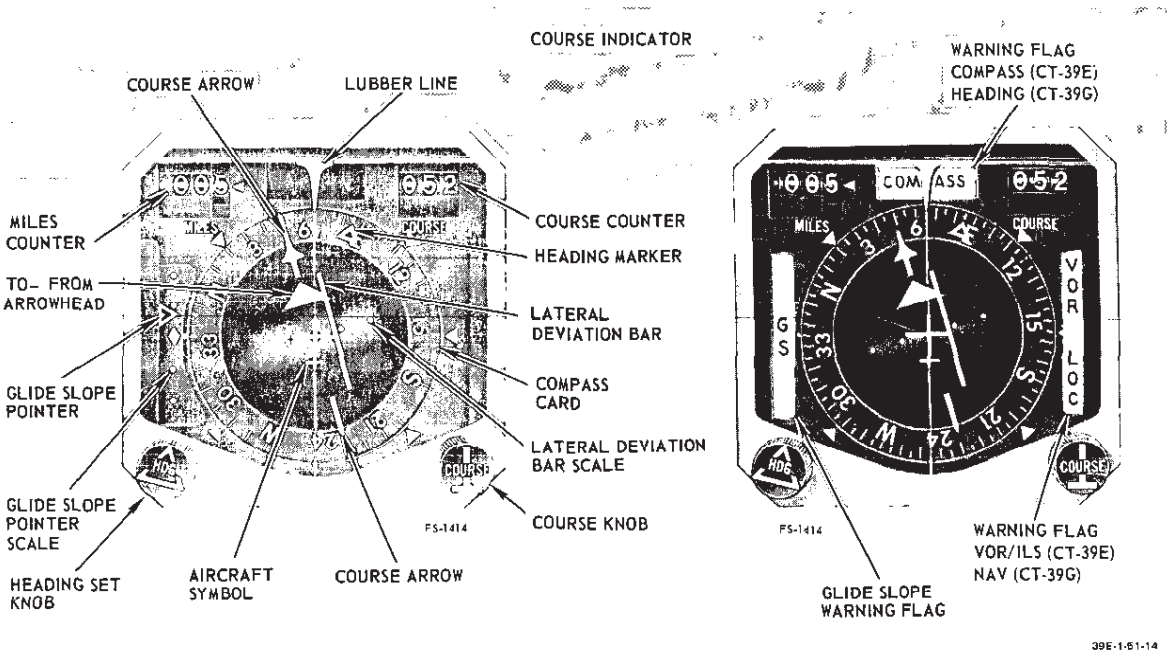

Figure 2-40. Flight Director and Course Indicators (Sheet 2 of 2)

eliminate the need to adjust the course indicator heading pointer for small corrections. In GYRO, the V-bars are not in view and the basic gyro attitude is displayed on the flight director indicator. In this mode, the annunciator lights do not function. In the HEADING mode, the V-bars command to steer the aircraft to the heading selected by the course indicator heading set knob and the bank angle called for decreases gradually as heading error decreases. In the HEADING mode with the autopilot engaged, the aircraft flies to obtain the heading selected by the pilot course indicator heading set knob. With the mode selector switch at NAV/LOC or N/L (navigation/ localizer), bearing information from the VOR or tacan receivers is presented on the flight director indicator and the course indicator. If the NAV/LOC (N/L) position is selected before radial interception, the HDG V/L light will illuminate and the V-bars will maintain the heading selected by the heading set knob. After radial capture, the light will go out and the flight director indicator V-bars will then indicate the direction to turn to maintain the selected radial or localizer.

The APPR position of the mode selector switch provides for automatic capture of the glideslope beam when intercepted for an ILS approach. In this mode, the system is in the armed condition (as indicated by the illumination of the GS ARM annunciator light) before glideslope capture, and in the engaged condition (as indicated by the illumination of the GS ENG annunciator light) after glideslope capture.

A pitch command control knob on the lower left edge of the instrument is used for adjusting the V-bars for a fixed angle of climb or descent when operating in the HDG, NAV/LOC, APPR modes and the autopilot is not engaged. The pitch command control is deactivated in the approach mode after glideslope capture to enable ILS tracking. The pitch command is variable from -10° to +15° in 5° increments. An adjustment scale is adjacent to the knob.

A glideslope pointer, representing the center of the glideslope beam, shows vertical displacement of the aircraft from the beam center. The pointer is seen only when the VOR-ILS receiver is tuned to a localizer frequency. The centerline of the glideslope pointer scale represents the aircraft position with respect to the glidepath. If the pointer is displaced upward, the aircraft is below the glidepath and vice versa.

A localizer deviation pointer below the indicator pictorial display moves to either side of the runway centerline symbol to show lateral displacement from the center of a localizer beam. It represents an expanded portion of the lateral deviation bar on the course indicator; the reference dots that bracket the localizer deviation pointer runway symbol are equivalent to the innermost dots on the course indicator (1-1/4° course deflection). The localizer deviation pointer is seen only in the approach mode when the navigation receiver is tuned to a localizer frequency and after glideslope capture.

A localizer deviation is driven vertically by an input from the radio altimeter during the last 200 feet of descent. The runway symbol just meets the aircraft symbol when the aircraft is at ground level.

There is no turn indicator on the flight director indicator. However, on these aircraft, separate pilot and copilot turn-and-slip indicators are provided.

The MDA and GA annunciator lights are on the pilot and copilot instrument panels. Refer to paragraph 2.18.1.15, Annunciator Lights, for information related to the MDA and GA lights.

The pilot flight director is powered by the dc essential bus and the ac primary bus. The copilot flight director is powered by the ac secondary bus.

2.18.1.2 Flight Director Indicator Warning Flags. The flight director indicator with power off and all (red) warning flags in view is shown in Figure 2-40. Limited system operation is possible with some of the flags in view. If any flag appears during the normal operation sequence, a check should be made to determine whether the apparent malfunction can be corrected by resetting circuit breakers, recycling an associated receiver, slaving an input gyro, or selecting the alternate position on the vertical gyro selector switch. The following paragraphs describe what causes a flag to appear.

1. The GYRO flag indicates failure of a vertical gyro, attitude display servo system, or power that drives the attitude display.

2. The COMPUTER flag indicates failure of the flight computer and V-bar display.

3. The GS (glideslope) flag indicates a malfunction of the VOR/ILS receiver or a weak or unreliable glideslope signal.

2.18.1.3 Course Indicator. The multipurpose indicator (Figure 2-40) on the pilot and copilot instrument panels presents course, heading, miles, and directional guidance in the various flight director modes. The pilot and copilot course indicators are identical except that the pilot indicator is also capable of presentations in the autopilot mode. Navigation data from the VOR and tacan receivers is used to present a symbolic display of the horizontal navigation situation.

A miniature aircraft symbol etched on the instrument glass shows aircraft position and heading with respect to the desired course. A rotating compass card, remotely driven by signals from the directional indicating system, displays aircraft magnetic heading against a lubber line. A triangular-shaped heading marker can be set to the desired magnetic heading around the outer edge of the compass card by the heading set knob. Clockwise knob rotation moves the heading marker clockwise. Once set, the marker will rotate with the compass card.

Desired course or track is shown by a course arrow that rotates within the compass card. The course arrow is manually set to a magnetic course that coincides with the desired VOR or tacan radial or localizer course by turning the course knob. Clockwise rotation moves the arrow clockwise. A three-digit course counter reflects the magnetic course to which the course arrow is pointing. Once set, the course arrow will rotate with the compass card. The course arrow has a segmented section called a lateral deviation bar that gives a plan view of the radial position of the desired tacan, VOR, or localizer course set on the course knob relative to the aircraft. The bar represents the centerline of the desired course. The aircraft symbol then simulates the aircraft position with respect to the desired course. A four-dot scale brackets the lateral deviation bar, two dots on each side of the centerline. In the navigation/localizer mode, a two-dot deflection of the bar represents a 10° course deviation. In the approach mode, a two-dot deflection represents a 2-1/2° course deflection. The localizer deviation pointer on the flight director indicator indicates a maximum one-dot deflection of 1-1/4° in the localizer mode.

The integrated flight system has the capability of intercepting the localizer back course. However, to obtain directional indications from the lateral deviation bar, the front approach course has to be set in the course indicator. Then, when the lateral deviation bar is offset, the aircraft has to be steered in the direction the bar is offset to center the bar and intercept the course.

A TO-FROM arrowhead indicates in the navigation/localizer mode (VOR-tacan only) whether the selected course is inbound (TO-FROM arrowhead pointing at head of course arrow) or outbound (TO-FROM arrowhead pointing at tail of course arrow) with respect to a VOR or tacan station. The TO-FROM arrowhead does not appear when turned to a localizer frequency. Range to a tacan station is shown on the miles counter. The three-digit counter covers 000 to 299 nautical miles.

A glideslope pointer, representing the center of the glideslope beam, shows vertical displacement of the aircraft from the beam center. This pointer, a repeat of the glideslope pointer on the flight director indicator, is effective only when the VOR/ILS receiver is tuned

to a localizer frequency. The center dot of the glideslope pointer scale represents the aircraft position. If the pointer is displaced upward, the aircraft is below the glidepath, and vice versa.

The course indicator is powered by the dc essential bus and the ac primary bus. The copilot course indicator is powered by the ac secondary bus.

2.18.1.4 Course Indicator Warning Flags. The course indicator with power off and all (red) warning flags in view is shown in Figure 2-40. Limited flight director system operation is possible with some of the flags in view. If any flag appears during the normal operation sequence, a check should be made to determine whether the apparent malfunction can be corrected by resetting circuit breakers, recycling the associated receiver, slaving an input gyro, or selecting the alternate position on the compass selector switch.

1. The GS flag indicates a malfunction of the VOR/ILS receiver or a weak or unreliable glideslope signal. The glideslope flag covers the glideslope pointer and scale to prevent accidental acceptance of faulty indications.

2. The VOR/LOC flag indicates a malfunction of the VOR/ILS receiver or reception of an unreliable signal.

3. The COMPASS flag indicates failure of a directional gyro.

2.18.1.5 IFS Light Rheostat Knob. This knob (7 and 14, Figure 2-39) on the pilot and copilot instrument panels is labeled IFS LTS and controls the light intensity of the flight director and course indicators. Turning the knob clockwise increases light intensity. The rheostat is powered by the ac secondary bus.

2.18.1.6 Radio Magnetic Indicator. This indicator (25 and 32, Figure 2-39) on the pilot and copilot instrument panels receives heading information from the directional indicating system. This is reflected on a rotating compass card that provides a magnetic heading displayed against a fixed reference marker at the 12 o'clock position on the dial. Signals from the ADF system and VOR and tacan receivers are fed into the instrument to drive two pointers, a single- and double-barred pointer. The indicator is labeled VOR on the left, TACAN on the right, and ADF on the lower center. A knob on the left side of the indicator has a single-barred arrow and a knob on the right side of the indicator has a double-barred arrow. When the left knob is turned to VOR or ADF, the single-barred pointer on the indicator reflects the bearing of the corresponding signal. Turning the right knob to TACAN or ADF causes the double-barred pointer on the indicator to reflect the corresponding signal bearing. If both knobs are turned to ADF, the pointers are servoed together and move as one to indicate ADF bearing. The pilot radio magnetic indicator is powered by the ac primary bus. The copilot radio magnetic indicator is powered by the ac secondary bus.

2.18.1.7 Glideslope Manual Switch. Spring-loaded to the NORM position, this switch (3 and 20, Figure 2-39) on the pilot and copilot instrument panels serves a twofold purpose in the approach mode. The first is backup in the event automatic glideslope capture malfunctions. If the glideslope pointer passes downward through the center of the scale and the GS ENG annunciator light does not come on to indicate automatic capture, the glideslope manual switch should be moved to the MAN (manual) position. This couples the V-bars on the flight director indicator to the glideslope for pitch commands. Secondly, the switch can be moved to MAN if it is desired to intercept the glideslope from above as the automatic capture functions only when intercepting from below. The switch is powered by the dc essential bus and the ac primary bus.

2.18.1.8 Go-Around Button and Light. A go-around button (Figure 2-29) on each control wheel, labeled G/A, is used in the approach mode when the radio altimeter MDA light comes on for a missed approach. Pressing the button turns on a light that reads G/A and causes the flight director indicator V-bars to command a wings-level, fixed pitch climbout. The lights are on the pilot and copilot instrument panels (6 and 15, Figure 2-39). Pressing the button on the pilot control wheel also turns off the autopilot. The G/A button and light are powered by the dc essential bus.

2.18.1.9 Altitude Hold Switch. The copilot altitude hold switch (19, Figure 2-39) is on the copilot instrument panel. The pilot altitude hold switch (27, Figure 2-39) on the autopilot control panel is channel-guarded to prevent inadvertent actuation. (Refer to paragraph 2.18.2, Autopilot System). The switch references pitch commands to barometric altitude so that the barometric altitude existing at the time of switch engagement will be maintained. The switch is engaged when it is moved from OFF to ON, and an annunciator light on the instrument panel comes on to read ALT. The switch should be moved to ON in the heading or navigation localizer mode or in the approach mode up to glideslope capture. With the altitude hold switch ON, the flight director indicator V-bars command a climb or descent as necessary to maintain the barometric altitude indicated on the altimeter at the time the switch is engaged to ON. The switch automatically returns to OFF in the approach mode after glideslope capture. The switch will likewise

return to OFF if the autopilot system is engaged or disengaged or when operating on autopilot when the autopilot pitch wheels are moved. The altitude hold switch is powered by the dc essential bus and the ac primary bus.

2.18.1.10 Course Selector Switch. The rocker-type course selector switch (9 and 11, Figure 2-39) on the left and right instrument panel shroud has NAV and TAC positions. When the NAV portion of the switch is pressed, VOR signals received by the VOR/ILS receiver are connected to the flight director, course, and radio magnetic indicators. With the TAC portion of the switch engaged, tacan signals are connected to the course and radio magnetic indicators. The switch is powered by the dc essential bus.

2.18.1.11 Course Select Fail Lights. Two amber, placard-type caution lights (26, Figure 2-39), one labeled PILOT C SEL FAIL and one COPILOT C SEL FAIL, are on the caution and warning light panel. If either light illuminates, a failure is indicated in the respective course select system and the associated flight director and course indicator should be considered unreliable.

2.18.1.12 Copilot Course Resolver Fail Light. An amber, placard-type caution light (12, Figure 2-33), labeled COPILOT C RSLVR INOP, is located on the caution and warning light panel. This light will illuminate if both pilots select the same NAVAID (either TAC or NAV) with their course select switch when there is a failure in the switching package. In the event of a relay failure in the switching package, the copilot flight director and course indicator should be considered unreliable. If the light is caused by both pilots selecting either NAV or TAC, the copilot can still monitor course deviation and to/from information as selected on the pilot course indicator if he selects a course that corresponds to the course set by the pilot.

2.18.1.13 Vertical Gyro Select Switch. This rocker-type switch (8, Figure 2-39), on the pilot instrument panel is labeled V/G SEL. The switch is used in selecting the vertical gyros, both of which are the source of pitch and roll information for the integrated flight system. When the NO. 1 position of the switch is pressed, the pilot systems are operating from the No. 1 vertical gyro and the copilot systems are operating from the No. 2 vertical gyro. When the NO. 2 position of the switch is pressed, the pilot and copilot systems are both operating from the No. 2 vertical gyro. If a vertical gyro failure is indicated by the appearance of the GYRO warning flag on the flight director indicator, the selector switch should be moved to the alternate position to select the operable vertical gyro. The vertical gyro selector switch is powered by the ac primary bus.

2.18.1.14 Vertical Gyro Fast-Erection Button. Pressing the button labeled V/G FAST ERECT (2 and 18, Figure 2-39) on the pilot and copilot instrument panels permits fast erection of the operating vertical gyro during flight. The button must be held until the gyro is erected, but should not be held longer than about 2 minutes without a 2-minute cooling period. The fast erection button should not be used until the aircraft is in a wings-level, unaccelerated condition. The GYRO flag on the flight director indicator may or may not appear momentarily during fast erection. The button is powered by the dc essential bus.

2.18.1.15 Annunciator Lights. The annunciator lights (5 and 17, Figure 2-39) on the pilot and copilot instrument panels are labeled as follows to indicate the corresponding operating functions and modes:

1. HDG V/L — Comes on (amber) when the flight director mode selector switch is at NAV/LOC (N/L) while in the navigation/localizer mode. Goes out upon navigation or localizer capture.

2. ALT — Comes on (green) when the altitude hold switch is moved to ON.

3. GS ARM — Comes on (amber) when the flight director mode selector switch is at APPR before glideslope capture. Goes out when the GS ENG light comes on.

4. GS ENG — Comes on (green) when the flight director mode selector switch is at APPR after glideslope capture.

5. G/S EXT — Comes on (green) when passing over the middle marker beacon (M marker beacon light on) to indicate the steering computer has switched to the glideslope extension phase. This eliminates undesirable glideslope noise.

6. MDA — Comes on (amber) when aircraft has descended to a preset radio altimeter decision height. At this point, a land or go-around decision must be made.

7. GA — Comes on (green) when the go-around button is depressed while operating in the approach mode.

The annunciator lights are powered by the essential bus and the ac secondary bus.

2.18.2 Autopilot System. The AP-103F autopilot system will hold the aircraft on any preselected course, maintain level flight, climb, or dive, and change course

in coordinated turns. The autopilot can be used in conjunction with the pilot integrated flight system for automatic control of heading, altitude and attitude, and acquisition of a selected magnetic heading or VOR localizer course and glideslope with crosswind corrections. The system includes rate and vertical gyros, an altitude hold controller, a three-channel (aileron, elevator, and rudder) amplifier, airspeed and rate sensors, and a primary control and trim coupler that positions the aircraft control surfaces. Airspeed and altitude sensing are received through the aircraft pitot-static system. A control panel in the cockpit center console contains the autopilot controls and indicators as well as the pilot integrated flight system-flight director mode selector switch. Autopilot control can be manually overridden at anytime. The autopilot system is powered by the dc essential bus, the ac primary bus, and the ac secondary bus.

2.18.2.1 Autopilot System Controls and Indicators

2.18.2.1.1 Emergency Disconnect Switch. This switch (28, Figure 2-39) on the center console has a handle labeled A/P EMERGENCY DISCONNECT. Moving the switch up to ENGAGE permits engagement of the autopilot system when the autopilot power switch is at ON. Moving the switch down to DISENGAGE electrically disconnects the autopilot engaging and servo power units and turns on a disengage indicator light. The emergency disconnect switch should not be used to turn off the autopilot in normal operation. The switch should be maintained at ENGAGE at all times except in an emergency when a malfunction occurs in the normal disengage circuits or in the autopilot disengage buttons, in which case, the emergency disconnect should be moved to DISENGAGE.

2.18.2.1.2 Power Switch. The autopilot power switch, on the autopilot control panel (27, Figure 2-39) is channel-guarded to prevent inadvertent switch actuation. Moving the switch up to A.P. POWER applies electrical power to all system components if the autopilot emergency disconnect switch is at ENGAGE.

2.18.2.1.3 Engage Button. The autopilot engage button on the autopilot control panel (27, Figure 2-39) is used for engaging the autopilot system. The autopilot can be engaged by pressing the button if the following conditions are met:

1. Power switch at A.P. POWER
2. Emergency disconnect switch at ENGAGE
3. Turn knob at center detent
4. Mode selector switch at GYRO or HEADING
5. Pilot flight director indicator GYRO and COMPUTER flags hidden.

Pressing the engage button turns off a disengage indicator light.

2.18.2.1.4 Disengage Buttons. Pressing the disengage button (Figure 2-29) labeled AUTOPILOT DISC on either control wheel disengages the autopilot and turns on the autopilot disengage light.

2.18.2.1.5 Disengage Light. This amber, dimmable, press-to-test light on the autopilot control panel (27, Figure 2-39) is labeled DISENGAGE. The light comes on if the autopilot power switch is on when the autopilot is disengaged by any means.

2.18.2.1.6 Go-Around Button. Pressing the G/A button (Figure 2-29) on the pilot control wheel shuts off the autopilot in addition to its function in the integrated flight system. For additional information on this switch and light, refer to paragraph 2.18.1.8, Go-Around Button and Light.

2.18.2.1.7 Turn Knob. The turn knob on the autopilot control panel (27, Figure 2-39) is used for commanding turns when the autopilot flight director system is in the GYRO mode. Labeled TURN L R, the knob is rotated counterclockwise for a left turn and clockwise for a right turn. Any degree of bank can be selected from a maximum of 25° in the heading and navigation/localizer modes to maximum of 15° in the approach mode. The aircraft will roll out in a coordinated turn and hold heading when the turn knob is centered at the detent position (white index line on the knob at 12 o'clock). The knob has to be at the center detent before the autopilot system can be engaged.

2.18.2.1.8 Pitch Wheels. Two pitch wheels on the autopilot control panel (27, Figure 2-39) are connected by a common shaft to control climb or dive in the autopilot mode. Each wheel has UP and DN positions and indicating arrows to each position. By rotating either wheel in the desired direction, selection can be made to climb 12° maximum, or in the go-around mode, 10° fixed pitchup. The pitchdown (dive) maximum is 2.2°. The autopilot may be engaged with the pitch wheels in any position and the aircraft will maintain the same pitch attitude existing at the time of engagement. The pitch wheels are inoperative in the approach mode (mode selector switch at APPR) after glideslope capture. If engaged, the altitude hold function is disconnected when the pitch wheels are moved.

2.18.2.1.9 Trim Indicators. Three indicators, on the autopilot control panel (27, Figure 2-39) are labeled RUD, AIL, and EL, on each for the rudder ailerons, and elevators. The rudder indicator is as viewed from aft to forward. Before autopilot engagement, the indicators show (in deflection) the direction and magnitude of torque that will be delivered by the autopilot servos at the instant of engagement. After autopilot engagement, the indicators show what the aircraft trim condition will be upon disengagement of the autopilot.

2.18.2.1.10 Trim Failure Light. This amber, placard-type light on the caution-warning light panel (12, Figure 2-39) comes on to read AP TRIM FAIL when the autopilot is engaged and automatic trimming becomes erratic or inoperative. Retrimming can be accomplished by first checking the horizontal stabilizer trim indicator for unusually large trim and manually trimming out control forces if necessary then disengaging the autopilot, manually retrimming aircraft, and reengaging the autopilot.

2.18.2.1.11 Pitot-Static Shutoff Switches. These switches (FO-2) on the right console control the pitot and static air supply to the autopilot system. With the A/S PITOT and A/S STATIC switches at the guarded ON position, the autopilot airspeed sensor and altitude controllers are supplied from the copilot normal pitot-static source (Figure 2-36) regardless of the position of the position of the copilot static selector switch. Raising the guard and moving the pitot shutoff or the static shutoff switch to OFF shuts off the respective system to the autopilot. These switches should be moved to OFF only when the respective pitot or static system is malfunctioning and it is suspected that the fault lies in the autopilot system.

WARNING

The autopilot must be turned to OFF when either of these switches is at OFF.

2.18.2.1.12 Altitude Hold Switch. The pilot altitude hold switch is on the autopilot control panel (27, Figure 2-39). The switch can be moved to ON to provide autopilot commands to maintain a constant barometric altitude. For additional information on this switch, refer to paragraph 2.18.1.9, Altitude Hold Switch.

2.18.2.1.13 Mode Selector Switch. The pilot mode selector switch is on the autopilot control panel (27, Figure 2-39). The autopilot may be engaged with the switch at GYRO. In this mode, turns are controlled by the turn knob and the pitch wheels may be used to command pitch corrections. The autopilot may likewise be engaged with the switch at HEADING. In this mode, the aircraft will enter a coordinated turn to a preset heading and the pitch wheels may be used to command pitch corrections. With the switch at NAV/LOC or APPR, the autopilot will track, maintain the selected headings, and move the aircraft in coordinated turns commanded by the integrated flight system. For additional information on this switch, refer to paragraph 2.18.1.1, Flight Director Indicator.

2.18.3 Operation of Integrated Flight and Autopilot Systems

2.18.3.1 Takeoff and Climbout. Check autopilot power switch on and disengage light illuminated. Set heading marker to runway magnetic heading and adjust pitch command control for desired climbout attitude (approximately 12° to 15°). When airborne and the climb is established, the autopilot may be engaged in either GYRO or HEADING mode.

If engaged in the GYRO mode, turns to the departure heading are controlled by the TURN control. The pitch control on the pedestal may be used to command attitude corrections to maintain proper climb airspeed. As level-off altitude is approached, the pitch control is adjusted to reduce the rate of climb to less than 500 fpm. The altitude switch may then be turned ON to provide autopilot commands to maintain a constant barometric altitude.

If the autopilot is engaged in the HEADING mode, the autopilot will maintain or turn to the preset heading.

2.18.3.2 VOR and Tacan. With the system operating in the HEADING mode, tune the VOR or tacan to the desired frequency. Set the course selector switch to correspond to the desired NAVAID. Set the course arrow to the desired radial and select an intercept heading of 85° or less with the heading marker on the course indicator. Change the mode selector to NAV/LOC. The HDG annunciator light will illuminate and the aircraft will maintain the selected intercept heading. As the desired radial is approached, this light will go out and the autopilot will control the aircraft in a coordinated turn to roll out on the radial.

After NAV/LOC capture, the system will provide automatic crosswind correction to track the radial. The altitude hold capability may be used to maintain constant barometric altitude during all phases of the radial intercept and capture.

If use of the autopilot is not desired, proceed as previously stated and manually fly the aircraft symbol to match the V-bars on the flight director indicator.

2.18.3.3 Holding. To establish a holding pattern over the outer marker, VOR station, or tacan fix, switch the mode selector to the HEADING mode. Turn the course arrow on the course indicator to the inboard front course or the appropriate VOR/TAC radial. The lateral deviation bar now represents the localizer or VOR/tacan radial. With the aircraft heading inbound to the holding fix, the radial is maintained by adjusting the heading marker to center the lateral deviation bar. When the holding fix is reached, turn the heading marker approximately 90° in the direction of the turn to command the aircraft into the outbound turn. As the turn progresses, continue moving the heading marker until the reciprocal heading of the inbound course is reached. If crosswind correction is required, compensate for the drift by adjusting the heading marker.

When the required time or distance has elapsed while tracking the outbound course, the inbound turn is initiated in the same manner as the outbound turn. Slight corrections may be necessary during the turn or rollout to aline the lateral deviation bar with the course arrow. Should automatic capture and tracking be desired, switch the mode selector to the NAV/LOC mode after approximately 90° of the inbound turn has been completed.

2.18.3.4 Letdown. For a typical approach and letdown to a VOR station, the autopilot, operating in the NAV/LOC mode, will track the selected radial to the station. The to-from pointer on the course indicator will reverse directly over the station. Special damping circuits allow station passage without excessive maneuvering. If the aircraft is approaching a VOR station on a course that requires maneuvering while over the station and prior to tracking the outbound radial, the maneuver may be executed in either the GYRO or HEADING mode.

When approaching the proper position to commence the procedure turn, move the heading marker under the lubber line and switch the mode selector to the HEADING mode; adjust the heading marker to maneuver through the procedure turn.

Note

For a 90° to 270° type of maneuver, do not move the heading marker past the tail of the aircraft or the flight director will reverse the direction of the turn.

If the letdown is continued during the turn, set the desired rate of descent using the pitch control. If altitude hold is desired, engage the altitude hold switch.

When the 180° phase of the procedure turn is completed and the inbound course is approached, move the course arrow to the inbound course and switch the mode selector to the NAV/LOC mode. The autopilot will capture and track the selected radial. Use the pitch control to establish the desired rate of descent. The crab angel established by the automatic crosswind correction is shown on the course indicator as the angle between the course arrow and the lubber line.

Note

Preset bank angle limits in the HEADING mode may require operation in the GYRO mode if standard rate turns are desired.

2.18.3.5 Approach. Both the localizer and glideslope are captured automatically on an ILS approach. The localizer is always captured before the glideslope and is captured from a selected heading.

If tracking a VOR radial in the NAV/LOC mode, set the heading marker to the desired intercept heading and switch the mode selector to the HEADING mode. After the VOR/ILS receiver is tuned to the localizer frequency, set the course arrow to the inbound localizer course and switch the mode selector to the APPR mode to arm the system for automatic localizer capture. The N/L ARM light will light.

The aircraft will turn to the selected heading if it is not the same as the aircraft heading and continue towards the localizer. As the aircraft approaches the localizer, the N/L ARM light will go out and the G/S ARM light will illuminate to show the computer has switched to localizer operation. The autopilot will control the aircraft in a smooth coordinated turn and rollout to track the localizer.

Note

Landing checklist should be completed prior to glideslope intercept.

As the aircraft approaches the glideslope, the glideslope pointers on the flight director and course indicator will move into view. The G/S ENG annunciator will light to indicate computer switching to glideslope operation. If altitude hold is being used, the altitude hold switch moves to off and the ALT annunciator light goes out.

When the aircraft passes over the middle marker, the G/S EXT annunciator lights to indicate that the glide-slope extension phase of the approach has started.

2.18.3.6 Go-Around. If the aircraft reaches decision height and the MDA annunciator light illuminates prior to visual contact with the airport environment, execute a missed approach maneuver utilizing the go-around mode.

Select the go-around mode by pressing the G/A mode button on the control wheel. The G/A annunciator illuminates and a wings-level, fixed pitch climbout is displayed by the command bars.

The missed approach heading is preset with the heading marker. Switching the mode selector to the HEADING mode at the completion of the go-around procedure immediately gives roll commands to turn to the preset missed approach heading.

The pitch control on the pedestal controller may be used to maintain a constant climb to the published missed approach altitude.

2.18.3.7 Back Course Approach. Set the mode selector to the HEADING mode, the course arrow to the inbound front course heading, and the heading marker to the tail of the course arrow. This ensures that the sensing of the lateral deviation bar in the course indicator will not give course readings.

Heading corrections to capture the back course are accomplished by adjusting the heading marker in the direction of the lateral deviation bar displacement. If there is a crosswind, it is necessary to reset the heading marker to aline the lateral deviation bar with the course arrow to keep the aircraft on the localizer.

Note

Fly the back-course approach in the GYRO mode for any procedure that requires bracketing of course or making frequent heading changes. The use of GYRO mode eliminates the extra workload involved in changing the heading marker for small heading corrections.

Azimuth information for the VOR is provided by the pilot directional gyro while azimuth information for the tacan is provided by the copilot directional gyro. If the pilot or copilot has NAV selected with his respective course select switch and the pilot gyro fails, the VOR pointer on the RMI will be inoperative and/or unreliable and must be disregarded. The course deviation indicator will continue to function.

If either pilot has TAC selected on his course selector switch and the copilot directional gyro fails, the tacan pointer on the RMI is inoperative but the course deviation indicator will continue to function.

2.19 INTEGRATED FLIGHT SYSTEM (CT-39G Aircraft 158843 and 159362 Through 159365)

The integrated flight system consists of the FD-109 flight director system combined with the AP-103F autopilot. The system provides the pilot automatic control of heading, attitude, altitude, VOR/ILS/tacan radial interception, and tracking with automatic crosswind correction. The flight director indicator presents a three-dimensional, parallax-free presentation to ensure safe aircraft control when the autopilot is not used.

Two complete and independent sets of components are provided, one set each for the pilot and copilot, including the flight director indicator, course indicator, instrument amplifier, a flight steering computer, vertical and directional gyros and inputs from the VOR/ILS, tacan, and marker beacon receivers. A switching package in the system (course select switch) controls selection of navigational signals to the appropriate indicators. Depending on the operating mode selected, the flight computers supply steering and attitude commands to the flight director and course indicators. The pilot flight director system operating mode is controlled from the controller located on the center pedestal (27, Figure 2-39). The mode selector switch on the pilot flight director indicator is inoperative. The copilot flight director system operating mode is controlled by the mode selector switch on the flight director control panel on the right-hand console. On these aircraft, there is no mode selector switch on the pilot or copilot flight director indicator.

The pilot has the option of coupling his flight director/course indicator presentation to the autopilot system or using the flight director system independently of the autopilot.

2.19.1 Integrated Flight System Controls and Indicators

2.19.1.1 Flight Director Indicator. The flight director indicators (Figure 2-40) on the pilot and copilot instrument panels provide a three-dimensional presentation of aircraft attitude combined with vertical and lateral steering commands from the command or V-bars. The indicators are identical and function the same except that the presentation on the pilot indicator can be coupled to the autopilot. If the autopilot is engaged, the pilot integrated flight system is in the same mode as the

autopilot. Each flight director receives attitude information from its own vertical and directional gyros.

The fixed delta-shaped symbol represents the aircraft. Pitch and bank (roll) are displayed by the relationship of the aircraft symbol to the movable attitude tape background. The background is colored with a blue sky above and black ground beneath with a white horizon line. A 360° roll presentation is possible. A scale on the same background displays pitch attitude from 0 to ±90°. A bank pointer moves in the direction in which the aircraft is rolled to show the amount of bank angle, indicated on a semicircular scale, indexed at 10°, 20°, 30°, 45°, and 60° each side of center on the upper half of the instrument. Bank is also indicated by the relationship of the aircraft symbol to the horizon line.

Tapered V-command bars that move across the indicator are servodriven by signals supplied by the flight computer. The V-bars supply autopilot command inputs when the autopilot system is engaged. The bars roll clockwise or counterclockwise to command a left or right bank to maintain a selected heading or course, and the bars move up or down to indicate pitch correction for proper altitude and attitude and to intercept and maintain the correct glideslope position. Thus, the effects of airspeed, distance, crosswind, and approach angle are resolved into computed commands and the aircraft is maneuvered so that the aircraft symbol is "flown" into the V-bars until the two are alined to satisfy these commands.

Commands generated in the flight computer for input to the flight director indicator are determined by the position of the mode selector switch. The switch has four positions: GYRO, HEADING (pilot switch) or HDG (copilot switch), NAV/LOC (pilot switch) and copilot switch (8A, FO-2), and APPR.

Note

The autopilot power switch must be on, the pilot mode selector switch must be at GYRO or HEADING, and the turn knob must be in the center detent before the autopilot can be engaged.

The GYRO position can be used in approaching a VOR or tacan station or when maneuvering is required while over the station and before tracking the outbound radial. It is also used in operations where frequent heading changes are required, such as orbiting a station, to eliminate the need to adjust the course indicator heading pointer for small corrections. In GYRO, the V-bars are not in view and the basic gyro attitude is displayed on the flight director indicator. In this mode, the annunciator lights do not function. In the heading mode, the V-bars command to steer the aircraft to the heading selected by the course indicator heading set knob, and the bank angle called for decreases gradually as heading error decreases. In the HEADING mode with the autopilot engaged, the aircraft flies to obtain the heading selected by the pilot course indicator heading set knob. With the mode selector switch at NAV/LOC or N/L (navigation/localizer), bearing information from the VOR or tacan receivers are presented on the flight director indicator and the course indicator. If the NAV/LOC (N/L) position is selected before radial interception, the HDG V/L light will illuminate and the V-bars will maintain the heading selected by the heading set knob. The light will go out and the N/L ENG light will illuminate when the selected VOR/tacan radial or localizer is captured. The flight director indicator V-bars will then indicate the direction to turn to maintain the selected radial or localizer.

The APPR position of the mode selector switch provides for automatic capture of the glideslope beam when intercepted for an ILS landing. In this mode, the system is in the armed condition (as indicated by the illumination of the GS ARM annunciator light) before glideslope capture, and in the engaged condition (as indicated by the illumination of the GS ENG annunciator light) after glideslope capture.

A glideslope pointer, representing the center of the glideslope beam, shows vertical displacement of the aircraft from the beam center. The pointer is seen only when the VOR-ILS receiver is tuned to a localizer frequency. The centerline of the glideslope pointer scale represents the aircraft position with respect to the glidepath. If the pointer is displaced upward, the aircraft is below the glidepath and vice versa.

A runway symbol below the indicator pictorial display moves to either side of the runway centerline symbol to show lateral displacement from the center of a localizer beam. It represents an expanded portion of the lateral deviation bar on the course indicator; the reference dots that bracket the runway symbol are equivalent to the innermost dots on the course indicator (1-1/4° course deflection). The runway symbol seen only in the approach mode when the navigation receiver is tuned to a localizer frequency and after glideslope capture. The runway symbol is driven vertically by an input from the radio altimeter during the last 200 feet of descent. The runway symbol just meets the aircraft symbol when the aircraft is at ground level.

A turn-and-slip indicator is located at the bottom of the flight director indicator. One standard needle width deflection provides a 360° turn in 4 minutes (1-1/2° per second rate of turn).

The MDA and GA annunciator lights are on the upper left and upper right corners, respectively, of the indicator. Refer to paragraph 2.18.1.15, Annunciator Lights, for information related to the MDA and GA lights.

A press-to-test button on the lower left corner of the indicator permits partial testing of the pitch and roll servo circuits. Pressing the button should provide a 10° pitchup and a 20° right bank attitude change. The GYRO and CMPTR flags should appear and, if in view, the command bars should disappear from view.

The pilot flight director is powered by the dc essential bus and the ac primary bus. The copilot flight director is powered by the ac secondary bus.

2.19.1.2 Flight Director Indicator Warning Flags. The flight director indicator with power off and all (red) warning flags in view is shown in Figure 2-40. Limited system operation is possible with some of the flags in view. If any flag appears during the normal operation sequence, a check should be made to determine whether the apparent malfunction can be corrected by resetting circuit breakers, recycling an associated receiver, slaving an input gyro, or selecting the alternate position on the vertical gyro selector switch. The following paragraphs describe what causes a flag to appear.

1. The GYRO flag indicates failure of a vertical gyro, attitude display servo system, or power that drives the attitude display.

2. The COMPUTER flag indicates failure of the flight computer and V-bar display.

3. The GS flag indicates a malfunction of the VOR/ILS receiver or a weak or unreliable glide slope signal.

4. The R/T flag indicates a malfunction in the rate-of-turn indicator.

5. The RUNWAY flag indicates a malfunction of the VOR/ILS receiver, an unreliable radio altimeter signal, or a failure in the localizer display.

2.19.1.3 Course Indicator. This multipurpose indicator (Figure 2-40) on the pilot and copilot instrument panels presents course, heading, miles, and directional guidance in the various flight director modes. The pilot and copilot course indicators are identical except that the pilot indicator is also capable of presentations in the autopilot mode. Navigation data from the VOR and tacan receivers is used to present a symbolic display of the horizontal navigation situation.

A miniature aircraft symbol etched on the instrument glass shows aircraft position and heading with respect to the desired course. A rotating compass card, remotely driven by signals from the directional indicating system, displays aircraft magnetic heading against a lubber line. A triangular-shaped heading marker can be set to the desired magnetic heading around the outer edge of the compass card by the heading set knob. Clockwise knob rotation moves the heading marker clockwise. Once set, the marker will rotate with the compass card.

Desired course or track is shown by a course arrow that rotates within the compass card. The course arrow is manually set to a magnetic course that coincides with the desired VOR or tacan radial or localizer course by turning the course knob. Clockwise rotation moves the arrow clockwise. A three-digit course counter reflects the magnetic course to which the course arrow is pointing. Once set, the course arrow will rotate with the compass card. The course arrow has a segmented section called a lateral deviation bar that gives a plan view of the radial position of the desired tacan, VOR, or localizer course set on the course knob relative to the aircraft. The bar represents the centerline of the desired course. The aircraft symbol then simulates the aircraft position with respect to the desired course. A four-dot scale brackets the lateral deviation bar, two dots on each side of the centerline. In the navigation/localizer mode, a two-dot deflection of the bar represents a 10° course deviation. In the approach mode, a two-dot deflection represents a 2-1/2° course deflection. The localizer deviation pointer on the flight director indicator indicates a maximum one-dot deflection of 1-1/4° in the localizer mode.

The integrated flight system has the capability of intercepting the localizer back course. However, to obtain directional indications from the lateral deviation bar, the front approach course has to be set in the course indicator. Then, when the lateral deviation bar is offset, the aircraft has to be steered in the direction the bar is offset to center the bar and intercept the course.

A TO-FROM arrowhead indicates in the navigation localizer/mode (VOR-tacan only), whether the selected course is inbound (TO-FROM arrowhead pointing at head of course arrow) or outbound (TO-FROM arrowhead pointing at tail of course

arrow) with respect to a VOR or tacan station. The TO-FROM arrowhead does not appear when turned to a localizer frequency. Range to a tacan station is shown on the miles counter. The three-digit counter covers 000 to 299 nautical miles.

A glideslope pointer, representing the center of the glideslope beam, shows vertical displacement of the aircraft from the beam center. This pointer, a repeat of the glideslope pointer on the the flight director indicator, is effective only when the VOR/ILS receiver is tuned to a localizer frequency. The center dot of the glideslope pointer scale represents the aircraft position. If the pointer is displaced upward, the aircraft is below the glidepath and vice versa.

The course indicator is powered by the dc essential bus and the ac primary bus. The copilot course indicator is powered by the ac secondary bus.

2.19.1.4 Course Indicator Warning Flags. The course indicator with power off and all (red) warning flags in view is shown in Figure 2-40. Limited flight director system operation is possible with some of the flags in view. If any flag appears during the normal operation sequence, a check should be made to determine whether the apparent malfunction can be corrected by resetting circuit breakers, recycling the associated receiver, slaving an input gyro, or selecting the alternate position on the compass selector switch.

1. The GS flag indicates a malfunction of the VOR/ILS receiver or a weak or unreliable glideslope signal. The GS flag covers the glideslope pointer and scale to prevent accidental acceptance of faulty indications.

2. The VOR/LOC flag indicates a malfunction of the VOR/ILS receiver or reception of an unreliable signal.

3. The COMPASS flag indicates failure of a directional gyro.

2.19.1.5 IFS Light Rheostat Knob. This knob (7 and 14, Figure 2-39) on the pilot and copilot instrument panels is labeled IFS/LTS (integrated flight system lights) and controls the light intensity of the flight director and course indicators. Turning the knob clockwise increases light intensity. The rheostat is powered by the ac secondary bus.

2.19.1.6 Radio Magnetic Indicator. This indicator (25 and 32, Figure 2-39) on the pilot and copilot instrument panels receives heading information from the directional indicating system. This is reflected on a rotation compass card that provides a magnetic heading displayed against a fixed reference marker at the 12 o'clock position on the dial. Signals from the ADF system and VOR and tacan receivers are fed into the instrument to drive two pointers, a single-barred and a double-barred pointer. The indicator is labeled VOR on the left, TACAN on the right, and ADF on the lower center. A knob on the left side of the indicator has a single-barred arrow and a knob on the right side of the indicator has a double-barred arrow. When the left knob is turned to VOR or ADF, the single-barred pointer on the indicator reflects the bearing of the corresponding signal. Turning the right knob to TACAN or ADF causes the double-barred pointer on the indicator to reflect the corresponding signal bearing. If both knobs are turned to ADF, the pointers are servoed together and move as one to indicate ADF bearing. The pilot RMI is powered by the ac primary bus. The copilot RMI is powered by the ac secondary bus.

2.19.1.7 Glideslope Manual Switch. Spring-loaded to the NORM position, this switch (3 and 20, Figure 2-39) on the pilot and copilot instrument panels serves a two-fold purpose in the approach mode. The first is backup in the event automatic glideslope capture malfunctions. If the glideslope pointer passes downward through the center of the scale and the GS ENG annunciator light does not come on to indicate automatic capture, the glideslope manual switch should be moved to the MAN (manual) position. This couples the V-bars on the flight director indicator to the glideslope for pitch commands. Secondly, the switch can be moved to MAN if it is desired to intercept the glideslope from above since the automatic capture functions only when intercepting from below. The switch is powered by the dc essential bus and the ac primary bus.

2.19.1.8 Localizer Reverse Switch. This switch (22B and 33A, Figure 2-39) on the pilot and copilot instrument panels reverses the polarity of the VOR/localizer deviation and course datum signals during a back course ILS approach (switch at LOC REVERSE position). In addition, when either switch is positioned to LOC REVERSE, the LOC REV annunciator lights come on.

2.19.1.9 Go-Around Button and Light. A go-around button (Figure 2-29) on each control wheel, labeled G/A, is used in the approach mode when the radio altimeter MDA light comes on for a missed approach. Pressing the button turns on a light that reads G/A and causes the flight director indicator V-bars to command a wings-level, fixed-pitch climbout. The lights are on the pilot and copilot flight director indicators (Figure 2-40). Pressing the button on the pilot control wheel also turns off the autopilot. The go-around

button and light are powered by the dc essential bus. Depressing either go-around button in the approach mode, prior to reaching minimum decision altitude, or in any other mode except gyro will provide synchronization of the V-bars with the existing pitch attitude of the aircraft.

2.19.1.10 Altitude Hold Switch. The copilot altitude hold switch (19, Figure 2-39) is on the copilot instrument panel. The pilot altitude hold switch (27, Figure 2-39) on the autopilot control panel is channel guarded to prevent inadvertent actuation. (Refer to paragraph 2.18.2, Autopilot System.) The switch references pitch commands to barometric altitude so that the barometric altitude existing at the time of switch engagement will be maintained. The switch is engaged when it is moved from OFF to ON, and an annunciator light on the instrument panel comes on to read F/S ALT. The switch should be moved to ON in the heading or navigation/localizer mode or in the approach mode up to glideslope capture. With the altitude hold switch ON, the flight director indicator V-bars command a climb or descent as necessary to maintain the barometric altitude indicated on the altimeter at the time the switch is engaged to ON. The switch automatically returns to OFF in the approach mode after glideslope capture. The switch will likewise return to OFF if the autopilot system is engaged or disengaged, or when operating on autopilot, when the autopilot pitch wheels are moved. The altitude hold switch is powered by the dc essential bus and the ac primary bus.

2.19.1.11 Course Selector Switch. The rocker-type course selector switch (9 and 11, Figure 2-39) on the left and right instrument panel shroud has NAV and TAC positions. When the NAV portion of the switch is pressed, VOR signals received by the VOR/ILS receiver are connected to the flight director, course, and RMIs. With the TAC portion of the switch engaged, tacan signals are connected to the course and RMIs. The switch is powered by the dc essential bus.

2.19.1.12 Course Select Fail Lights. Two amber, placard-type caution lights (26, Figure 1-27), one labeled PILOT C SEL FAIL and one COPILOT C SEL FAIL, are on the caution and warning light panel. If either light illuminates, a failure is indicated in the respective course select system and the associated flight director and course indicator should be considered unreliable.

2.19.1.13 Copilot Course Resolver Fail Light. An amber, placard-type caution light (12, Figure 2-37), labeled COPILOT C RSLVR INOP, is located on the caution-warning light panel. This light will illuminate if both pilots select the same NAVAID (either TAC or NAV) with their course select switch when there is a failure in the switching package. In the event of a relay failure in the switching package, the copilot flight director and course indicator should be considered unreliable. If the light is caused by both pilots selecting either NAV or TAC, the copilot can still monitor course deviation and to/from information as selected on the pilot course indicator if he selects a course that corresponds to the course set by the pilot.

2.19.1.14 Vertical Gyro Selector Switch. This rocker-type switch (8, Figure 2-39) on the pilot instrument panel is labeled V/G SEL. The switch is used in selecting the vertical gyros, both of which are the source of pitch and roll information for the integrated flight system. When the No. 1 position of the switch is pressed, the pilot systems are operating from the No. 1 vertical gyro and the copilot systems are operating from the No. 2 vertical gyro. When the No. 2 position of the switch is pressed, the pilot and copilot systems are both operating from the No. 2 vertical gyro. If a vertical gyro failure is indicated by the appearance of the GYRO warning flag on the flight director indicator, the selector switch should be moved to the alternate position to select the operable vertical gyro. The vertical gyro selector switch is powered by the ac primary bus.

2.19.1.15 Annunciator Lights. The annunciator lights (5 and 17, Figure 2-39) on the pilot and copilot instrument panels are labeled as follows to indicate the corresponding operating functions and modes:

1. N/L ARM — Comes on (amber) when the flight director mode selector switch is at NAV/LOC (N/L) while in the navigation/localizer mode. Goes out upon navigation or localizer capture.

2. N/L ENG — Comes on (green) after navigation or localizer capture.

3. F/S ALT — Comes on (green) when the altitude hold switch is moved to ON.

4. GS ARM — Comes on (amber) when the flight director mode selector switch is at APPR before glideslope capture. Goes out when the GS ENG light comes on.

5. GS ENG — Comes on (green) when the flight director mode selector switch is at APPR after glideslope capture.

6. GS/EXT — Comes on (green) when passing over the middle marker beacon (M marker beacon light on) to indicate the steering computer

has switched to the glideslope extension phase. This eliminates undesirable glideslope noise.

7. MDA — Comes on (amber) when aircraft has descended to a preset radio altimeter decision height. At this point, a land or go-around decision must be made.

8. GA — Comes on (green) when the go-around button is depressed while operating in the approach mode.

9. LOC REV — Comes on (white) when localizer reverse switch is placed at LOC REVERSE (navigation receiver tuned to localizer frequency and mode selector switch at HEADING).

The annunciator lights are powered by the essential bus and the ac secondary bus.

2.19.2 Autopilot System. The AP-103F autopilot system will hold the aircraft on any preselected course, maintain level flight, climb, or dive, and change course, in coordinated turns. The autopilot can be used in conjunction with the pilot integrated flight system for automatic control of heading, altitude and attitude, and acquisition of a selected magnetic heading or VOR localizer course and glideslope with crosswind corrections. The system includes rate and vertical gyros, an altitude hold controller, a three-channel (aileron, elevator, and rudder) amplifier, airspeed and rate sensors, and a primary control and trim coupler that positions the aircraft control surfaces. Airspeed and altitude sensing are received through the aircraft pitot-static system. A control panel in the cockpit center console contains the autopilot controls and indicators as well as the pilot integrated flight system-flight director mode selector switch. Autopilot control can be manually overridden at any time. The autopilot system is powered by the dc essential bus, the ac primary bus, and the ac secondary bus.

2.19.2.1 Autopilot System Controls and Indicators

2.19.2.1.1 Emergency Disconnect Switch. This switch (28, Figure 2-39) on the center console has a handle labeled A/P EMERGENCY DISCONNECT. Moving the switch up to ENGAGE permits engagement of the autopilot system when the autopilot power switch is at ON. Moving the switch down to DISENGAGE electrically disconnects the autopilot engaging and servo power units and turns on a disengage indicator light. The emergency disconnect switch should not be used to turn off the autopilot in normal operation. The switch should be maintained at ENGAGE at all times except in an emergency when a malfunction occurs in the normal disengage circuits or in the autopilot disengage buttons, in which case, the emergency disconnect should be moved to DISENGAGE.

2.19.2.1.2 Power Switch. The autopilot power switch on the autopilot control panel (27, Figure 2-39) is channel guarded to prevent inadvertent switch actuation. Moving the switch up to A.P. POWER applies electrical power to all system components if the autopilot emergency disconnect switch is at ENGAGE.

2.19.2.1.3 Engage Button. The autopilot engage button on the autopilot control panel (27, Figure 2-39) is used for engaging the autopilot system. The autopilot can be engaged by pressing the button if the following conditions are met:

1. Power switch at A.P. POWER

2. Emergency disconnect switch at ENGAGE

3. Turn knob at center detent

4. Mode selector switch at GYRO or HEADING

5. Pilot flight director indicator GYRO and COMPUTER flags hidden.

Pressing the engage button turns off a disengage indicator light.

2.19.2.1.4 Disengage Buttons. Pressing the disengage button (Figure 2-29) labeled AUTOPILOT DISC on either control wheel disengages the autopilot and turns on the autopilot disengage light.

2.19.2.1.5 Disengage Light. This amber, dimmable, press-to-test light on the autopilot control panel (27, Figure 2-39) is labeled DISENGAGE. The light comes on if the autopilot switch is on when the autopilot is disengaged by any means.

2.19.2.1.6 Go-Around Button. Pressing the go-around button (Figure 2-29) on the pilot control wheel shuts off the autopilot in addition to its function in the integrated flight system. For additional information on this switch and light, refer to paragraph 2.18.1.8, Go-Around Button and Light.

2.19.2.1.7 Turn Knob. The turn knob on the autopilot control panel (27, Figure 2-39) is used for commanding turns when the autopilot-flight director system is in the GYRO mode. Labeled TURN L/R, the knob is rotated counterclockwise for a left turn and clockwise for a right turn. Any degree of bank can be selected from

a maximum of 25° in the heading and navigation/localizer modes to a maximum of 15° in the approach mode. The aircraft will roll out in a coordinated turn and hold heading when the turn knob is centered at the detent position (white index line on the knob at 12 o'clock). The knob has to be at the center detent before the autopilot system can be engaged.

2.19.2.1.8 Pitch Wheels. Two pitch wheels on the autopilot control panel (27, Figure 2-39) are connected by a common shaft to control climb or dive in the autopilot mode. Each wheel has UP and DN positions and indicating arrows to each position. By rotating either wheel in the desired direction, selection can be made to climb 12° maximum, or in the go-around mode, 10° fixed pitchup. The pitchdown (dive) maximum is 2.2°. The autopilot may be engaged with the pitch wheels in any position and the aircraft will maintain the same pitch attitude existing at the time of engagement. The pitch wheels are inoperative in the approach mode (mode selector switch at APPR) after glideslope capture. If engaged, the altitude hold function is disconnected when the pitch wheels are moved.

2.19.2.1.9 Trim Indicators. Three indicators on the autopilot control panel (27, Figure 2-39) are labeled RUD, AIL, and EL, one each for the rudder, ailerons, and elevators. The rudder indicator is as viewed from aft to forward. Before autopilot engagement, the indicators show (in deflection) the direction and magnitude of torque that will be delivered by the autopilot servos at the instant of engagement. After autopilot engagement, the indicators show what the aircraft trim condition will be upon disengagement of the autopilot.

2.19.2.1.10 Trim Failure Light. This amber, placard-type light on the caution-warning light panel (12, Figure 2-39) comes on to read AP TRIM FAIL when the autopilot is engaged and automatic trimming becomes erratic or inoperative. Retrimming can be accomplished by first checking the horizontal stabilizer trim indicator for unusually large trim and manually trimming out control forces if necessary, then disengaging the autopilot, manually retrimming aircraft, and reengaging the autopilot.

2.19.2.1.11 Pitot-Static Shutoff Switches. These switches, (FO-2) on the right console control the pitot and static air supply to the autopilot system. With the A/S PITOT and A/S STATIC switches at the guarded ON position, the autopilot airspeed sensor and altitude controllers are supplied from the copilot normal pitot-static source (Figure 2-36) regardless of the position of the copilot static selector switch.

Raising the guard and moving the pitot shutoff or the static shutoff switch to OFF shuts off the respective system to the autopilot. These switches should be moved to OFF only when the respective pitot or static system is malfunctioning and it is suspected that the fault lies in the autopilot system.

WARNING

The autopilot must be turned to OFF when either of these switches is at OFF.

2.19.2.1.12 Altitude Hold Switch. The pilot altitude hold switch is on the autopilot control panel (27, Figure 2-39). The switch can be moved to ON to provide autopilot commands to maintain a constant barometric altitude. For additional information on this switch, refer to paragraph 2.18.1.9, Altitude Hold Switch.

2.19.2.1.13 Mode Selector Switch. The pilot mode selector switch is on the autopilot control panel (27, Figure 2-39). The autopilot may be engaged with the switch at GYRO. In this mode, turns are controlled by the turn knob and the pitch wheels may be used to command pitch corrections. The autopilot may likewise be engaged with the switch at HEADING. In this mode, the aircraft will enter a coordinated turn to a preset heading and the pitch wheels may be used to command pitch corrections. With the switch at NAV/LOC or APPR, the autopilot will track, maintain the selected headings, and move the aircraft in coordinated turns commanded by the integrated flight system. For additional information on this switch, refer to paragraph 2.18.1.1, Flight Director Indicator.

The YAW DAMPER switch is on the right side of the instrument panel shroud (11A Figure 2-39). The switch when positioned to ON engages the rudder channel of the autopilot system independently. The YAW DAMPER switch must be at OFF during takeoff and landing to obtain sufficient rudder movement for nosewheel steering.

2.19.3 Operation of Integrated Flight and Autopilot Systems

2.19.3.1 Takeoff and Climbout. Check autopilot power switch on and disengage light illuminated. Set heading marker to runway magnetic heading and adjust pitch command control for desired climbout attitude (approximately 12° to 15°). When airborne and at least 1,500 feet AGL, the autopilot may be engaged in either GYRO or HEADING mode.

If engaged in the GYRO mode, turns to the departure heading are controlled by the TURN control. The pitch control on the pedestal may be used to command attitude corrections to maintain proper climb airspeed. As level-off altitude is approached, the pitch control is adjusted to reduce the rate of climb to less than 500 fpm. The altitude switch may then be turned ON to provide autopilot commands to maintain a constant barometric altitude.

If the autopilot is engaged in the HEADING mode, the autopilot will maintain or turn to the preset heading.

2.19.3.2 VOR and Tacan. With the system operating in the HEADING mode, tune the VOR or tacan to the desired frequency. Set the course selector switch to correspond to the desired NAVAID. Set the course arrow to the desired radial and select an intercept heading of 85° or less with the heading marker on the course indicator. Change the mode selector to NAV/LOC. The N/L ARM annunciator light will illuminate and the aircraft will maintain the selected intercept heading. As the desired radial is approached, this light will go out and the autopilot will control the aircraft in a coordinated turn to roll out the radial. When the N/L ARM light goes out, the N/L ENG light illuminates.

After NAV/LOC capture, the system will provide automatic crosswind correction to track the radial. The altitude hold capability may be used to maintain constant barometric altitude during all phases of the radial intercept and capture.

If use of the autopilot is not desired, proceed as previously stated and manually fly the aircraft symbol to match the V-bars on the flight director indicator.

2.19.3.3 Holding. To establish a holding pattern over the outer marker, VOR station or tacan fix, switch the mode selector to the HEADING mode. Turn the course arrow on the course indicator to the inboard front course or the appropriate VOR/TAC radial. The lateral deviation bar now represents the localizer or VOR/tacan radial. With the aircraft heading inbound to the holding fix, the radial is maintained by adjusting the heading marker to center the lateral deviation bar. When the holding fix is reached, turn the heading marker approximately 90° in the direction of the turn to command the aircraft into the outbound turn. As the turn progresses, continue moving the heading marker until the reciprocal heading of the inbound course is reached. If crosswind correction is required, compensate for the drift by adjusting the heading marker.

When the required time or distance has elapsed while tracking the outbound course, the inbound turn is initiated in the same manner as the outbound turn. Slight corrections may be necessary during the turn or rollout to align the lateral deviation bar with the course arrow. Should automatic capture and tracking be desired, switch the mode selector to the NAV/LOC mode after approximately 90° of the inbound turn has been completed.

2.19.3.4 Letdown. For a typical approach and letdown to a VOR station, the autopilot, operating in the NAV/LOC mode, will track the selected radial to the station. The to-from pointer on the course indicator will reverse directly over the station. Special damping circuits allow station passage without excessive maneuvering. If the aircraft is approaching a VOR station on a course that requires maneuvering while over the station and prior to tracking the outbound radial, the maneuver may be executed in either the GYRO or HEADING mode.

When approaching the proper portion to commence the procedure turn, move the heading marker under the lubber line and switch the mode selector to the HEADING mode; adjust the heading marker to maneuver through the procedure turn.

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Note

For a 90° to 270° type of maneuver, do not move the heading marker past the tail of the aircraft or the flight director will reverse the direction of the turn.

If the letdown is continued during the turn, set the desired rate of descent using the pitch control. If altitude hold is desired, engage the altitude hold switch.

When the 180° phase of the procedure turn is completed and the inbound course is approached, move the course arrow to the inbound course and switch the mode selector to the NAV/LOC mode. The autopilot will capture and track the selected radial. Use the pitch control to establish the desired rate of descent. The crab angle established by the automatic crosswind correction is shown on the course indicator as the angle between the course arrow and the lubber line.

Note

Preset bank angle limits in the HEADING mode may require operation in the GYRO mode if standard rate turns are desired.

2.19.3.5 Approach. Both the localizer and glideslope are captured automatically on an ILS approach. The localizer is always captured before the glideslope and is captured from a selected heading.

If tracking a VOR radial in the NAV/LOC mode, set the heading marker to the desired intercept heading and switch the mode selector to the HEADING mode. After the VOR/ILS receiver is tuned to the localizer frequency, set the course arrow to the inbound localizer course and switch mode selector to the APPR mode to arm the system for automatic localizer capture. The N/L ARM light will light.

The aircraft will turn to the selected heading if it is not the same as the aircraft heading and continue towards the localizer. As the aircraft approaches the localizer, the N/L ARM light will go out and the G/S ARM light will illuminate to show the computer has switched to localizer operation. The N/L ENG light will also illuminate. The autopilot will control the aircraft in a smooth coordinated turn and rollout to track the localizer.

Note

Landing checklist should be completed prior to glideslope intercept.

As the aircraft approaches the glideslope, the glideslope pointers on the flight director and course indicator will move into view. The G/S ENG annunciator will light to indicate computer switching to glideslope operation. If altitude hold is being used, the altitude hold switch moves to off and the ALT annunciator light goes out.

When the aircraft passes over the middle marker, the G/S EXT annunciator lights to indicate that the glideslope extension phase of the approach has started.

2.19.3.6 Go-Around. If the aircraft reaches decision height and the MDA annunciator light illuminates prior to visual contact with the airport environment, execute a missed approach maneuver utilizing the go-around mode.

Select the go-around mode by pressing the go-around mode button on the control wheel. The G/A annunciator illuminates and a wings-level, fixed pitch climbout is displayed by the command bars.

The missed approach heading is preset with the heading marker. Switching the mode selector to the HEADING mode at the completion of the go-around procedure immediately gives roll commands to turn to the preset missed approach heading.

The pitch control on the pedestal controller may be used to maintain a constant climb to the published missed approach altitude.

2.19.3.7 Back Course Approach. Set the mode selector to the HEADING mode, the course arrow to the inbound front course heading, and the heading marker to the tail of the course arrow. Position the localizer reverse switch to the LOC REVERSE. The LOC REV light will come on. After about a 20-second delay (time required for capture computing), set the mode selector to NAV/LOC. The N/L ARM light will come on. At localizer capture, the N/L ARM light will go off and the N/L ENG light will come on. The flight director indicator runway symbol provides normal indications during the back course approach.

2.20 INTEGRATED FLIGHT SYSTEM (CT-39G Aircraft 158844 and 160053 Through 160057)

The flight control system consists of an FD-109 flight director system combined with the AP-105 autopilot. The system provides the pilot automatic control of heading, attitude, altitude, VOR/ILS/tacan radial interception, and tracking with automatic crosswind correction. The flight director indicator presents a three-dimensional, parallax-free presentation to ensure safe aircraft control when the autopilot is not used.

Two complete and independent sets of components are provided, one set each for the pilot and copilot, including the flight director indicator, course indicator, flight computer, mode coupler, turn rate sensor, and a mode selector panel. The system also contains two vertical gyros, acceleration sensor, and remote control that are common to both sets of components. A switching package in the system controls selection of navigational signals to the appropriate indicators. Depending on the operating mode selected, the flight computers supply steering and attitude commands to the flight director and course indicators. The pilot and copilot flight director system operating mode is controlled from the mode selector panels located on the center pedestal. The pilot mode selector is the top panel.

The pilot has the option of coupling his flight director to the autopilot system or using the flight director system independently of the autopilot.

2.20.1 Integrated Flight System Controls and Indicators. A flight director indicator is mounted in the pilot and copilot instrument panel and displays aircraft attitude and steering commands on a symbolic three-dimensional display. Attitude information is supplied to the pilot and copilot flight director systems by two vertical gyros (Figure 2-41).

A fixed delta-shaped symbol represents the aircraft. Aircraft pitch and roll attitude is displayed by the relationship of the aircraft symbol to the movable attitude tape. Pitch indexes are shown on the attitude tape. The attitude tape is colored with a blue sky above a brown earth separated by a white horizon line. Roll indexes show 10°, 20°, 30°, 45°, and 60° of right and left bank. A full 360° roll presentation about the horizon is possible.

The command bars display computed pitch and bank commands. These bars move up or down to command a climb or descent and roll clockwise or counterclockwise to command a right or left bank. The aircraft is maneuvered so that the aircraft symbol is flown into the command bars until the two are aligned to satisfy the commands. With no modes selected on the mode selector, the command bars are out of view. In this condition, the autopilot is in a command or gyro mode and any flight commands are accomplished by use of the autopilot controller.

The glideslope pointer represents the center of the glideslope beam and displays vertical aircraft displacement from the glideslope beam center. The center of the glideslope scale represents the aircraft. When the pointer is deflected upward, the aircraft is below the glidepath. This is displacement information only; pitch commands are presented by the command bars. The glideslope pointer is in view when the navigation receiver is tuned to a localizer frequency and the APPR button on the flight mode selector is depressed.

The runway symbol represents the runway centerline and displays aircraft lateral deviation from the center of the localizer. The runway scale represents an expanded portion of the lateral deviation scale on the course indicator. The outside marks on the scale are equivalent to the inner dots on the course indicator (approximately 1-1/4° from the localizer centerline). The radio altimeter drives the runway symbol vertically during the last 200 feet of descent as it continues to display localizer deviation.

The radio altimeter lights the DH annunciator at a preselected radio altitude, indicating that a decision to land or go around must be made. The radio altimeter is functional from 2,000 feet to touchdown.

The inclinometer or slip indicator displays aircraft slip or skid. The inclinometer consists of a weighted ball in a liquid-filled, curved glass tube.

The rate-of-turn pointer displays aircraft rate of turn about the yaw axis. A two-needle width deviation shows a standard rate of turn of 3° per second; a one-needle width deviation shows a standard rate of turn of 1-1/2° per second.

The flight director indicator test switch permits partial testing of the flight director indicator pitch and roll servo systems. Depressing this switch causes the attitude display to change by 10° pitchup and 20° right bank. The GYRO and COMPTR flags appear and, if the command bars are in view, they are retracted. The pilot flight director is powered by the ac primary and dc essential bus and the copilot is powered by the 115-vac secondary and 28-vdc essential buses.

2.20.1.1 Flight Director Indicator Warning Flags. The flight director indicator with power off and all (red) warning flags in view is shown in Figure 2-41. Limited system operation is possible with some of the flags in view. If any flag appears during the normal operation sequence, a check should be made to determine whether the apparent malfunction can be corrected by resetting circuit breakers, recycling an associated receiver, slaving an input gyro, or selecting the number two vertical gyro. The following paragraphs describe what causes a flag to appear.

1. The GS flag indicates a malfunction of the glideslope receiver or an unreliable glideslope signal. Vertical commands in approach mode are unusable; however, attitude, heading, and VOR/LOC position information are correct. The GS flag is out of view during back localizer course.

2. The COMPTR flag monitors the active inputs to the flight control system. When the COMPTR flag appears, command information is unreliable and the command bars go out of view. Attitude, heading, VOR/LOC position, and glideslope information may be correct.

3. The RUNWAY flag indicates a malfunction of the navigation receiver, an unreliable localizer signal, or a radio altimeter malfunction. When the RUNWAY flag appears and the navigation flag on the course indicator does not appear, the failure is in the radio altimeter. Attitude, heading, and vertical pitch commands may be correct.

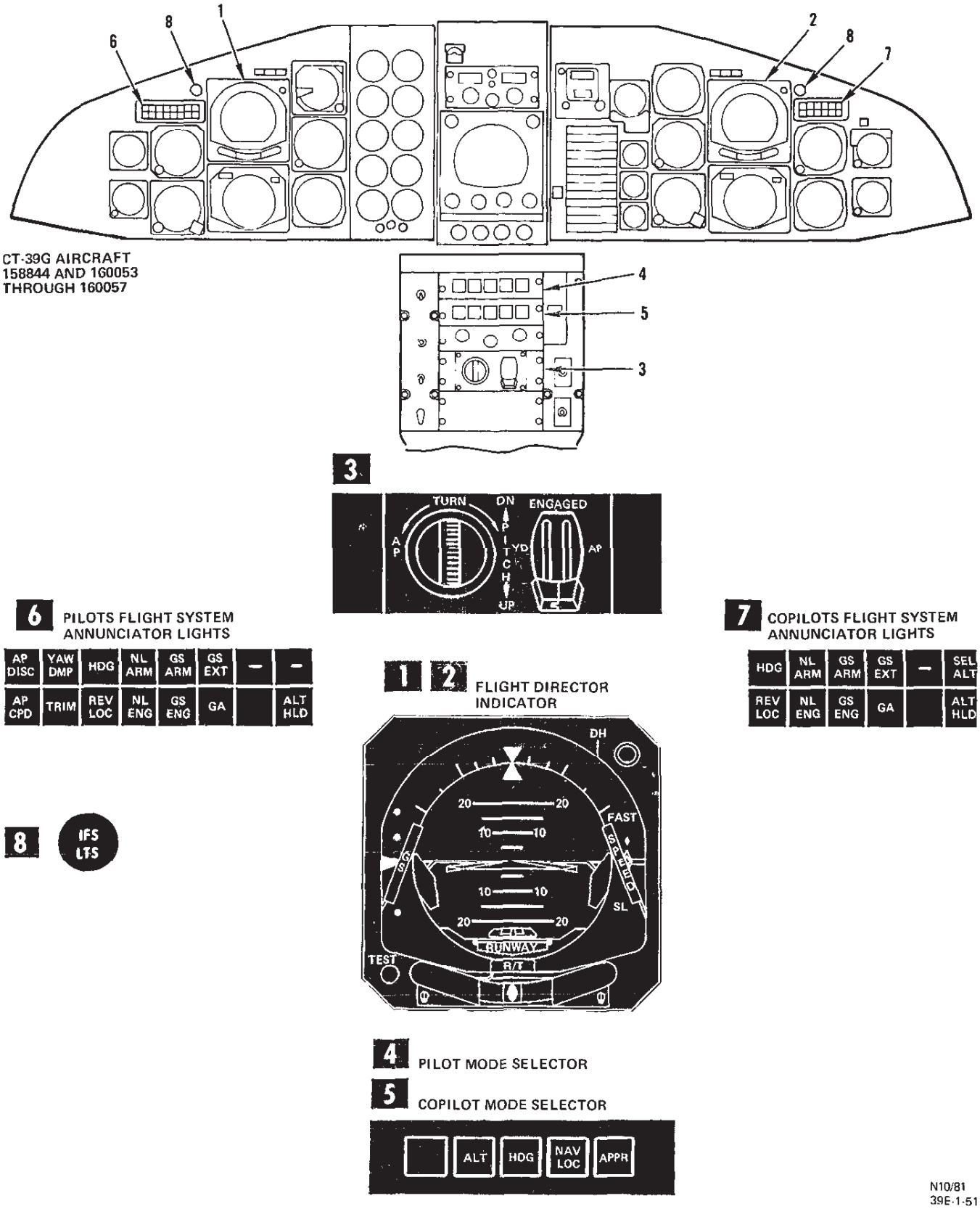

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Figure 2-41. Flight Director and Course Indicators — CT-39G (Sheet 1 of 2)

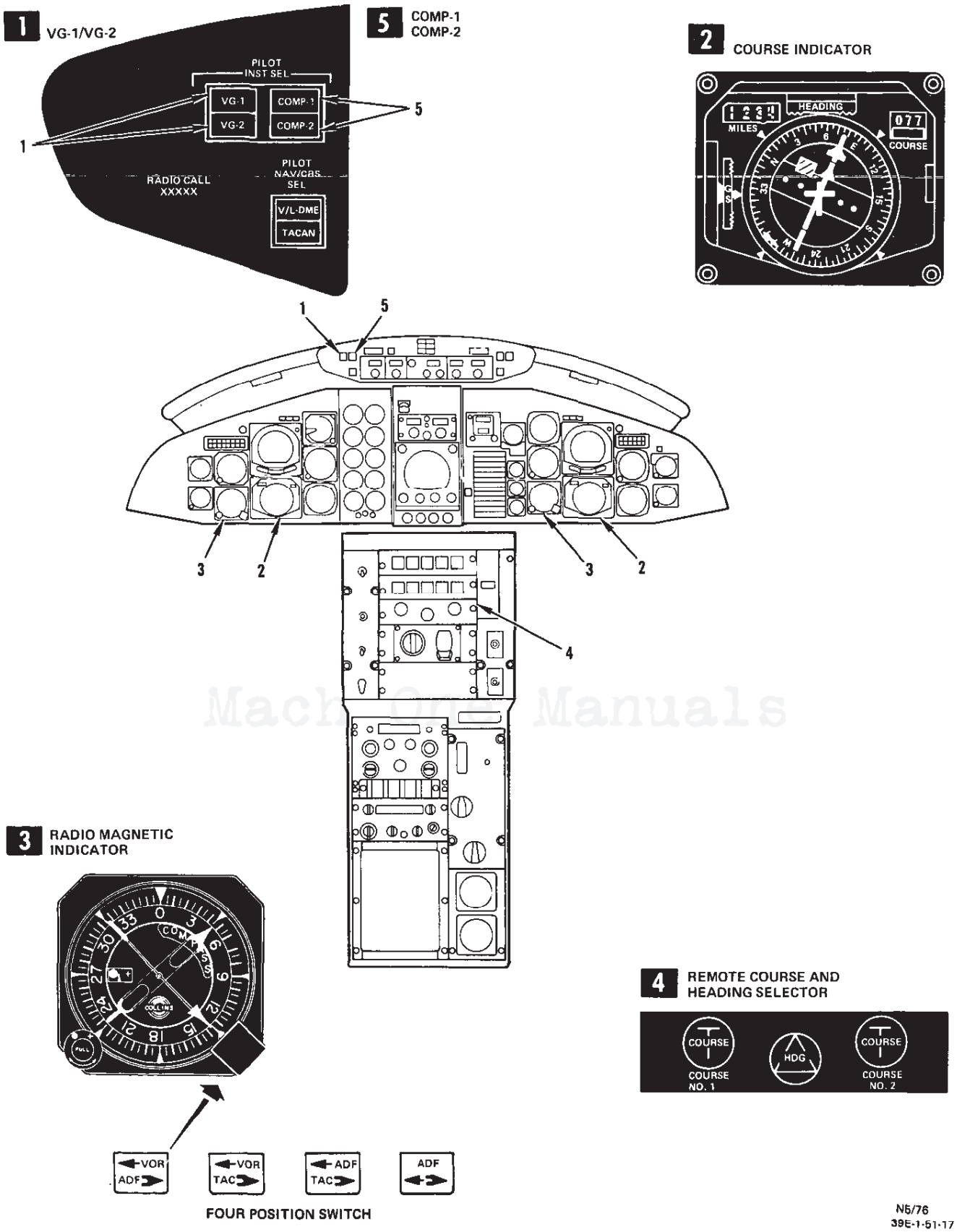


Figure 2-41. Flight Director and Course Indicators — CT-39G (Sheet 2 of 2)

4. The GYRO flag indicates a malfunction in the vertical reference or display system. Attitude information could be incorrect. Heading, VOR/LOC, and glideslope position displays are correct.

5. The R/T flag indicates a malfunction in the rate-of-turn sensor. The rate-of-turn display is unreliable. Steering commands, attitude, heading, VOR/ LOC, and glideslope position displays are correct.

2.20.1.2 Flight Mode Selector. There are two flight mode selectors on the center pedestal; pilot mode (selector No. 1 — top panel) and copilot mode (selector No. 2 — bottom panel). The flight mode selector (Figure 2-41) consists of four pushbutton switches, each designated by a given control mode. The pushbuttons are backlighted with the intensity controlled through the cockpit dimmers. The pushbuttons actuate push-on, push-off magnetically held solenoid switches. When a mode selection is made, it is routed through the mode coupler to the flight computer. The select panel contains four modes of operation.

2.20.1.3 ALT (Altitude Hold). Steering commands maintain the altitude at which altitude hold was selected. The altitude hold mode is automatically released when the glideslope is captured or when the flight control system is engaged and coupled and the autopilot controller pitch control is moved from the center position.

2.20.1.4 HDG (Heading). When HDG is selected on the mode selector and the mode selector and the autopilot is engaged, the autopilot maneuvers the aircraft to turn to and maintain the magnetic heading as set on these course indicator with the HDG marker.

To complete a turn of more than 135°, it is best to set the HDG marker on the course indicator approximately 135° in direction of the desired turn. After completing about 120° of the turn, reselect up to 135 additional degrees.

2.20.1.5 NAV/LOC. Steering commands control the aircraft to follow the radio course as set by the heading marker on the course indicator. The heading marker is set to produce the desired intercept angle (less than 90°). While flying on the selected heading, the flight computer computes a capture point based on deviation from the selected radio course, the course intercept angle, and the rate the aircraft is approaching the VOR or localizer course. When the selected course is captured, the flight control system turns the aircraft to track the selected radio course with automatic crosswind correction.

2.20.1.6 APPR (Approach). Steering commands control the aircraft to fly to and maintain the localizer and glideslope capture. The vertical mode automatically switches off at glideslope capture and the vertical steering command is based on the glideslope signal and pitch data.

2.20.1.7 Flight Director/Autopilot Annunciator Lights. The flight director/autopilot annunciator lights (Figure 2-41) are located on the pilot and copilot instrument panel. The lights are identified and function as follows:

1. NL-ARM — Illuminates after NAV/LOC or APPR is selected and prior to VOR, NAV, or localizer course capture.

2. NL-ENG — Illuminates after NAV VOR or localizer capture in either NAV/LOC or APPR mode.

3. GS-ARM — Illuminates after APPR is selected and prior to glideslope capture.

4. GS-ENG — Illuminates after glideslope capture in APPR mode.

5. GS-EXT — Illuminates when radio ALT reaches 200 feet or middle marker is received from marker beacon receiver.

6. SEL-ALT — Illuminates when the aircraft deviates 300 feet from the altitude that is set on the altitude alerter (on copilot annunciator panel only).

7. GA — Illuminates when the go-around button on the control wheel is activated.

8. AP-DSC — Illuminates when the autopilot is disengaged (on pilot annunciator panel only).

9. AP-CPD — Illuminates whenever the autopilot is coupled to flight director system and a mode selected (on pilot annunciator only).

10. ALT-HLD — Illuminates when the altitude hold mode is selected.

11. REV/LOC — Illuminates when the aircraft heading is within 75° of the back localizer bearing. The reverse localizer is automatically switched to a back localizer course by the flight computer.

12. HDG — Illuminates after the HDG button is depressed on the mode selector. Will maintain a

heading as selected on the course indicator heading marker.

13. YAW DAMP — Illuminates when the yaw damper is engaged by the yaw damper engage lever on the autopilot control panel.

14. TRIM — Illuminates anytime the autopilot is trimming the horizontal stabilizer. The light is disabled when the autopilot is disengaged.

2.20.1.8 IFS LTS Light Rheostat Knobs. Two light rheostat knobs (Figure 2-41) on the pilot and copilot instrument panels are labeled IFS LTS and control the light intensity of the flight director and course indicators. Turning the knobs clockwise increases light intensity. The rheostat is powered by the ac secondary bus.

2.20.1.9 G/A (Go-Around Button). A go-around button on each control wheel, labeled G/A, is used to initiate the go-around mode. Pressing the button illuminates the G/A annunciator light and causes the flight director indicator V-bars to command a wings-level, fixed-pitch climbout. The annunciator lights are on the pilot and copilot instrument panels (Figure 2-41). The go-around button and light are powered by the dc essential bus. Go-around may be selected anytime after the APPR mode is selected.

2.20.1.10 Autopilot System/Controls and Indicator. The AP105/FD-109 flight control system automatically controls the aircraft throughout the entire flight regime from immediately after takeoff, through cruise, to the minimum decision height. The flight control system presents a visual display to the pilot for use in monitoring the system when it is engaged or in manually controlling the aircraft in response to computed steering commands when the system is not engaged. The flight control system consists of a flight computer, autopilot system, cockpit displays, and the yaw damper system. The flight computer generates lateral and vertical steering commands from data received from the attitude reference, compass system, navigation systems, and air data system. The autopilot system positions the aircraft elevator and aileron control surfaces in response to flight computer steering commands or the pilot manual commands as selected on the controller. The yaw damper senses yaw accelerations and generates a corrective rudder command.

2.20.1.11 Autopilot Controller. The autopilot controller (Figure 2-41) is the control center for the autopilot and is part of the pilot equipment complement. It provides the autopilot engage and command functions. The engage lever is a dual section lever that engages or disengages the autopilot and yaw damper. As it is the master disconnect for both systems, the yaw damper may be engaged independently from the autopilot. Each section of the switch is independently solenoid held and spring loaded to the DISENGAGE position. The autopilot controller is also equipped with a pitch and turn control knob. When the autopilot is engaged, the pitch control supplies pitch commands to the autopilot system. The pitch control supplies a pitch rate command that is proportional to control displacement. Pitchdown is commanded when the control is rotated towards DN and pitchup is commanded when the control is rotated towards UP.

The pitch control is spring loaded to a center detent. When engaged, the autopilot maintains the aircraft pitch attitude that exists when the control is returned to the detent. Moving the pitch control causes the vertical mode (as selected on the flight mode selector) to release, placing the system in pitch hold mode.

The turn control supplies roll commands to the autopilot when no lateral modes are selected on the flight mode selector. The turn control is also spring loaded to a center detent. When the aircraft bank attitude is greater than 2° and the turn control is returned to the center detent, the autopilot holds bank attitude. When the aircraft bank attitude is less than 2° and the control is returned to the detent, the autopilot remains in heading hold.

2.20.1.12 Autopilot Amplifier. The autopilot amplifier positions the aircraft aileron and elevator servos in response to guidance signals from the flight computer or manual command from the autopilot controller. The autopilot amplifier is divided into two basic channels: aileron and elevator command. The amplifier accepts input data from the flight computer for automatic flight control and from the autopilot controller for coordinated manual control.

2.20.1.13 Aileron Servo. The aileron servo is connected to the aileron control system and accepts commands from the autopilot amplifier. The autopilot amplifier positions the aircraft aileron servo in response to guidance signals from the flight computer or manual command from the autopilot controller or airspeed sensor. A command signal from the flight control system energizes the aileron servo to correct the aircraft attitude. When the aileron is properly oriented, a feedback signal from the rate generator within the servo cancels the command signal. Voltage to the dc torque motor is then cut off and the aircraft stabilizes until the flight control system commands a change course.

2.20.1.14 Elevator Servo. The elevator servo principle of operation is similar to the aileron servo. The

autopilot amplifier provides an output that energizes the servo and drives the aircraft trim system to accomplish automatic elevator trimming. The trim circuit is functionally divided into two identical, electrically isolated channels (a trim-up and a trim-down channel). The two channels are interlocked to prevent simultaneous outputs of the trim-up and trim-down signals.

The yaw damper computer generates a rudder command signal from two inputs: yaw acceleration rate and roll data. Yaw acceleration rate is derived from a self-contained angular accelerometer. Roll data is derived from the vertical gyro. The angular accelerometer signal is applied to a mixing point where it is combined with filtered, gain-programmed (with indicated airspeed), and gain-adjusted roll data. The resulting rudder command signal is applied to the servo.

2.20.1.15 Air Data Controls (Nos. 1 and 2). The air data controls supply the altitude hold error signal, indicated airspeed, and altitude programming voltages to the flight control system. The air data control provides this data to the flight computer and to the yaw damper computer. The altitude hold mode can be selected after the aircraft has obtained the desired altitude. When the altitude hold mode is selected, the air data control provides an altitude hold error signal to the flight control system that indicates when the aircraft deviates from the selected altitude.

2.20.1.16 Vertical Gyros (Nos. 1 and 2). The vertical gyros are a source of attitude reference information used in integrated flight systems and automatic flight control systems. The vertical gyro provides pitch and roll data to the flight system. Fast erection of the gyro is automatic during starting. The angular displacement of the aircraft from the vertical reference is measured by means of pitch and roll synchro transmitters within the gyros. Selection of the vertical gyros is accomplished by the VG-1 and VG-2 switches (Figure 2-11) on the left side of the instrument panel shroud. Pitch and roll data are furnished to pilot flight director system from the selected gyro.

2.20.1.17 Airspeed Sensor. The airspeed sensor senses airspeed and provides a signal representing the sensed airspeed to the automatic flight control system for continuous gain changing. The signal provided by the airspeed sensor changes the gain of the automatic flight control circuits. The gain change will control the magnitude of the command signals generated by the automatic flight control system. This control is required to provide greater signals at slower rates of speed since a greater deflection of control surfaces are required for aircraft maneuvering and to provide lesser signals to prevent maneuvering the aircraft in a hazardous manner while flying at higher rates of speed.

2.20.1.18 Rate-of-Turn Sensors (Nos. 1 and 2). The rate-of-turn sensors provide a rate-of-turn and monitor signals to the rate-of-turn pointer and flag in the flight director indicator. This signal is derived from an internal gyro that supplies a voltage proportional to the aircraft rate of turn. The rate-of-turn flag in the flight director indicator will appear when the internal gyro revolutions per minute drops to 80 percent of its full rated speed.

2.20.1.19 Mode Couplers (Nos. 1 and 2). The mode couplers integrate the flight director and autopilot systems. It performs interface and mode selection functions necessary for a complete flight control system. The mode couplers supply the flight control system with back localizer switching and annunciation, steering command analog adjustment, vertical mode pulse, attitude power, and vertical mode selection power.

2.20.1.20 Flight Computers (Nos. 1 and 2). The flight computer furnishes vertical and lateral steering computation for the autopilot system. The computers derive vertical and lateral steering commands from inputs supplied by the vertical gyros, navigation receivers, and magnetic compass system. With the flight director system in the HDG mode, the computer furnishes computed steering information to fly to and maintain the selected heading. In this mode of operation, heading and roll input signals are supplied by the compass system and the selected vertical gyro. In the ALT mode of operation, the computer maintains the altitude at the time the ALT mode was selected. If a new attitude is desired, the aircraft is positioned to the new attitude and the PITCH SYNC button is depressed. The computer will provide the necessary signals to the flight director system to maintain the new attitude. The No. 1 flight computer is powered by the 115-vac primary bus and the No. 2 flight computer is powered by the 115-vac secondary bus.

2.20.1.21 Pitot-Static Shutoff Switches. These switches (FO-5) on the right console control the pitot and static air supply to the autopilot system. With the A/S PITOT and A/S STATIC switches at the guarded ON position, the autopilot airspeed sensor and altitude controllers (air data controls) are supplied from the copilot normal pitot-static source (Figure 2-37). Raising the guard and moving the pitot shutoff or the static shutoff switch of OFF shuts off the respective system to the autopilot. These switches should be moved to OFF only when the respective pitot or static system is malfunctioning and it is suspected that the fault lies in the autopilot system.

2.20.2 Operation of Integrated Flight and Autopilot Systems

2.20.2.1 Disengagement of Autopilot. In case of erratic or uncontrollable autopilot maneuvers, the autopilot can be disengaged by any of the following methods:

1. Pressing the AUTOPILOT DISC button on the pilot and copilot control wheel.
2. Moving the autopilot ENGAGE lever to the DISENGAGE position.
3. Depressing the G/A button on the control wheels (only when in the approach mode).
4. Pulling (open) the autopilot ac or dc circuit breaker.
5. Any interruption or failure of power.

If necessary, the autopilot may be overpowered by either pilot. The following conditions will cause the autopilot to disengage automatically:

1. Vertical gyro failure
2. Autopilot power or circuit failure.

The autopilot shall be disengaged for takeoff and landing. When using the autopilot, the pilot or copilot shall be in his seat with his seatbelt fastened.

2.20.2.2 Preflight Check. Before beginning the check, turn on aircraft power, check that the vertical gyro is erected (GYRO flag out of view), and that the gyro magnetic compass is slaved. Set the heading marker on the course indicator under the lubber line and press the HDG button of the mode selector.

Note

The pressure of airflow that normally opposes movement of control surfaces is absent during the preflight check. It is possible to get a hardover control surface deflection if an autopilot command is allowed to remain active for any appreciable time. If it is desired to check operation of the turn or pitch controls, move them only as required to check control operation and then return them to the center position.

1. Engage the autopilot. Check that the controls resist movement.
2. Push forward on the control column and hold. After a few seconds, the trim annunciator should light.
3. Press the TRIM DN button while holding force on the control column in the nosedown direction. Observe that the trim annunciator lights. Press the TRIM UP button while holding force on the control column in the noseup direction. Again the AP TRIM annunciator should illuminate.
4. Press the AP disconnect button on the control column. Observe that the engage lever moves to the DISENGAGE position and that the flight controls operate freely.
5. Engage the autopilot. With the navigation receiver tuned to a localizer frequency, press the APPR button, then press the G/A button. Observe that the autopilot disengages and the command bars move to command a wings-level, 10° pitchup attitude.

2.20.2.3 Takeoff. Check that autopilot is disengaged. Press HDG button on the mode selector and set the heading marker on the course indicator to the runway heading. Establish climb airspeed and begin the climbout to cruise altitude. After climb is established, press the PITCH SYNC button on the control wheel. The command bars then command flight to the pitch attitude at which the PITCH SYNC button was pressed. A lateral mode must be selected to obtain PITCH SYNC.

2.20.2.4 Climbout. Engage the autopilot. Make heading changes by moving the heading marker on the course indicator. The flight control system maneuvers the aircraft in a smooth, coordinate turn to the new heading.

2.20.2.5 Cruise Altitude. Just before reaching cruise altitude, lower vertical speed to less than 500 feet per minute. When on a cruise altitude, press the ALT button. The flight control system then commands the aircraft to maintain the selected barometric altitude.

Note

The aircraft may overshoot the selected altitude by as much as 10 percent of the vertical speed.

2.20.2.6 VOR and Tacan. With the system operating in HDG mode, turn the navigation or tacan receiver to

the desired VOR station. Depress the applicable navigation selector (V-L DME or TACAN). Check the PLT and C PLT NAV SEL FAIL annunciator lights to verify that proper switch has taken place. Set the course arrow on the course indicator to the desired course. Select NAV/LOC mode on mode selector; the N/L ARM annunciator lights, indicating the system is armed for VOR or tacan radial capture. After radial capture, the system maintains automatic crosswind correction and the N/L ARM light goes out and the N/L ENG annunciator lights, indicating the system is tracking the VOR or tacan radial.

Note

- The course may be changed over a VOR station when operating in NAV/LOC mode as long as the course change is not more than 10°. If the course change is more than 10°, the HDG mode should be selected to establish a new intercept and then NAV/LOC mode reselected to set up a new capture.
- Limit all capture angles to 90° or less.
- When flying inbound toward a VOR, do not set up a VOR capture within approximately 15 miles of the VOR (depending on the speed of the aircraft). The system does not have time to compute the new capture point and stabilize on course before passing over station.

If it is desired to manually place the aircraft on a VOR radial, select VOR/LOC mode after the course indicator lateral deviation bar indicates one bar width deflection or less. The system immediately goes to N/L ENG and begins to track the center of the beam.

When passing a VOR station, bank commands are limited to 3° or less. If large error is shown on the deviation bar after passing a VOR station, corrective commands are limited to 8° or less. If faster corrections are desired, select a new intercept heading and then select NAV/LOC mode. This puts the system back in N/L ARM and a new capture cycle is commanded.

Use the course indicators to make a position fix. Leave the pilot flight director system tracking in the NAV/LOC mode; select HDG mode on the copilot mode selector; tune the copilot NAV receiver to the position fix VOR; select the position fix radial on the copilot course indicator. When the copilot lateral deviation bar is centered, the aircraft is over the position fix. Retune the copilot navigation receiver to the en route VOR; select the en route radial on the copilot course indicator; select NAV/LOC mode.

2.20.2.7 Holding. To establish a holding pattern over the outer marker or VOR intersection, select HDG mode on the mode selector. Tune the navigation receiver to the inbound VOR or localizer frequency. Set the course arrow to the inbound or localizer course. Maintain flight to the holding point by adjusting the HDG marker. When the aircraft reaches the holding point, turn the heading marker 135° in the direction of the outbound turn. After completing 90° of the turn, continue moving the heading marker until the reciprocal heading of the inbound course is reached. If crosswind correction is needed, adjust the heading marker for the appropriate crab angle.

After the required time or distance has elapsed while tracking the outbound heading, set the heading marker 135° in the direction of the inbound turn. After completing 90° of the turn, continue moving the heading marker to the inbound localizer with crosswind correction. If automatic capture and tracking of the inbound radial is desired, select NAV/LOC of APPR mode after 90° of the inbound turn has been completed.

2.20.2.8 Letdown. To fly a typical VOR letdown, track into the station in NAV/LOC mode. The TO-FROM pointer will reverse at station passage and the course arrow may be set to the published outbound heading at this time. If the VOR is approached from a heading that requires maneuvering to the outbound leg, select HDG mode and use the HDG marker to alter the aircraft course, then select HDG when the outbound leg is completed. Adjust the HDG marker 135° or less in the direction of the 180° turn. Set the course arrow to inbound radial. Continue adjusting the HDG marker until the 180° turn is completed. If continuing the letdown during the turn, use the pitch knob to set the desired rate of descent. When the 180° turn is completed, select NAV/LOC on the mode selector. Set up the desired rate of descent with the pitch knob on the autopilot controller.

Any crab angle that exists is displayed on the course indicator as the difference between the course arrow and the lubber line.

2.20.2.9 Front Course Approach. On an ILS front course approach, the localizer and glideslope are automatically and independently captured. The localizer is always captured from a selected heading, but the glideslope can be captured from the vertical mode and from above or below the glideslope. To perform a front course approach, set the heading marker to the desired intercept heading. Tune the navigation receiver to the

localizer frequency and set the course arrow to the published inbound course. The runway symbol at the bottom of the flight director indicator appears when the localizer frequency is tuned and displays localizer deviation and altitude from 200 feet to touchdown provided the radio altimeter is operating. The HDG, NL ARM, and GS ARM annunciators illuminate to show the system is armed for ILS capture. As the aircraft nears the localizer, the HDG and NL ARM annunciators go out and the NL CAP annunciator illuminates. The aircraft makes a smooth coordinated rollout on the localizer.

Note

The published go-around heading may be set at this time with the HDG marker on the course indicator. The Landing checklist should be completed prior to glideslope intercept.

When the glideslope is captured, the GS ARM annunciator goes out and the GS CAP annunciator illuminates. The vertical mode will automatically release at glideslope capture. All steering commands are to maintain the center of the localizer and glideslope. The decision height annunciator illuminates at preset radio altitude. The decision to land or go around must be made at this time.

2.20.2.10 Go-Around. If the aircraft reaches decision height and the DH annunciator light illuminates prior to visual contact with the airport environment, execute a missed approach maneuver utilizing the go-around mode.

Select the go-around mode by pressing the go-around mode button on the control wheel. The GA annunciator illuminates and a wings-level, 10° fixed-pitch climbout is displayed by the command bars.

The missed approach heading is preset with the heading marker. Switching the mode selector to the HDG mode at the completion of the go-around procedure immediately gives roll commands to turn to the preset missed approach heading.

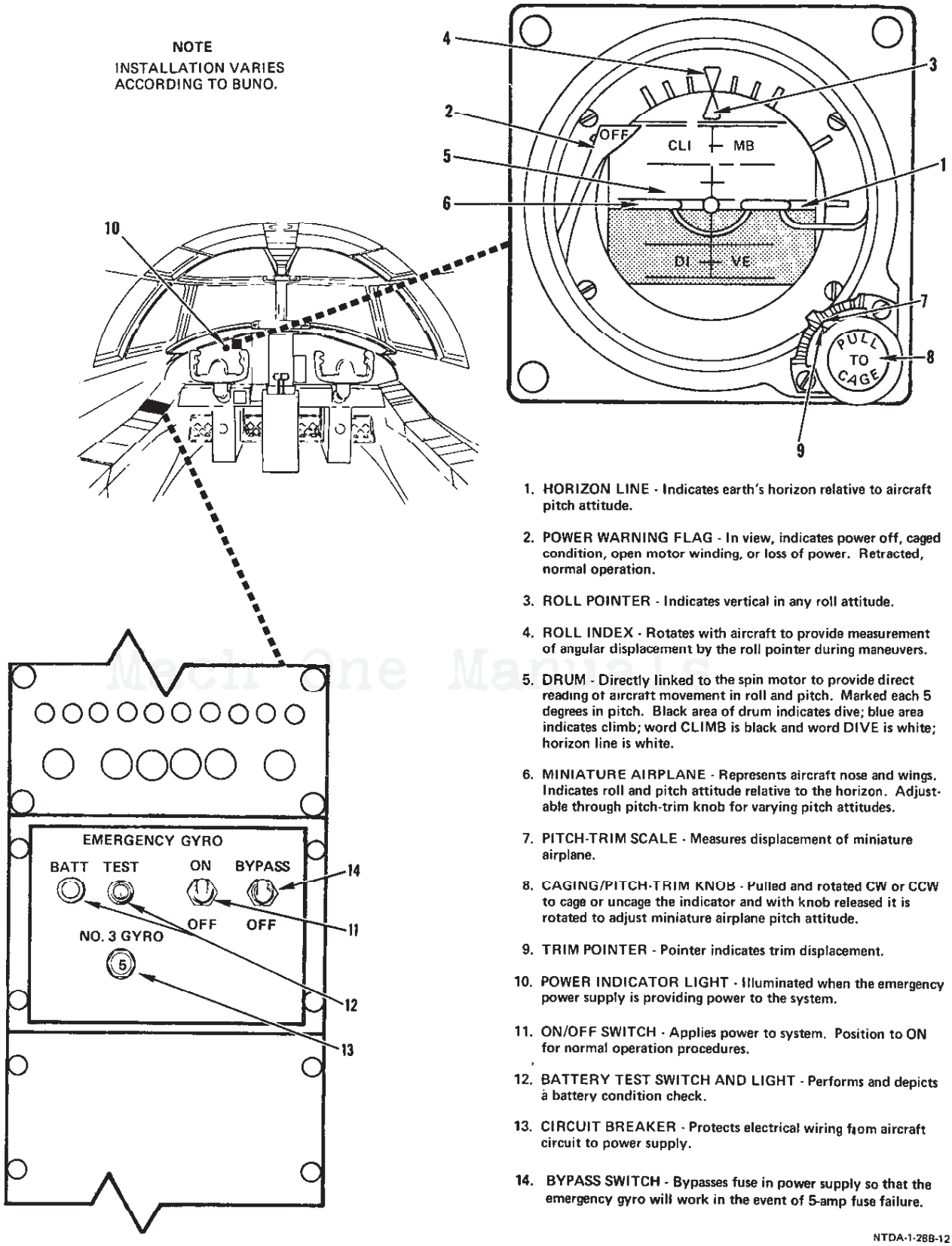
The pitch control on the pedestal controller may be used to maintain a constant climb to the published missed approach altitude.

2.20.2.11 Back Course Approach. Set the course arrow to the inbound front course bearing and the heading marker to the desired intercept heading, then select HDG mode and APPR or NAV/LOC on the mode selector. When the aircraft heading is within 75° of the back course localizer bearing, the system automatically switches to back course operation and the REV LOC annunciator illuminates. The NL ARM annunciator light will come on. At localizer capture, the NL ARM light will go off and the NL ENG light will illuminate. The flight director indicator runway symbol provides normal indications during the back course approach. The go-around operation is the same as in a front course approach.

2.20.2.12 Standby Attitude System. The standby attitude system (Figure 2-42) provides aircraft attitude in the event of a power and/or flight director indicator failure. The system consists of a two-position switch (placarded ON-OFF) and an AI804 vertical reference indicator located on the pilot instrument panel and a PS823BT emergency power supply located behind the pilot seat. See Figure 2-42.

2.20.2.13 Emergency Power Supply. The emergency power supply consists of a dc to ac inverter, a battery pack, and a battery charger. In normal operation, the inverter and battery charger are supplied by the aircraft 18-vdc power system. If the aircraft power fails, the battery pack supplies dc power to the inverter. The inverter converts the dc power to 115-vac to operate the indicator with the standby attitude switch in the ON position. In the event of an aircraft electrical failure, the emergency power supply ON light on the instrument panel will illuminate anytime the emergency power supply is providing electrical power to the indicator. A fully charged battery pack will provide power to the gyro for 4 hours. The gyro rotor will continue to operate the indicator providing useful attitude information for a minimum of 9 minutes in the event of an electrical power failure of the emergency power supply.

2.20.2.14 Vertical Reference Indicator. The vertical reference indicator displays aircraft pitch and roll attitude. Pitch indications range from 92° climb to 78° dive, marked every 5°. Roll indications are presented through 360° with left and right bank 10° indices through 30° and major indices at 30°, 60°, and 90°. A flag labeled OFF appears in the event of power loss and when the PULL TO CAGE knob is in the caged position. The PULL TO CAGE knob is located on the lower right-hand corner of the instrument. To cage the gyro, pull the knob and rotate it clockwise. Uncaging is accomplished by pulling the knob and rotating it counterclockwise. The desired pitch axis presentation can be obtained by adjusting the knob in the appropriate position. The indicator does not normally require any adjustments during flight. However, in the event of errors caused by extended steep maneuvers, the aircraft should be brought to level flight and the indicator should be momentarily caged.



1. HORIZON LINE - Indicates earth's horizon relative to aircraft pitch attitude.
2. POWER WARNING FLAG - In view, indicates power off, caged condition, open motor winding, or loss of power. Retracted, normal operation.
3. ROLL POINTER - Indicates vertical in any roll attitude.
4. ROLL INDEX - Rotates with aircraft to provide measurement of angular displacement by the roll pointer during maneuvers.
5. DRUM - Directly linked to the spin motor to provide direct reading of aircraft movement in roll and pitch. Marked each 5 degrees in pitch. Black area of drum indicates dive; blue area indicates climb; word CLIMB is black and word DIVE is white; horizon line is white.
6. MINIATURE AIRPLANE - Represents aircraft nose and wings. Indicates roll and pitch attitude relative to the horizon. Adjustable through pitch-trim knob for varying pitch attitudes.
7. PITCH-TRIM SCALE - Measures displacement of miniature airplane.
8. CAGING/PITCH-TRIM KNOB - Pulled and rotated CW or CCW to cage or uncage the indicator and with knob released it is rotated to adjust miniature airplane pitch attitude.
9. TRIM POINTER - Pointer indicates trim displacement.
10. POWER INDICATOR LIGHT - Illuminated when the emergency power supply is providing power to the system.
11. ON/OFF SWITCH - Applies power to system. Position to ON for normal operation procedures.
12. BATTERY TEST SWITCH AND LIGHT - Performs and depicts a battery condition check.
13. CIRCUIT BREAKER - Protects electrical wiring from aircraft circuit to power supply.
14. BYPASS SWITCH - Bypasses fuse in power supply so that the emergency gyro will work in the event of 5-amp fuse failure.

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Figure 2-42. Standby Attitude Indicator

2.20.2.15 Battery Test Switch and Light. The battery test switch and light are mounted on the left console panel (see Figure 2-42) or the pilot instrument panel. The switch and light provide a method to determine the condition of the batteries within the power supply. With the switch depressed and after approximately 1.5 seconds, the light will illuminate indicating that the batteries are charged to the proper level. If the light does not illuminate, the power supply should be replaced or repaired as required prior to flight.

CAUTION

The emergency battery test should only be made with a ground power unit attached to the aircraft or an aircraft generator operating. Testing the system with only aircraft batteries on the essential bus may damage the emergency power supply and the test light will not illuminate.

2.20.2.16 Emergency Gyro ON-OFF Switch. The emergency gyro ON-OFF switch is mounted on the left console panel (see Figure 2-42) or the pilot instrument panel. The switch is used to apply electrical power to the system. During normal operating procedures, the switch shall be in the ON position. With the switch in the ON position and in the event of a failure of the aircraft electrical system, the power supply will continue to operate the indicator without interruption of the attitude information.

2.20.2.17 Operation of Standby Attitude System. To operate the standby attitude system, proceed as follows:

1. Aircraft electrical system — Energize.
2. Emergency gyro switch — ON.
3. Caging knob — Subsequent to a 1-minute warmup period, pull caging knob and rotate clockwise to cage the gyro. Turn knob counterclockwise to uncage the gyro. Slowly release caging knob.
4. Warning flag — Out of View.
5. Caging knob — Rotate as required to obtain the desired pitch axis presentation.

2.20.2.18 Directional Indicating System (CT-39G Aircraft 158844 and 160053 Through 160057). Two independent MC-103 magnetic compass systems (Nos. 1 and 2) are installed in the aircraft. The systems combine the rigidity characteristic of a directional gyro and the ability of a flux detector to sense magnetic north to produce a stable, magnetic heading signal. Each system consists of a flux detector, a 328A-5 compass amplifier, a 332E-5 directional gyro, and a 332C-8 RMI.

The systems provide magnetic information to the pilot and copilot course and RMI indicators.

The center of the MC-103 magnetic compass system is the 328A-5 compass amplifier. The primary function of the compass amplifier is to process and distribute magnetic heading information. The compass amplifier also monitors the system components, provides power and excitation voltages, and contains self-test circuits. The compass amplifier is located in the nose avionics section of the aircraft and the No. 1 amplifier is powered by the 115-vac bus and the No. 2 amplifier is powered by the 115-vac secondary bus.

The 332E-5 directional gyro and the flux detector are the data sources for the system. The directional gyro is located in the fuselage nose section and the flux detector is located on the wingtip (the flux detector for No. 1 system is on the left wingtip and No. 2 system is on the right wingtip). The directional gyro provides the short-term portion of the compass data output and the flux detector provides the long-term portion. The flux detector couples the magnetic compass to the Earth's magnetic field and conveys magnetic flux data to the compass amplifier. The directional gyro azimuth reference is compared to the flux detector output. If an error exists, the error signal is fed to the compass amplifier, phase-detected and used to control the slaving torque motors in the gyro. If the automatic slaving mode has been selected, the directional gyro error is corrected at a rate depending upon whether the system is being fast or slow slaved. Fast slaving occurs within 10 seconds after power is applied or when D/G mode is momentarily selected. Slow slaving is automatically initiated whenever the directional gyro error has been corrected. The phase-detected error signal is also amplified and fed to the slave indicator on the RMI. The slave indicator shows whether or not an error exists between the directional gyro and the flux detector.

The 332C-8 RMI (Figure 2-41) serves as the heading information readout for the system and permits manual operation of the system. Magnetic heading information is presented on the rotatable compass card.

The position of the mode selector determines the system mode of operation. When the mode selector is

depressed (normal position), the system is in the automatic slaving mode. To select DG mode, the mode selector is pulled away from the indicator front panel. To select manual slaving mode, the selector is rotated clockwise or counterclockwise. If the slave indicator indicates an error in the + direction, the mode selector should be rotated counterclockwise to slave the directional gyro in the negative (indicated by a dot) direction. If the slave indicator indicates an error in the negative direction, the mode selector should be rotated clockwise to slave the directional gyro in the + direction. Having been rotated, the mode selector will return automatically to the center position.

In the D/G mode operation, the slaving control circuits are disabled and the gyro is used as a free drift reference source. The compass data output then reflects the position of the gyro azimuth gimbal without reference to the Earth's magnetic field. This mode of operation is valuable in polar regions or other areas where coupling to the Earth's magnetic field would create confusing or inaccurate compass data.

The four-position selector switch (Figure 2-41) on the bottom right corner of the indicator determines the information to be indicated by the double and single bar pointers. In one position, ADF bearing data is indicated by both pointers. In another position, ADF bearing data is indicted by the double bar pointer and VOR bearing data is indicated by the single bar pointer. In another position, VOR is indicated by the single pointer and TAC is indicated by the double bar pointer. In another position, ADF is indicated by the single pointer and TAC is indicated by the double bar pointer. Additional monitoring of the RMI is accomplished within the RMI by a monitoring circuit that checks for valid signals from the compass amplifier. The compass flag on the RMI displays the results of the RMI monitoring. When the COMPASS flag is in view, the compass data is invalid.

The COMP-1 and COMP-2 selector switches are located on the left side of the instrument panel shroud (Figure 2-41). With COMP-1 selected, the No. 1 compass amplifier heading signals are supplied to the No. 1 course indicator, RMI, navigation receiver, and autopilot and tacan systems. With COMP-2 selected, the No. 2 compass amplifier heading signals are supplied to the No. 2 course indicator, RMI, navigation receiver, autopilot and tacan systems and all the No. 1 system components.

2.20.2.19 Course Indicator (Horizontal Situation Indicator). The course indicator is an aircraft navigation unit that displays a pictorial view of an aircraft with respect to magnetic north, selected course, and selected heading. Selected heading, selected course, and relative bearing are read against a servo-driven azimuth card. A digital DME display and course display are provided. Meter movements display course deviation, to-and-from indication, and glideslope deviation. Warning flags monitor the azimuth card and navigation and glideslope signals. A stationary aircraft symbol presents a pictorial display of aircraft position and heading.

2.20.2.20 Controls and Indicators. The aircraft symbol shows aircraft position and heading with respect to the azimuth card, lateral deviation bar, heading marker, and bearing pointer. The rotating azimuth card displays heading information from the magnetic compass system. Aircraft heading is read on the compass card beneath the lubber line at the top of the instrument. The orange heading marker is set to the desired magnetic heading as read on the azimuth card. The course arrow is set to the selected VOR/TAC or localizer course. The course deviation bar represents the centerline of the selected VOR/TAC or localizer course and depicts a right or left deviation. A digital readout on the upper left corner displays the distance to go to the selected station in nautical miles. The glideslope pointer represents the center of the glidepath and displays aircraft position above or below the glideslope. When the pointer is above the glideslope scale centerline, the aircraft is below the glideslope. The pointer is in view only when the navigation receiver is tuned to a localizer frequency. The TO-FROM arrow indicates the direction toward the VOR or tacan station. The arrow is not visible when a localizer frequency is selected.

2.20.2.21 Course Indicator Warning Flags. The heading flag indicated an unreliable heading signal. The navigation flag indicates an unreliable navigation signal, either VOR/LOC or TAC. The GS flag indicates an unreliable glideslope signal. The GS warning flag circuit is energized only when the navigation receiver is tuned to a localizer frequency. The miles display blank out and flag out indicates an unreliable distance signal.

2.20.2.22 Remote Heading and Course Selector. The remote heading and course selector (Figure 2-42) controls two course indicators. The selector is located on the center pedestal. The HEADING control drives both the pilot and copilot course indicator heading markers. The COURSE NO. 1 control drives the pilot course indicator course arrow. The COURSE No. 2 control drives the copilot course indicator course arrow. The remote heading and course selector is powered by the aircraft 115-vac primary bus. Clockwise rotation of the knobs rotates the course arrows of the pilot and copilot course indicators in the increased direction. Clockwise rotation of the HEADING knob

rotates the heading markers of the course indicators in the clockwise direction.

2.20.3 Directional Indicating System (CT-39E, CT-39G Aircraft 158843 and 159362 Through 159365). The MC-102 directional indicating system is a gyro-stabilized magnetic compass system that supplies heading information for the integrated flight system. The system includes a remote flux detector in each wing, compass amplifiers, directional gyros, and a gyrocompass mode switch and slaving indicator for the pilot and copilot. Directional information is transmitted from each gyro to the associated course indicator and radio magnetic indicator compass cards. The system is powered by the dc essential bus, the ac primary bus, and the ac secondary bus.

2.20.3.1 Directional Indicating System Controls and Indicators

2.20.3.1.1 Gyrocompass Mode Switch. This guarded switch (1 and 21, Figure 2-39) on the pilot and copilot instrument panels has four positions. With the guard down, the switch is at the maintained AUTO position and the automatic mode (gyro-stabilized magnetic compass) is in operation. Here, the directional gyro is slaved to the flux detector and the aircraft magnetic heading is presented on the compass cards. When the guard is raised and the switch is moved to the UP FOR D/G position, the system operates in the directional gyro mode and functions as an uncorrected directional gyro free of the flux detector. Raising the guard and holding the switch at the spring-loaded DECR HDG of INCR HDG position manually slaves the gyro to the flux detector in a fast-slaving cycle of 2° to 6° per second. After the gyro is properly slaved to the flux detector, the system automatically reverts to a slow-slaving cycle of 2° to 3° per minute. When the switch is released from the DECR HDG of INCR HDG position and is returned to the AUTO position, the system will fast-slave until synchronized, then revert to the slow rate.

2.20.3.1.2 Compass Slaving Indicator. The compass slaving indicator (22 and 34, Figure 2-39) on the pilot and copilot instrument panels is used in the magnetic mode to show whether the directional gyro is properly slaved to the flux detector (pointer centered), the amount of error, and the direction in which such error exists. Displacement of the pointer from center indicates the magnitude and direction of azimuth correction required to bring the directional gyro and the flux detector into synchronization. Full-scale pointer displacement to the left toward the dot or to the right toward the + sign indicates about 10° of error between gyro and flux detector. If the indicator pointer moves to the plus side, the mode switch must be held at DECR HDG to center the pointer. If the pointer moves to the dot side, the switch must be held at INCR HDG to center the pointer. A slight midscale oscillation of the pointer indicates proper operation of the directional indicating system.

2.20.3.1.3 Compass Selector Switch. This rocker-type switch (8, Figure 2-39) on the pilot instrument panel is labeled COMP SEL. The switch is used in selecting the MD-102 directional indicating system directional gyro, the source of heading information for the course and radio magnetic indicator compass cards.

When the NO. 1 position of the switch is pressed, the pilot systems are operating from the No. 1 directional gyro and the copilot systems are operating from the No. 2 directional gyro. When the NO. 2 position of the switch is pressed, the pilot and copilot systems are both operating from the No. 2 directional gyro. If a directional gyro failure is indicated by the appearance of the COMPASS warning flag on the course indicator, the selector switch should be moved to the alternate position to select the operable directional gyro. The compass selector switch is powered by the dc essential bus and the ac primary bus.

2.20.3.1.4 Course Indicator and Radio Magnetic Indicator. The compass card on each of these indicators (25 and 32, Figure 2-39; Figure 2-40) on the pilot and copilot instrument panels reflects directional information transmitted from the associated directional gyro. For additional information on these indicators, refer to paragraph 2.18.1, Integrated Flight System Controls and Indicators.

2.20.3.1.5 Operation of Directional Indicating System in Directional Gyro Mode. The D/G mode is designed for use at latitudes above 75° for grid navigation where magnetic compass indications are unreliable. Gyro "induced" precession rate, "apparent" precession from Coriolis force (Earth's rotation effect), and meridian convergency will affect the accuracy of navigation in the polar latitudes. It is necessary to know grid navigation and the use of the sextant to obtain reliable headings in polar latitudes.

"Induced" precession rate can be a maximum of 2° per 15 minutes. The significance of precession can be shown by the following example: Assume a constant heading for 1 hour at a ground speed of 400 knots. Also assume the "induced" precession rate of 1 hour is

5°. At the end of the hour, it is possible to be about 34 nm off course because of precessional error alone. For this reason, the gyro should be rated for at least 10-minute legs on the intended headings and the average precession applied to reduce this error.

The error because of Coriolis force ("apparent" precession) is equal to 15° per hour times the sine of the latitude. Thus, at a latitude of 80°, the error would be 15° times 0.98481 or approximately 14°, 46 minutes per hour. This force moves the aircraft to the right of course in the Northern Hemisphere and to the left in the Southern Hemisphere. This aircraft does not have a latitude balancing mechanism to eliminate apparent precession.

The convergency of successive meridians crossed results in an error that is a function of groundspeed, ground track, and flight latitude.

The sum of the errors caused by Coriolis force, precession, and convergency can be significant during prolonged flight in the D/G mode. Consequently, it is important to frequently verify position by visual reference, radar fix, and/or celestial fix. The system can then be slaved to a corrected reading.

In latitudes where the magnetic compass is reliable, the D/G mode may be used to maintain accurate heading should both flux detectors fail. This is accomplished by slaving to the heading obtained from the magnetic compass after applying compass card correction and gyro precession, if known.

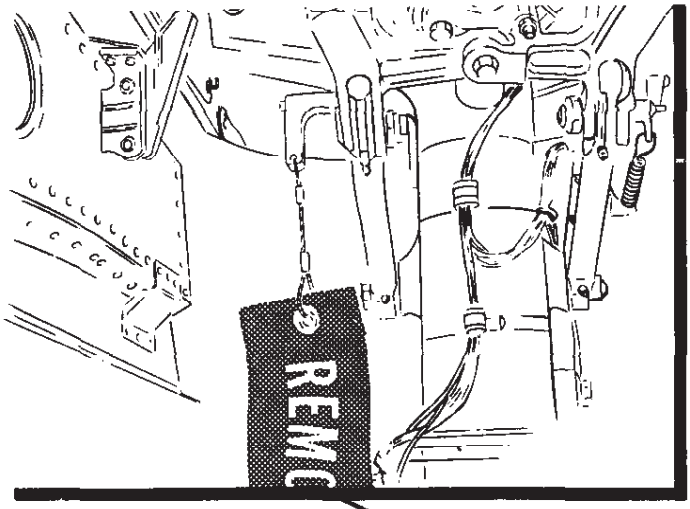
2.20.3.1.6 Emergency Operational of Directional Indicating System. Failure of the directional indicating system in the automatic mode will be indicated by erratic presentation of the course indicator or radio magnetic indicator compass card, "freezing" of the compass cards, or the compass slave indicator showing other than a nominal center position. If failure is indicated, it may be possible to use the system in the D/G mode. When the D/G mode is selected, a check should be made to see if the compass cards respond to turns. The heading information on the compass cards should be checked periodically against the slaving indicator and the system manually slaved as necessary to correct gyro drift errors. The compass card indications should also be checked against the magnetic compass to determine whether the system gyro is properly slaved to the flux detector.

2.21 LANDING GEAR SYSTEM

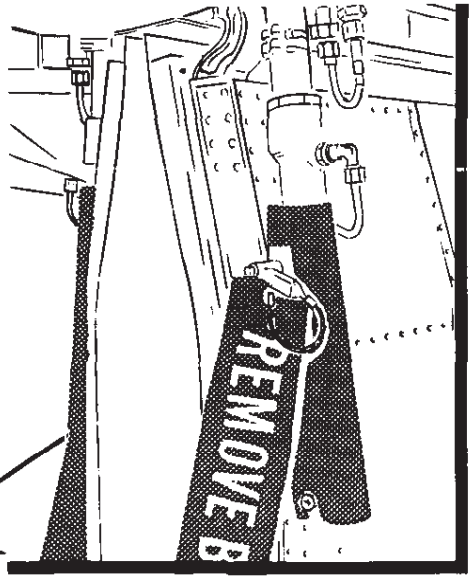
The fully retractable tricycle landing gear is electrically actuated and hydraulically operated. The nosegear doors are mechanically controlled by the operation of the nosegear. The main landing gear wheelwell cover doors are electrically sequenced and hydraulically actuated. The main gear retracts inward into the lower surface of the wing; the dual-wheel nosegear retracts forward into the fuselage. The wheelwell doors fair the gear flush with the aircraft contour when the gear is retracted and cycle closed after the landing gear is extended. A safety lock prevents normal landing gear handle operation while the aircraft is on the ground. This lock can be overridden if necessary. An emergency means of lowering the gear in case of hydraulic or electrical failure is provided. Nosewheel steering is accomplished by a nosewheel steering cylinder attached to the aft side of the nosegear strut. Nosewheel shimmy is eliminated by corotation of the dual nosewheels and the installation of a balance weight mounted between the nose landing gear tires. The main wheels are equipped with hydraulically operated, dual-disk type brakes and an antiskid system. Mechanical actuated ice breaker pins are installed on the main landing gear downlock switch linkage.

2.21.1 Landing Gear Maintenance Safety Locks. Removable maintenance safety locks (Figure 2-43) may be installed in the main and nosegear assemblies during maintenance operations. These locks are to be used when the aircraft is on jacks to protect maintenance personnel from an inadvertent gear retraction when work is being performed in the wheelwells. Installation of the maintenance safety locks will not ensure a gear-safe condition when the weight of the aircraft is on the gear. The locks have red warning streamers. All gear safety locks must be removed before flight.

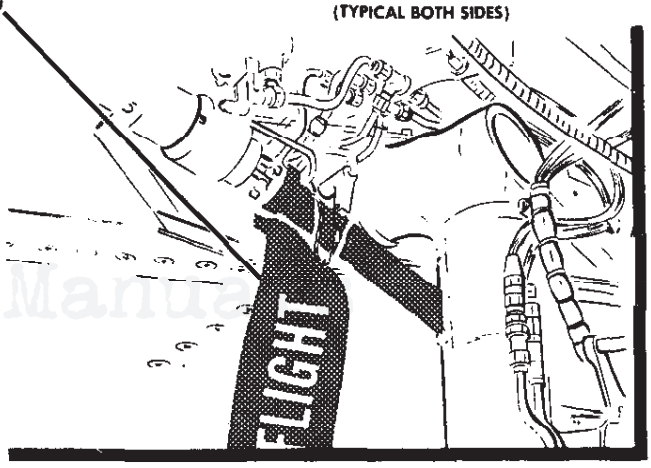
2.21.2 Landing Gear Handle. The landing gear handle on the landing gear control panel (Figure 2-44) electrically controls the gear and gear door hydraulic selector by dc essential bus power. When the aircraft is airborne, moving the handle to UP positions the landing gear selector valve so that hydraulic pressure pulls the landing gear downlocks and retracts the landing gear. The main gear wheelwell doors are electrically sequenced and hydraulically powered to close when each main gear is in its uplock. The nosegear doors are locked open by means of an overcenter action of the linkage between the doors. As the nosegear retracts, the shock strut piston strikes a cam plate to break the overcenter condition and mechanically closes the doors behind the gear. Once the uplock hooks close on the strut rollers and the wheel doors are closed, the hydraulic pressure in the landing gear system is deenergized and the uplock hooks are held in place by means of an overcenter linkage and spring.



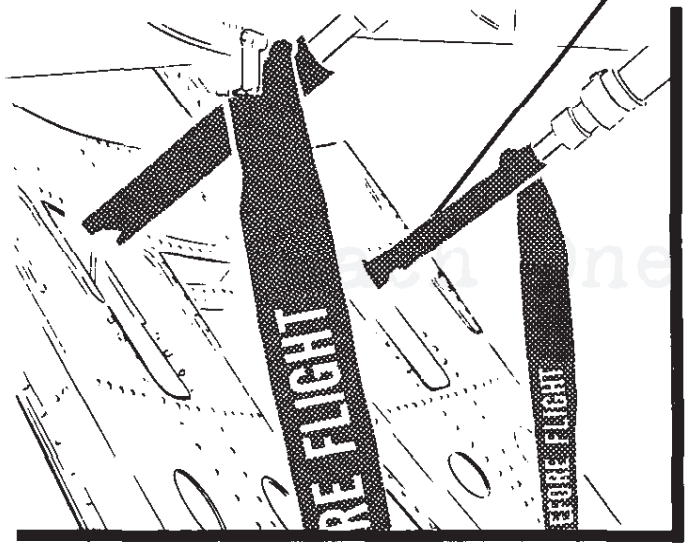
NOSE GEAR



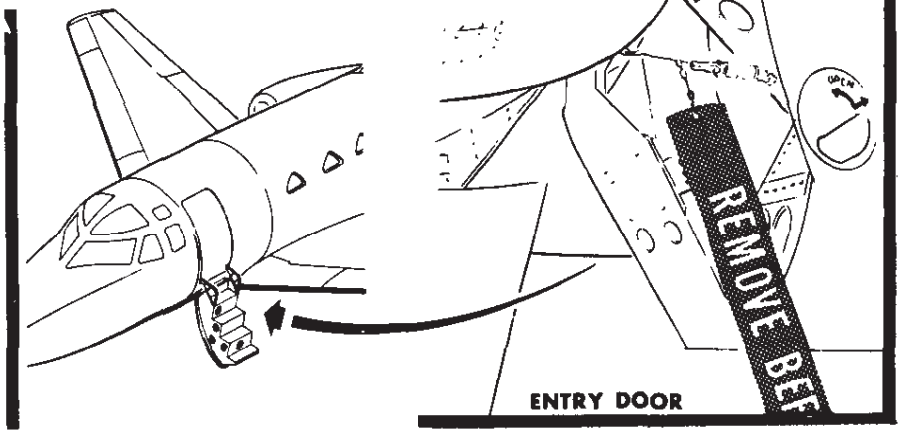
MAIN GEAR DOORS
(TYPICAL BOTH SIDES)



MAIN GEAR
TYPICAL BOTH SIDES)



SPEED BRAKE

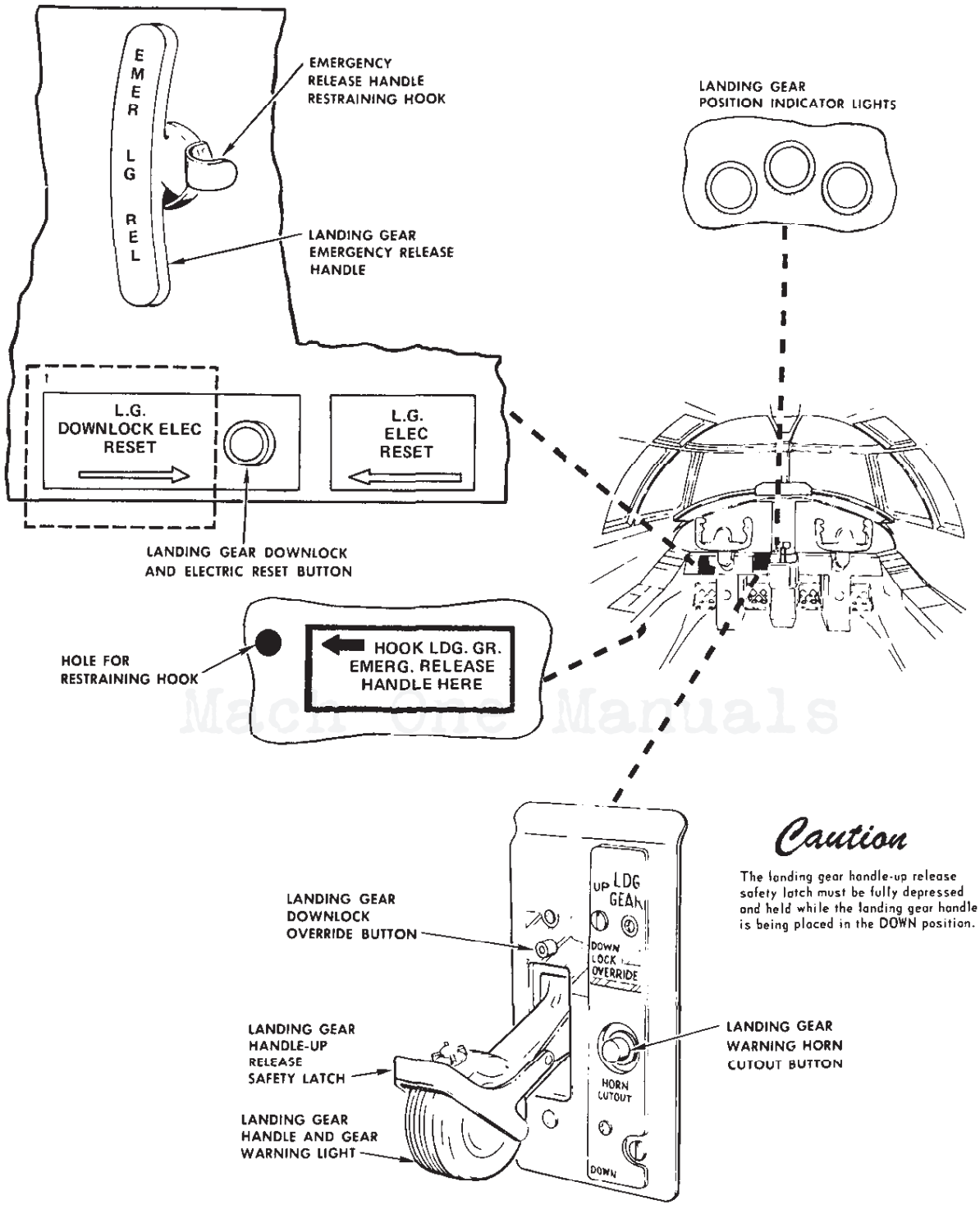


ENTRY DOOR

WARNING

- Make sure all maintenance safety locks are removed before flight.
- If any maintenance safety lock is installed, do not remove it until checking status of the system with maintenance personnel.

39E-1-73-5



T-39E Aircraft

39E-1-33-2A

Figure 2-44. Landing Gear Controls Indicators

CAUTION

When the weight of the aircraft is on the main gear, a mechanical lock prevents the landing gear handle from being moved to UP. The handle can be placed at UP by pressing the downlock override button. When the gear handle is at UP and with hydraulic and electrical power on the aircraft, the landing gear will retract on the ground.

The landing gear handle incorporates an external release safety latch. The latch controls a mechanical lock that, when engaged, maintains the landing gear handle in the UP position and prevents inadvertent movement of the handle by accidental bumping or by an aircraft g-load condition. The latch must be depressed when the handle is up in order to retract the lock and permit movement of the handle to the DOWN position.

CAUTION

The latch or button must be fully depressed before the handle is moved from the UP position and held depressed until the handle is full down. This is necessary to prevent damaging or breaking the handle and to prevent unnecessary wear on the latching or locking mechanism.

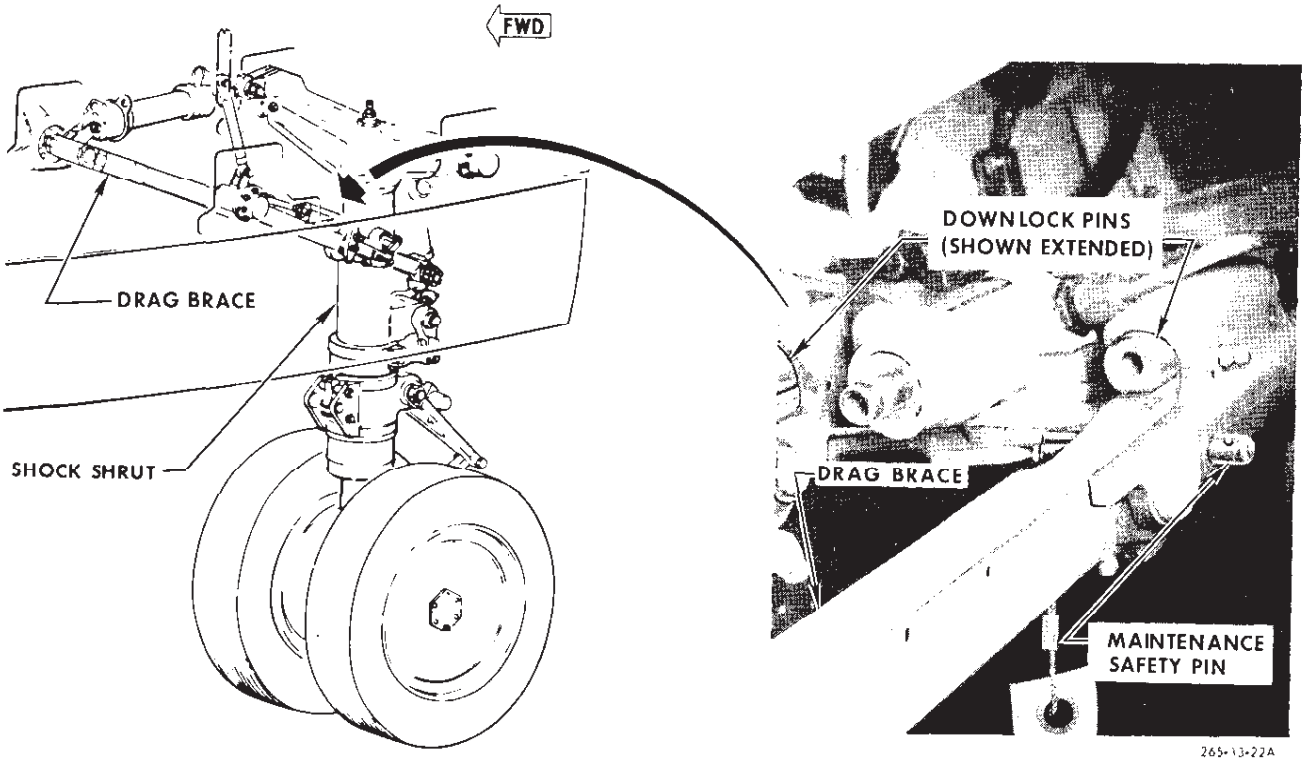
The latch need not be depressed when the landing gear handle is moved to UP after takeoff. When the handle is moved to DOWN, the hydraulic selector valve is energized to route hydraulic pressure to the gear-down side of the main gear cylinders. These cylinders release the main gear door uplock hooks and put the doors down. The landing gear uplock hooks are hydraulically released, and the landing gear cylinders extend the landing gear.

When the gear is down, spring-loaded downlocks slide into place and lock the gear down. (See Figure 2-45.) When all three gears are down and locked, hydraulic pressure is maintained on the downside of the landing gear cylinders as a safety factor to prevent gear collapse in case the gear downlock pin fails. When the weight of the aircraft is on the landing gear, the landing gear handle is mechanically locked in the DOWN position. If it is necessary to retract the landing gear while the weight of the aircraft is on the gear, the override pushbutton has to be depressed before the landing gear handle can be moved to UP. If the handle is moved to UP with the weight of the aircraft on the gear and electrical and hydraulic power is available, the gear downlocks are retracted and hydraulic pressure is applied to the upside of the gear actuating cylinders for retraction.

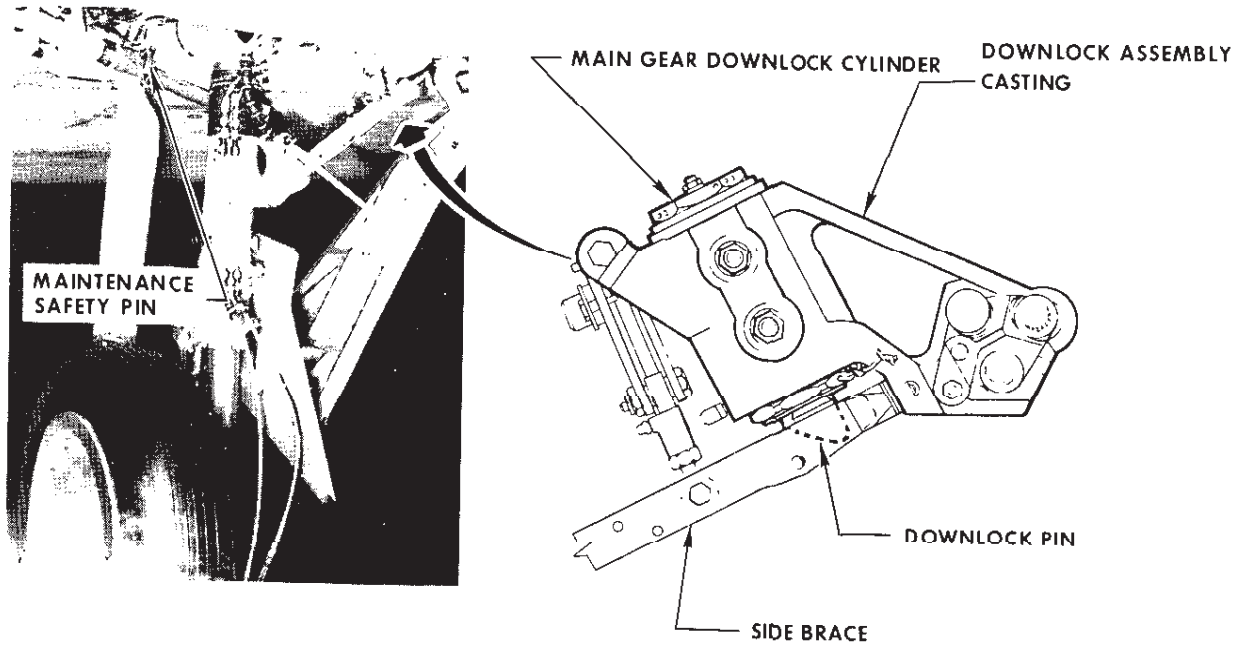
2.21.2.1 Auxiliary Power Assist for Emergency Gear Retraction (Aircraft Modified by SOK 11). The installation of this kit provides for utilization of both normal and auxiliary hydraulic accumulators to decrease the landing gear and fairing door retraction time. With the gear handle in the UP position and normal system pressure below 1,900 ±100 psi, the auxiliary accumulator shutoff valve opens and provides auxiliary system pressure to assist the gear retraction sequence. The system works only on the gear retraction cycle unless, on placing the landing gear handle down, the auxiliary hydraulic switch is in the ON position, at which time the auxiliary accumulator pressure is routed to the landing gear downside.

2.21.2.2 Landing Gear Downlock Override Button. When the weight of the aircraft is on the landing gear, the gear handle is locked in the DOWN position. If it is necessary to retract the gear during takeoff or landing and hydraulic pressure is available, the landing gear handle may be moved to UP while the downlock override button is pressed. The downlock override pushbutton is above the gear handle. (See Figure 2-44.) When the aircraft is airborne, a solenoid removes the mechanical lock and the gear handle may be moved to UP without actuating the override button. The gear handle can, if necessary, be moved from UP to DOWN before landing by first depressing and holding the downlock override button and then moving the handle to DOWN.

2.21.3 Landing Gear Emergency Release Handle. The landing gear emergency release T-handle (Figure 2-44) is used to lower the gear in case hydraulic or electrical system failure prevents normal gear lowering. When the handle is extended (pulled to its full extension of about 11 inches with a maximum pull of about 35 pounds), all fairing doors and the gear uplocks are unlocked mechanically, a dump valve is opened to return hydraulic fluid from the actuating cylinders, and the gear lowers by gravity.



NOSE LANDING GEAR



MAIN LANDING GEAR

39E-1-33-3

Figure 2-45. Landing Gear Downlock Pins

CAUTION

The emergency handle must be pulled to full extension to ensure release of all uplocks and proper positioning of the dump valve. When the handle is restowed, manually replace the handle to prevent damage to the surrounding equipment.

A restraining hook on the emergency handle may be inserted into a placarded hole below the pilot console to facilitate keeping the handle at its fully extended position. If the emergency handle has been used for an emergency, it should be stowed only after the gear malfunction has been corrected by maintenance personnel.

A switch in the landing-gear-up circuit is opened when the emergency handle is pulled and prevents possible gear retraction following release of the emergency handle. The switch shall be reset for normal operation of the gear control circuit by the landing gear downlock and electric reset button in the cockpit.

CAUTION

When demonstrating landing gear extension by the landing gear emergency release handle, the main gear doors will remain extended. Prior to landing, the landing gear should be retracted and extended normally to ensure proper sequencing of landing gear doors.

2.21.4 Landing Gear Downlock and Electric Reset Button (CT-39E Aircraft). This pushbutton (Figure 2-44) serves a dual purpose as landing gear electric reset button and as landing gear downlock reset button. After emergency lowering of the gear, it has to be pressed (to reset the landing gear-up circuit which is opened when the gear emergency release handle is pulled) to permit subsequent gear retraction. Also, it can be pressed to neutralize the landing gear selector valve, relieving back pressure on the landing gear downlocks and allowing the spring-loaded downlock pins to complete the gear-safe indicating circuit.

2.21.5 Landing Gear Electrical Reset (L.G. ELEC RESET) Button (CT-39G Aircraft). This pushbutton (Figure 2-44) resets the landing-gear-up circuit that is opened when the gear emergency release handle is pulled to permit subsequent landing gear retraction.

Note

Extending the landing gear by using the emergency extension system does not render the nosewheel steering system inoperative.

2.21.6 Landing Gear Position Lights. The three green landing gear position lights (Figure 2-44) are on the pilot inboard instrument panel. The lights are arranged to represent the right and left gear and the nosegear. Each light comes on when the respective gear is down and locked. The indicator lights are powered by the dc essential bus. The dimming of the landing gear position indicator lights for night flight is controlled by the flight instrument lights indirect rheostat on the pilot console.

2.21.7 Landing Gear Warning System. The landing gear warning system consists of an audible warning in the pilot and copilot headsets amplified through the audio system; a warning horn; and a flashing or steady red light in the landing gear handle. The warning tone is not reproduced through the loudspeakers. The system warns of an unsafe landing configuration. The landing gear warning system is powered by the dc essential bus or the battery bus if both dc generators fail or are off.

2.21.7.1 Landing Gear Audio Warning Signal. The landing gear audio warning signal is heard in the pilot and copilot headsets to warn of below-preset limits that may require the extension of the landing gear. The audio warning can be silenced by increasing airspeed to exceed 140 ±10 KIAS, advancing both throttles above 95-percent rpm, or pressing the horn cutout button.

2.21.7.2 Landing Gear Warning Horn. The landing gear warning horn provides an audible indication of a gear-unsafe condition under the same basic conditions as the audible warning system. However, the warning horn is wired through the flap travel monitor switch. As the wing flaps are moved down from the approach flap setting (66 percent) toward the DOWN position (100 percent), the flap travel monitor switch is actuated a approximately 80-percent flap position and the landing gear warning horn sounds regardless of airspeed, altitude, or throttle setting. The warning horn cannot be silenced by the landing gear warning horn cutout button

when the wing flaps are extended beyond the approach flap setting.

CAUTION

Do not press warning horn cutout button on deck. Substantial damage to the warning system may result.

2.21.7.3 Landing Gear Warning Light. The landing gear warning light is in the landing gear handle. A steady red light indicates an unsafe landing gear condition when the landing gear position does not correspond to the landing gear handle position. In addition, a flashing red light in the handle with the landing gear in the UP position indicates that the aircraft pressure altitude is less than 12,000 ±600 feet, the airspeed is less than 140 ±10 KIAS, and either throttle position is less than 95-percent rpm. The warning light goes out when the landing gear is locked in the DOWN position or the altitude, IAS, and engine revolutions per minute are above limits. The lamps are tested by actuating the caution light test switch.

2.21.7.4 Landing Gear Warning Horn Cutout Button. The landing gear warning horn cutout button is to the right of the landing gear handle. (See Figure 2-44.) This button is pushed to silence the landing gear warning horn. However, if the wing flaps are extended beyond the approach position, the horn signal cannot be silenced. The horn circuit is reset automatically when both throttles are advanced above 95 percent. The red light in the gear handle is not affected by the horn cutout button.

2.21.8 Landing Gear Door Ground Control Switch. A two-position switch powered by the dc essential bus is in the left main landing gear strutwell. The function of this switch is to open the main landing gear wheel well doors for inspection or maintenance. The switch is labeled NORMAL and OPEN and should be checked to ensure that it is in the NORMAL position before flight. If the switch is left in the OPEN position, actuation of the landing gear control handle after takeoff will override the ground control switch and normal gear retraction and door closing will occur after takeoff; however, before landing when the landing gear is extended, the main gear doors will not close after the gear is extended. Without electrical or normal system hydraulic power on the aircraft, the main landing gear doors may be extended by actuation of the landing gear emergency release T-handle.

CAUTION

If the main landing gear wheel doors are opened with the landing gear emergency release T-handle, first place the ground control switch in the OPEN position. This is a safety precaution that will prevent the wheel doors from inadvertently closing if electrical power is subsequently turned on.

If the landing gear emergency release T-handle is used to open the doors, the landing gear downlock and electric reset button shall be depressed before the next flight.

2.21.9 Ground Transfer Switches. Three ground transfer switches (squat switches) differentiate the flight and ground mode for the aircraft. They affect the electrical system, door seal, emergency pressurization, fuel jettison, nosewheel steering disconnect, landing light angle, landing gear safety solenoid, ram and dump valves, pilot/EPR/stall warning heat, stall warning, flight hour meter, and ground blower systems.

2.22 LIGHTING EQUIPMENT

2.22.1 Exterior Lighting. Exterior lighting consists of a position light on each wingtip, two position lights faired into the tail cone, one anticollision light on the vertical stabilizer and one on the bottom of the fuselage just forward of the speedbrake, three strobe lights (one in each wing tip and one on the top of the tail cone), and an ice-check light on each side of the fuselage directed at the wing leading edge. Retractable landing-taxi lights, when in use, extend below the nose into the airstream.

2.22.1.1 Position Light Switch. The POSITION switch on the exterior lighting control panel (Figure 2-46) controls the lights on the wingtips and tail. Moving the switch to ON turns on the lights. The OFF position turns the lights off. The circuit is powered by the dc essential bus and is protected by a circuit breaker on the overhead circuit breaker panel.

2.22.1.2 Anticollision Light Switch. The ANTI-COLLISION switch is on the exterior lighting control panel (Figure 2-46). Moving the switch to ON supplies power to the rotating mechanism and lights of the unit on the top of the vertical stabilizer and under the fuselage just forward of the speedbrake. These lights rotate within a streamlined enclosure to provide a flashing-type, high-intensity red warning light. When the switch is moved to OFF, power to the motor and lights is turned

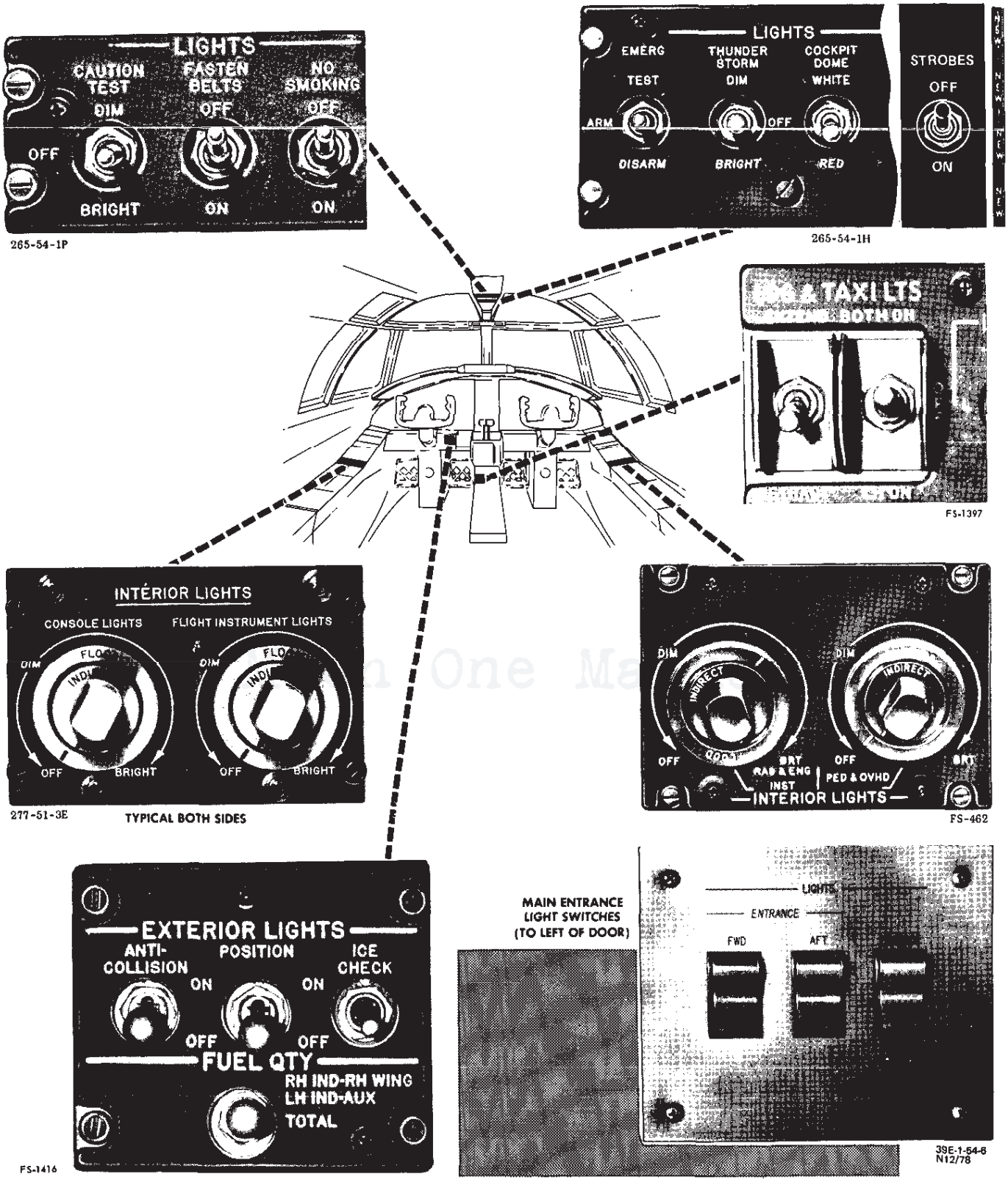


Figure 2-46. Lighting Control Panels (Typical)

off. The upper anticollision light is powered by the dc essential bus and the lower anticollision light (CT-39E aircraft 158843 and 158844) by the dc secondary bus. The circuits are protected by two circuit breakers on the overhead circuit breaker panel.

Note

The rotating anticollision lights should be turned off during flight through conditions of reduced visibility where the pilot could experience vertigo as a result of the rotating reflections of the lights against the clouds. In addition, the lights would be ineffective as anticollision lights during these conditions since they could not be observed by other aircraft.

2.22.1.3 Landing-Taxi Light Switches. The LDG & TAXI LTS switches (Figure 2-46 and FO-6) on the center pedestal control the position and illumination of the landing-taxi lights. When the left switch is placed at EXTEND, both lights extend to the landing position. During icing conditions, the lights can be deiced by turning them on before extending. When the weight of the aircraft is on the nosegear, both lights automatically extend farther to the taxi position, providing properly directed beams for taxiing. If a touch-and-go landing is made and the switch is left at EXTEND, both lights return to the landing position as the weight of the aircraft is removed from the nosegear. Placing the switch at RETRACT causes both lights to retract. Placing the right-hand switch at BOTH ON turns both lights on. When the switch is placed at LH ON, the left light only is turned on. Both lights go out when the switch is placed at OFF. The circuit is powered by the dc essential bus and is protected by four circuit breakers on the overhead circuit breaker panel.

2.22.1.4 Landing Light Extended Caution Light. The round amber LDG LIGHTS EXT caution light is mounted on the instrument panel shroud (17, FO-1 or 16, FO-4). This light is powered by the dc essential bus and comes on anytime either landing light is not fully retracted.

2.22.1.5 Wing Ice-Check Light Switch. The ICE CHECK switch on the exterior light control panel (Figure 2-46) controls the ice-check lights on each side of the fuselage. Placing the switch at ON turns both lights on to illuminate the wing leading edges. Placing the switch at OFF turns off the lights. The circuit is powered by the dc secondary bus and is protected by a circuit breaker on the right-hand console circuit breaker panel.

2.22.2 Strobe Lights. In addition to the aircraft position light system, three high intensity, white (strobe) lights are installed. The wing-tip strobe lights are an integral part of the position light assembly. The tail strobe light is a single lamp assembly and is installed on the upper surface of the tailcone. The system consists of three individual power supplies and a timing circuit that sequences the lights. The lights are powered by the 28-vdc essential bus and actuated by the STROBE LIGHTS switch located on the LIGHTS switch panel (center overhead).

2.22.3 Interior Lighting. In the cockpit, all instruments and control panels are indirectly illuminated. Direct lighting of the consoles, instrument panels, and center pedestal is provided by fixed floodlights. A combination portable map and utility light is provided above both the left and right consoles. A map light is incorporated under the medallion of both control wheels. Two thunderstorm lights and a dome light are also provided in the cockpit. Cabin lighting consists of passenger reading lights, forward and aft entrance lights, a passenger floor step light, and baggage compartment lights.

2.22.3.1 Pilot and Copilot Console Light Rheostats. Indirect lighting and floodlighting of the left and right consoles are controlled by dual CONSOLE LIGHTS rheostats on the interior lights control panels (4 and 11, FO-2) of the consoles. The rheostats sweep from OFF to BRIGHT. The inner INDIRECT rheostat knob controls power from the 5-vac indirect light bus for indirect lighting of the console and instrument subpanels. The outer FLOOD rheostat ring controls power from the dc essential bus for the console floodlights.

2.22.3.2 FLIGHT INSTRUMENT LIGHTS Rheostats. Indirect lighting and floodlighting of the pilot and copilot flight instruments are controlled by dual FLIGHT INSTRUMENT LIGHTS rheostats on the interior lights control panels (4 and 11, FO-2). The rheostats sweep from OFF to BRIGHT. The outer FLOOD rheostat ring controls power from the dc essential bus for the respective instrument floodlights. The inner INDIRECT rheostat knob controls power form the 5-vac indirect light bus for indirect lighting of the respective instruments. In addition, as the INDIRECT rheostat knob on the left console is rotated toward BRIGHT, it turns on the passenger floor step light and dims the caution-warning lights, landing gear position indicator lights, and marker beacon lights operate at full brilliance.

2.22.3.3 Radio and Engine Instrument Light Rheostats. Indirect lighting and floodlighting of the radio control panels, engine instruments, and miscellaneous instruments are controlled by a dual RAD & ENG INST

rheostat on the left console interior light control panel (4, FO-2). The rheostat sweeps from OFF to BRT. The inner INDIRECT rheostat knob controls power from the 5-vac indirect light bus for indirect lighting. The outer FLOOD rheostat ring controls power from the dc essential bus for the center instrument panel floodlights.

2.22.3.4 Pedestal and Overhead Light Rheostat. Indirect lighting and floodlighting of the center pedestal and overhead panels are controlled by a dual PED & OVHD rheostat on the left console interior lights control panel (4, FO-2). The rheostat sweeps from OFF to BRT. The inner INDIRECT rheostat knob controls power from the 5-volt indirect light bus for indirect lighting of the panels and magnetic compass. The outer FLOOD rheostat ring controls power from the dc essential bus for the center pedestal floodlights.

2.22.3.5 Thunderstorm Light Switch. The THUNDERSTORM switch on the light control panel (Figure 2-46) controls dc secondary bus power to the two white thunderstorm lights in the cockpit. Placing the switch at either DIM or BRIGHT turns the lights on to the desired brightness. Placing the switch at OFF turns the lights off.

2.22.3.6 Cockpit Dome Light Switch. The COCKPIT DOME switch on the light control panel (Figure 2-46) controls dc essential bus power to the dual cockpit dome light. Placing the switch at either WHITE or RED turns on the respective light. Placing the switch at OFF turns the light off.

2.22.3.7 Map and Utility Light Controls. A combination portable map and utility light for the pilot and copilot is above each console. These lights may be adjusted in color from white to red, in light pattern from spot to flood, and in brilliance from off to bright. On an adjustable collar at the forward end of the light barrel is a series of dots, coded as follows:

Large white — White Flood
Small white — White Spot
Small red — Red Spot
Large red — Red Flood

Light pattern and color are obtained by rotating the collar to align the dot (corresponding to the light pattern and color desired) with a white index marker on the forward end of the light barrel. A red rheostat knob at the aft end of the light has OFF, DIM, and BRIGHT positions. When one of these positions is aligned with a white index marker at the aft end of the barrel, corresponding brilliance is obtained. Depressing a pushbutton switch on the aft face of the rheostat knob provides momentary full brilliance regardless of the rheostat setting. The pilot utility light is powered by the dc essential bus, and the copilot utility light is powered by the dc secondary bus.

On CT-39G aircraft 159361 through 159365 and 160053 through 160057, a map light is installed on the right and left overhead forward circuit breaker panels. The lights are powered by the dc essential bus and controlled by a rheostat immediately adjacent to the light.

2.22.3.8 Control Wheel Map Light Switch. A map light is installed under the medallion of the pilot and copilot control wheels. Each light is controlled individually by a toggle switch which protrudes from the lower edge of the medallion. The lights are powered by the dc essential bus.

2.22.3.9 Fasten-Seatbelts and No-Smoking Cabin Sign Switches. The FASTEN BELTS and NO SMOKING switches on the interior light control panel (Figure 2-46) control dc essential bus power to the respective cabin sign. Placing either switch at ON turns on the respective sign light. Placing the switch at OFF turns the light off. In addition, the switches are bypassed and the signs come on automatically under certain oxygen mask use requirements.

2.22.3.10 Forward Entrance Light Switch. The two-position FWD ENTRANCE light rocker switch (Figure 2-46) controls dc essential bus power to the overhead entrance light just inside the entry door.

2.22.3.11 Aft Entrance Light Switch. The two-position AFT ENTRANCE light rocker switch (Figure 2-46) controls dc essential bus power to the entrance lights on the aft cabin bulkhead.

2.22.3.12 Baggage Compartment Light Switch. The two-position BAGGAGE compartment light rocker switch (Figure 2-46) controls dc essential bus power to the two overhead lights in the baggage compartment.

2.22.3.13 Passenger Reading Lights and Switches. An adjustable "eyeball" type reading light and pushbutton switch is above each passenger seat except for the center seat of the rear couch. Two fixed lights with individual pushbutton switches are on the overhead for this seat position. Power is received from the dc essential bus. Depressing the switch turns the respective light on or off.

2.23 NAVIGATION EQUIPMENT

2.23.1 Tacan System (CT-39E/G Aircraft With AN/ARN-91 Installed). The AN/ ARN-91 (tactical air navigation) tacan system operates in conjunction with a tacan surface beacon or another airborne tacan set that has air-to-air capability. The system enables continuous reception of bearing and slant range from any selected station within line-of-sight. It can also receive and transmit air-to-air ranging information. Magnetic bearing from the aircraft to the transmitting station is displayed by a double-barred pointer on the radio magnetic indicator when the double-barred pointer knob is at TAC. Distance (in nautical miles) from the aircraft to the transmitting station, TO-FROM indications, and radial position (lateral deviation) are displayed on the integrated flight system course indicator. The surface beacon transmits Morse-coded identification signals every 37.5 seconds. These are amplified in the audio system.

In the air-to-air mode, the system operates as an interrogator and transponder with other compatible airborne tacan systems operating in the air-to-air mode. The tacan system in the second aircraft must be set to a channel that is 63 channels from the tacan system in the first aircraft. Bearing information is not provided in the air-to-air mode. An aircraft operating in other modes, in an area with two aircraft using the air-to-air mode, will not be disturbed even though they may be on the same channels. The tacan system is powered by the dc essential bus and the ac secondary bus.

2.23.1.1 Tacan Controls and Indicators

2.23.1.1.1 Function Switch. When the function switch (Figure 2-47), on the tacan control panel is moved from OFF to REC, the system is energized and bearing and identification information is presented. With the switch at T/R, the system presents bearing,

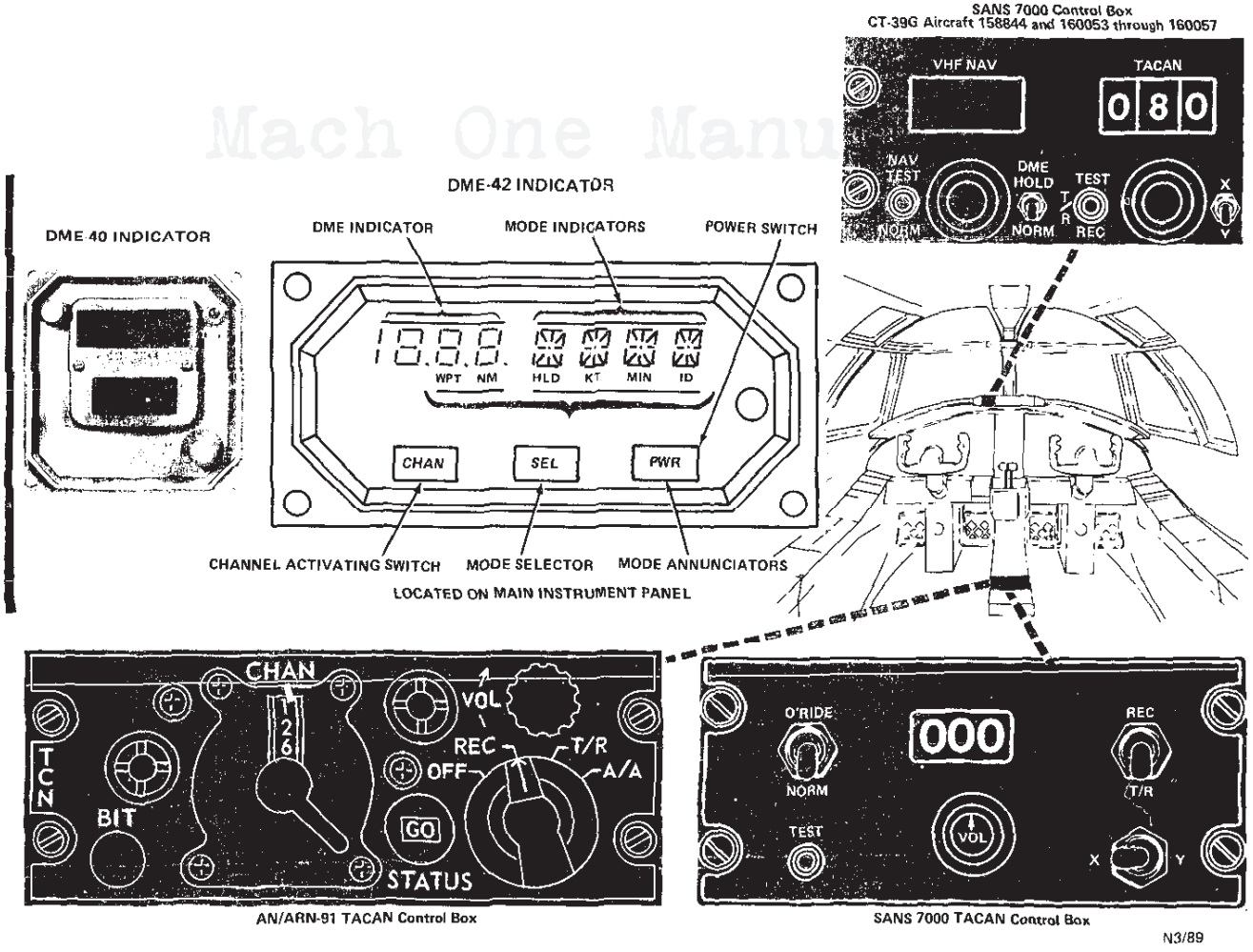


Figure 2-47. TACAN Control Panels and DME-40 Indicator

identification, and slant range. The A/A position provides range information only between two aircraft operating in this mode.

2.23.1.1.2 Channel Selector Switch. The CHAN selector switch (Figure 2-47) on the TACAN control panel permits selection of frequency for the mode in use. The switch includes a circular knob for selecting the first two digits and a concentric handle for selecting the third digit of a desired channel.

2.23.1.1.3 Volume Knob. This knob (Figure 2-47) on the TACAN control panel adjusts the audio signal strength of the beacon identification tone. Clockwise knob rotation increases volume.

2.23.1.1.4 Self-Test Button and Status Indicator. A pushbutton (BIT) switch and a GO/NO-GO status indicator (Figure 2-47) on the control panel are used to check the operational status of the system. When the button is pressed and released, a signal is inserted in the system to start the self-test cycle. (While the button is depressed, GO appears on the status indicator.) The test cycle takes about 15 to 20 seconds with the system status displayed on the indicator only for the last 10 seconds of the test cycle. The test occurs on the channel selected. During the test, the range and bearing indicators slew to zero to show proper test response. If the function switch is at REC during the test cycle, only the receive functions of the system are checked.

2.23.1.1.5 Course Selector Switch. Pressing the TAC portion of this switch (9 and 11, Figure 2-39) on the left and right instrument panel shroud connects tacan signals to the course and radio magnetic indicators. For additional information on this switch, refer to paragraph 2.18.1, Integrated Flight System Controls and Indicators.

2.23.1.1.6 Radio Magnetic Indicator. Tacan receiver signals drive the double-barred pointer on the RMI (25 and 32, Figure 2-39) when the doubled-barred pointer knob is set at TAC. For additional information on the RMI, refer to paragraph 2.18.1, Integrated Flight System Controls and Indicators.

2.23.1.1.7 Course Indicator. Range to a tacan station is shown on the miles counter on the course indicator. TO-FROM arrows indicate whether the selected tacan course is inbound or outbound. (See Figure 2-43.) For additional information on the course indicator, refer to paragraph 2.18.1, Integrated Flight System Controls and Indicators.

2.23.1.1.8 Operation of Tacan. The following procedure are for setting the system into operation only. Refer to paragraph 2.18.3, Operation Of Integrated Flight and Autopilot Systems for additional operating procedures.

1. Function switch — As Desired.
2. Course selector switch — TAC.
3. Channel selector switch — Desired Frequency.
4. RMI double-barred pointer knob — TAC.
5. Volume knob — As Desired.

2.23.2 Tacan System (CT-39E/G Aircraft with SANS 7000 Tacan Installed). The SANS 7000A tacan system operates in conjunction with a surface beacon. The system provides the pilot with range and bearing information to the selected station. The system displays range information in nautical miles and is displayed on the course indicators. Bearing information is displayed on the RMI double pointer when the course indicators. positioned at TAC. The TO-FROM indications and radial position (lateral deviation) are displayed on the course indicator. Two antennas are provided for the tacan system. One is installed on top of the fuselage and one on the bottom of the fuselage. The antennas are automatically switched by the tacan bearing unit. In aircraft with the ARN-91 tacan, the system is powered by the 115-vac secondary bus and the 28-vdc essential bus. In aircraft with the SANS 7000 tacan, the system is powered by the 115-vac secondary bus for the indicator.

2.23.2.1 Tacan Controls and Indicators

2.23.2.1.1 Tacan System Power Switch. The tacan system power switch (Figure 2-47) provides control of system electrical power.

This switch is the aft knob of the stacked control switch. When rotated clockwise, electrical power is applied to the tacan system. This switch also varies the volume of the audio identification tone.

2.23.2.1.2 Channel Selector Switch. The channel selector switch (Figure 2-47) on the TACAN control panel permits selection of the desired channel. These switches are the two forward knobs of the stacked control switch.

2.23.2.1.3 Self-Test Button. A pushbutton switch (Figure 2-47) on the tacan control panel is used to check the operational status of the system. This switch is used in conjunction with the T/R-REC switch. Depressing the TEST switch allows the range display in the course indicator to display a 000.0 ±0.1 nautical mile. The display is accompanied by a steady audio tone. A 180°

bearing is also displayed on the course indicators with the left/right bar at center (with a north heading) and the TO/FROM indicator will read FROM. If the heading is set to south, the TO/FROM indicator will indicate TO.

2.23.2.1.4 O'RIDE/NORM Switch. The O'RIDE/NORM switch (Figure 2-47) functions in the range capabilities of the system. When NORM is selected, the range is limited to 199 nm. The O'RIDE position allows the full 399-nm capacity.

2.23.2.1.5 Mode Selector Switch. The X/Y mode selector switch (Figure 2-47) allows the tacan transmitter to operate in either the X or Y mode channels. Each position has the capability of operating on 126 channels.

2.23.2.1.6 Operation of Tacan. The following procedures are for setting the system into operation only. Refer to paragraph 2.18.3, Operation Of Integrated Flight and Autopilot Systems, for additional operating procedures.

1. System power switch — Rotate Clockwise.
2. Pilot and copilot NAV/CRS SEL — TACAN.
3. Channel selector switch — Set to Desired Frequency.
4. X/Y mode selector — As Required.
5. RMI indicator — Select for TAC on Double-Barred Pointer.

2.23.3 Distance Measuring Equipment System (CT-39E/G Aircraft With DME-40 or DME-42 Installed). The DME-40/DME-42 system measures the slant range (line-of-sight) distance from the aircraft to a ground station and displays a continuous distance readout in nautical miles. The system also displays aircraft ground speed in knots or TTS in minutes. The groundspeed and time to station are accurate only if the aircraft is flying directly toward the ground station at a sufficient distance that the slant range and ground range are nearly equal. The system consists of a transceiver, antenna, and a DME indicator. The transceiver is tuned by the VHF navigation receiver. When a VORTAC or ILS localizer frequency is selected, the DME ground station which is associated with the VORTAC or localizer is automatically selected. The DME system is powered by the 26-vac primary bus and the 28-vdc essential bus.

2.23.3.1 DME-40 Indicator. The DME indicator (Figure 2-47) computes the groundspeed from the rate of change of distance. The time to station is computed in the indicator from the rate of change of distance and the analog signal from the transceiver. The groundspeed is displayed in knots and the time to station is displayed in minutes. A switch on the front panel of the indicator is used to select either knots or minutes. The groundspeed and TTS indications are accurate only if the aircraft is flying directly toward the ground station and the distance is sufficient for the slant range and ground range to be nearly equal. For example, at an altitude of 12,000 feet above the VORTAC station and at a slant range of 10 nm from the station, the groundspeed indication is lower than aircraft groundspeed by approximately 2 percent. The TTS indication will have a corresponding error. The test button on the indicator provides a self-test of the system. When the test switch is depressed, both windows will display dashes after 8 to 12 seconds. The three-position function switch provides control of system electrical power and the selection of knots (distance) display or the minutes to station display. The DIM controls the intensity of the displays.

2.23.3.1.1 Operation DME-40. The following procedures are for setting the system into operation and performing a self-test.

Mach One Manuals

1. AC and dc power — ON.
2. Function selector switch — MIN.
3. NAV receiver — Tune to Local VORTAC or ILS Frequency. Allow 1 minute for System Warmup.
4. TEST button — Depress.
5. Display windows — Displays Dashes After 8 to 12 Seconds.

2.23.3.2 DME-42 Indicator. The DME-42 (Figure 2-47) is capable of providing complete DME information from three ground stations simultaneously. The DME-42 indicator displays distance in nautical miles (0 to 300 nmi) on the three-and-one-half character left-side display. The distance is accurate to 0.1 nmi. The display will show tenths of a mile from 0 to 199.9 nmi and 1-mile increments from 200 to 300 nmi. The four characters on the right side display indicate station identifier (two-, three-, or four-letter station identifier), TTS (0 to 199 minutes), groundspeed (0 to 999 knots), or diagnostic code in the self-test mode. This right-side alphanumeric display and annunciation are selectable by means of the SEL pushbutton which, when repeatedly pressed, will sequentially select the information desired in the alphanumeric display. The momentary

pushbutton CH switch sequentially selects the information from the next DME channel and lights the appropriate channel annunciator "1," "2," or "3." Power to the unit is controlled by the pushbutton PWR. The groundspeed and TTS indications are accurate only if the aircraft is flying directly toward the ground station and the distance is sufficient for the slant range and ground range to be nearly equal. For further systems information, refer to the Collins Pilot Guide.

2.23.3.2.1 Operations of DME-42

1. VOR — ON (DME-42 is powered through the VOR).

2. DME-42 PWR switch — ON.

If the DME-42 is within range of the selected station, the numerical display will show slant range distance. After two DME idents have been received, the alphanumeric display will show the two-, three-, or four-letter station identifier.

3. Utilize the SEL switch to change the alphanumeric display from KT (groundspeed), to MIN TTS, to ID (station identifier), whichever is desired.

4. When DME HOLD is desired, select HOLD on the DME HOLD switch directly below the VOR on the instrument panel. An amber light will illuminate, retaining the current VOR station DME until DME HOLD is turned off or signal is lost.

5. For system test, depress the TEST button (all display segments and annunciators illuminate for lamp test). If an out-of-tolerance condition is detected, the word DIAG is displayed in the numeric (left) display, the diagnostic code is displayed in the alphanumeric (right) display, and the test sequence ends. If no out-of-tolerance condition is detected, the numeric display shows a test distance of 100 nmi. The alphanumeric display will depend on what is selected: KT, MIN, or ID, and will show AOK at the completion of the test sequence.

2.23.4 DF-203 Automatic Direction Finding System (CT-39E Aircraft and CT-39G Aircraft 158843). The DF-203 ADF system continuously indicates the magnetic bearing to a selected station for homing, navigation, or position finding, and simultaneously provides aural reception of audio transmission from that station. The system can also be used in a loop mode for manual direction finding. The ADF system consists of a receiver, a loop antenna and a sense antenna, and a control panel. Bearing information is presented on the RMI. Operating frequency range of the system is 190 to 1750 kHz and power is supplied by the dc essential bus and the ac secondary bus. Aural signals are received and amplified through the audio system.

2.23.4.1 DF-203 ADF System Controls and Indicators. The ADF control panel (Figure 2-48) on the instrument panel contains the system controls and indicators that function as follows.

2.23.4.1.1 Function Knob. Turning this knob to ADF automatically positions the loop antenna circuits in the receiver for bearing or position fixing. Magnetic bearing to the selected station can be indicated by either or both pointers on the radio magnetic indicator. With the knob at ANT, the station selected will be received, but the radio magnetic indicator will not indicate station bearing. This knob position can be used for broadcast monitoring, maximum tuning, or range navigation. With the knob set at LOOP, aural-null orientation is made possible by using the loop knob. Turning the function knob to OFF turns off the ADF system.

2.23.4.1.2 Loop Knob. While the function switch is at LOOP, this knob can be turned to L or R to position the loop antenna in the corresponding direction to obtain aural-null orientation. The "true-null" magnetic bearing to the station may be indicated by either end of the radio magnetic indicator pointer or pointers.

2.23.4.1.3 Band Knob. The three frequency bands available in this system are labeled in MHz above the knob as follows: .19-.40 (190 to 400 kHz), .40-.84 (400 to 840 kHz), and .84-1.75 (840 to 1750 kHz). The selected range is displayed on a frequency indicator adjacent to the knob.

2.23.4.1.4 Frequency Indicator. This indicator displays the frequencies in MHz as selected by the band knob. For conversion to kHz, the decimal point should be moved three places to the right.

2.23.4.1.5 GAIN Knob. The GAIN knob permits selection of the desired volume in the headsets. Signal output increases as the knob is turned clockwise.

2.23.4.1.6 Tuning Meter. The tuning meter is a pointer that moves to the right across a scale to indicate relative signal strength and how well the selected station is tuned in.

2.23.4.1.7 Tuning Knob. Labeled TUNE, this knob is used in adjusting receiver frequency within the band selected by the band knob. Five turns of the tuning knob

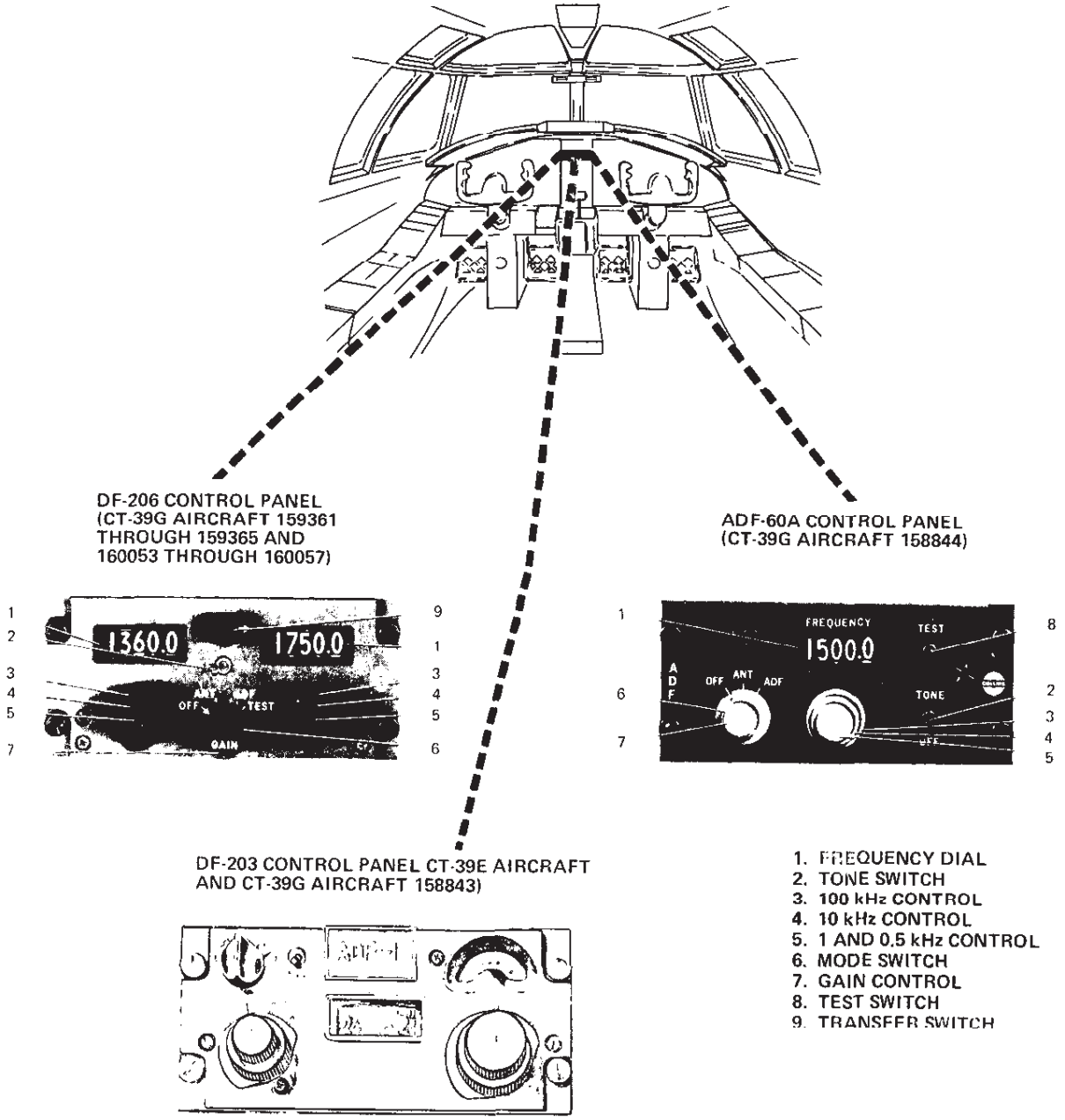


Figure 2-48. ADF Control Panel

covers the selected band. About 5 seconds is required to completely scan any one frequency band.

2.23.4.1.8 Beat Frequency Oscillator Switch. Moving the BFO switch from OFF to BFO allows the oscillator in the receiver to provide a zero-beat frequency to the carrier. This creates an audio note on unmodulated or on-keyed continuous-wave stations.

2.23.4.1.9 Radio Magnetic Indicator. Either the single-barred pointer or the double-barred pointer on the RMI (25 and 32, Figure 2-39) on the pilot and copilot instrument panels reflects the magnetic bearing of the selected station. This depends upon the setting of the pointer knobs on the indicator. If both knobs are set at ADF, both pointers move as one. For additional information on this indicator, refer to paragraph 2.18.1, Integrated Flight System Controls and Indicators.

2.23.4.1.10 Operation of DF-203 ADF System

1. ADF mixer switches — As Desired.
2. Function knob — ANT.
3. Band knob — Desired Band on Frequency Indicator.
4. Tuning knob — Adjust Frequency. Check Maximum Deflection of Tuning Meter Pointer.
5. Gain knob — Comfortable Volume.
6. BFO switch — BFR. Check for Continuous Tone.
7. RMI single-barred knob, double-barred knob, or both knobs — ADF.
8. Function knob — ADF. Check Station Bearing on RMI.
9. Function knob — LOOP.
10. Loop knob — L or R To Obtain Aural Null. Check Station Bearing or Reciprocal on RMI.
11. Function Knob — OFF.

2.23.5 DF-206 ADF System (CT-39G Aircraft 159362 Through 159365 and 160053 Through 160057). The DF-206 ADF system provides continuous relative bearing readings to low-frequency homing stations, radio beacons, and broadcast stations in the 190- to 1750-kHz frequency range and simultaneously provides aural reception of audio transmission from the station. The ADF system consists of a receiver, a fixed loop antenna, a sense antenna, and a control panel. Bearing information is presented on the RMI. Aural signals are received and amplified through the aircraft audio system. This ADF system provides simplified operation using digital tuning. The frequency is selected on the digital control unit and the receiver automatically switches to proper band and tunes to the proper 0.5-kHz increment. The control unit (panel) is a single-radio, dual-control unit allowing switching or transfer between two preset frequencies.

2.23.5.1 DF-206 System Controls and Indicators. The ADF control panel (Figure 2-48) on the center instrument panel contains the system control and the indicators that function as described in the following paragraphs.

2.23.5.1.1 Mode Control Switch. Turning this knob to the ADF mode allows the system receiver to process the selected radio signal from the loop and sense antennas into relative bearing information for use by the RMI. The indicator pointers are synchronized with the revolving compass card to simultaneously indicate relative bearings and magnetic bearings to the selected station. In the ANT mode, the system receiver functions as a conventional low-frequency and broadcast band aural receiver. In this position, the station selected will be received; the RMI will not indicate station bearing, but will remain in a horizontal position. Positioning the mode control to the momentary TEST position activates the functional (self-test) feature of the system. The bearing indicator will display a test bearing indication of 45°, and a 1020-Hz tone will be audible through the audio system. Turning the mode control to OFF removes power from the ADF system receiver.

2.23.5.1.2 Frequency Selection Controls. There are two sets of frequency selection controls with independent frequency selected displays. The frequency selection controls are the same and provide frequency selection control from 0.5 to 1700 kHz using a set of three control knobs. The inner knob (nearest the panel) is the 100-kHz control and controls frequency selection from 100 to 1700 kHz in 100-kHz steps. The middle knob is the 10-kHz control and controls frequency selection from 10 to 90 kHz in 10-kHz steps. The outer knob is the 1- and 0.5-kHz control and controls frequency selection from 0.5 kHz to 9.5 kHz in 0.5-kHz steps. The selected range is displayed on the frequency indicator above the respective frequency selection control.

2.23.5.1.3 Frequency Indicators. There are two frequency indicators on the control panel. The indicators display the frequency in kHz selected by the respective frequency selection control knobs.

2.23.5.1.4 Gain Control. The gain control knob controls the audio output level from the system receiver permitting selection of desired volume in the headset. Audio output level increases as the knob is turned clockwise.

2.23.5.1.5 TONE Switch. Placing the TONE switch at ON controls the operation of the tone oscillator (1020 Hz) in the system receiver, making identification of CW signals possible.

2.23.5.1.6 Transfer Switch. The TFR switch selects one of the two preset frequencies on the control panel and illuminates the frequency display selected.

2.23.5.1.7 Radio Magnetic Indicator. Either the single-barred pointer or the double-barred pointer on the RMI (25 and 32, Figure 2-39), on the pilot and copilot instrument panels, reflects the magnetic bearing of the selected station. This depends upon the setting of the pointer knobs on the indicator. If both knobs are set at ADF, both pointers move as one.

2.23.5.1.8 Operation of DF-206 ADF System. The following procedures are for checking and setting the system in operation.

1. Audio system ADF mixer switches — As Desired.
2. RMI single-barred knob, double-barred knobs, or both knobs — ADF.
3. Mode control switch — ANT. Check That Panel and Indicator Display Lamp Are Lit. Check That ADF Indicator Is Parked in Horizontal Position.
4. Mode control switch — TEST. Check That Within Approximately 4 Seconds ADF Indicator Displays a Test Bearing Indication of 45° and That a 1020-Hz Tone Is Audible Through Audio System.
5. Frequency selection control — Select Frequency of Known Outer Marker or Broadcast Station.
6. Mode control switch — ADF. Check That Indicator Points to a Bearing Suitable for Station Received.
7. Tone switch — ON. Check That a 1020-Hz Tone Is Audible When Station Is Received.
8. Gain control — Comfortable Volume.
9. Mode control switch — OFF. Position RMI System Selection Knobs As Required.

2.23.6 ADF-60A System (CT-39G Aircraft 158844). The ADF-60A system is identical to the previously described DF-206 ADF system with two exceptions:

1. The control unit panel is a single-radio, single-frequency control unit. Only one frequency can be selected at anytime.
2. The test switch replaces the TEST position on the mode control switch. Pressing the test switch causes the RMI indicator to swing 90° and a 1000-Hz tone is heard.

All other functions of the ADF-60A system are identical to the DF-206.

2.23.7 Marker Beacon System (CT-39E, CT-39G Aircraft 158843 and 159362 Through 159365). The 51Z-4 marker beacon system provides visual indication and aural indication through the audio system of the passage of the aircraft over marker beacon transmitters. Used as a navigational and instrument landing aid, the system includes a receiver, a sensitivity control, indicator lights, and a button for self-testing the system. The marker beacon system is powered by the dc essential bus.

2.23.7.1 Marker Beacon Lights. Three lights (16 and 29, Figure 2-39) on the pilot and copilot instrument panels present a visual indication of passage over a marker beacon. The upper light comes on with the letter A appearing within the white light to identify passage over an airway or inner marker. This passage is accompanied by an audible signal tone of 3000 Hz. The center light comes on with the letter O appearing within the blue light to identify passage over an outer marker. This passage is accompanied by an audible signal tone of 400 Hz. The lower light comes on with the letter M appearing within the amber light to identify passage over a middle marker. This passage is accompanied by an audible signal tone of 1300 Hz.

2.23.7.2 Marker Beacon Sensitivity Switch. This rocker-type switch (11, Figure 2-39) on the instrument panel shroud turns on the marker beacon receiver and permits marker beacon frequency selection. When the HI portion of the switch is pressed, the receiver has maximum sensitivity to the marker beacon signals. Pressing the LO portion of the switch reduces receiver signal sensitivity. With the switch at the center OFF position, power is removed from the marker beacon receiver.

2.23.7.3 Marker Beacon Test Button. This button (11, Figure 2-39) on the instrument panel shroud is for testing the marker beacon receiver and lights. Pressing and holding the button energizes a self-test cycle that first turns on the O light and produces a 400-Hz tone for approximately 1 second. Then the M light comes on accompanied by a 1300-Hz tone for approximately 1 second. This is followed by the illumination of the A light and the sounding of a 3000-Hz tone for approximately 1 second. The test function is inoperable with the autopilot engaged.

2.23.7.4 Operation of Marker Beacon System. Refer to paragraph 2.18.3, Operation Of Integrated Flight and Autopilot Systems.

2.23.8 Marker Beacon System (CT-39G Aircraft 158844 and 160053 Through 160057). The marker beacon system is a function of the VIR-30A navigation system and provides visual indication and aural indication through the audio system of the passage of the aircraft over marker beacon transmitters. Used as a navigational and instrument landing aid, the system includes modules (part of navigation receiver), sensitivity pushbutton switches, and indicator lights. The marker beacon system is powered by the dc essential bus and the 26-vac primary bus.

2.23.8.1 Marker Beacon Lights. Three lights (7 and 32, FO-4) on the pilot and copilot instrument panels present a visual indication of passage over a marker beacon. The right-hand light comes on with the letter A/I appearing within the light to identify passage over an airway or inner marker. This passage is accompanied by an audible signal tone of 3000 Hz. The left-hand light comes on with the letter O appearing within the light to identify passage over an outer marker. This passage is accompanied by an audible signal tone of 400 Hz. The middle light comes on with the letter M appearing within the light to identify passage over a middle marker. This passage is accompanied by an audible signal tone of 1300 Hz.

2.23.8.2 Marker Beacon Sensitivity Pushbutton Switches. The marker beacon sensitivity pushbutton switches are located on the right side of the instrument panel shroud (24, FO-4). With the navigation receiver energized and the MKR-HI switch depressed, the marker system module has maximum sensitivity to the marker beacon signals. Depressing the MKR-LO switch reduces receiver signal sensitivity. Both switches incorporate lights which are illuminated when the applicable switch is depressed.

2.23.8.3 Marker Beacon Self-Test. To perform a marker beacon self-test, refer to paragraph 2.23.10, VHF Navigation System (CT-39G Aircraft 158844 and 160053 Through 160057).

2.23.9 Variable Omnirange Navigation/ Instrument Landing System (CT-39E, CT-39G Aircraft 158843 and 159362 Through 159365). The 51RV-1 VOR/ILS navigation receiver provides reception of VHF/omnirange, tone localizer, lateral guidance, and voice facilities in the 180.0 to 135.9 MHz range. In providing vertical guidance for the instrument landing approach, the system covers glideslope frequency ranges between 329.3 and 335.0 MHz. The VOR/ILS signals received are amplified in the audio system and are displayed on the integrated flight system director and course indicators. Indication of magnetic bearing to the VOR station is shown on the RMI single-barred pointer when the single-barred pointer knob is at VOR. A control panel (10, Figure 2-39) labeled VHF/NAV is on the instrument panel shroud. Volume for the VOR/ILS receiver is controlled by the volume knobs on the audio system control panels. The system is powered by the dc essential bus and ac primary bus.

2.23.9.1 VOR/ILS Control Panel

2.23.9.1.1 Power and Frequency Selector Knobs. A set of three concentric, rotary knobs (10, Figure 2-39) is used to turn on and select the frequency of the VOR/ILS receiver. The smallest and uppermost knob is rotated clockwise to turn on the receiver. The lower knobs are rotated to select the desired frequency in MHz displayed by counters above the selector knobs.

2.23.9.1.2 Test Buttons. Three test buttons (10, Figure 2-39) are used in initiating a self-test circuit for quickchecking VOR/ILS receiver-integrated flight system operation. A VOR/ILS frequency must first be tuned so that in the performance of the self-test, adequate signals can be received. Pressing the center button (labeled VOR) tests deviation and direction functions of the course and radio magnetic indicators, respectively. Pressing the left button (labeled ILS) at UP/L of the right button (also labeled ILS) at DN/R tests the glide slope functions of the flight director and course indicators.

2.23.9.1.3 Course Selector Switch. Pressing the NAV portion of this switch (9 and 11, Figure 2-39) on the left and right instrument panel shroud connects VOR/ILS signals to the flight director, course, and radio magnetic indicators. For additional information on this switch, refer to paragraph 2.18.1, Integrated Flight System Controls and Indicators.

2.23.9.1.4 Flight Director Indicator, Course Indicator, and Radio Magnetic Indicator. These indicators (25 and 32, Figure 2-39; Figure 2-40), on the

pilot and copilot instrument panels, reflect various phases of VOR/ILS system operation. For additional information, refer to paragraph 2.18.1, Integrated Flight System Controls and Indicators.

2.23.9.1.5 Operation of VOR Navigation/Instrument Landing System. Refer to paragraph 2.18.3, Operation Of Integrated Flight and Autopilot System.

2.23.10 VHF Navigation System (CT-39G Aircraft 158844 and 160053 Through 160057). The VHF VIR-30 navigation system provides reception of VHF omnirange, tone localizer, and lateral guidance in the 108.00 to 117.95 MHz range. In providing vertical guidance for the instrument landing approach, the system covers glideslope frequency ranges between 329.15 and 335.0 MHz. The system consists of a frequency control panel, three antennas, a receiver, and a RMI adapter. The system is powered by the 28-vdc essential bus and the 26-vac primary bus. The receiver consists of three separate modules that provide VOR/LOC, glideslope, and marker beacon signals to the applicable indicating devices. A separate antenna is provided for each module. The VOR/LOC and marker beacon modules also provide audio identification signals to the audio system in the aircraft. The audio level of these identification signals are controlled by the volume control on the audio control panel. Indication of magnetic bearing to the VOR station is shown on the RMI single-barred pointer with the selector switch on the RMI positioned at VOR. Glideslope and VOR deviation signals are displayed on the course and flight director indicators. VOR/TAC bearing signals may also be provided by the tacan system as determined by the pilot and copilot navigation/course selector switches.

2.23.10.1 VHF Navigation Control Panel. A set of three concentric, rotary knobs (Figure 2-46) is used to apply electrical power to the system and select the frequency of the navigation system. With a localizer frequency selected on the control panel, the glideslope frequency is automatically tuned and processed by the glideslope module in the navigation receiver. Output signals (glideslope deviation signals) from the glideslope module are applied to the flight director and course indicators. The RMI indicators display bearing information to the selector VOR station. Localizer and glide warning flags in the flight director and course indicators indicate when either signal is too weak for reliable operation or when a malfunction develops in the receiver or the ground station.

2.23.10.2 NAV TEST Button. The NAV TEST button (Figure 2-40) on the VHF control panel is used to initiate a self-test of the navigation receiver. To test the VOR/LOC module within the receiver, a VOR frequency must be selected on the control panel and the course knob set to 00°. As the NAV TEST button is depressed, the warning flags are out of view and the deviation bar should be approximately centered. To test the localizer function of the VOR/LOC module, a LOC frequency must be selected on the VHF navigation control head. As the NAV TEST button is depressed, the deviation bars should indicate a right deflection (approximately one dot on the course indicator) and the glideslope pointers should indicate a down deflection (approximately one dot on the flight director indicator) and the glideslope flags should be out of view. During both tests, the marker beacon lights should be illuminated and flickering. This provides a test of the marker beacon module within the receiver.

2.23.10.3 PILOT and COPILOT NAV/CRS SEL Switches. The PILOT and COPILOT NAV/CRS SEL switches are located on the outboard sections of the instrument panel shroud. These switches select either tacan or the V/L — DME system. Depressing either switch cancels the previous selection. A light within the individual switch illuminates to depict the function selected. With both switches positioned to TACAN or both at V/L — DME, the CPLT RSLVR INOP annunciator light illuminates, indicating that the same selection has been made on both the pilot and copilot switches. In this condition, the course selection is maintained by the pilot; however, the copilot should set his course indicator to correspond with the setting on the pilot course indicator. This provides the same pictorial presentation on both course indicators. The PLT NAV SEL FAIL AND CPLT NAV SEL FAIL annunciator lights on the caution-warning light panel indicate failure of V/L — DME and TACAN switching system. The respective light will illuminate if the switching system does not transfer from TACAN to V/L — DME or from V/L — DME to TACAN. Changing positions of the select switches could restore the system to normal operation.

2.23.10.4 DME HOLD/NORM Switch. The DME HOLD/NORM switch provides a frequency selection hold of the DME receiver. When the switch is positioned at DME HOLD, the DME receiver will remain locked on the selected frequency while the navigation receiver is tuned to a different localizer frequency. When in the NORM position and the navigation receiver is tuned to a different localizer frequency, the DME receiver will immediately tune to the selected frequency.

2.23.10.5 Operation of VHF Navigation System. Refer to paragraph 2.20.2, Operation Of Integrated Flight and Autopilot Systems (CT-39G Aircraft 160053 Through 160057).

2.23.11 PRIMUS-400 Weather Radar (CT-39G Aircraft 158843, 158844, 160053, and 160057). The PRIMUS-400 coloradar system is a lightweight, X-band digital radar with alphanumerics designed for weather detection and analysis and ground mapping. The primary purpose of the system is to detect storms along the flightpath and give the pilot a visual indication in color of their rainfall intensity. After proper evaluation, the pilot can chart a course to avoid these storm areas.

WARNING

The system performs only the functions of weather detection or ground mapping. It should not be used nor relied upon for proximity warning or anticollision protection.

In weather detection mode, storm intensity levels are displayed in three bright colors contrasted against a deep black background. Areas of heaviest rainfall appear in red, less severe rainfall in yellow, moderate rainfall in green, and little or no rainfall in black (background).

Range marks and identifying numerics, displayed in cyan, are provided to facilitate evaluation of storm cells. The cyclic function further emphasizes the heavy rainfall areas (red) of the weather display by flashing them approximately once per second.

Selection of the map function causes system parameters to be optimized on the lowest ranges so as to improve resolution and enhance identification of small targets at short ranges. The reflected signals from various ground surfaces is displayed as magenta, yellow, or cyan (most to least reflective).

The radar system consists of three units: indicator, receiver-transmitter, and antenna. The indicator is mounted in the cockpit and contains all the controls used to operate the radar. The receiver-transmitter is located on a radio rack or other convenient location. The antenna that is fully stabilized for aircraft pitch and roll is mounted within a radome in the nose of the aircraft.

All controls used to operate the system are located on the indicator front panel. These controls and display features are indexed and identified in Figure 2-49.

1. STBY — Pushbutton that selects standby function. Standby is useful for keeping radar in ready state while taxiing, loading, etc. In standby, antenna does not scan, transmitter is not enabled, and display memory is erased. STBY and selected range are displayed.

2. WX — Pushbutton that selects weather detection function. WX is displayed.

3. CYC — Pushbutton that selects cyclic weather display. Displayed red targets will flash on and off once per second. Automatically selects preset level for radar gain. CYC is displayed.

4. MAP — Pushbutton that selects ground mapping function. MAP is displayed.

5. OFF — Pushbutton that turns system off.

6. GAIN — Rotary control with one fixed-gain detented position: PRESET. Used to adjust sensitivity of radar receiver, primarily to resolve nearby strong target signals usually while ground mapping. Sensitivity increases with clockwise rotation. Full counterclockwise rotation to detent sets gain at preset level. When control is not in detented position, VAR is displayed unless pre set gain has been automatically selected.

7. RANGE, 10/25/50/100/200/300/ TEST — Rotary control with seven detented positions that is used to select one of six ranges or test function. Five range marks are displayed for each range.

In test position, special test pattern is displayed to allow verification of system operation/100-mile range and preset range gain are automatically selected and TEST is displayed.

WARNING

Transmitter output power is radiated in test position.

8. STAB ON/OFF — Rotary control used to turn the stabilization function on and off.

9. INT — Rotary control used to regulate brightness of display. Full counterclockwise rotation reduces intensity to zero (black screen).

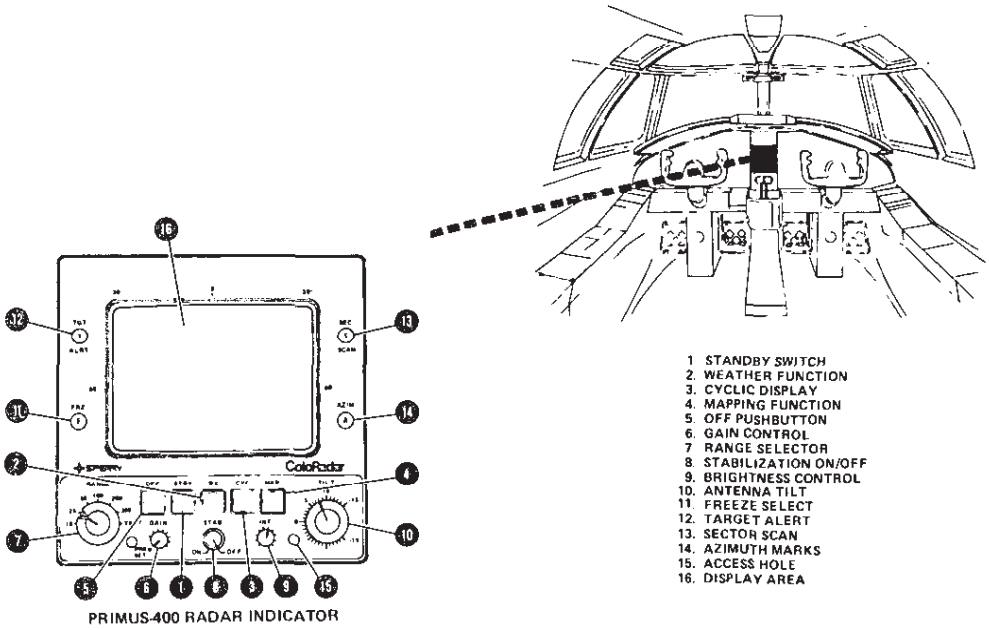


Figure 2-49. Weather Radar PRIMUS-400

Mach One Manuals

10. TILT — Rotary control used to select tilt angle of antenna beam with relation to Earth (with stabilization on) or with relation to airframe (with stabilization off). Clockwise rotation tilts beam upward 0° to 15°; counterclockwise rotation tilts beam downward 0° to 15°.

11. FRZ — Momentary pushbutton that operates alternately to turn freeze function on or off. When "freeze" is selected, display is not updated with incoming target return data. To alert pilot, FRZ label is displayed and is flashed on and off once each second. FRZ is automatically deactivated whenever selection of different control settings dictate a change in displayed data. At system turn-on, FRZ is automatically off.

12. TGT ALRT (target alert) — Momentary pushbutton that operates alternately to turn target alert function on or off. When enabled, letter T in red rectangle is displayed to identify that target alert function is active. Target alert is active only when radar gain is calibrated (i.e., in WX with GAIN PRESET or in CYC or TEST). The symbol TGT in red rectangle is displayed and flashes once each second whenever a red-level target is detected within the target alert sector (range from 60 to 160 nm and within ±7.5° of the aircraft heading). Target alert is deactivated automatically if MAP is selected or if variable GAIN is used, but is reactivated automatically when operating controls are restored to valid alert settings. At system turn-on, the alert function is automatically off.

13. SEC SCAN — Momentary pushbutton that selects an antenna scan sector of either 60° or 120°. At system turn-on, an antenna scan angle of 120° is automatically selected.

14. AZIM — Momentary pushbutton that operates alternately to either display or not display azimuth markers at 30° intervals. At system turn-on, azimuth markers are automatically not displayed.

15. Access Holes — Two holes that provide access to locking pawl screws that secure indicators in aircraft panel.

16. Display area.

2.23.11.1 Initial Control Settings. Place the system controls in the following positions before applying power from the aircraft electrical system:

1. OFF — Depressed.
2. GAIN — PRESET (fully counterclockwise in detent).
3. RANGE — TEST.
4. TILT — +15 (fully clockwise).
5. INT — Fully clockwise.

2.23.11.2 Precautions. If the radar system is to be operated in any mode other than STBY while the aircraft is on the ground:

1. Direct nose of aircraft such that antenna scan sector is free of large metallic objects such as hangars or other aircraft for a distance of 100 feet (30 meters) and tilt antenna fully upwards.
2. Avoid operation during refueling of aircraft or other refueling operations within 100 feet (30 meters).
3. Avoid operation if personnel are standing too close in the 270° forward sector of the aircraft.

Distinctive test patterns are displayed for each system operating mode when the TEST function is selected. The Pilot's Handbook for the PRIMUS-400 radar gives detailed instructions for all ground tests.

WARNING

Transmitter output power is radiated throughout any checkout procedure.

2.23.11.3 Fault Displays. Operation of the radar system is monitored by circuits that warn the pilot if abnormal operation occurs. Visual fault displays are provided for four types of monitored faults: display colors, display sensitivity, transmitter and receiver integrity, and antenna synchronization.

2.23.11.4 Weather Avoidance. Recommended procedures when using the radar for weather avoidance are:

1. Use CYC to flash red areas if additional distinction is desired.
2. Use SCAN to increase the update rate for storms within ±30° of flightpath.
3. Use FRZ to assess movement of rapidly moving storms.
4. Keep TGT ALRT enabled when using short ranges to be alerted if a new storm cell develops in your flightpath.
5. Remember that hail and turbulence present the greatest thunderstorm hazards to aircraft. Therefore, care must be exercised to avoid inadvertently flying into areas or either hail or turbulence.
6. When flying at low altitudes, rotate tilt upward frequently to avoid flying under a thunderstorm.
7. When flying at high altitudes, tilt downward frequently to keep track of low altitude storms so as to avoid flying above storm tops.

Note

If tilt control is left in a fixed position at the higher flight levels, a storm detected at long range may appear to become weaker and actually disappear as it is approached.

8. Use increased gain (rotate GAIN control near its maximum clockwise position) when flying near storm tops. This will help display the normally weaker returns that could be associated with hail.

CAUTION

- Dry hail may be prevalent at higher altitudes within, near, or above storm cells and, since its radar reflectivity is poor, it may not be detected.
- Storms situated behind intervening rainfall may be more severe than depicted on the display.

2.23.11.5 Ground Mapping. Ground mapping operation is selected by pressing the MAP pushbutton. Turn the TILT control down until the desired amount of terrain is displayed. The degree of downtilt will depend upon the type of terrain, aircraft altitude, and selected range. For the low ranges (10, 25, and 50 nm), the transmitter pulse width is narrowed and the receiver bandwidth is widened to enhance the identification of

small targets. In addition, the receiver STC characteristics are altered to better equalize ground target reflections versus range. As a result, the preset gain position will generally provide the desired mapping display. However, the pilot may desire to manually decrease the gain by rotating the GAIN control from its PRESET position to aid in eliminating unwanted clutter.

With experience, the pilot will develop the ability to interpret the color display patterns that indicate water regions, coastlines, hilly or mountainous regions, cities, or even large structures. It is recommended that the pilot practice ground-mapping from time to time during flights in clear visibility where the radar display may be compared with the terrain.

2.24 NOSEWHEEL STEERING SYSTEM

The nosewheel steering system (Figure 2-50) is hydraulic powered and electrically actuated. The steering system receives electrical power from the dc essential bus and hydraulic power from the normal hydraulic system. The nosewheels will steer with normal pilot effort approximately 45° each side of center. If normal hydraulic pressure is lost, an auxiliary accumulator can be selected by a switch in the cockpit to supply auxiliary pressure for the steering system (in addition to pressure for the wheelbrakes and the speedbrake). The steering system is operable when the weight of the aircraft is on the gear and is engaged and disengaged by a pushbutton switch on the pilot or copilot control wheel. When either button is actuated, a solenoid valve is opened, directing hydraulic pressure to the steering system. When the system is engaged, movement of either the pilot or copilot rudder pedals mechanically varies the position of the electrical control potentiometer in the steering system. This energizes the steering control valve to route the hydraulic pressure to the proper side of the steering cylinder piston, moving the nosewheels right or left. Movement of the nosewheels actuates a followup system; and the wheels continue to travel in the desired direction as long as the portion of the steering control system responding to the pedal position remains ahead of the followup system. When the nosewheels reach a position corresponding to the position of the rudder pedals, the steering control and followup systems become electrically balanced. This balance condition deenergizes the solenoid valve and hydraulic pressure to the steering cylinder is shut off. Subsequent movement of the pedals upsets the electrical balance so that the control valve becomes energized and hydraulic pressure to the steering cylinder moves the nosewheels to the new position demanded by the position of the rudder pedals. When the nosewheel steering system is disengaged (steering control valve deenergized), both sides of the steering cylinder are ported to the hydraulic system return through the steering control valve.

The steering system has a dual electrical control system, consisting of a main system and a standby system. Each of these systems functions essentially the same as the single-type steering electrical control system. A switch in the cockpit permits selection of either the main or standby system for operational or test purposes; however, normally MAIN should be engaged so that steering control will be transferred automatically to the standby system if the main system fails.

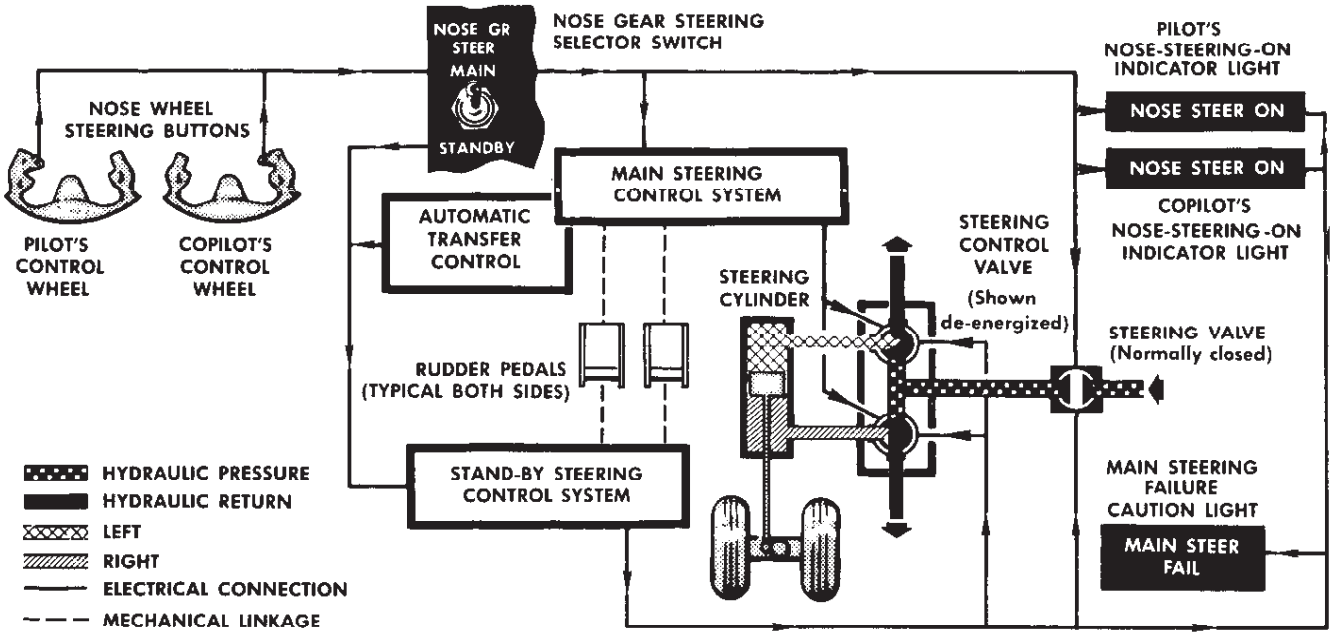
Note

Automatic transfer will not take place unless a nosewheel position out of neutral is selected.

Following automatic or selective transfer to the standby system, the steering remains engaged through the standby control and, during this condition, the steering buttons on the control wheels do not function. The steering selector switch must be used to return steering control from the standby system to the main system, but the main system cannot be operated if a malfunction still exists within the main system even though the main system is selected because the main system will continue to transfer to the standby system. If the standby system fails, there is no automatic transfer to the main system.

2.24.1 Nosewheel Steering Buttons. Two pushbutton switches, one on the outboard grip of each control wheel (Figure 2-29) and powered by the dc essential bus, control hydraulic pressure for nosewheel steering. The nosewheel steering circuit is energized when the weight of the aircraft is on either main gear. However, the nosewheel cannot be turned until the weight of the aircraft is on the nosegear. Pressing and releasing either button engages the steering system by opening a solenoid valve that supplies hydraulic pressure to the steering system. Movement of the rudder pedals will then turn the nosewheels. Pressing and releasing either button again shuts off hydraulic pressure to the system and the nosewheels go to a trail position. The steering buttons do not function when the standby system is operating. (The steering system remains engaged when it is on standby control.)

2.24.2 Nosewheel Steering System Selector Switch. The NOSE GR STEER selector switch (11, FO-3) on the center pedestal is used for selecting main or standby control of the nosewheel steering system. During normal operations, the switch should be at the



CONDITION	COCKPIT INDICATION	REMARKS
Normal operation (main system engaged). Steering selector switch at **MAIN**.	NOSE STEER ON indicator lights on. Normal nose wheel steering.	• Engage or disengage with either steering button. • Automatically disengaged at lift-off.
Main system failure (automatic transfer to stand-by system). Steering selector switch at **MAIN**. **Note** Subsequent failure of stand-by system will not result in an automatic transfer to main system.	NOSE STEER ON indicator lights on. MAIN STEER FAIL caution light on. MASTER CAUTION lights on. Steering buttons have no effect on disengagement. Nose wheels turn in response to rudder pedals.	• If the use of steering is contemplated during the landing roll-out, place the steering selector switch at **STANDBY** only after the nose wheels are on the runway and with the rudder pedals at or about neutral.
Operation with a known failed main system. (Operation in this condition is not recommended.) Prior to operation, pull NOSE GR STEER —MAIN circuit breaker. Steering selector at STBY for ground operations.	NOSE STEER ON indicator lights on. MAIN STEER FAIL caution light on. MASTER CAUTION lights on. Nose wheels turn in response to rudder pedals. Subsequent failure of stand-by system will not result in free-castering nose wheels. Free castering after failure of stand-by system can be accomplished by placing the selector switch back at **MAIN**.	• Disengage steering by placing steering selector switch at **MAIN**. • Automatically disengaged at lift-off. • To avoid premature steering, the steering selector switch should be placed at **MAIN** prior to landing. Place the steering selector switch at **STANDBY** for steering only after the nose wheels are on the runway and with the rudder pedals at or about neutral.
Operation with a known failed stand-by system. (Operation in this condition is not recommended.) Steering selector switch at **MAIN**. Prior to operation: a. Disconnect and stow electrical plug for control box CN515 from nose wheel steering stand-by control box. b. Leave steering selector switch at **MAIN**.	NOSE STEER ON indicator lights on. Nose wheels turn in response to rudder pedals. Subsequent failure of main system will result in free-castering nose wheels.	• Engage or disengage with either steering button. • Automatically disengaged at lift-off. **WARNING** Do not pull NLG STEER – STBY circuit breaker. This would prevent free castering in the event of a main nose wheel steering system failure.

Figure 2-50. Nosewheel Steering

MAIN position so that main system control will be energized when nosewheel steering is engaged. If the main system fails when the selector switch is at MAIN, control of nosewheel steering is transferred automatically to the standby system. (The selector switch remains at the MAIN position when transfer to the standby system occurs.) When the selector switch is moved to STANDBY, the standby steering system is engaged and the main steering system failure caution light comes on. The STANDBY position should be used for test purposes only. When the selector switch is at STANDBY (or when the standby system has been engaged automatically because of main system failure), there is no automatic transfer to the main system if the standby system fails. To return steering control from standby to main, the selector switch should be placed at MAIN. If the standby system has been engaged by the automatic transfer system, it is necessary to place the switch momentarily at STANDBY and then at MAIN to disengage the standby system. The nosewheel steering system selector switch is powered by the dc essential bus.

2.24.3 Main Steering System Failure Caution Light. The placard-type MAIN STEER FAIL amber caution light is on the caution-warning light panel (12, Figure 2-33). It is powered by the dc essential bus and illuminates if the nosewheel steering main system fails and automatic transfer to the standby system has occurred or if the nosewheel steering system selector switch is at STANDBY.

2.24.4 Nosewheel Steering-On Indicator Lights. Two placard-type NOSE STEER ON green indicator lights (1 and 7, Figure 2-33) are on the instrument panel shroud, one on the left side for the pilot and the other on the right for the copilot. These lights are powered by the dc essential bus. Both lights come on to show that the nosewheel steering system is on, whether the steering system is controlled by the main or by the standby system.

2.24.5 Nosewheel Steering Lock Lever. A nosewheel steering lock lever (Figure 2-51) on the front of the nosegear assembly just above the wheel disengages the nosewheel steering cylinder from the nosewheel when the aircraft is towed. The lock lever has been designed so that it is impossible to connect the tow bar (part No. 265-810052) to the aircraft without actuating the lock lever. In case maintenance personnel fail to engage the spring-loaded lock lever when the tow bar is disconnected, the lock lever automatically drops into place when the rudder pedals are actuated during the initial phase of taxiing with nosegear steering engaged. This is a design safety feature only and is not recommended for normal operation.

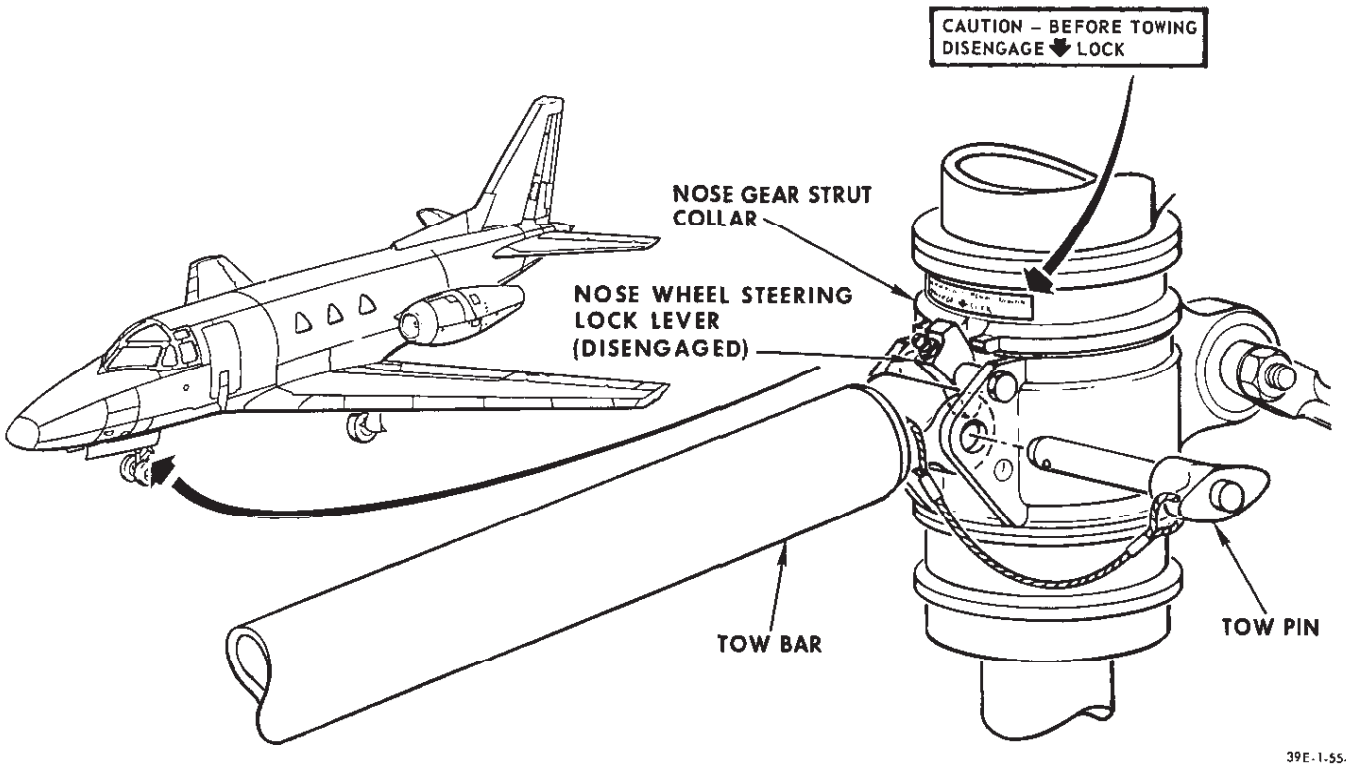


Figure 2-51. Tow Bar Connection

CAUTION

If other than the North American-designed tow bar (part No. 265-810052) is connected to the aircraft, ensure that the lock lever is disconnected from the nosegear strut collar. Otherwise, damage to the collar will occur during towing.

2.24.6 Operation of Nosewheel Steering. After landing when the weight of the aircraft actuates either or both main landing gear load switches and the nosegear load switch, the nosewheel steering may be engaged as follows:

1. Nosewheel steering selector switch — Check MAIN.
2. Either nosewheel steering button — Depress and Release.

   When the nosewheel steering system is energized, the two green nosewheel steering indicator lights come on.
3. Movement of the rudder pedals will then turn the nosewheels.

Note

If during taxiing, the nosegear extends enough to allow the load switch to be actuated, the nosewheel steering is inoperative and steering is again obtained by depressing and releasing the steering button.

2.25 OXYGEN SYSTEM

The pilot and passenger gaseous oxygen system is supplied by a single 74-cubic-foot (CT-39E aircraft and CT-39G aircraft 158843 and 158844) or two 38-cubic-foot, high-pressure (1,800 psi) oxygen cylinder(s) located at the rear of the cabin. A gauge in the cockpit and one on the oxygen system shutoff valve indicate cylinder pressure. Cylinder pressure is converted to operating pressure by an oxygen pressure regulator. Individual diluter-demand regulators and oxygen outlets are provided on the pilot and copilot side consoles. Oxygen is also supplied to the passenger compartment. When required, the passenger oxygen mask compartment door at each passenger position opens automatically and the mask is presented for use. A walkaround bottle is stowed behind the copilot seat. An external filler valve is provided for servicing the system. The oxygen system is shown schematically in Figure 2-52, the oxygen system controls and indicators are shown in Figure 2-53, and the oxygen duration chart is presented in Figure 2-54.

2.25.1 Oxygen Cylinder Pressure Gauges. One oxygen cylinder pressure gauge powered by the No. 1 26-vac bus is on the passenger oxygen control panel. A second gauge is mounted on the cylinder. The gauges indicate cylinder pressure in pounds per square inch before reduction to using pressure.

2.25.2 Oxygen Overboard Dump Indicator. Normal servicing pressure of the oxygen bottle is 1,800 psi. Should bottle pressure rise to 2,500 to 2,700 psi, a blowout plug in the bottle releases and the oxygen supply is dumped overboard. A green indicator disk in the outlet of the oxygen overboard dump line is blown out when this occurs.

2.25.3 Pilot Oxygen Regulators. The pilot and copilot each have a combination pressure-breathing, diluter-demand-type oxygen regulator (Figure 2-53) that mixes cabin air with oxygen in varying amounts according to altitude. At high altitudes, the regulator supplies oxygen at continuous positive pressure. The delivery pressure automatically changes with cabin altitude. Each oxygen regulator control panel includes an oxygen pressure gauge, a flow indicator, and control levers.

Note

Above 30,000 feet cabin altitude, a vibration wheezing sound may sometimes be noticed in the mask. This is a normal characteristic of regulator operation.

2.25.3.1 Supply Lever. The green supply lever controls oxygen to the regulator and has an ON and an OFF position.

2.25.3.2 Diluter Lever. The white diluter lever has a NORMAL OXYGEN and a 100% OXYGEN position. With the lever at NORMAL OXYGEN, the regulator supplies a mixture of cabin air and oxygen up to about 30,000 feet that is equivalent to a normal breathing mixture at sea level. Above 30,000 feet, 100-percent oxygen is automatically supplied. The lever is placed at 100% OXYGEN for emergency use. These operating characteristics are related to cabin altitude only.

2.25.3.3 Emergency Lever. The red emergency lever has three positions: EMERGENCY, NORMAL, and TEST MASK. The lever should be at NORMAL (center position) at all times. When oxygen pressure

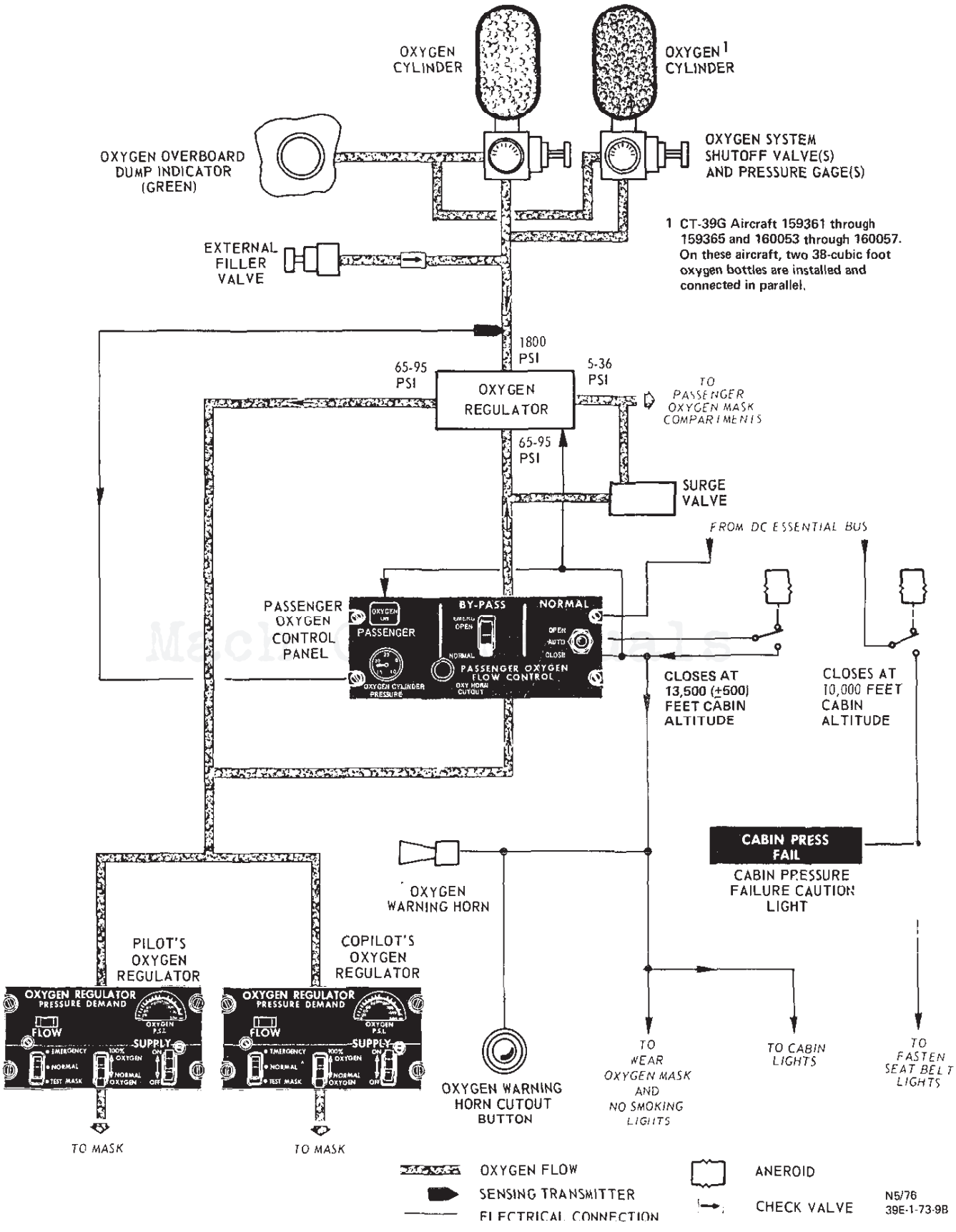


Figure 2-52. Oxygen System

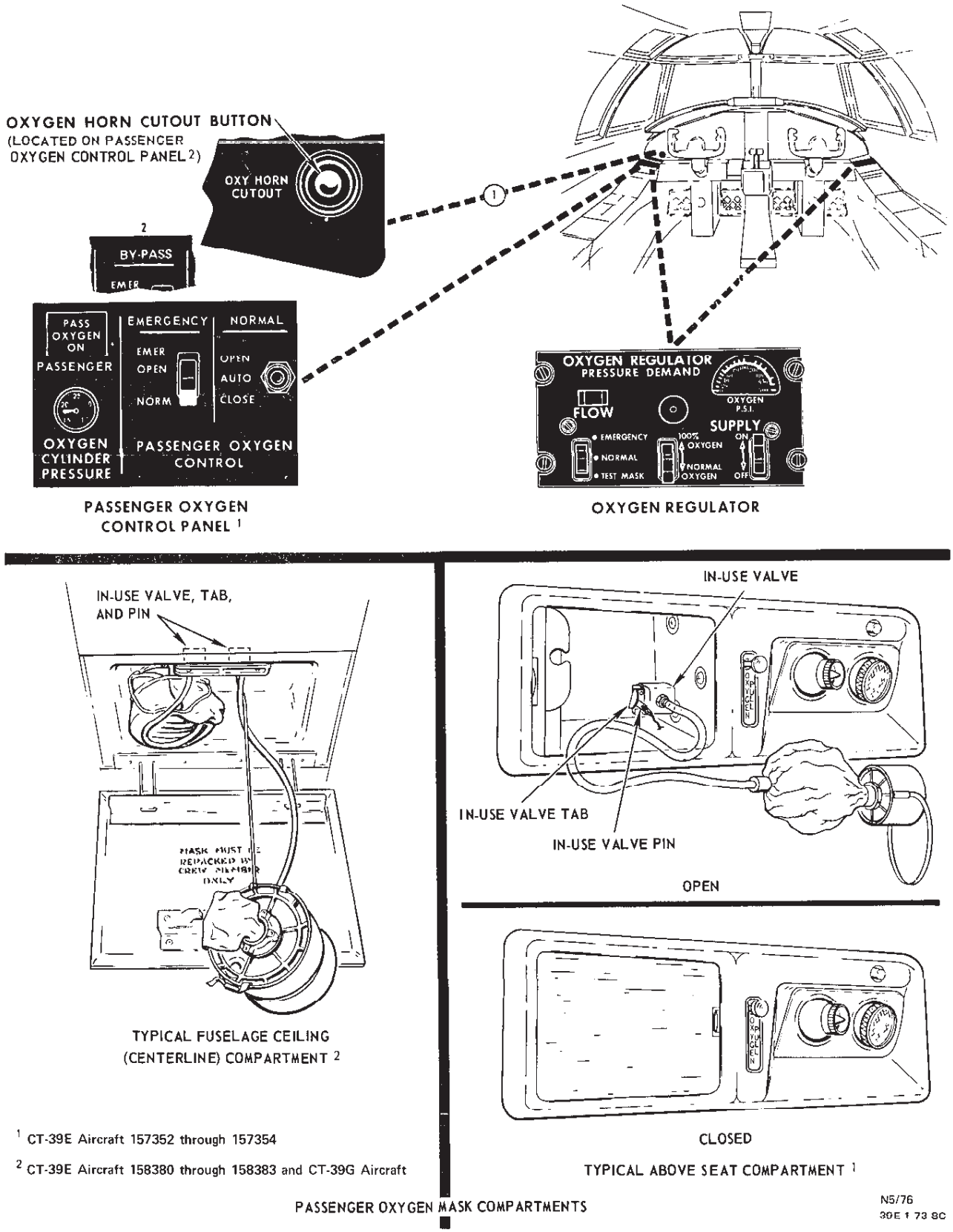


Figure 2-53. Oxygen System Controls and Indicators

Figure 2-54. Oxygen Duration

ORIGINAL

increase is desired, moving the lever to EMERGENCY provides continuous positive pressure to the mask. When the lever is held at TEST MASK, oxygen at positive pressure is provided to test the pilot masks for leaks.

2.25.3.4 Pressure Gauge and Flow Indicator. The pressure gauge shows reduced oxygen system pressure available at the regulator. The flow indicator (blinker) consists of an oblong opening that shows black and white alternately during the breathing cycle.

2.25.4 Oxygen Warning Horn. The oxygen warning horn, mounted on the rear pressure bulkhead of the cabin, is powered by the dc essential bus and is armed when oxygen pressure is applied to the passenger oxygen system. The warning horn will sound when the cabin altitude reaches 13,500 ±500 feet. At that time, the cabin lights and WEAR OXYGEN MASK and NO SMOKING signs will also come on. The oxygen warning horn cutout button is used to silence the oxygen warning horn.

2.25.4.1 Oxygen Warning Horn Cutout Button. The oxygen warning horn cutout button (Figure 2-53) is powered by the dc essential bus. On CT-39E aircraft and CT-39G aircraft 158843 and 158844, the button is on the pilot outboard instrument panel. On CT-39G aircraft 159361 through 159365 and 160053 through 160057, the button is on the passenger oxygen control panel. Pressing this button momentarily when the oxygen warning horn sounds will silence the horn.

2.25.5 Passenger Oxygen Control Panel. The passenger oxygen control panel (Figure 2-53) on the pilot left console contains an oxygen cylinder pressure gauge, a passenger-oxygen-on indicator light, an emergency toggle valve, and a normal switch. On CT-39G aircraft 159361 through 159365 and 160053 through 160057, the panel also contains the oxygen warning horn cutout button.

2.25.5.1 Oxygen Cylinder Pressure Gauge. The OXYGEN CYLINDER PRESSURE gauge is powered by the 26-vac bus. This gauge indicates oxygen cylinder pressure (0 to 1,800 psi) before the oxygen pressure is reduced to using pressure.

2.25.5.2 Passenger-Oxygen-On (PASS OXYGEN ON) Indicator Light. The red press-to-test PASS OXYGEN ON indicator light illuminates whenever the solenoid shutoff valve in the oxygen pressure regulator is open, indicating that oxygen is being supplied to the passenger oxygen system.

2.25.5.3 EMERGENCY (BY-PASS) Toggle Valve. The two-position EMERGENCY (BY-PASS) toggle valve controls a plumbing loop that bypasses the solenoid shutoff valve in the oxygen pressure regulator. When the valve is positioned at NORMAL, the valve is closed and there is no effect on the oxygen flow. When the valve is positioned at EMER OPEN, the valve is open and 65 to 95 psi flow from the pilot side of the system bypasses the solenoid shutoff valve and flows directly to the aneroid-type regulator.

2.25.5.4 NORMAL Switch. The three-position NORMAL switch controls the passenger oxygen system solenoid shutoff valve that controls oxygen flow to the passenger oxygen system. With the NORMAL switch in the normal AUTO position, the solenoid shutoff valve is controlled by an aneroid switch that closes at 13,500 ±500 feet cabin altitude and opens the solenoid shutoff valve. With the NORMAL switch in the OPEN position, the aneroid switch is bypassed and the solenoid shutoff valve is energized open directly. With the NORMAL switch in the CLOSED position, the circuit is deactivated, keeping the solenoid shutoff valve closed.

2.25.6 Oxygen Pressure Regulator. The oxygen pressure regulator, mounted just above and outboard of the oxygen cylinder, regulates oxygen pressure to the pilot and copilot oxygen system, and controls and regulates oxygen pressure to the passenger oxygen system. Included in the regulator are a pressure regulator, an electrically actuated solenoid shutoff valve, and an aneroid-type flow regulator. The pressure regulator reduces oxygen pressure from 1,800 psi to 65 to 95 psi for delivery to the pilot and copilot system. The solenoid shutoff valve is powered by the dc essential bus through a circuit controlled by the oxygen normal switch and the actuation of an aneroid-type altitude switch. When the solenoid shutoff valve is energized open, oxygen is supplied to the passenger system. The aneroid-type flow regulator further reduces the oxygen pressure as necessary (0 to 36.5 psi) to the passenger oxygen system.

2.25.7 Passenger Oxygen Masks. A passenger oxygen mask is enclosed in a compartment above each seat position on CT-39E aircraft and CT-39G aircraft 159361 through 159365. The passenger oxygen masks on CT-39G aircraft 160053 through 160057 are contained in compartments in the cabin ceiling (along the fuselage centerline). Latched doors (Figure 2-53) hold the mask(s) within the compartments until oxygen is needed. When oxygen is required, the latches are released by oxygen pressure and the doors open to allow the oxygen masks to drop out. Pulling the mask down for use pulls a pin from the in-use valve tab and provides

oxygen flow to the mask. Oxygen flow to the mask is stopped by raising the in-use tab and reinstalling the pin. On CT-39E aircraft and CT-39G aircraft 158843 and 158844, the oxygen compartment doors can also be manually opened by pulling the door-opening button.

2.25.8 Cabin Signs. Three white cabin signs can be seen from each passenger position. The FASTEN SEAT BELT sign comes on simultaneously with the cabin pressure failure light any time the cabin altitude exceeds 10,000 feet. The WEAR OXYGEN MASK and NO SMOKING signs come on when the cabin altitude is 13,500 ±500 feet. At this time, the cabin lights also come on and the oxygen warning horn sounds. The FASTEN SEAT BELT and NO SMOKING signs can also be turned on by switches in the cockpit.

Note

WEAR OXYGEN MASK signs not installed in all aircraft.

2.25.9 Oxygen Walkaround Bottle. An oxygen walkaround bottle is mounted behind the copilot seat. The bottle is complete with a constant flow regulator, a pressure gauge, an ON-OFF valve, a relief valve, a charging valve, a hose, a mask, and sling strap attached to the bottle. The bottle has a capacity of 11 cubic feet at 1,800 psi. To recharge the walkaround oxygen bottle, it must be removed from the aircraft.

2.25.10 Oxygen System Preflight Check

Note

Any reference to oxygen regulators applies to the pilot or copilot oxygen regulators.

1. Oxygen system shutoff valve — Open.
2. Oxygen cylinder pressure gauge — Check.

   Fully charged cylinder is 1,800 psi at 70 °F.
3. Passenger oxygen normal switch — CLOSE.
4. Oxygen regulator supply lever — Safetied ON.
5. Oxygen regulator gauge — Check.

   Gauge should read 80 ±15.
6. Oxygen regulator diluter lever — NORMAL OXYGEN.
7. Oxygen regulator — Blow check.

   Blow gently into end of oxygen regulator hose as during normal exhalation. There should be resistance to blowing. Little or no resistance to blowing indicates a leak or faulty operation.
8. Oxygen regulator diluter lever — 100% OXYGEN.
9. Oxygen regulator — Blow Check.

   Blow gently into end of oxygen regulator hose as during normal exhalation. There should be resistance to blowing. Little or no resistance to blowing indicates a leak or faulty operation.
10. Oxygen mask hose — Connect.
11. Oxygen mask — Put On.
12. Oxygen flow indicator — Check.
13. Oxygen regulator emergency lever — EMERGENCY.

    Hold breath to determine whether there is leakage around mask.
14. Oxygen regulator emergency lever — NORMAL.
15. Passenger oxygen normal switch — AUTO.
16. Passenger oxygen emergency toggle valve — NORMAL.

2.26 PILOT SEATS

Two crew seats are provided in the pilot compartment. Each seat is equipped with a quick-disconnect combination lap belt and shoulder harness, two armrests which are raised upward and aft to a stowed position, and adjustment levers for vertical as well as fore-and-aft movement.

2.26.1 Pilot Seat Adjustment. The pilot and copilot seats may be individually adjusted manually by two levers beneath each seat on the inboard side. Fore-and-aft adjustment is accomplished by aft movement of the lever that extends upward from the floor. Vertical adjustment is accomplished by raising the horizontal lever and holding it in that position until the proper adjustment is made. Releasing the lever allows the locking pin to enter an adjustment hole. Both the vertical and horizontal adjustments are in 1/2-inch increments with a full travel of 5 inches.

2.26.2 Safety Belt and Shoulder Harness. The pilot and copilot seats are equipped with adjustable seatbelts that are attached to the seat frames. The entire harness assembly consists of a lap belt, a crotch strap, and a shoulder harness. The lap belt and crotch strap are attached separately to the seat. Each strap has an attachment fitting that is inserted into a single, quick-release buckle attached to one side of the seatbelt. Turning the face of the buckle approximately one-eighth turn in either direction immediately releases all plug-in connections. In addition, there is an alternate release at the top of the buckle for the shoulder straps only. This feature permits the convenience of extended forward movement of the head and shoulders without releasing the complete harness assembly. A dual inertia reel for the shoulder harness straps is provided within the bulkhead immediately aft of each seat. No manual locking device is provided for the shoulder straps as the automatic locking device within the inertia reel locks at a maximum forward acceleration of 1.4g and functions with the reel in any position. When the shoulder straps are released from the belt buckle, they retract, eliminating the hazard of loose straps in the pilot compartment.

2.27 PILOT SLIDING WINDOW

The forward side window on the pilot side may be opened on the ground. If necessary, the window may be opened in flight to aid in evacuation of smoke and fumes or during landing if forward vision is obscured. The pilot sliding window has a positive lock system that includes a hinge-type lock attached to the forward edge of the window frame. The lock has a tab (OPEN HINGE BEFORE OPENING WINDOW. CLOSE HINGE AFTER CLOSING WINDOW.) for easy opening. Moving the hinge outboard locks the window.

1. To unlock and open the window: Pull hinge lock tab inboard. Grasp lock handle (at forward lower edge of window) and pull inboard to clear lock detent. Pull aft until window is fully opened, then push handle outboard into lock detent.

2. To close and lock the window: Pull lock handle inboard to clear lock detent, then move handle forward until window is fully closed. Push handle outboard into lock detent. Move hinge lock tab outboard.

2.27.1 Radio Altimeter (CT-39E, CT-39G Aircraft 158843 and 159362 Through 159365). The radio altimeter (Figure 2-55) on the pilot instrument panel measures terrain clearance from an altitude of 2,500 feet to touchdown by a white pointer that rotates against a calibrated dial. Calibrations are linear from 0 to 500 feet and nonlinear above 500 feet to 2,500 feet. A black mask is fixed above the 2,500-foot mark on the dial. A pushbutton is for self-testing the altimeter. A red warning flag comes into view in case of instrument malfunction or failure to indicate the instrument is unreliable. The flag will also appear when the radio altimeter switch is at OFF. The test function is inoperable with the autopilot engaged. A minimum decision altitude knob on the indicator is used for setting a yellow minimum decision altitude index on the dial. Descending below the altitude set by this knob causes the MDA light (4 and 13, Figure 2-39), labeled MDA on the pilot and copilot instrument panels, to come on and remain on as long as the aircraft is below the selected altitude. On CT-39G aircraft, the MDA light is on the pilot and copilot flight director indicator. The minimum decision altitude index disappears when the aircraft is above the maximum indicating altitude of 2,500 feet or when failure occurs. On CT-39G aircraft during an ILS approach, the radio altimeter provides signals to drive the flight director indicator runway symbol vertically during the last 200 feet of descent. The radio altimeter is powered by the dc essential bus and by the ac primary bus through the radio altimeter switch.

2.27.1.1 Radio Altimeter Switch. The rocker-type radio altimeter switch (9, Figure 2-39) on the instrument panel shroud has ON and OFF positions. When the ON portion of the switch is pressed, power is applied to the radio altimeter. Pressing the lower portion of the switch disconnects power to the radio altimeter. The switch is powered by the ac primary bus.

2.27.2 Radio Altimeter (CT-39G Aircraft 158844 and 160053 Through 160057). The radio altimeter (Figure 2-55) on the pilot instrument panel measures terrain clearance from an altitude of 2,000 feet to touchdown by a yellow pointer that rotates against a calibrated dial. Calibrations are linear from 0 to 500 feet and nonlinear above 500 feet to 2,000 feet. A black mask is fixed above the 2,000-foot mark on the dial. A pushbutton is for self-testing the altimeter. An orange warning flag comes into view in case of instrument malfunction or failure to indicate the instrument is unreliable. The flag will also appear when the RADIO ALTM ON switch is depressed. An MDA knob on the indicator is used for setting a yellow minimum decision altitude index on the dial. Descending below the altitude set by this knob causes the decision height light, labeled DH on the pilot and copilot flight director indicators, to come on and remain on as long as the aircraft is below the selected altitude. During an ILS approach, the radio altimeter provides signals to drive the flight director indicator runway symbol vertically during the last 200 feet of descent. The radio altimeter is powered by the dc essential bus.

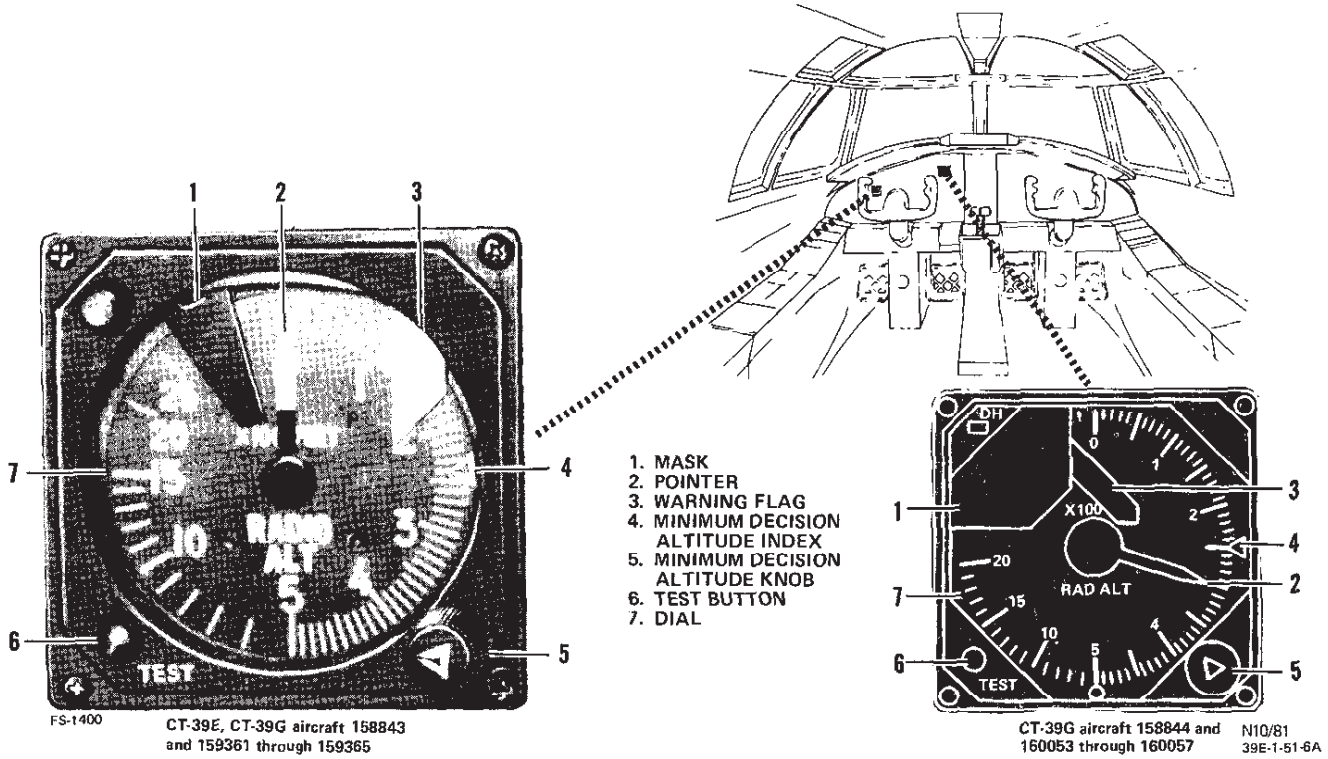


Figure 2-55. Radio Altimeter

2.27.2.1 RADIO ALTM ON Switch. The RADIO ALTM ON switch (25, FO-4) on the instrument panel shroud is a pushbutton type switch. When the switch is depressed, power is applied to the radio altimeter system. A light within the switch illuminates when the radio altimeter system is energized. The switch is powered by the 28-vdc essential bus.

2.27.2.2 Radio Altimeter TEST Switch. The radio altimeter TEST switch (Figure 2-55) is installed on the radio altimeter indicator. With the DH set to 25 feet and the TEST button depressed, the DH indicator should light and the indicator should depict 50 ±5 feet and the flag should be visible.

2.28 SPEEDBRAKE SYSTEM

A hydraulically operated, electrically actuated speedbrake is on the lower surface of the fuselage aft of the nosegear well. The speedbrake panel is hinged at its leading edge and, when open, extends down and forward to increase drag. The speedbrake may be operated at any airspeed by a switch on the left throttle.

2.28.1 Speedbrake Maintenance Safety Locks. Two removable maintenance safety locks may be installed to keep the speedbrake in the extended position when the aircraft is on the ground. (See Figure 2-43.) A lock assembly is inserted on each of the two speedbrake actuating cylinders. The lock assemblies which have conventional red warning streamers shall be removed before flight.

2.28.2 Speedbrake Switch. A speedbrake switch (Figure 2-23) on the left side of the left throttle controls speedbrake operation. Moving the switch aft to OUT opens the speedbrake; moving the switch forward to IN closes it. The switch supplies dc essential bus power to a control valve that directs hydraulic system pressure to the speedbrake actuating cylinders. When the switch is returned to the OFF (center) position, the control valve is closed (trapping cylinder pressure) and the brake is held in the desired position. After the brake has been retracted, the switch should be placed in the forward IN position to prevent the speedbrake from creeping open in flight.

2.28.3 Speedbrake Emergency Dump Switch. The two-position emergency dump switch (79, FO-1) on the pilot instrument panel permits the speedbrake to be retracted in flight if hydraulic pressure fails. When the switch is moved to DUMP (after first placing the speedbrake switch in the OFF position), dc essential bus power opens a dump valve that relieves hydraulic pressure from the speedbrake actuating cylinders. Air loads then force the speedbrake to a trail position. The dump switch is spring loaded to NORM. Releasing the switch to NORM allows the dump valve to close, trapping hydraulic fluid in the actuating cylinder to hold the speedbrake in the trail position. The dump valve is reset,

allowing normal operation of the speedbrake system when the emergency dump switch is returned to NORM.

2.28.4 SPEED BRAKE OPEN Caution Light. The amber SPEED BRAKE OPEN caution light on the caution warning light panel (12, Figure 2-33) is powered by the dc essential bus. The light comes on when the speedbrake is not closed and if either throttle is at a setting of approximately 95-percent rpm or greater. The light goes out when the speedbrake is closed or when both throttles are retarded below 95-percent power. The function of the light is to alert the pilot if the speedbrake is not closed during takeoffs and go-arounds. When the SPEED BRAKE OPEN caution light is on, the master caution lights come on.

2.29 WHEELBRAKE SYSTEM

The hydraulic brake system consists of a combination power system with a manual emergency system and incorporates an antiskid system. Hydraulic pressure for the wheelbrake system is available for (1) the normal hydraulic system, (2) the auxiliary hydraulic system, and (3) the manual emergency system when the emergency brake T-handle is pulled out. The braking units on each main landing gear wheel consist of rotor and spot brake assemblies. Braking effect is obtained by toe action on the rudder pedals that meters hydraulic pressure to apply the brakes. The wheelbrake system operates from 3,000 psi; but because of the metering of the brake control valves, the maximum pressure to the wheelbrakes is 1,800 psi. The emergency braking system has its own fluid supply and separate brake lines to brake-mounted shuttle valves. If no hydraulic pressure is available from the normal or auxiliary hydraulic system, pulling the emergency brake control T-handle in the cockpit will permit the brake master cylinders to function as conventional brake master cylinders when enough toe pressure is applied to the top of the rudder pedals.

WARNING

The brake pedals are mechanically interconnected. Emergency braking procedure (pumping action) cannot be accomplished if the opposite set of brake pedals is being depressed.

An auxiliary accumulator, selected by a switch in the cockpit, supplies auxiliary hydraulic pressure for the nosewheel steering, wheelbrakes, and speedbrake if normal hydraulic pressure is lost.

Note

Each main gear wheel is equipped with three fusible plugs that melt and release tire pressure before the wheel and tire become dangerously overheated.

2.29.1 Parking Brake Handle. The parking brake T-handle (77, FO-1) is on the pilot instrument subpanel to the right of the control column. To set the parking brakes, toe action is applied to the rudder pedals and the parking brake handle is pulled out. The parking brake may be set as long as hydraulic pressure is available in either accumulator. A pressure compensator in the parking brake system valve maintains approximate parking brake pressure regardless of any thermal expansion or contraction of hydraulic fluid when the parking brakes are set. Hydraulic pressure required to release the parking brakes is obtained by toe pressure applied to the rudder pedals. The release of the parking brake is indicated by the T-handle returning to the normal (in) position. The emergency brake control handle must be pushed in if it has been used before the parking brakes can be applied.

2.29.2 Emergency Brake Control Handle. The emergency brake control T-handle (9, FO-3) is on the center pedestal. To select emergency braking, the control handle must be pulled to full travel which converts the brake valves from a power type to a manual type. Then, by pushing on the top of the rudder pedals, braking is proportional to the amount of effort exerted. The emergency brake system is supplied hydraulic fluid from two small reservoirs in the nose wheelwell. The two reservoirs are connected to the wheelbrake system by a common supply line.

The emergency brake control handle is the rotate-to-stow type. This T-handle incorporates a spring-loaded blade that provides a positive lock when the handle is in the extended position. By rotation of the handle 90° counterclockwise, the handle is unlocked and may then be pushed in. The handle rotates 90° clockwise as it returns to the normal stowed position.

2.29.3 Antiskid System. An electrically controlled, hydraulically operated antiskid system prevents excessive skidding. A skid detector for each main wheel senses electrically any sudden change in wheel rotation speed. As the skid is detected, an electrical signal is supplied to remove hydraulic pressure from either brake. With brake pressure removed, the wheel speed will increase and hydraulic pressure will be restored to the brakes. The antiskid system continues this on-off cycling as long as braking pressure is sufficient to cause the skidding condition. This is not an automatic braking

system; therefore, maximum braking should not be applied at touchdown and held throughout landing roll. Should antiskid cycling be felt, relax pressure on brakes slightly, then reapply brakes, attempting to remain below the pressure that results in cycling. Brake pressure may be increased as the aircraft slows down with less chance of cycling.

If a malfunction develops during operation, the entire antiskid system is electrically rendered inoperative and the antiskid brake failure caution light illuminates. Complete control of braking is returned to the pilot. This will occur whenever brake pressure is held off by the antiskid system for a continuous period of approximately 3 seconds.

The antiskid switch may be placed at OFF and then returned to ON in order to reset the antiskid system. If the malfunction has been corrected, normal antiskid operation will result; if not, the system will again be rendered inoperative and normal pilot braking should be used during the remainder of the landing.

If a landing is being made using the auxiliary hydraulic system, it is recommended that the antiskid system be turned off to conserve hydraulic accumulator pressure.

Note

- Skid protection is not available whenever the emergency brake system is in operation or at aircraft speeds less than 10 knots.
- As wheelbrakes are applied, to stop wheel rotation after takeoff, the ANTISKID BRAKE FAIL and MASTER CAUTION light may illuminate. This is a normal condition, and the lights may be extinguished by momentarily placing the antiskid switch at OFF and then back to ON.
- Landing with the antiskid system on using auxiliary hydraulic accumulator pressure can result in rapid depletion of available auxiliary hydraulic pressure.

2.29.3.1 ANTI-SKID Switch. The ANTI-SKID switch (25, FO-3) on the overhead control panel controls 28-vdc essential bus power to the antiskid system. Placing the switch at ON makes electrical power available to the system to cycle the antiskid control valves. Placing the switch at OFF renders the antiskid system and caution light inoperative.

2.29.3.2 Antiskid Brake Failure Caution Light. The ANTISKID BRAKE FAIL caution light on the caution-warning light panel (12, Figure 2-33), illuminates whenever a malfunction exists in the system. This occurs any time brake pressure is removed by the antiskid system for a continuous period of approximately 3 seconds. When the caution light comes on, the system is in a fail-safe condition and complete control of braking through the normal brake system is returned to the pilot.

2.30 WING FLAP SYSTEM

Electrically operated, slotted-type wing flaps extend from the aileron to within approximately 2 feet of the fuselage on each wing panel. Each flap is operated by an individual electrical actuator through an individual electric circuit. The actuators are mechanically interconnected by a flexible shaft to synchronize the flap travel. Each actuator is capable of driving the opposite flap through the synchronizing shaft along with the flap to which it is attached. Therefore, in case of a power or motor failure to one actuator, flap positioning (at a reduced speed) is still obtainable. The actuators are of the screw-jack type and are mechanically irreversible, thus preventing air loads from moving the flaps. No emergency system is provided as there is enough protection in the normal system through the mechanical interconnection, individual electric motors, and individual electric circuits.

2.30.1 Wing Flap Handle. The wing flap handle (8, FO-3) is recessed in the center pedestal to prevent inadvertent operation. The flap handle is powered by the dc essential bus. On CT-39E aircraft, the flap handle moves in a quadrant marked UP, HOLD, and DN. To position the flaps full up or full down, the flap handle is moved to the corresponding position and left there. (Power is automatically removed from the actuators when the flaps reach the extreme position.) The flaps will extend to the full-down position in 10 to 15 seconds during flight and will retract in 5 to 10 seconds. Intermediate flap positions may be selected by monitoring the flap position indicator during flap operation and placing the flap handle at HOLD when the desired flap position is reached.

On CT-39G aircraft 158843, 158844, and 159361 through 159365, the wing flap quadrant is marked UP, APPR, and DN. On CT-39G aircraft 160053 through 160057, the wing flap quadrant is marked UP, PART, and DN. When the approach (APPR) or (PART) flap setting is selected, power is automatically removed from the flap actuators when the flaps extend (or retract) to the 66-percent setting. The flap handle has a lever lock feature that allows unimpeded movement

between the UP and APPR or PART positions. To move the handle to the DN position requires that the handle be pulled out against a spring load to clear a one-way gate. The switch may be freely returned from the DN to the APPR, PART, or UP position. To permit the flap handle to be grasped and pulled out to clear the gate, its shape incorporates a bulge at the trailing edge.

2.30.2 Wing Flap Position Indicator. A wing flap position indicator (63, FO-1) on the copilot instrument panel is powered by the dc essential bus. The instrument is calibrated in percent of flap extended and reads from UP to DOWN in increments of 10 percent. Full flap travel is 25° (100 percent). This indicator facilitates the selection of intermediate flap positions.

2.31 WING SLATS

Wing slats extend along the leading edge of each wing. Slat operation is automatic and no pilot-operated controls are provided. Aerodynamic forces, a function of indicated airspeed and angle of attack, cause the slats to open and close. As the direction of forces exerted on the wing leading edge is changed, the slats move forward along a curved track to open. The slot formed by the opened slat redirects airflow over the wing upper surface, thereby restoring wing efficiency. The slatted wing may then be flown at an angle of attack higher than or at stall speeds lower than those for the same unslatted wing. When the angle of attack is reduced, the aerodynamic forces again change direction, causing the slats to close. Some slate movement occurs during the takeoff run; however, it is not until rotation that conditions prevail that maintain the slats open. After takeoff, the slats begin to close at approximately 140 KIAS and are fully closed at approximately 180 KIAS. During climb and 1g cruise, the slats remain closed. In flight, slat operation is dependent upon a combination of g forces, airspeed, gross weight, and altitude. The slats begin to open during 1g flight as airspeed is reduced below 180 KIAS.

2.32 MISCELLANEOUS EQUIPMENT

2.32.1 Windshield Wipers. Electrically operated windshield wipers keep the exterior of the windshield panels in front of the pilot and copilot clear during adverse weather conditions. The multispeed wipers are controlled by a switch in the cockpit. The windshield wipers are designed to be used for ground operations and at airspeeds below 180 KIAS.

Note

- Do not operate the wipers if the windshield is dry or dirty as the windshield surfaces may be damaged.
- The windshield wiper low-speed position is not recommended for in-flight use. Do not operate continuously on the low-speed position (5 to 10 minutes maximum).

2.32.1.1 Windshield Wiper Knob. The five-position windshield wiper knob (29, FO-3) on the overhead control panel is powered by the dc essential bus and has PARK, OFF, HIGH, MED, and LOW positions.

Turning the knob to PARK powers the wipers to a stowed position. The HIGH, MED, and LOW positions are used to select the desired speed of the wipers. Turning the knob to OFF shuts off all power to the wiper motor.

2.32.2 Passenger Seats. On CT-39E aircraft, the passenger seats in the aft cabin area consist area consist of two forward-facing seats, two aft-facing seats, and a three-passenger couch and a center facing belted lavatory seat on the left side of the intermediate cabin area just aft of the entry door. A chemical toilet is installed under the seat.

The aft cabin area of CT-39G aircraft 158843, 158844, and 159361 through 159365 and 160053 through 160057 contains six individual cabin seats, an aft three-seat couch, and a belted lavatory seat for a maximum seating arrangement of 10 passengers. The lavatory seat is located immediately aft of the entry door bulkhead.

Each seat position is equipped with a seatbelt, and the forward- and aft-facing seats each have an arm rest on the aisle side of the seat. The aft-facing seats also have padded headrests.

Note

All seat folding armrests must be stowed for takeoff and landing.

2.32.3 Plane Captain/Passenger Seat (Toilet). A plane-captain/passenger seat is on the left side of the intermediate cabin area just aft of the entry door. This center-facing seat is equipped with a seatbelt, and a

chemical toilet is installed under the seat. A sliding curtain is provided to isolate the area from the cockpit and a sliding door isolates the area from the aft cabin. In addition, a paper dispenser, a waste paper container, and a 110-vac outlet for an electric razor are located in the area.

2.32.4 Jump Seat. A removable jump seat, installed in the cockpit aisle for in-flight use only, is provided for a cockpit observer. The seat is supported at each side by pins that fit into slotted guides attached to the cockpit doorway frame. When the seat is installed, the plate with the instructions for "INFLIGHT USE ONLY" should be on the right side. The jump seat is to be stowed during takeoff and landing. It may be stowed in the baggage compartment when not in use or it may be left hanging from its right-hand side support pin attached to the doorway frame.

2.32.5 Baggage Compartment. A baggage compartment is on the right side of the intermediate cabin area just aft of the cockpit and across from the entry door. Provisions are included in the baggage compartment for hanging up coats.

2.32.6 Flight Report Holders. A combination aircraft flight report holder and map pocket is on the back of the pilot and copilot seats.

2.32.7 Seat Warning Indicator Light (CT-39G Aircraft 158843, 158844, and 159362 Through 159365). The warning light is in the right-hand corner of the instrument panel (10A, Figure 2-33) and is powered by the dc essential bus. The SEAT WRN placard-type amber warning light illuminates when the fasten (seat) belt switch is placed at ON if the tracked cabin seats are not in the proper position for takeoff and landing. The left-hand aft-facing seat must be in full forward position (against the forward bulkhead), and the middle right-hand seat must be in the forward (outboard) position.

2.32.8 Cabin Call Indicator Light and Reset Button (CT-39G Aircraft 158843 and 158844). The placard-type cabin call (CBN CALL) indicator light is on the electrical control panel between the dc loadmeters (11A, Figure 2-33 and 61A, FO-1) and is powered by the dc secondary bus. The amber indicator light and a buzzer can be activated by a CKPT CALL button at the right-hand center passenger seat. A PRESS TO RESET reset button above the light on the top of the electrical control panel (61A, FO-1) is provided to turn the light off.

2.32.9 Mooring and Jacking Kit. Provisions are made on the aircraft for mooring and jacking. All necessary fittings are included in a mooring and jacking kit.

2.32.10 Protective Covers. Removable covers include pitot covers, exhaust duct covers, engine air inlet weather shields, and a crew compartment canopy cover.

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2.32.11 Lightning Protection. Damage to the aircraft by lightning is minimized by the use of metal lightning protection strips mounted on the radome and wingtips. Two whip-type lightning arrester rods, one mounted above and one mounted below the fuel vent outlet, prevent damage to the fuel vent outlet flame suppresser.

Mach One Manuals

CHAPTER 3

Servicing

3.1 GENERAL INFORMATION

The servicing data contained in this chapter is provided to lend assistance to the crewmembers during transit maintenance operations in the event the maintenance crews are unfamiliar with the aircraft. This chapter is divided into four categories: general information, starting requirements, danger areas, and external power requirements. The general information category contains information necessary to ensure proper servicing of the aircraft. (See Figure 3-1.)

3.1.1 Refueling

3.1.1.1 Single-Point Refueling System. The single-point refueling system permits all internal fuel tanks to be filled on the ground by means of a pressure refueling system connected to a single-point refueling receptacle (Figure 3-2) on the right wing, inboard, below the leading edge. If a refueling pressure of approximately 50 psi is maintained, the wing tanks fill in approximately 11 minutes and all tanks fill in about 12 minutes. The system consists of one remote-control shutoff valve and one level control and vent valve in each wing; one dual-level control valve in the fuselage tank; a pressure refueling test and fuselage tank refueling valve; and fuel pressure, venting, and sensing lines.

During single-point refueling, fuel flows from the receptacle to the remote-control shutoff valve and from there the fuel is discharged into each wing tank. As each wing tank becomes full, the level control and vent valve senses the full condition and signals (through fuel pressure-sensing lines) the remote-control shutoff valve to close. To prevent inadvertent filling of the fuselage tank during single-point refueling, the fuselage tank dual-level control valve closes when refueling pressure is applied to the system. The fuselage tank dual-level control valve opens to allow fuel to enter the fuselage tank only after the fuselage tank refueling control buttons are pressed. In single-point refueling of the fuselage tank, fuel is routed from the single-point refueling receptacle to the dual-level control valve within the fuselage tank. When fuel to the fuselage tank rises to full height, the valve float acts to shut off fuel flow. If single-point pressure refueling equipment is not available, the aircraft can be filled with conventional refueling equipment through the gravity-fill access to each tank. (See Figure 3-3.) The single-point refueling system is shown in Figure 3-4.

3.1.1.2 Single-Point Refueling Test and Fuselage Tank Refueling Control Buttons. The single-point refueling test and fuselage tank refueling control buttons (Figure 3-2) are flush-mounted in the undersurface of the right-wing leading edge, adjacent to the single-point refueling receptacle. Two buttons, PRI TEST and SEC TEST, govern primary and secondary refueling valve test procedures for all tanks.

Note

Single-point refueling can be accomplished without turning on any electrical power.

Satisfactory valve operation is indicated by the shutoff of fuel flow accompanied by the stopping of vibration and the stiffening of the refueling hose. A more positive indication of fuel shutoff can be obtained by observing the counter on the ground refueling equipment. A FILL AUX button controls flow to the fuselage tank. A SHUT OFF AUX button locks and holds the FILL AUX button during fuselage tank refueling. To lock the AUX FILL button, depress both the FILL AUX and SHUT OFF AUX buttons and then release the SHUTOFF AUX button; the FILL AUX button will remain depressed.

Note

With AUX FILL button depressed, the PRI/SEC test features become inoperative.

When refueling operations are completed, the AUX FILL button sometimes remains depressed. If this happens, there is no cause for concern.

CAUTION

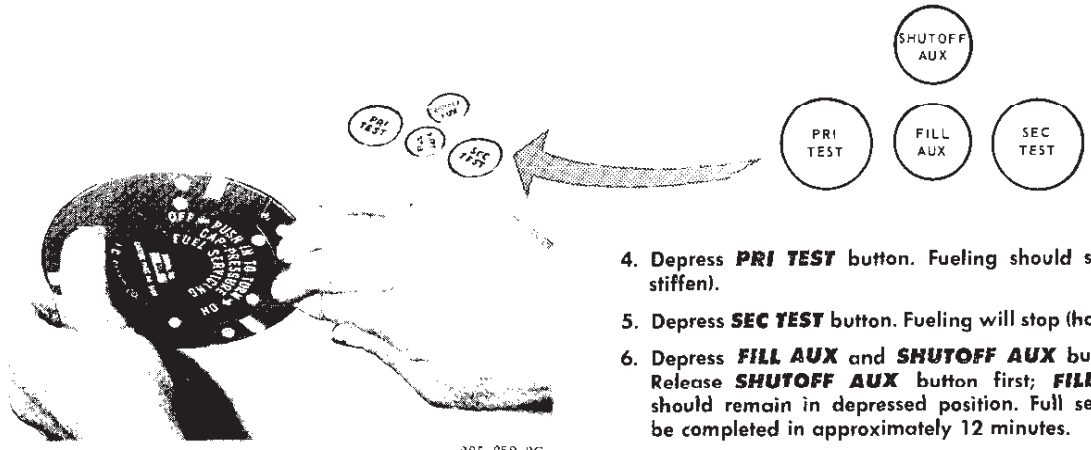
- If MIL-L-23699 engine oil is not available, drain and flush oil tank and fill with MIL-L-7808 oil.
- The batteries are nickel-cadmium batteries. Do not service these batteries using equipment that has previously been used to service lead-acid batteries.

	MILITARY SPECIFICATION	NATO SYMBOL	COMMERCIAL DESIGNATION	QUANTITY OR PRESSURE
FUEL	JP-4 MIL-T-5624 JP-5 MIL-T-5624 JP-8 MIL-T-83133 NONE	F-40 F-44 F-34 F-35	JET B+ NONE JET A1+ JET A1	1056 US GALLONS (USABLE FUEL)
ENGINE OIL	MIL-L-23699 MIL-L-7808	O-156 O-148	—	6 QUARTS (EACH TANK)
HYDRAULIC FLUID	MIL-H-83282A		—	2 US GALLONS

	MILITARY SPECIFICATION	QUANTITY OR PRESSURE
OXYGEN SYSTEM PORTABLE OXYGEN SYSTEM	MIL-O-27210	1,500 TO 1,800 PSI
FIRE EXTINGUISHING AGENT (ENGINE)	MIL-B-12218	REMOVE AND REPLACE CONTAINER.
LANDING GEAR STRUTS NITROGEN HYDRAULIC FLUID	BB-N-411 MIL-H-83282A	—
TIRES NITROGEN MAIN NOSE	BB-N-411	185 + 5, - 0 PSI 75 + 5, - 0 PSI
ACCUMULATOR NITROGEN — THRUST REVERSER	BB-N-411	900 PSI ± 50
ACCUMULATOR NITROGEN — HYDRAULIC SYSTEM	BB-N-411	1,500 PSI AT 70°F
BATTERY DISTILLED WATER	—	ELECTROLYTE LEVEL 1/16 TO 1/8 INCH ABOVE NYLON BARRIER (With battery fully charged)

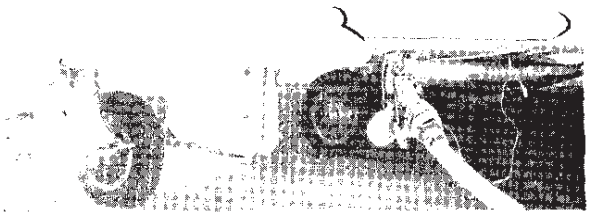
N3/89
39E-1-00-22B

Figure 3-1. Servicing Chart



1. Remove pressure refueling cap.

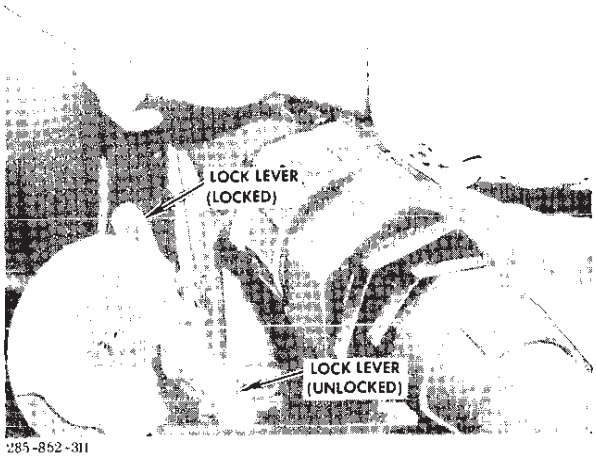
285-852-3G
285-852-4F



2. Insert nozzle grounding plug into jack on aircraft.

CAUTION

- Check to be sure that the single-point refueling receptacle is completely engaged prior to refueling. If it is not, damage may be done to the refueling receptacle.
- Before start of refueling, ensure that refueling hose is free from kinks.



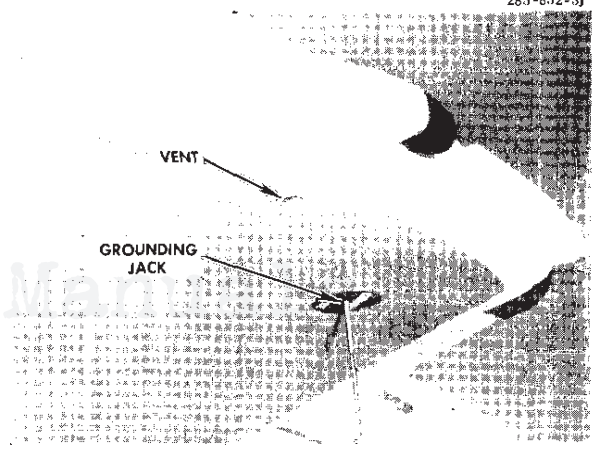
285-852-3H

3. Connect and lock refueling nozzle. Start ground servicing unit. Refueling pressure should be 50 psi.

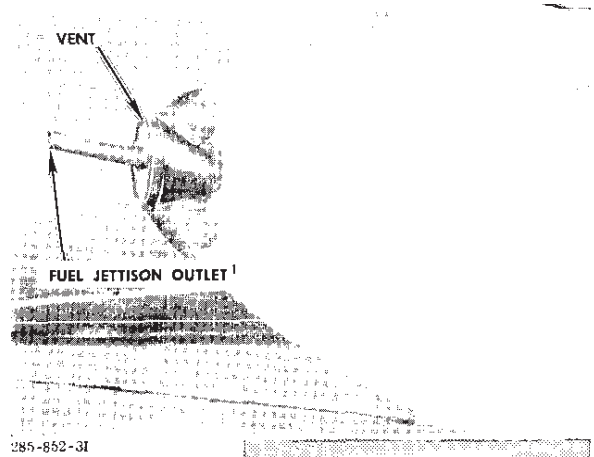
39D-1-48-6D

4. Depress ***PRI TEST*** button. Fueling should stop (hose will stiffen).
5. Depress ***SEC TEST*** button. Fueling will stop (hose will stiffen).
6. Depress ***FILL AUX*** and ***SHUTOFF AUX*** buttons together. Release ***SHUTOFF AUX*** button first; ***FILL AUX*** button should remain in depressed position. Full servicing should be completed in approximately 12 minutes.
7. Check wing fuel tank vents at both wing tips for obstructions and passage of air to ensure proper wing tank venting. If no passage of air is noted, secure the fueling operation and have discrepancy investigated.

285-852-3J



8. Fuel may come out wing tip vents. If this occurs, turn off refueler momentarily. Valves will reseat.



285-852-3I

[1] CT-39E, CT-39G Aircraft 158843, 158844 and 159361 through 159365

N5/76
39E-1-48-3A

Figure 3-2. Single-Point Refueling

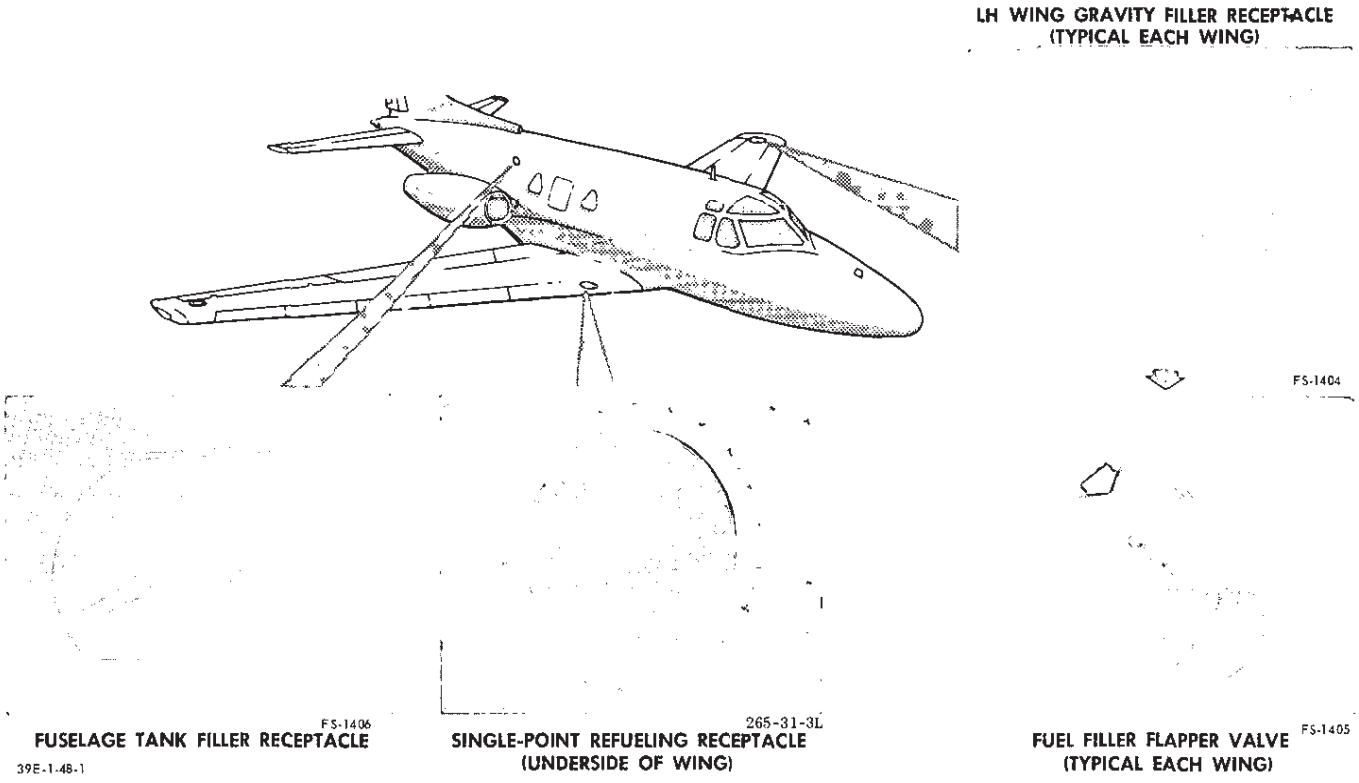


Figure 3-3. Fuel Servicing

3.1.1.3 Gravity Refueling. Gravity refueling of the aircraft is possible by filling through the fuselage tank filler while manually holding the spring-loaded fuel filler flapper on each wingtip open. (See Figure 3-3.) The flappers shall be released just before the fuel overflows from the opening. Then continue filling the fuselage tank until full. By using this method of refueling, it will take a minimum of 30 minutes to refuel the aircraft.

Note

- Observe all refueling precautions since some fuel may be spilled before the fuel filler flappers are released.
- Total usable fuel after gravity refueling will be about 8 U.S. gallons (52 pounds) less than after pressure refueling.

3.1.2 Engine Oil Systems. The engine oil tank has a capacity of 1.6 U.S. gallons, and the level shall be checked and serviced within 1 hour after engine shutdown.

WARNING

Prolonged contact with engine oil is very irritating to the skin and eyes. Wash hands with soap and water before touching food.

CAUTION

The engine oil used to service this aircraft is harmful to some substances. Be careful not to spill on electrical wiring, painted areas, rubber parts, and other units. Synthetic oil spilled on painted surfaces or on rubber components shall be wiped up immediately. Use a clean cloth saturated with aliphatic naphta.

Note

When an accurate reading of the oil tank level is needed following engine shutdown, the engine should be operated at approximately 75-percent rpm for not less than 15 nor more than 30 seconds immediately before shutdown in order to scavenge oil properly from inside the engine.

1. Open oil tank filler access door.
2. Remove oil tank filler cap by raising locking tab and rotating tab counterclockwise until filler cap is released; then remove cap. (See Figure 3-5.)

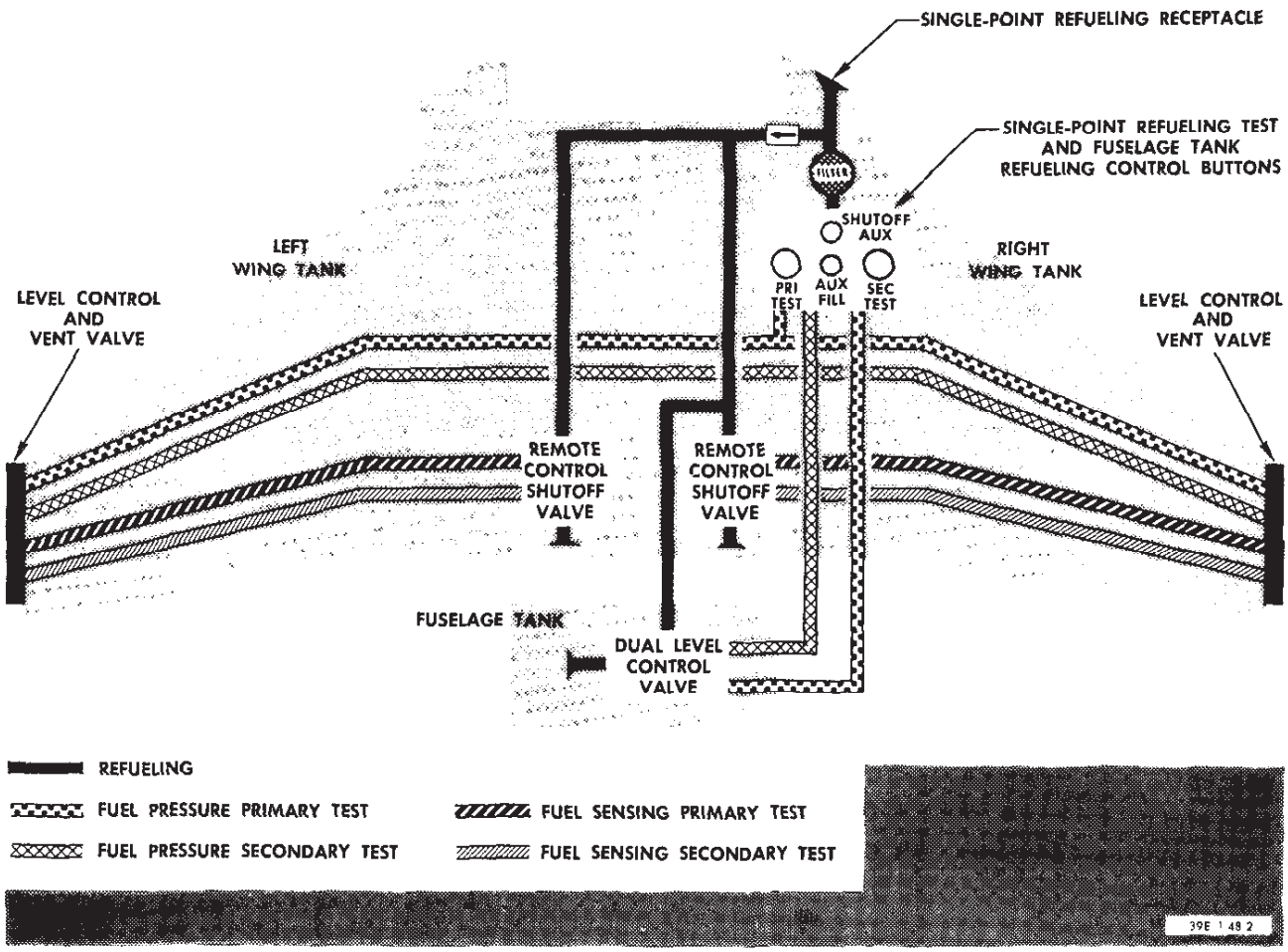


Figure 3-4. Single-Point Refueling System

CAUTION

- Do not mix oil types or brands. If mixing has occurred or if oil type is changed, drain and flush the system.
- Use care that wind or jet/prop blast from another aircraft does not blow dirt into the oil tank filler when the cap is removed.

3. Remove dipstick, wipe with a clean cloth, and reinstall in oil tank. Remove dipstick and check oil level. The oil level should be at the "FULL" mark; however, it is permissible to fly with a minimum of 4 quarts in the tank.

4. When oil is low, add lubricating oil to obtain proper oil level; do not overfill.

5. Insert dipstick and install oil tank filler cap by placing cap in position and rotating locking tab clockwise to lock cap; then press tab down into locked position.

Note

Before closing oil tank filler access door, check for proper installation and security of filler cap. If the cap is properly installed, it will be flush with the edges of the recess in the oil tank.

6. Close engine oil tank filler access door.

3.1.3 Hydraulic System. The hydraulic system is filled through the fluid reservoir. (See Figure 3-6.) The hydraulic power system, including all of the subsystem components, holds about 4.6 U.S. gallons. The reservoir contains 2.0 U.S. gallons when serviced to the "FULL" mark.

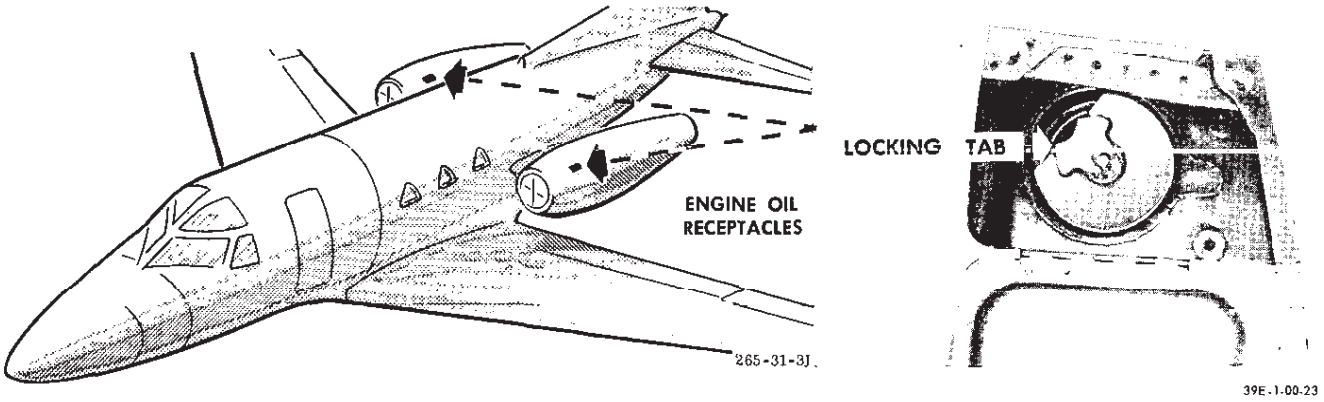


Figure 3-5. Engine Oil Servicing

3.1.4 Checking Reservoir Fluid Level. To check the reservoir fluid level, the following procedure should be used:

Note

The fluid in both accumulators shall be dumped into the reservoir in order to check proper fluid level.

1. With reservoir filler cap closed and locking tab down, hold accumulator dump valves partially open, one at a time, and slowly bleed off all hydraulic pressure.

CAUTION

Too rapid dumping of the accumulators *may exceed the reservoir pressure relief* valve rate and cause possible damage.

2. Check fluid level in reservoir sight gauge. It shall be maintained between the "FULL" and "REFILL" marks.

3. Add fluid if necessary.

Note

The accumulators shall be properly serviced before the system is filled. (Refer to paragraph 3.1.5, Servicing Accumulator.)

3.1.5 Servicing Accumulators. The normal, auxiliary, and thrust reverser hydraulic accumulators and servicing panel are in the aft equipment compartment. The accumulators are serviced through the hinged access doors on the underside of the aft fuselage section. (See Figures 3-7 and 3-8.)

1. With reservoir filler cap closed and locking tab down, hold accumulator dump valves partially open, one at a time, and slowly bleed off all hydraulic pressure.

CAUTION

Too rapid dumping of the accumulators may exceed the reservoir pressure relief valve rate and cause possible damage.

2. Check normal and auxiliary accumulator air pressure gauges for 1,500 ±50 psi. Check thrust reverser accumulator air pressure gauge for 900 ±50 psi.

3. If air servicing is required, attach nitrogen or dry-air supply source chuck to filler valve stem and loosen small hexnut on filler valve of affected accumulator not more than two turns.

4. Precharge normal and auxiliary accumulators with dry air or nitrogen to 1,500 ±50 psi. Precharge thrust reverser accumulator with dry air or nitrogen to 900 ±50 psi.

5. Tighten filler valve hexnut and open accumulator dump valve. Recheck precharge and service as required.

6. Check filler valve hexnut for tightness and disconnect air supply source.

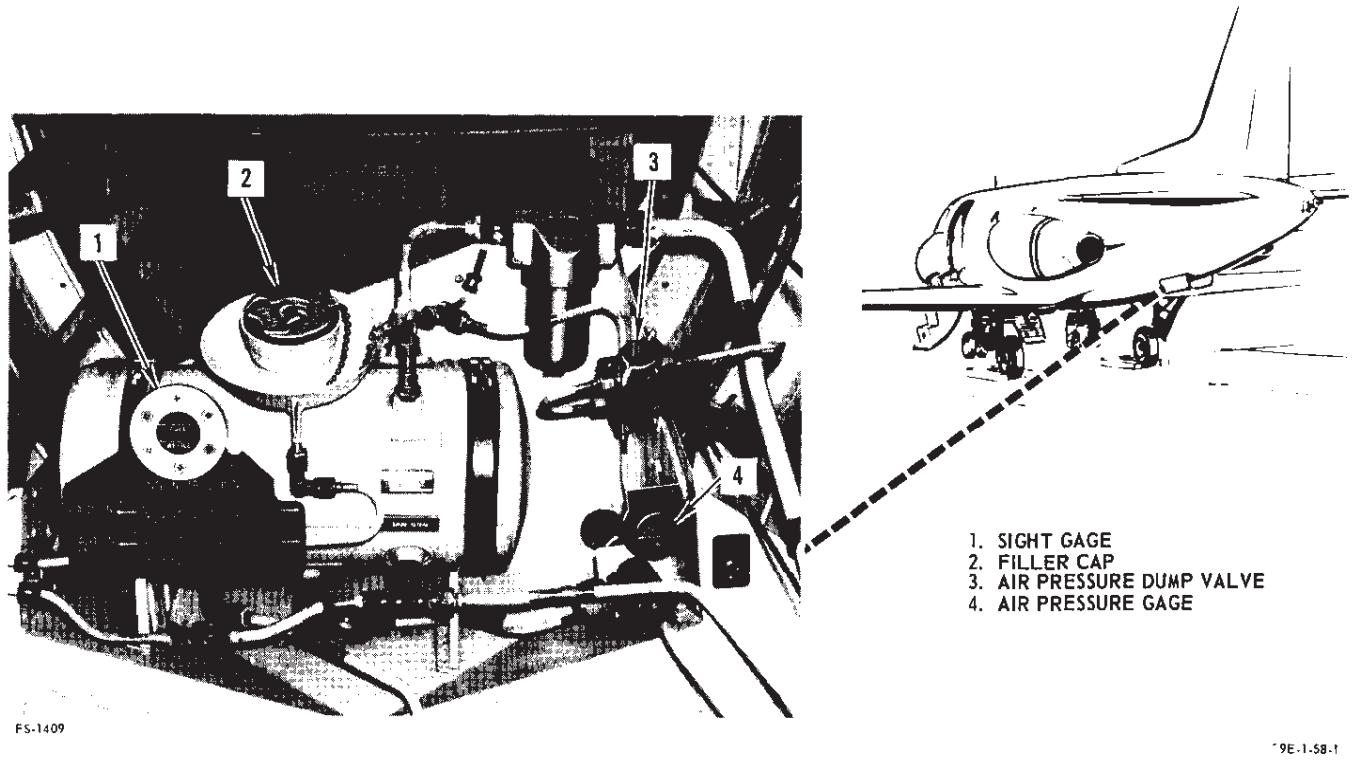


Figure 3-6. Hydraulic Reservoir

3.1.6 Servicing Hydraulic Reservoir. To service the hydraulic reservoir, the following procedures should be used:

1. Dump reservoir air pressure as follows:

   a. Discharge air pressure from storage tank by operating drain valve under tank.

   b. Discharge air pressure from reservoir by operating reservoir pressure dump valve until air discharge stops.

2. Remove reservoir filler cap and pour hydraulic fluid from a clean container into reservoir. Fill reservoir to "FULL" mark on reservoir sight gauge.

3. Replace filler cap on reservoir and position locking tab down.

4. Precharge of the hydraulic reservoir may be accomplished by either of the following procedures:

   a. Before operating hydraulic pump, connect a source of dry air or nitrogen (120 psi maximum) to "SEAL REG" fitting on air-conditioning ground test panel. Apply pressure until reservoir pressure gauge indicates at least 35 psi. Disconnect air source and cap fitting.

WARNING

When this procedure is being used, emergency brakes shall be used or the wheels shall be adequately chocked.

Note

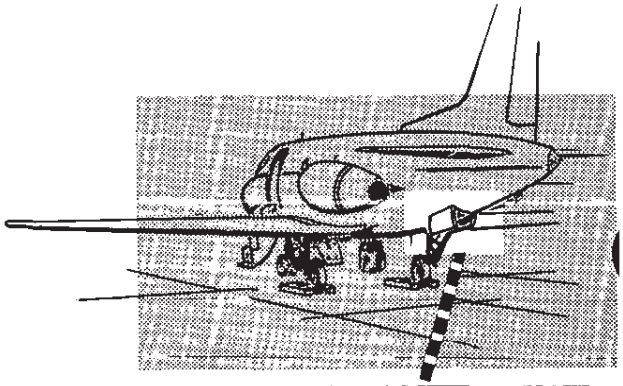
During precharge, air will be discharging from the jet pump in the forward fuselage section.

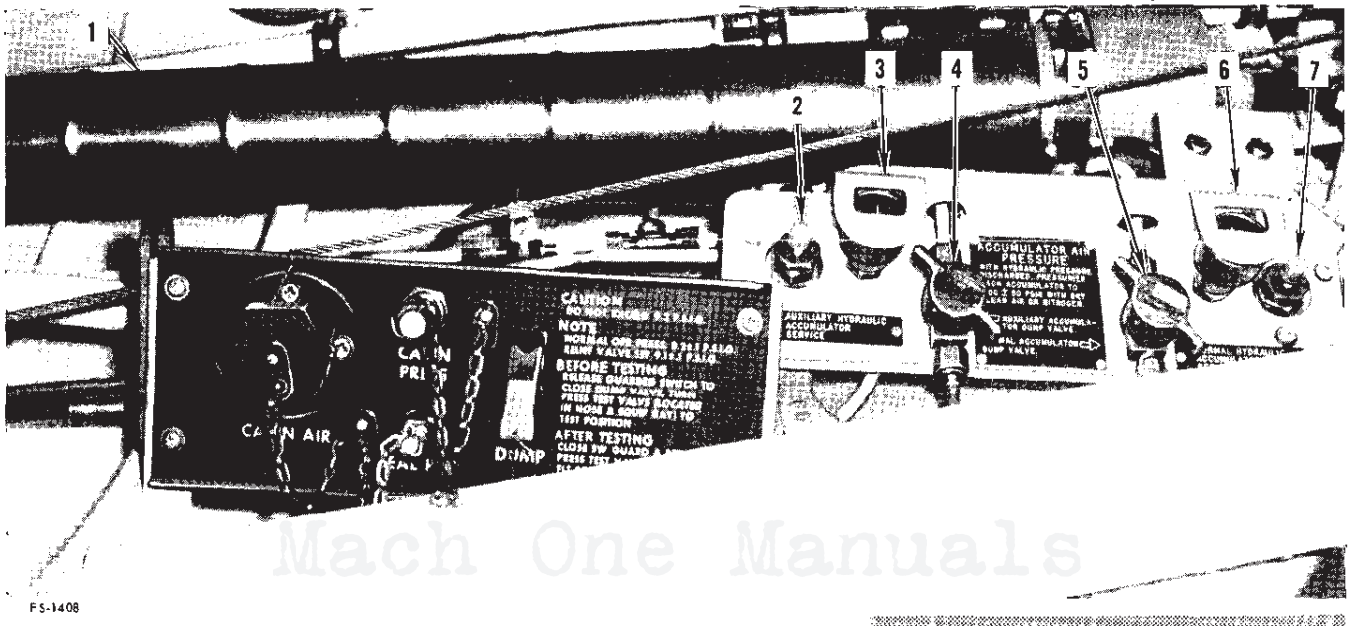
   b. Before operating hydraulic pump, precharge reservoir by operating one engine at 70-percent rpm for one-half minute to supply reservoir air pressure.

3.1.7 Batteries. Two 25.2-volt, 42-ampere-hour, nickel-cadmium batteries are in the aft equipment compartment and are accessible through aft equipment compartment doors. Quick-disconnect plugs connect the battery cables to the receptacles on the batteries.

NOTE

An improperly serviced accumulator may give the impression that it is leaking. To avoid replacing a serviceable accumulator, make sure all hydraulic pressure is bled off before servicing. If this is not done, a false reading will result. Also, if accuracy of servicing air gage is doubtful, it should be removed and calibrated or replaced.





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1. AUXILIARY ACCUMULATOR
2. AUXILIARY ACCUMULATOR AIR FILLER
3. AUXILIARY ACCUMULATOR AIR PRESSURE GAGE
4. AUXILIARY ACCUMULATOR PRESSURE DUMP VALVE
5. NORMAL ACCUMULATOR PRESSURE DUMP VALVE
6. NORMAL ACCUMULATOR AIR PRESSURE GAGE
7. NORMAL ACCUMULATOR AIR FILLER VALVE

This temperature chart should be used for proper servicing of either accumulator. The following is an example of how to use the chart: On a 21.1°C (70°F) day, the accumulator precharge would be 1500 psi. If the temperature at the accumulator were 48.9°C (120°F), the accumulator should be charged to 1640 psi.

TEMPERATURE	SPECIFIED ACCUMULATOR PRECHARGE (PSI)
48.9°C (120°F)	1640
43.3°C (110°F)	1610
37.8°C (100°F)	1585
32.2°C (90°F)	1555
26.7°C (80°F)	1530
21.1°C (70°F)	1500
15.6°C (60°F)	1470
10.0°C (50°F)	1445
4.4°C (40°F)	1415
−1.1°C (30°F)	1380
−6.7°C (20°F)	1360
−12.2°C (10°F)	1330
−17.8°C (0°F)	1300
−23.3°C (−10°F)	1270
−28.9°C (−20°F)	1245
−34.4°C (−30°F)	1215

39E-1-58.2

Figure 3-7. Accumulator Servicing

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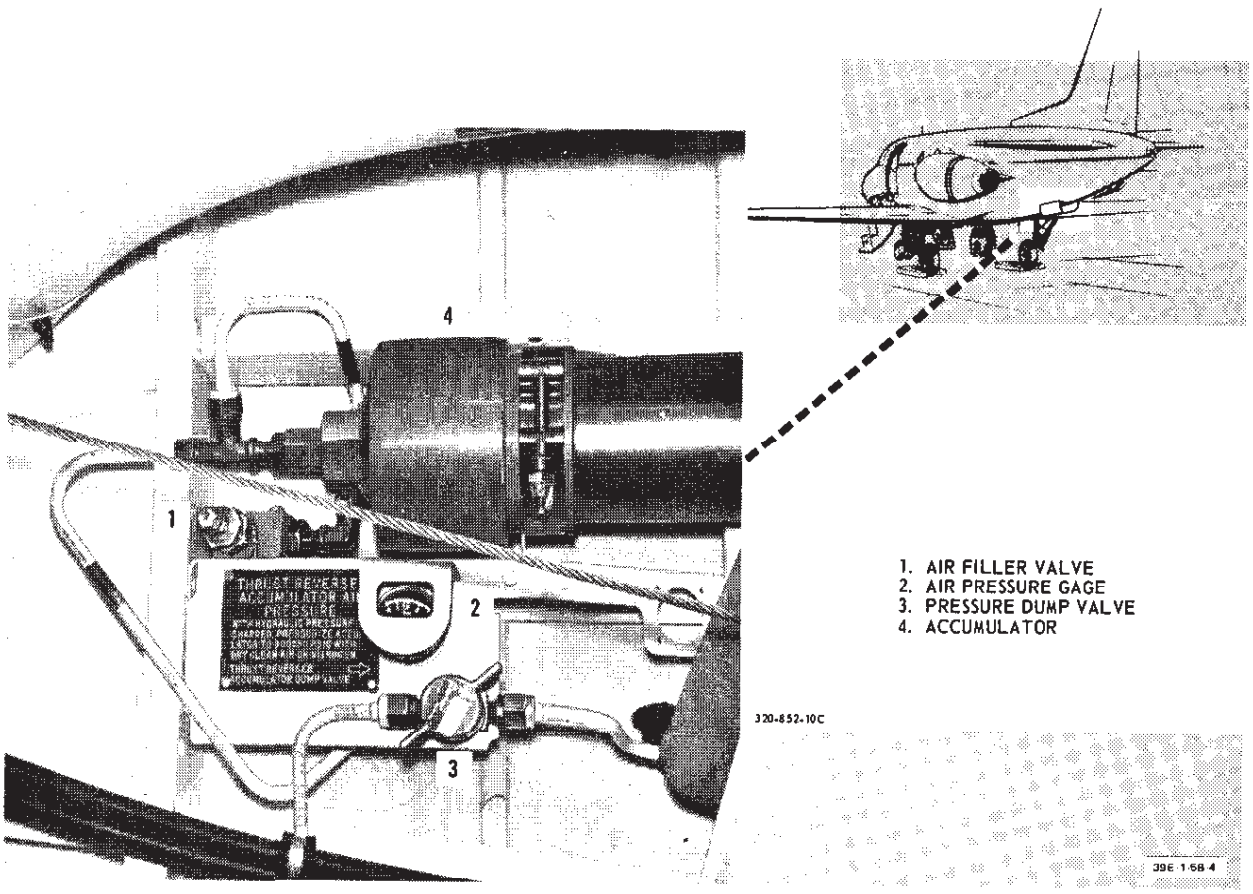


Figure 3-8. Thrust Reverser Hydraulic Accumlator

CAUTION

Never use a hydrometer (or other servicing tool) that has been used to service lead-acid batteries. Such practice will contaminate the nickel-cadmium battery electrolyte and render the battery useless in a very short time.

3.1.8 Pitot-Static System. On CT-39E aircraft and CT-39G aircraft 158843 and 158844, there are nine pitot-static drains installed at low points in the system. Two pitot-static drains are on each side of the forward fuselage just below the pitot heads; two drains are in the nose wheelwell, one on each side, and another drain is above the pilot left rudder pedal. Two EPR system drains are in the aft equipment compartment above the hydraulic reservoir. If the aircraft has been parked during inclement weather, the two static drains in the nose wheelwell and the four pitot and static drains on the forward fuselage should be drained before flight. If it is suspected that there is water in other parts of the system, the remaining three drains should also be serviced.

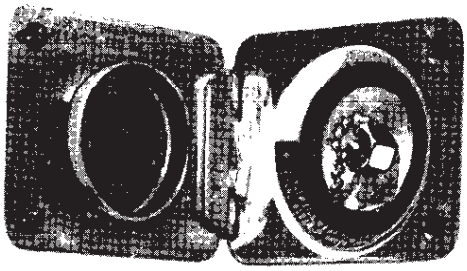
On CT-39G aircraft 159361 through 159365 and 160053 through 160057, pitot-static drains are located as follows: two static drains on each side of the fuselage near the pitot-static tube, a static drain in the stall warning computer line on the right side of the cabin, pitot drain holes in each pitot-static tube, and two EPR system drains in the aft equipment compartment above the hydraulic reservoir.

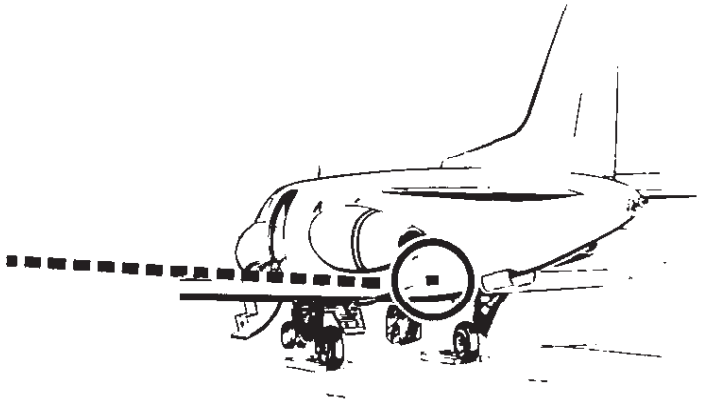
3.1.9 Oxygen System. The high-pressure (1,800 psi) oxygen storage cylinder is filled through a high-pressure filler valve on the lower left side of the fuselage at the wing trailing edge. (See Figure 3-9.) When the pressure in system has been depleted to 1,500 psi, it shall be recharged.

Note

Ac electrical power must be on for cockpit indication.

3.1.10 Oxygen Walkaround Bottle. The oxygen walkaround bottle may be removed from the aircraft to be recharged. When the pressure in the bottle has been





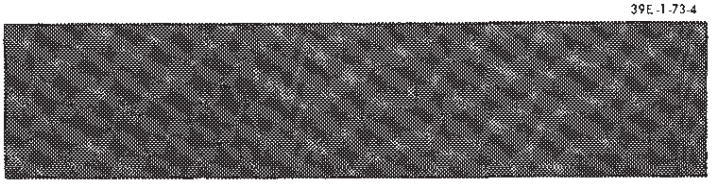


Figure 3-9. Oxygen Filler Valve

used or found to have less than 1,500 psi, it shall be recharged.

3.1.11 Landing Gear Struts. The nosegear and main gear struts are inflated through a high-pressure valve at the top of each strut. The air pressure required to inflate the struts varies with aircraft gross weight.

3.1.12 Towing Aircraft. When towing aircraft, the following procedure should be used:

1. Check nosegear strut extension. Do not tow aircraft when strut is completely deflated or extended beyond maximum limits.

2. Station man in cockpit to operate brakes if necessary.

Note

The entry door should be closed during towing to prevent any unnecessary bouncing of the door against its supports.

3. Ensure that gust lock T-handle in cockpit is engaged.

4. Disconnect electrical quick-disconnect plug.

5. Turn antiskid switch off.

6. Connect tow bar assembly to nosegear, using tow pin provided.

CAUTION

- Ensure that nosegear steering lock lever is disengaged by use of proper tow bar before towing.

- Use of a universal tow bar may not disengage the nosegear steering lock lever. Subsequent towing may cause damage to the nosegear steering collar.

7. After aircraft has been towed and spotted, remove tow bar and check that nosegear steering lock lever engages.

Note

The lock lever will not engage unless the nosewheels are properly centered.

8. Check that electrical quick-disconnect plug at right side of nose strut is connected and undamaged.

AMBIENT TEMP (°F)	CONTAINER GAGE PRESS. (PSIG)	
	MINIMUM	MAXIMUM
-65	90	175
-60	95	180
-50	103	188
-40	113	198
-30	122	207
-20	133	218
-10	145	230
0	158	243
10	173	258
20	190	275
30	209	294
40	230	315
50	258	343
60	285	370
70	320	405
80	357	442
90	395	480
100	437	522
110	484	569
120	530	615
130	580	665
140	630	715
150	680	765
160	730	815

Figure 3-10. Fire-Extinguishing Agent Container Pressure Chart

3.1.13 Overboard Drains. The overboard drains are fuel, oil, water, and battery fumes (Figure 3-11). The wing tanks fuel sump drains must be drained after each flight and before refueling to clear tanks of any water condensed in flight. A water drain valve is flush-mounted in the fuselage skin below the left side of the cockpit. A second drain valve is installed in the panel just aft of the speedbrake. These valves can be used before flight to determine whether water has accumulated in the under cockpit area or below the intermediate cabin area aft of the speedbrake. The presence of trapped water can be checked by pressing in on the spring-loaded center of the valve. The valve is slotted for a screwdriver and is opened by a quarter turn of the slot. After the water has completely drained, the valve shall be closed. (When the valve is closed, the slot in the center of the valve is aligned with the slot in the outer ring of the valve assembly.)

CAUTION

If the aircraft has been recently washed, exposed, to or parked outside during inclement weather, water may accumulate in the lower cabin area. This water must be drained prior to flight because of the formation of ice and possibly frozen controls resulting from extremely cold temperatures experienced at high altitude.

The static drains are drained periodically and after a wet weather flight or aircraft wash.

Note

If the aircraft has been parked outside during inclement weather, the valve shall be used before flight to make sure the areas below the cockpit and cabin floor are free of water.

3.1.14 Ice Removal. Ice formation on parked aircraft may be removed by spraying affected areas with isopropyl alcohol (specification TT-I-735).

3.2 STARTING REQUIREMENTS

The rectangular dc power receptacle is within an access door on the bottom left side of the fuselage just forward of the aft fuselage lower access door. (See Figure 3-12.) Before the external power source is turned on, the cockpit should be checked to ensure that all switches are at OFF and the throttles are closed.

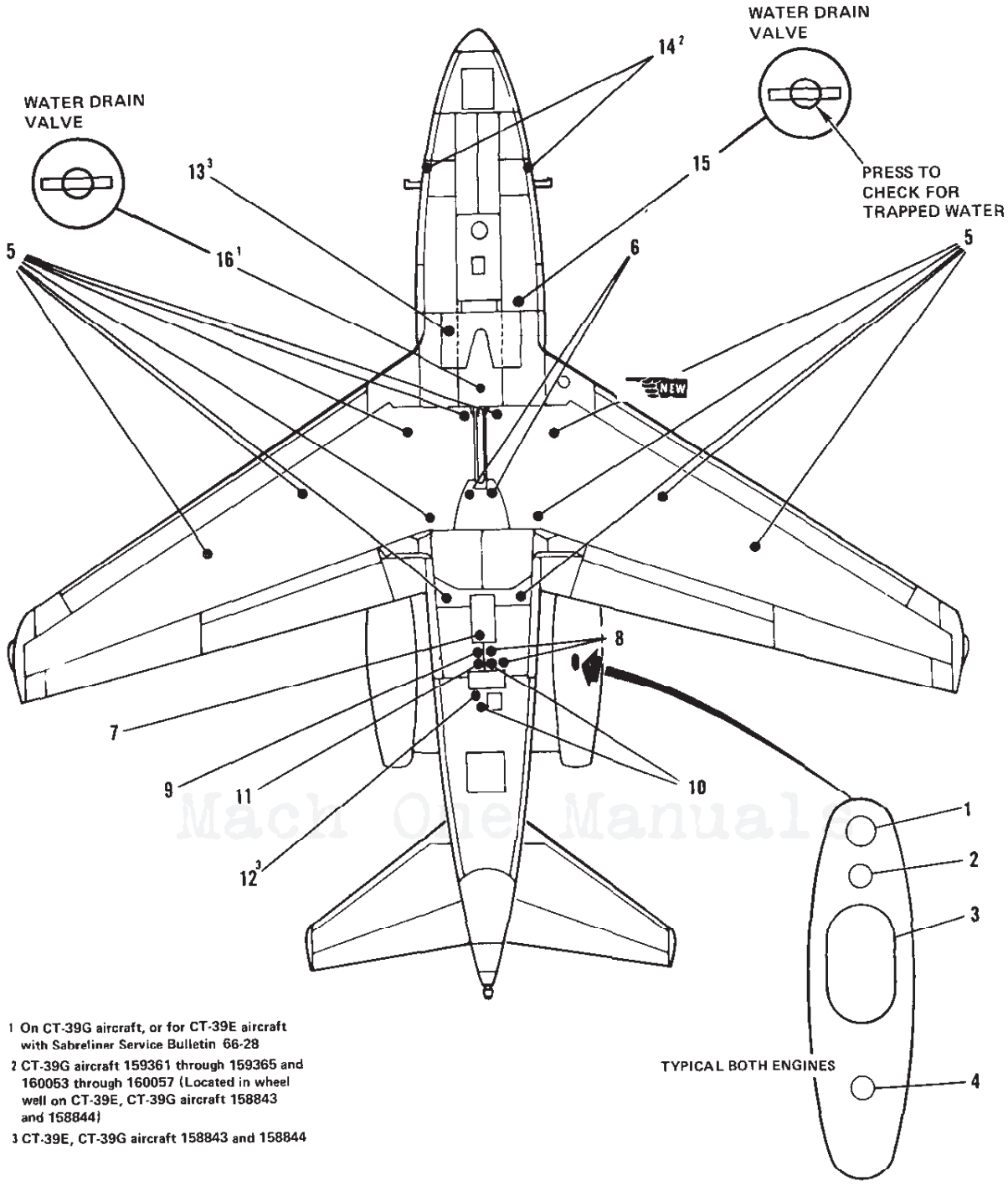
CAUTION

- The external power source shall be turned off whenever the dc power supply plug is connected to or disconnected from its receptacle.
- When the external power source is connected to the JET START receptacle, ensure that the plug is fully seated and that the engine start external power monitor microswitch has been actuated. (See Figure 3-12).

For engine start using external power, use a 28-vdc (800-ampere, constant-current minimum) supply.

3.3 DANGER AREAS

See Figure 3-13.



1 On CT-39G aircraft, or for CT-39E aircraft with Sabreliner Service Bulletin 66-28

2 CT-39G aircraft 159361 through 159365 and 160053 through 160057 (Located in wheel well on CT-39E, CT-39G aircraft 158843 and 158844)

3 CT-39E, CT-39G aircraft 158843 and 158844

1 OIL TANK SCUPPER DRAIN
2 FUEL PRESSURE AND DUMP VALVE DRAIN, FUEL PUMP DRAIN
3 ACCESSORY DRIVE GEAR CASE OIL BREATHER
4 COMBUSTION CHAMBER FUEL DRAIN
5 FUEL SUMP DRAIN
6 WATER DRAIN
7 FUEL SUMP DRAIN
8 FUEL DRAIN
9 BATTERY DRAIN
10 FUEL VENT
11 FUEL DRAIN
12 AIR CONDITIONING WATER DRAIN[3]
13 STATIC DRAIN[3]
14 STATIC DRAIN
15 COCKPIT AREA WATER DRAIN
16 CABIN FLOOR WATER DRAIN

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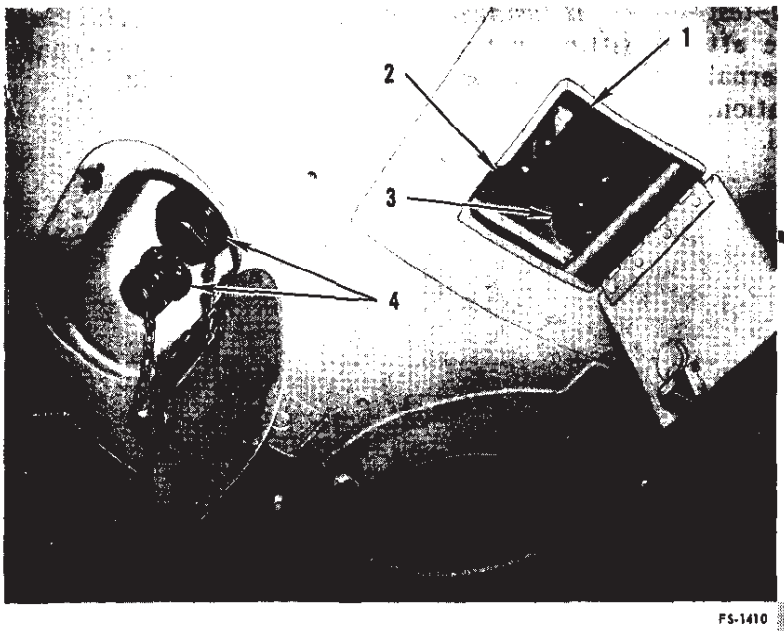
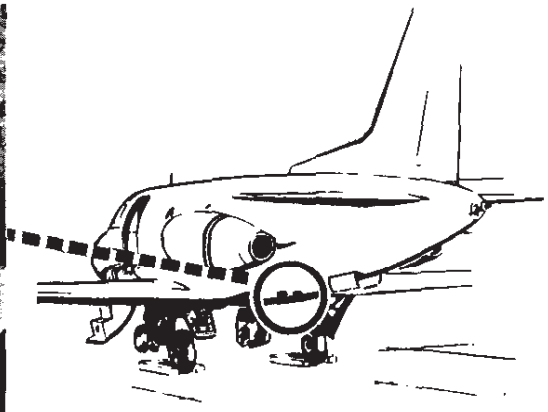
Figure 3-11. Overboard Drains





1. POWER MONITOR MICROSWITCH
2. JET START RECEPTACLE
3. UTILITY RECEPTACLE
4. HYDRAULIC GROUND TEST CONNECTIONS

FS-1410

CAUTION

To prevent damage to aircraft structure, do not flex any of the electrical leads when engaging or disengaging power cart.

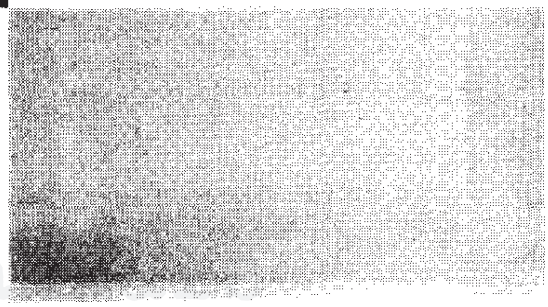


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Figure 3-12. External Electrical and Hydraulic Power Receptacles

3.4 EXTERNAL POWER REQUIREMENTS

The oval dc power receptacle is within an access door on the bottom left side of the fuselage just forward of the aft fuselage lower access door. (See Figure 3-12.)

CAUTION

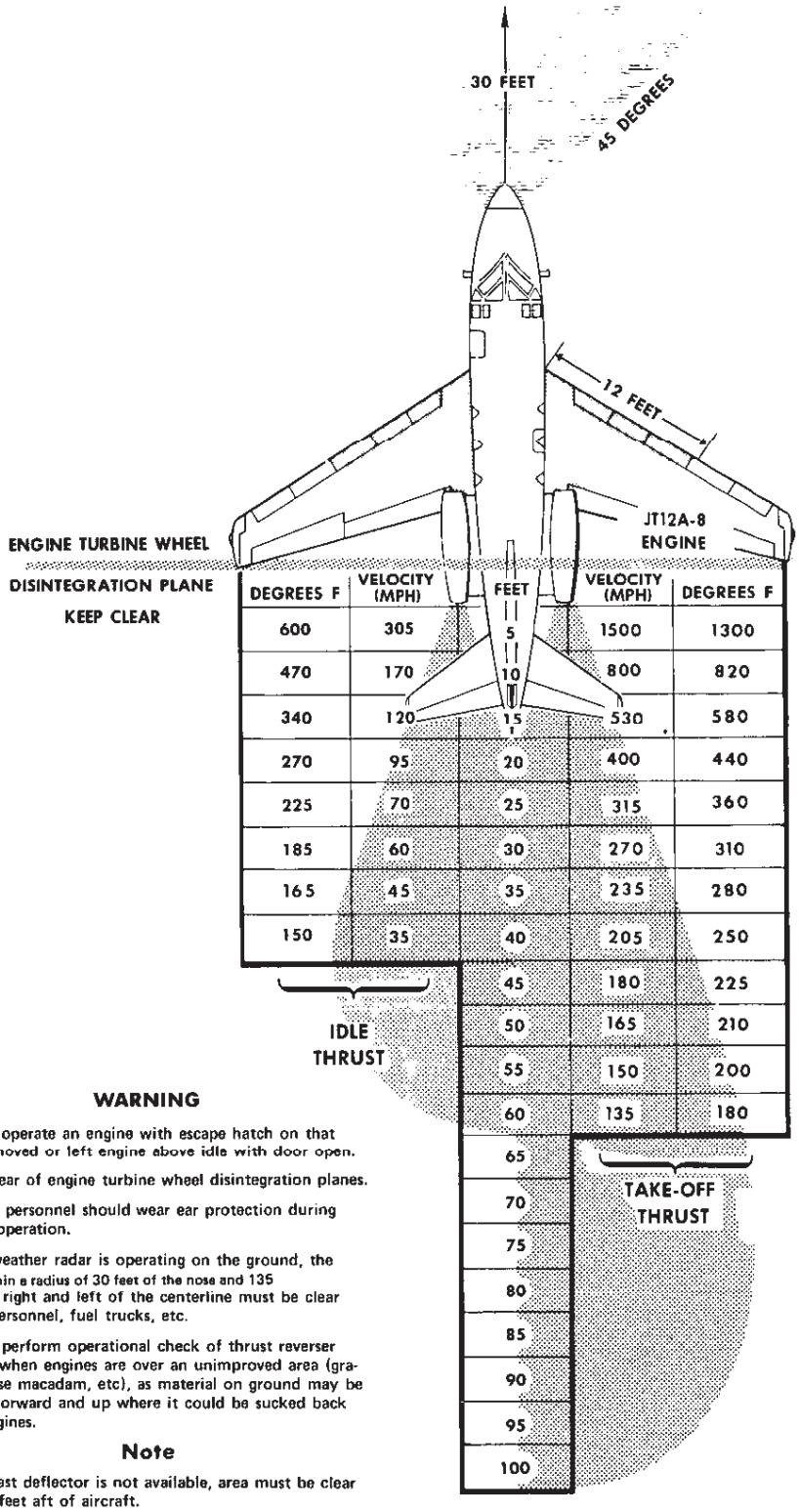
The external power source shall be turned off whenever the dc power supply plug is to be connected or disconnected from its receptacle.

Before the dc power source is turned on, the cockpit should be checked to be sure all switches are at OFF and the throttles are closed.

Connecting an external 28-vdc (300-ampere minimum current) power supply to the UTILITY power receptacle powers all dc buses except the starter and battery buses for the system. Positioning the electrical master and battery switches at ON permits external power to be directed to the battery bus for system operation and battery charging.

Two hydraulic system ground-test connections are on the lower side of the aft fuselage. (See Figure 3-12.) For external hydraulic power supply operation information, refer to the applicable maintenance manual.

If installed, a cabin ground air-conditioning fitting in the aft equipment bay is for connection of an air-conditioning ground unit.



DEGREES F	VELOCITY (MPH)	FEET	VELOCITY (MPH)	DEGREES F
600	305	5	1500	1300
470	170	10	800	820
340	120	15	530	580
270	95	20	400	440
225	70	25	315	360
185	60	30	270	310
165	45	35	235	280
150	35	40	205	250
		45	180	225
		50	165	210
		55	150	200
		60	135	180
		65		
		70		
		75		
		80		
		85		
		90		
		95		
		100		

Figure 3-13. Danger Areas

CHAPTER 4

Operating Limitations

4.1 MINIMUM FLIGHTCREW

The minimum flightcrew required for performing all functions necessary to comply with the airworthiness standards is two: one pilot and one copilot.

4.2 INSTRUMENT MARKINGS

Careful attention shall be given to the instrument markings (Figure 4-1) because the limitations shown on these instruments and noted in the captions are not necessarily repeated in the text of this or any other part.

4.3 ENGINE LIMITATIONS

All normal engine limitations are shown in Figure 4-1 and are based on grade JP-4 or JP-5 fuel. The engines should be trimmed for the grade fuel that is normally being used. However, if takeoff thrust can be obtained using the precomputed EPR, the grade (JP-4/JP-5) or density of fuel is not limiting and retrimming is not necessary.

Note

Engine trimming is to be accomplished by qualified maintenance personnel only.

The red-line limits, shown for revolutions per minute and EGT, are established by tests to ensure the mechanical integrity of the engine. As there is a considerable margin between these values and those values obtained by a normal engine, readings at or near these limits usually indicate a malfunction. Therefore, when operating at or slightly below the revolutions per minute or EGT limits, it is recommended in the interest of improved engine service life that thrust be reduced to obtain more nearly normal readings if flight conditions will permit. This condition should be reported to maintenance personnel for their possible action.

4.3.1 Thrust Definitions and Time Limits

4.3.1.1 Takeoff Thrust. Takeoff thrust is defined as the maximum thrust that may be used and is time-limited to 5 minutes continuous operation. Takeoff thrust is selected by adjusting the throttle to obtain a specific EPR that will vary with altitude, air temperature, and Mach number. The charts for determining static takeoff thrust and in-flight takeoff thrust are contained in Part XI.

4.3.1.2 Maximum Continuous Thrust. Maximum continuous thrust is defined as the maximum thrust that may be used continuously. Maximum continuous thrust is selected by adjusting the throttle to obtain a specific engine pressure that will vary with altitude, air temperature, and Mach number. The chart for determining maximum continuous thrust is contained in Part XI. Maximum continuous thrust EPR is for use in climb and acceleration and should not be used for normal operations.

4.3.2 Engine Overspeed. The maximum allowable engine speed is 104.2-percent rpm. If this speed is exceeded by either engine while the aircraft is on the ground, the engines shall be shut down. If this speed is exceeded in flight, use minimum thrust (on the engine that encountered the overspeed) to sustain flight and land as soon as possible. In either case, the engine that encountered the overspeed shall be inspected for damage.

Note

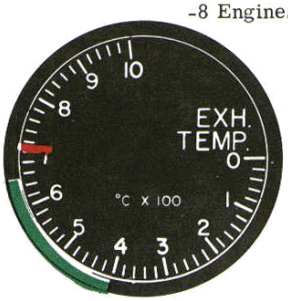
The amount and duration of any engine overspeed shall be entered in the aircraft flight record (yellow sheet) so that the prescribed engine inspection can be performed.

4.3.3 Exhaust Temperature Limits. Exhaust temperature limits are shown in Figure 4-1.

BASED ON JP-4 AND JP-5 FUEL

EXHAUST TEMPERATURE GAGE

-8 Engines



MARKING	RANGE	CONDITION
	400° C to 655° C	
	718° C	Maximum at Takeoff Thrust (5 minutes continuous operation). During engine acceleration (2 minutes continuous operation)
None	525°C	Maximum allowable during start up to idle.
None	515°C	Maximum allowable at idle rpm.

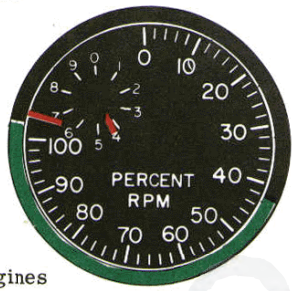
REMARKS

During Takeoff Thrust operation climbs, retard throttle if necessary to maintain exhaust temperatures at or below the red line. Exhaust temperature above the following values requires shutdown and correction of cause of overtemperature.

760° C (or 730° C for more than 5 seconds)

During the starting cycle, exhaust temperature above 525°C requires shutdown to determine and correct cause of overtemperature. Above 630°C (or 595°C for more than 5 seconds) requires more extensive maintenance inspection.

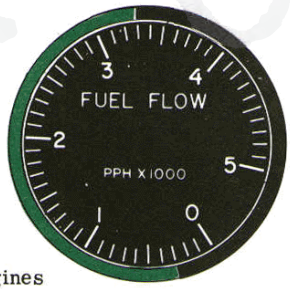
TACHOMETER



-8 Engines

MARKING	RANGE	CONDITION
None	42 ± 1	Idle
	42% to 103%	Normal operating
	104.2%	Maximum

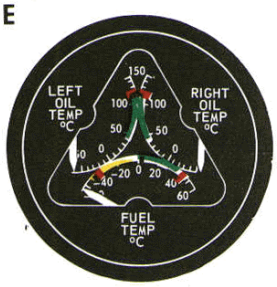
FUEL FLOW INDICATOR



-8 Engines

MARKING	RANGE	CONDITION
	300 to 3300 lb/hr	Continuous
None	260 lb/hr	Minimum during start
None	600 lb/hr	Maximum during start

FUEL OIL TEMPERATURE GAGE



MARKING	RANGE	CONDITION
OIL		
	121° C	Maximum
	40° to 120° C	Normal operating
FUEL		
	-40° C	Minimum
	+43° C	Maximum
	-40° C to 0° C	Caution Use fuel heaters
	0° C to +43° C	Normal

OIL PRESSURE GAGE

35 psi Minimum at idle rpm

40 to 50 psi Normal

N12/78
39E-1-51-2A

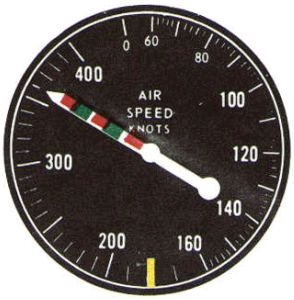
Figure 4-1. Instrument Markings (Sheet 1 of 2)

MACH/AIRSPEED INDICATOR

MACH INDICATOR

.808 indicated Mach maximum allowable above 21,500 feet pressure altitude (CT-39G aircraft 159361 through 159365 and 160053 through 160057).

AIRSPEED INDICATOR



180 KIAS — Maximum for gear down, flaps fully extended and for windshield wiper operation.

Note

- On CT-39G and CT-39E aircraft with Service Bulletin 68-5 incorporated, the maximum allowable airspeed for flaps extended at approach (APPR) or part (PART) is 225 KIAS.
- Landing-taxi lights may be extended at speeds up to 250 KIAS.

Maximum allowable speed pointer varies automatically with altitude. (Refer to "Airspeed Limitations" in this chapter.)

HYDRAULIC PRESSURE GAUGES

NORMAL SYSTEM

1,500 to 2,650 psi Caution

Shows malfunction if pressure remains in this range. Intermittent fluctuation into this range during transient conditions is permissible.

2,650 to 3,200 psi Normal

3,200 psi Maximum

Note

Momentary overshoot permissible during transient conditions such as large flow demands by the various hydraulically operated systems.

AUXILIARY SYSTEM

1,500 psi Minimum

1,500 to 2,800 psi Caution

Operation of hydraulic systems should be limited to minimum requirements to conserve accumulator capacity.

2,800 to 3,600 psi Normal

Indicates system accumulator is fully charged and system can be utilized as required.

3,600 psi Maximum

N3/89
39E-1-51-3B

1. Oil pressure fluctuations of 5 psi within the normal operating range are permissible. During ground operation, if fluctuations occur and exceed 5 psi, the cause should be investigated before flight. During flight, if fluctuations occur and exceed 5 psi, reduce thrust on the affected engine and land as soon as practicable.

2. If oil pressure is stable within the range of 35 to 40 psi or 50 to 60 psi during ground operation above engine idle revolutions per minute, the condition should be remedied before flight. If oil pressure is stable within either of these ranges during flight, reduce thrust on the affected engine.

4.4 THRUST REVERSER LIMITATIONS

1. The thrust reversers are limited to ground operation only.

2. The thrust reverser control switch shall not be in the ON position except:

   a. During reverser ground operational check

   b. After throttle reduction during rejected takeoff

   c. After touchdown for reverse thrust operation.

3. The thrust reverser stow procedure must be initiated prior to decelerating to 60 KIAS.

4. Reverse thrust shall not be used to back up the aircraft. It should not normally be used to control taxi speed. When operating on icy runways, thrust reversers may be armed and used if conditions warrant.

CAUTION

If thrust reversers are used in snow, slush, or icy conditions, wheelwells and flaps shall be checked prior to takeoff.

5. A visual check of the secondary latch S-hook for proper engagement with the door striker must be performed prior to each flight.

6. During takeoff and landing, the thrust reverser control switch must be in the EMERG STOW position except:

   a. Following takeoff after attaining 1,500 feet AGL or until obstacle clearance (whichever is higher), place thrust reverser control switch in OFF position.

   b. Prior to landing, place thrust reverser control switch in EMERG STOW position before placing landing gear handle at DOWN.

7. A preflight (prior to first flight of the day) check of the thrust reverser system must be made in accordance with the instructions contained in the Sabreliner Maintenance Manual.

8. Thrust reversers are not to be used during touch-and-go landings.

9. A malfunctioning reverser should be repaired before flight or flyaway locks shall be installed on the malfunctioning reverser.

WARNING

Use of single-engine reversing under wet or slippery runway conditions requires judicious application of reverse throttle. Otherwise, loss of directional control may occur.

4.5 AIRSPEED LIMITATIONS

4.5.1 Maximum Allowable Airspeed. On CT-39E, CT-39G aircraft 158843, and 158844 the maximum allowable airspeed varies from 356 KIAS at sea level to 370 KIAS at 21,500 feet pressure altitude and .808 indicated Mach number above 21,500 feet pressure altitude. On CT-39G aircraft 159361 through 159365 and 160053 through 160057, the maximum allowable airspeed varies from 353 KIAS at sea level to 367 KIAS at 21,500 feet pressure altitude and .808 indicated Mach number above 21,500 feet pressure altitude. The maximum allowable airspeed pointer varies automatically with altitude to follow these schedules.

4.5.2 Landing Gear Lowering Speed. The maximum allowable airspeed with the landing gear extended is 180 KIAS. If the landing gear is extended above this speed, structural damage can occur.

4.5.3 Landing and Taxi Light Extension Speed. The maximum allowable airspeed with the landing and taxi lights extended is 250 KIAS.

4.5.4 Wing Flap Extension Speed (CT-39G and CT-39E With Service Bulletin 68-5 Incorporated). The maximum allowable airspeed with the wing flaps at APPR or PART is 225 KIAS. The maximum allowable airspeed with the wing flaps fully extended is 180 KIAS. On all other CT-39E, the maximum allowable airspeed with wing flaps extended at any position is 180 KIAS.

4.5.5 Windshield Wiper Operating Speed. The maximum allowable airspeed for windshield wiper operation is 180 KIAS.

4.6 OXYGEN SYSTEM LIMITATIONS

At altitudes above 41,000 feet, all cabin occupants shall wear oxygen masks.

4.7 ESSENTIAL BUS CIRCUIT BREAKERS

All 28-vdc essential bus circuit breakers (cockpit and cabin) must be checked and engaged (pushed in) during the preflight (prior to first flight of the day) inspection.

4.8 STANDING WATER AND/OR SLUSH

Chine tires shall be installed on the nosewheel to permit takeoff in standing water and/or slush.

4.9 AUTOPILOT LIMITATIONS

1. The maximum autopilot operating speed is M_{MO}/V_{MO}.
2. The autopilot shall be disengaged for takeoff and landing.
3. When using the autopilot, the pilot or copilot shall be in his seat with his seatbelt fastened.

4.10 HIGH-ALTITUDE FLIGHT LIMITATIONS

The maximum operating altitude is 45,000 feet. The maximum unpressurized operating altitude is 25,000 feet.

If the stall warning and/or continuous ignition feature fails, engine flameout may occur above 25,000 feet altitude under certain conditions of high angle of attack and low airspeed. The critical high angle of attack can be encountered during unaccelerated aircraft (1g) stall and accelerated aircraft stalls. The flameout is caused by turbulent airflow from the wing entering the engine inlet ducts. Above 35,000 feet altitude, engine flameout may be encountered at indicated airspeeds below 180 KIAS. In addition, above 35,000 feet, rapid throttle movements at any airspeed may cause engine flameout.

To prevent these flameout phenomena, the following restrictions shall be observed:

1. Above 25,000 feet altitude, do not perform stalls.
2. Above 35,000 feet altitude, maintain at least 180 KIAS; do not make rapid throttle movements.

4.11 MANEUVERING LIMITATIONS

4.11.1 Prohibited Maneuvers. The aircraft is restricted from performing the following maneuvers:

1. Spins. (Refer to paragraph 10.5, SPINS AND RECOVERY.)
2. Acrobatics.
3. Rolls through a bank angle change of more than 180°.
4. Lateral wheel deflections in excess of one-half throw at airspeeds above 225 KIAS.
5. Do not purposely operate in stall warning buffet above 25,000 feet.
6. Yawing maneuvers in other than 1.0g flight.

4.12 ACCELERATION LIMITATIONS

In order to ensure correct interpretation of the acceleration limitations imposed on this aircraft, the following definitions apply:

1. Symmetrical applies where the aircraft bank angle is constant (no roll). Therefore, the aircraft may be in other than a wing-level attitude but shall not be rolling during the period of accelerated flight.
2. Unsymmetrical applies where the aircraft is rolling during the period of accelerated flight.

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Acceleration limitations are imposed on this aircraft for either structural or aerodynamic reasons. The structural limitations apply at all altitudes, but may be less restrictive than the aerodynamic limits. The aerodynamic limits are imposed to prevent encountering flight conditions where a sudden up-float of the ailerons will occur, resulting in an abrupt change in lateral controllability. The composite structural aerodynamic symmetrical limits are presented on a scribed plate along the upper edge of the accelerometer and are as follows:

4.12.1 Positive Limits

1. Sea level to 30,000 feet — 3.0g.
2. 30,000 to 35,000 feet — 2.5g.
3. 35,000 to 40,000 feet — 2.0g.
4. 40,000 to 45,000 feet — 1.5g.

4.12.2 Negative Limits

1. All altitudes — -1.0g.
2. Max g-limit with landing gear and/or flaps extended — +2.0g.

4.12.3 Zero- and Negative-G Flight Time Limit.

The aircraft is limited to 10 seconds of continuous flight at zero and negative g because of fuel, oil, and hydraulic system limitations. If this time limit is exceeded, engine flameout may occur, the engines may be damaged because of oil starvation, or hydraulic pump failure may occur.

4.13 CENTER-OF-GRAVITY LIMITATIONS

The following paragraphs outline specific limitations that shall be observed in order to maintain the aircraft center of gravity within limits.

1. The pilot and copilot seats shall be occupied and all passengers shall be seated during takeoff and until each fuel quantity gauge indicates 3,000 pounds of fuel or less.
2. During takeoff, landing, and all prolonged climbs and dives, the pilot and copilot seats shall be occupied and all passengers shall be seated. For all other flight conditions, only one person at a time is permitted to stand in the cockpit/intermediate cabin area. The jump seat is intended for in-flight use only; it must not be occupied during takeoff or landing.
3. The center-of-gravity limits are provided in the applicable weight and balance manual for the individual aircraft.

4.14 WEIGHT LIMITATIONS

Allowable gross weight figures for the particular aircraft shall be displayed in the cockpit and are as follows:

	CT-39E 19,922	CT-39G 158843, 158844 20,372	CT-39G 159361-159365 & 160053-160057 20,372
Ramp and taxi			
Start of takeoff	19,612	20,172	20,172
Zero fuel	12,800	13,800	13,800
Landing	17,500	17,500	17,500
Airborne over water beyond glide range of land	18,650	20,172	20,172

It is the responsibility of the pilot to determine that these weights are not exceeded. Three factors will affect the actual gross weight of the aircraft. They are:

1. The basic weight of the aircraft, that is given in the applicable weight and balance manual for the individual aircraft.
2. Useful load includes personnel and baggage. Weight of personnel to be carried (standard weight of 200 pounds per person includes clothing, personal equipment, and appropriate survival equipment).
3. Weight of fuel to be carried.

4.14.1 Baggage Compartment Weight Limitations. Baggage compartment weight limitations are 270 pounds maximum for CT-39E aircraft and 300 pounds maximum for CT-39G aircraft.

4.15 FUEL SYSTEM LIMITATIONS

4.15.1 Asymmetric Fuel Limitation — Takeoff and Landing

1. During takeoff or landing, lateral fuel balance must be within 200 pounds.

4.15.2 Asymmetric Fuel Limitation — In Flight

1. For all normal flight operations, maintain a lateral fuel balance within 400 pounds. If fuel imbalance should occur, balancing may be accomplished by use of the crossfeed and tank selector switch. Refer to paragraph 8.16, FUEL SYSTEM MANAGEMENT.

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PART II

Indoctrination

Chapter 5 — Indoctrination

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CHAPTER 5

Indoctrination

5.1 INDOCTRINATION

The information set forth here is intended as a guide to aircraft indoctrination. The requirements are considered the minimum and in no way are intended to compromise any operational or training standards.

5.2 GROUND TRAINING

The ground training requirements for an activity will vary according to local conditions and requirements and with crewmember experience. The following subjects are essential to ensure safe operation of this aircraft.

5.2.1 Flight Support Lectures or Briefings

1. Aircraft general
   a. Electrical system
   b. Hydraulic system
   c. Flight control system
   d. Environmental system
   e. Communication systems
   f. Navigation systems
2. Engines
3. Fuel system
4. Preflight inspection and line operations
5. Normal procedures
6. Emergency procedures and equipment
7. Aircraft limitations
8. Aircraft flight characteristics
9. Performance data and mission planning

5.2.2 NATOPS Flight Manual Examination. Open- and closed-book examination, stressing normal and emergency procedures and aircraft limitations, is required.

5.3 FLIGHT TRAINING

Before any flight training, all pilots will have completed the ground training program. An instructor pilot will occupy the right-hand seat during flight training.

Local conditions and requirements may affect the actual sequence; however, the following are considered essential to be accomplished:

1. Aircraft familiarization
   a. Takeoff and landing
   b. Takeoff and maximum continuous climb
   c. Approach to stall
   d. Slow flight
   e. System emergencies
   f. Acceleration to Mach limit
   g. Simulated single-engine cruise
   h. Simulated single-engine landing
   i. Simulated single-engine takeoff
   j. Simulated single-engine waveoff
   k. High- and low-altitude maneuvering

2. Instrument

   a. Basic instrument

   b. Penetration and approach

   c. Navigation (day and night)

3. Flight hours

   a. Aircraft familiarization — 5 hours.

   b. Instrument — 5 hours.

	TAC	T2P	T3P	AIRCREW
Monthly				
Hours in Model	4	4		4
Landings in Type	2	2		
Inst App in Type	1	1		

Figure 5-1. Minimum Currency Requirements

   c. Night familiarization — 2 hours.

   d. NATOPS evaluation flight — 2 hours.

5.4 PILOT QUALIFICATIONS

TAC, T2P, and T3P will be in accordance with OPNAVINST 3710.7 with the following exceptions:

1. TAC — 700 Total (minimum) 100 in Model.

2. T2P — 500 Total 100 in Model.

3. Transition Pilot — A pilot having held a transport aircraft commander designation may be initially designated a T2P consummate to his abilities.

5.4.1 Requalification. One refresher flight and an evaluation with a designated NATOPS instructor. This may be flown as a single, combined flight in which general performance is checked and emergency procedures are thoroughly reviewed.

5.4.2 Flight Requirements in Line Operations. Pilot must have completed the familiarization syllabus of at least 10 hours in model. Shall be designated a third pilot or higher before controlling the aircraft with passengers embarked.

Note

The flight hours and flight requirements specified here may be waived at the discretion of the unit commander when pilot experience warrants.

5.5 PERSONAL FLYING EQUIPMENT REQUIREMENTS

Personal flying equipment requirements delineated in OPNAVINST 3710.7 shall be adhered to on each flight.

5.5.1 Minimum Currency Requirements. Pilots and aircrew who fail to comply with the currency requirements (Figure 5-1) shall be scheduled to meet the requirement in which there is a delinquency. Delinquency in monthly currency may be satisfied by completion of an approved simulator course or NATOPS evaluation.

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PART III

Normal Procedures

Chapter 6 — Scheduling

Chapter 7 — Mission Planning

Chapter 8 — Shore-Based Procedures

Chapter 9 — Functional Checkflight Procedures

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CHAPTER 6

Scheduling

6.1 GENERAL

These procedures are designed to aid the operating crew in the conduct of an efficient and orderly flight.

6.2 CREW DUTY TIME

Crew duty time begins when a crew reports to a designated place of duty to begin preparation for a mission. However, in the event a pilot/crewmember is assigned other duties commencing before his reporting for preparation for a flight, his crew duty time will be computed from the time he initially reports for such assigned duties. Deadhead time and passenger time immediately before acting as a crewmember will count as crew duty time. Crew duty time ends when the engines are cut at the end of a mission or series of missions. Maximum crew duty time is 14 hours with a functioning autopilot and 12 hours without a functioning autopilot. Maximum crew duty time may be extended under special circumstances by the unit commander.

6.3 EN ROUTE CREW REST

Crew rest should be a minimum of 12 hours and should commence when the aircraft blocks in and shall end when the crew reports to the designated place of duty to begin preparation for the next mission. This shall permit the crew to have 8 hours of uninterrupted rest, plus time for postflight, transportation, and meals. The aircraft commander may declare a 10-hour crew rest period to expedite the aircraft's return to home base. Aircraft commanders will delay a mission to take crew rest whenever this action is in the interests of flight safety.

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CHAPTER 7

Mission Planning

7.1 MISSION PLANNING

Mission planning for training flight will be conducted in accordance with current training syllabuses. In order to allow thorough planning and guidance for the trainee, the mission planning/briefing shall commence 1 hour before scheduled departure.

Mission planning for the line operations will be conducted in strict compliance with current FLIPs, ICAO procedures, and applicable local directives. Refer to Part XI to determine fuel consumption, thrust settings, airspeeds, optimum altitudes, etc., to safely complete the assigned mission.

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CHAPTER 8

Shore-Based Procedures

8.1 PREPARATION FOR FLIGHT

8.1.1 Flight Restrictions. Refer to Chapter 4 for aircraft and engine limitations.

8.1.2 Flight Planning. Refer to XI to determine fuel quantity, engine settings, and airspeeds required to complete the proposed mission.

8.1.3 Weight and Balance. Refer to Chapter 4 for weight and balance limitations.

8.1.4 NATOPS Pilot Card Checklist. The numbered items (line items) in this chapter correspond to identically numbered items in the NATOPS Pilot's Pocket Checklist, NAVAIR 01-60GBE-1B.

8.2 USE OF CHECKLISTS

The following checklists are basic for all CT-39E/G aircraft operations. A CT-39E/G NATOPS Normal Procedures Card Checklist (NAVAIR 01-60GBE-1C) is provided for use as a condensed cockpit reference document.

8.3 PREFLIGHT CHECK

8.3.1 Before Exterior Inspection

1. ADB — Check

   Determine aircraft status by consulting the ADB. Review MAFS, noting discrepancies and corrective action.

2. Gear handle — Check Down.

3. Gustlock — Off (optional in gusty wind conditions).

8.3.2 Exterior Inspection. Starting at entry door while making exterior inspection, check for signs of fuel, oil, and hydraulic leaks; check that all access doors and panels are secured; check tires for general condition, that main wheels are chocked, and that both main wheel struts are extended evenly.

Note

The inspections required by the flightcrew are the minimum requirements for a safe flight; however, additional checks may be made at the discretion of the flightcrew.

In a clockwise path, check the following items:

1. Entry door ground lock — Removed.
2. Left pitot head — Cover Removed.
3. Static ports — Clear.
4. Water drain valves — Check.
5. Radome — Condition.
6. Landing and taxi lights — Condition.
7. Nosewheel strut — Check Extension.
8. Nosewheel steering release latch — Engaged.
9. Steering disconnect plug — Properly Engaged.
10. Nosegear maintenance safety pin — Removed.
11. Nosegear wheelwell — For General Condition.
12. Right pitot head — Cover Removed.
13. Static ports — Clear.
14. Speedbrake — For Condition (locks removed).
15. Single-point refueling cap — Installed and Tight.
16. Main wheel — Chocked.
17. Engine inlet duct — Clear.

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18. Wing slats — Freedom of Movement.
19. Gravity refueling cap — Installed and Tight.
20. Wingtip and light — General Condition.
21. Static wicks — Condition.
22. Wheelwell — For General Condition.
23. Wheel strut — Check Extension.
24. Starboard gear maintenance safety lock — Removed.
25. Engine pod latches — Secure.
26. Engine tailpipe — Clear.
27. Thrust reverser secondary latch S-hook — Check for Proper Engagement With Door Striker and That Door Striker Is Not Bent.
28. Hydraulic reservoir — Properly Serviced and Pressurized.
29. Battery connection — Secure.
30. Accumulator pressure — Check.
31. Aft equipment compartment — General Condition.
32. Empennage — For General Condition.
33. Engine tailpipe — Clear.
34. Thrust reverser secondary latch S-hook — Check for Proper Engagement With Door Striker and That Door Striker Is Not Bent.
35. Engine pod latches — Secure.
36. Fire extinguisher and oxygen dump indicators — Check.
37. Port gear maintenance safety lock — Removed.
38. Fire extinguisher gauges — Check.
39. Gear door ground control switch — Normal.
40. Wheel strut — Check Extension.
41. Wheelwell — General Condition.
42. Wingtip and light — General Condition.
43. Static wicks — General Condition.
44. Gravity refueling cap — Installed and Tight.
45. Wing slats — Freedom of Movement.
46. Engine inlet duct — Clear.
47. Main wheel — Chocked.

See figure 8-1 for entry door operation.

8.3.3 Interior Check

8.3.3.1 Cabin Area. The following checks are to be performed upon entering the aircraft.

1. Intermediate cabin area — Check That Loose Articles Are Properly Secured. Check That Jump Seat Is Stowed.
2. Cabin — Check for Loose articles.
3. Cabin circuit breakers (station 333) — In.
4. Ground escape hatches — Check That Ground Escape Hatches Are Properly Latched, Secured, and Unobstructed.
5. Baggage compartment — Check That Baggage, Cargo, Loose Articles, and Survival Equipment Are Properly Secured.
6. Entry door — As Required.
7. Hand fire extinguisher — Check for Condition and Pressure.
8. Oxygen walkaround bottle — Check for Condition and Pressure.
9. Oxygen bottle — Open Shutoff Valve and Check Pressure.
10. Crew briefing — Complete.

Before all missions, the aircraft commander will ensure that his crew is properly briefed regarding route weather, surface conditions, destination and alternate weather, ships and aircraft transiting the proposed route, ADIZ procedures, SAR facilities, and any other material pertaining to the proposed flight, as applicable.

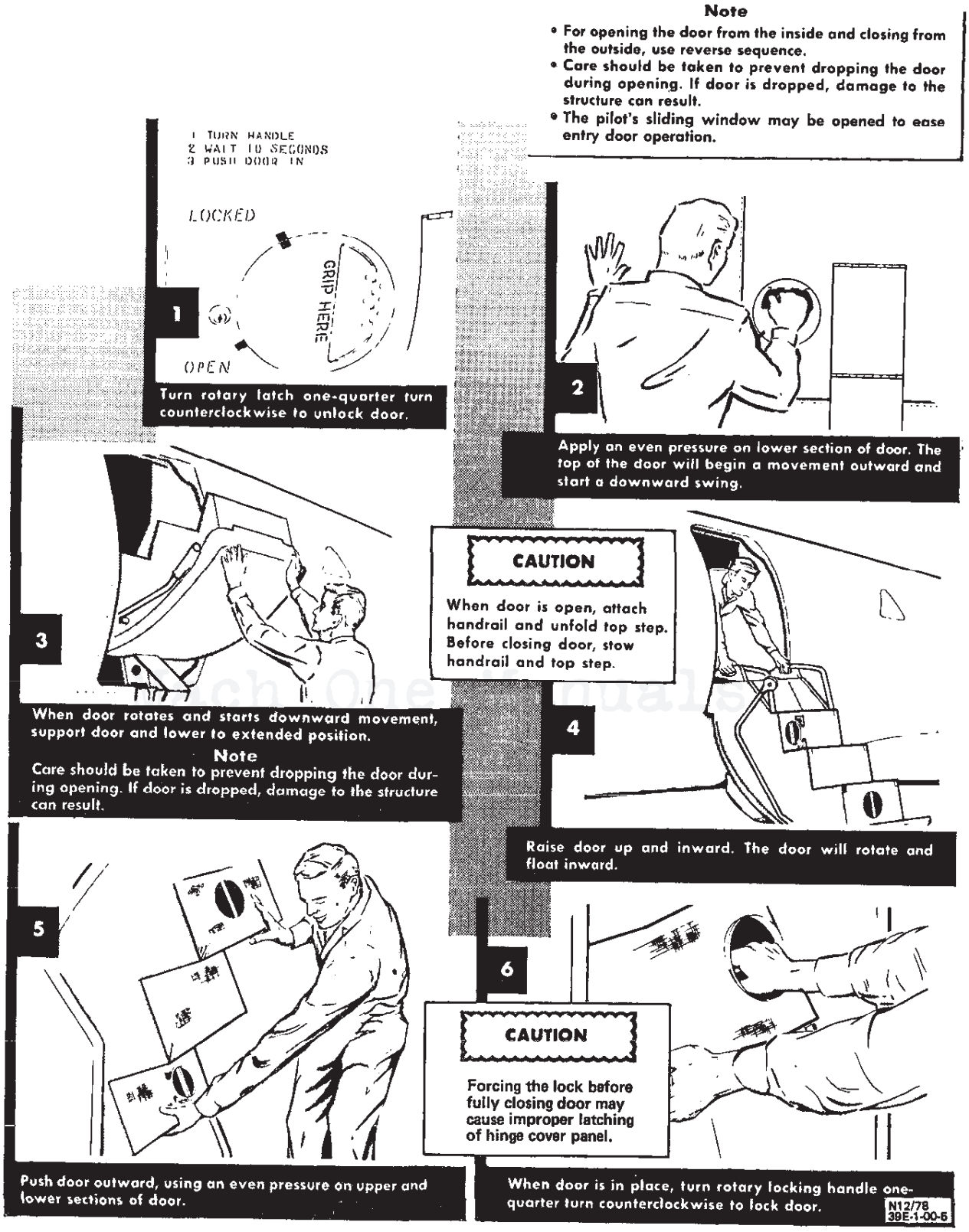

Figure 8-1. Entry Door Operation

8.3.3.2 Cockpit Area. After strapping in, check gustlock, adjust seatbelt and rudder pedals, and proceed to the applicable before-starting-engines procedure.

8.4 BEFORE STARTING ENGINES

1. Shoulder and seatbelt — Set.
2. Gear handle — DOWN.
3. Gear pin/pitot covers — Removed.
4. Circuit breakers (cockpit and cabin) — In. Check aft voltage regulator vent for proper ventilation and station 333 circuit breakers for condition.
5. Seatbelt, no smoking/lights — ON.
6. Strobe lights — OFF.
7. Thunderstorm, dome, and map lights — OFF.
8. Anti-icing control panel switches — OFF.
9. Antiskid switch — ON.
10. Electrical master switch — ON.
11. Fuel jettison switch (CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365) — OFF (if applicable).
12. Hydraulic pump/auxiliary switch — AUTO/OFF.
13. Windshield wiper switch — OFF.
14. Fire pull handles — In.
15. Radios, RAD ALT, radar, NAVAIDs, and transponder — OFF.
16. Engine master switches — OFF.
17. Starter buttons — OUT.
18. X-feed — NORMAL.
19. Thrust reverser control switch — OFF.
20. Ignition — NORMAL.
21. AIR START switch — OFF.
22. Throttles/thrust reverser levers — OFF/FORWARD.
23. Speedbrake — Centered OFF.
24. Autopilot — OFF (if applicable).
25. Emergency brake handle — IN.
26. Landing/taxi lights — RETRACT/OFF.
27. Cabin temperature — AUTO/SET.
28. Cabin air selector — OFF.
29. Nosegear steering switch — MAIN.
30. Autopilot emergency disconnect lever — ENGAGE (if applicable).
31. Cabin altitude/rate — Set.
32. Pilot/copilot cockpit lights — Set.

Note

To ensure maximum brilliance of the indicator, caution, and warning lights during daylight operations, the pilot flight instrument indirect light rheostat shall be at OFF.

33. Pilot, copilot oxygen — ON/100%/TEST.
34. Passenger oxygen — NORMAL/AUTO.
35. Speedbrake dump — NORMAL.
36. Landing gear emergency T-handle — IN/STOWED.
37. Pilot-copilot static source — Guarded (CT-39E aircraft and CT-39G aircraft 158843 and 158844); BOTH (CT-39G aircraft 159361 through 159365 and 160053 through 160057).
38. Pilot altimeter mode selector switch — NORM.

Note

On CT-39E aircraft and CT-39G aircraft 158843 and 158844, if it becomes necessary to select the pilot alternate static pressure source, the pilot altimeter mode selector switch must be positioned by STBY. This is because the static defect correction module does not compensate for alternate static pressure installation error.

On CT-39G aircraft 159361 through 159365 and 160053 through 160057, the static defect correction module is not required because the pitot-static tube is aerodynamically compensated for static pressure position error.

39. Anticollision/external lights — Set.

40. Auxiliary equipment switches — Guarded.

41. Batteries — ON.

42. Generators — OFF.

43. Inverter switch — Check in #1/Main Position.

44. Individual battery voltage — Checked.

45. Auxiliary power — As Desired.

46. Fire warning/aft fuselage hot lights — Tested.

47. Caution emergency lights — Test/ARM.

48. Battery temperature/isolation system — Tested.

49. Essential bus circuit breaker — Tested.

50. Parking brake — Set.

51. Landing gear electrical reset — RESET.

52. Passenger oxygen light — Tested.

53. Chocks — Removed.

WARNING

Before starting engines, make certain that areas fore and aft of the aircraft are clear of foreign objects, personnel, and equipment. (See Figure 3-13 for danger areas.)

8.5 STARTING ENGINES — EXTERNAL OR BATTERY POWER

The procedures for starting are the same with external or battery power except that during battery starts battery temperature is a consideration. The dc voltmeter should be monitored to ensure a minimum of 22 volt prior to start and a minimum of 10 volts during the start cycle.

Battery temperatures increase during battery starts and also when heavy charging is in progress. Do not attempt a battery start if the initial temperature of one of the batteries is over 120 °F. If connected, the GPU should be secured during battery starting because it may charge the batteries, thereby raising the temperature. This also prevents potential damage to the GPU.

The amber 120 °F light may come on during a battery start. If the red 150 °F "hot" light comes on, the master caution will be actuated. This can be canceled in the normal manner. If the 150 °F point is reached, observe the rate the needle is rising. Thermal runaway may begin to occur in the above 150 °F range. At 160 °F, immediately isolate the battery and observe that cooling is taking place. Report any temperature over 150 °F on the yellow sheet.

Following battery start, when generators are on the line, battery temperature should be noted. If over 150 °F and climbing, isolate the battery. Do not take off until the red light has gone out and the temperature is decreasing.

Note

Certain aircraft listed in Part I have an automatic disconnect feature set at 150 °F.

1. Radio — ON (as appropriate).

2. Danger areas — Clear.

3. Jet start unit (if used) — Connect.

4. Engine master switch — ON.

5. Inverter — ON.

6. Engine starter button — Depress. Hold depressed until engine revolutions per minute reaches approximately 30 percent. If button remains engaged at idle revolutions per minute, manually disengage.

CAUTION

If 10 percent is not obtained in 15 seconds or there is no indication of oil pressure within 20 seconds, the starter button should be pulled out to stop the starting cycle.

CAUTION

- The starter is limited to a total of 120 seconds in a 20-minute period (four 30-second cycles) with a 2-minute cooling period between cycles.
- If EGT rises rapidly to 475 °C, anticipate a hot start. Attempting a start with greater than 10 knots of tailwind component increases probability of a hot start.

7. Throttle — IDLE at 10-percent rpm.

Note

If revolutions per minute stabilizes between 8 (minimum) and 10 percent, advance throttle to IDLE without further delay. Exercise caution to preclude engine overtemperature.

8. Engine instruments — Check.

   a. Fuel flow gauge — Check (if less than 260 pounds or if more than 600 pounds per hour, abort the start and clear engine). The right engine fuel flow indication may be in a negative direction below approximately 10-percent rpm.

   b. EGT gauge — Check (525 °C, maximum start EGT, 515 °C maximum idle EGT).

CAUTION

If no light-up is obtained within 20 seconds or a hot start occurs, return the throttle to OFF and clear the engine. Do not attempt another start after a hot start until engine is inspected by maintenance personnel.

   c. Engine tachometer — Check (IDLE revolutions per minute, 42 ±1 percent).

   If the engine fails to accelerate to IDLE revolutions per minute within 1 minute after lightup, the throttles should be returned to OFF.

9. Generator — Checked/ON.

8.5.1 Second-Engine Start Using Jet Star Unit

10. Engine master switch — ON.
11. Second engine start — Repeat Steps 6 Through 8.
12. Generator — Checked/ON.
13. External power (if used) — Disconnect.

8.5.2 Immediate Second-Engine Battery Start

10. Battery switch(es) — OFF.
11. Isolation switch (if installed) — OVERRIDE.
12. Engine master switch — ON.
13. Second engine start — Repeat Steps 6 Through 8.
14. Battery/generator switches — ON.
15. Isolation switches (if installed) — Normal.
16. External power (if used) — Disconnect.

8.5.3 Delayed Second-Engine Battery Start

10. External power (if used) — Disconnect After First Engine Start.
11. Note loadmeter for operating dc generator. 0.5 or less ensures adequate battery charge.
12. Advance operating engine as required. Sixty-five-percent rpm guarantees adequate generator output to power the electrical system. The operating generator is then isolated from the starting cycle by moving the battery switches to OFF.
13. Engine master switch — ON.
14. Battery switch(es) — OFF.
15. Isolation switches (if installed) — OVERRIDE.
16. Second engine start — Repeat Steps 6 Through 8.
17. Batteries — ON.
18. Isolation switches (if installed) — Normal.
19. Generator — Checked/ON.

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8.5.4 Clearing Engine. When it is necessary to clear an engine of trapped fuel or vapor during ground operation, proceed as follows:

1. Electrical master switch — ON.
2. Battery switches — ON.

Note

Batteries are disconnected from the starter bus whenever the jet starter is plugged in regardless of battery switch position.

3. Engine master switches — ON.
4. Throttles — OFF.
5. Engine starter button — Depress; Motor Engine for 30 Seconds.
6. Engine starter button — Pull (up); Allow Engine To Stop Rotating Before Attempting Another Start. A Visual Inspection May Be Appropriate.

CAUTION

The starter is limited to 2 minutes of operation during any 20-minute period to prevent damage to the starter with a 2-minute cooling period between cycles.

8.6 BEFORE TAXIING

1. Ac/dc voltage/load — Check. Test Each Generator Output Voltage (27.2 to 29.5 volts).

   On CT-39E aircraft, check as follows:

   a. INVERTER switch — STBY/MAIN Checked.

   b. Ac voltmeter — Check.

      Normal standby inverter voltage reading should be 107 to 123 volts.

   c. INVERTER switch — MAIN.

   d. Ac voltmeter — Check.

      Normal main inverter voltage reading should be 107 to 123 volts.

   On CT-39G aircraft, check as follows:

   a. INVERTER switch — NO. 2.

   b. Ac voltmeter — Check.

      Normal inverter voltage reading should be 107 to 123 volts.

   c. Ac frequency meter — Check.

      Normal inverter frequency reading should be 390 to 410 Hz.

   d. INVERTER switch — NO. 1.

   e. Ac voltmeter — Check.

      Normal inverter voltage reading should be 107 to 123 volts.

   f. Ac frequency meter — Check.

      Normal inverter frequency reading should be 390 to 410 Hz.

2. Radios/IFF/autopilot — ON/Checked.
3. Standby attitude indicator switch — ON.

*4. Oxygen pressure — Checked.

5. Pilot/copilot RMI — Check and Slave.

*6. Mach/airspeed and stall warning — Checked. Check Mach Warning, Outboard and Inboard Vane Stall Warning. Observe Red Light(s) in Starter Button(s).

*7. Alternate/normal trim — Checked and Set.

In alternate trim position, run elevator trim with alternate trim switch. Select OFF with trim control selector switch to stop elevator travel. Set NORMAL and check operation of trim system. On CT-39G aircraft 160053 through 160057 and aircraft changed by Service Bulletins 74-6 and 74-7, operate elevator trim with normal trim switch on pilot control wheel and ensure audible trim warning horn sounds. While operating elevator trim, depress

* Items may be omitted if conducted by flightcrew plane captain.

TRIM DISCONNECT button. Trim action should stop, audible trim warning horn should silence, and TRIM DIS caution light should illuminate. Reset trim system by positioning trim switch from NORMAL to OFF, then select ALTERNATE position and repeat check. Perform entire check utilizing copilot control wheel. Return settings for takeoff:

a. Aileron — Set to 0.

b. Rudder — Set to 0.

c. Horizontal stabilizer — Set.

(1) Wing fuel only — 5.0°.

(2) Full fuel — 4.5°.

*8. Trim disconnect (if applicable) — Pilot/Copilot Checked.

*9. Fuel quantity — TEST-TOTAL.

a. Note total fuel quantity and if it is within balance limits.

b. Place test switch in up position until fuel quantity on left hand indicator stabilizes and indicates fuel in auxiliary tank; then hold switch down in TEST position and check that caution lights come on at 300 pounds for left and right hand tanks. If auxiliary tank is empty, switch to TEST position when left-hand indicator reads 1,500 pounds and ensure that low level caution lights come on at 300 pounds for left- and right-hand tanks. If auxiliary tank is empty, switch to TEST position when left-hand indicator reads 1,500 pounds and ensure that low level caution lights come on at 300 pounds for left- and right-hand indicators.

*10. Hydraulic system — Checked.

With aux switch ON, extend speedbrake; note drop in auxiliary hydraulic pressure. Turn auxiliary switch OFF; retract speedbrake; note drop in normal pressure. Check both normal and auxiliary hydraulic pressure gauges within normal limits. Select audible warning TEST (if applicable) (CT-39E, CT-39G aircraft 158843 and 158844).

*11. Flaps — Checked/UP.

Select flaps down; note speed of flap travel. Select flaps UP, stop at APPR, retract flaps fully, note flap position indicator shows full up.

12. YAW damper — OFF (if applicable).

13. Entry door — Closed/Locked/Light Out.

14. Cabin air select — As required.

15. Altimeter and clocks — Tested and Set.

CAUTION

If altimeter error is in excess of 75 feet, do not accept the aircraft.

16. Radios — Check.

Test operation of VOR/ILS receiver, tacan receiver, radio altimeter, and marker beacon receiver. (See Figures 8-2 or 8-3 if power was not available for aircraft preflight inspection.)

17. Caution lights — Check.

18. Gustlock — OFF.

19. Controls — Free.

Note

If any jamming or binding of flight or engine controls is experienced on the ground, check autopilot OFF, maintain controls in the jammed position, and request an immediate inspection of the controls.

20. Nosewheel steering — ENGAGED.

8.7 GROUND OPERATION

No engine warmup is necessary. As soon as the engine stabilizes at idle revolutions per minute with normal gauge readings, the throttle may be advanced to TAKEOFF THRUST.

* *Items may be omitted if conducted by flightcrew* plane captain.

CT-39E, CT-39G AIRCRAFT 158043
AND 159361 THROUGH 159365

STEP	MODE SEL	PROCEDURE	RESULT
1	HEADING	Set the A.P. POWER switch to ON. Adjust the pitch control for zero pitch command on the command bars.	The DISENGAGE indicator should light.
2	GYRO	Center the TURN control in the detent position and press the autopilot ENGAGE switch.	The command bars should move from view. The DISENGAGE light should go out, and the controls should resist manual movement.
3	GYRO	Move the TURN control to the right and left and return to center. Move the pitch control toward UP and DN and return to center.	The control wheel and column should follow the command.
4	GYRO	Set the ALTITUDE switch to ON and move the pitch control in either direction.	The ALTITUDE switch should move to the OFF position.
5	GYRO	Press the autopilot release button on the control wheel.	The DISENGAGE light should come on and the controls move freely.
6	GYRO	Press the ENGAGE button to engage the autopilot. Overpower the rudder and aileron controls. Overpower the elevator control.	The trim indicators should move in the proper direction. The trim indicator and trim control should move in the proper direction.
7	GYRO	Set the A/P EMERGENCY DISCONNECT to DISENGAGE. Return the A/P EMERGENCY DISCONNECT to ENGAGE.	The airplane controls should move freely.
8	HEADING	Set the heading marker 10 degrees to either side of the lubber line. Reset the heading marker under the lubber line.	The command bars should command a bank to the new heading. The command bars should return to the zero bank position.
9	HEADING	Turn the pitch control through its full range. Set the ALTITUDE switch to the ON position. Turn the pitch control through its full range. Return the ALTITUDE switch to the OFF position.	The command bars should deflect up and down as the control is turned. The command bars should remain stationary.
10	HEADING	Set the course arrow to a course approximately 25 degrees to either side of the course to the VOR station. Set the mode mode selector to the NAV/LOC mode. Turn the HDG knob several degrees in both directions.	The N/L ARM annunciator should light. The command bars should indicate commands to follow changes in heading.

Figure 8-2. Flight Director/Autopilot Check (CT-39E, CT-39G Aircraft 158443 and 159362 Through 159365) (Sheet 1 of 2)

STEP	MODE SEL	PROCEDURE	RESULT
11	HEADING	Set the course arrow to the course to the VOR station.	The lateral deviation bar should center, the N/L CAPT annunciator should light, and the bank commands should follow the course arrow.
12	AS REQUIRED FOR FLIGHT	Set all controls for subsequent flight operations or turn all systems off.	
13	RADAR ALTIMETER	Turn on, set MDA index to desired altitude, perform self test.	MDA light should light.

NTDA-CT39E/G-NFM-3/1A-5/76

Figure 8-2. Flight Director/Autopilot Check (CT-39E, CT-39G Aircraft 158443 and 159362 Through 159365) (Sheet 2 of 2)

CAUTION

- If a takeoff thrust engine runup is to be made during ground operation, be sure the main wheels are chocked and hold the brakes on because at light gross weights the aircraft wheels may slide.
- Prolonged ground operations at high power settings can result in the scorching and/or buckling of the engine cowlings.
- If either throttle is inadvertently retarded to off, even if only momentarily, do not reopen the throttle because relight may be impossible and the resultant flow of unburned fuel into the engine can create a fire hazard in the engine.

8.8 TAXIING

Perform the following while taxiing:

1. Normal and emergency brakes — Check and Handle IN.
2. Nosewheel steering check
   a. Nosewheel steering test switch — TEST.
   b. Deflect rudder — Check Fail Down to STBY.
   c. Cycle nosewheel steering switch from STBY to MAIN reengage normal steering.
   d. Test faildown in opposite direction.
   e. Test switch normal; reengage normal steering.
3. *Flight instruments* — Check. Check all Flight Instruments for Proper Operation.

8.9 BEFORE TAKEOFF

After taxiing to the takeoff area, complete the following checks:

1. Speedbrake switch — IN.
2. Thrust reverser control switch — EMERG STOW (if applicable).
3. Thrust reverser stops — Set (if applicable).
4. Wing flaps — UP.
5. Engine instruments — Check That Both Engines Are Operating Normally.
6. Generators/batteries — Check/ON.
7. Cabin air selector switch — RH-OFF/LH-OFF/BOTH.
8. Pilot sliding window — Closed/Locked.
9. Windshield anti-icing switches — ON.
10. Anti-icing control panel switches — As Required.

    If fuel temperature is at or below freezing, the fuel heater should be turned on for 1 minute before takeoff.

STEP	MODE SEL	PROCEDURE	RESULT
1	NONE (ALL BUTTONS OFF)	Engage AP and YD levers.	Autopilot system is energized.
2	HDG AND ALT	Move the TURN control to right and left and return to center. Move PITCH control toward UP and DN and return to center.	Control wheel and column should follow the left, right, up and down command. HDG and ALT modes disengage (OFF).
3		Move control colum and wheel manually in all directions.	Resistance to movement should be noticed.
4		Hold control column and manually trim nose down.	A-P TRIM FAIL annunicator light should illuminate.
5	HDG	Set heading marker on course indicator to 10 degrees on either side of lubber line.	Command bars should depict a bank angle
6		Set heading marker on course indicator under lubber line.	Command bars should return to zero bank position.
7	HEADING	Set the course arrow to a course approximately 25 degrees to either side of the course to the VOR station. Depress the NAV/LOC mode button. Turn the HDG knob several degrees in both directions.	The N/L ARM annunciator should light. The command bars should indicate commands to follow changes in heading.
8		Depress autopilot disconnect button on either control wheel.	AP and YD levers should move to the DISENGAGE position. The AP DISC annunciator should light and the control column and wheel should move freely.
9	AS REQUIRED FOR FLIGHT	Set all controls for subsequent flight operations or turn all systems off.	
10	RADIO ALTIMETER	Turn on, set to desired DH altitude, perform self test.	DH light should light.

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Figure 8-3. Flight Director/Autopilot Check (CT-39G Aircraft 158444 and 160053 Through 160057)

WARNING

Do not take off with the fuel heater on. If fuel heat is necessary before takeoff, complete operation before takeoff.

11. Fuel quantity — Balance Check.
12. Flight directors/emergency gyro — Set.
13. Takeoff trim — Check.
14. COMM/NAV/radios — SET.
15. Aircraft with KNS-660 installed — NAV/TAC Selected.
16. Radar/radar stab — STANDBY/ON.
17. Takeoff data — Reviewed.

    Sufficient takeoff data will be reviewed to ensure safe normal takeoff and abort/emergency return.

18. Crew and passenger briefings — Completed.

8.10 LINEUP

1. Nosegear steering test switch — NORMAL (if applicable).

CAUTION

Do not takeoff with nosegear steering test switch in test position. This may cause undesired failure to STBY.

2. Strobe lights — ON.

WARNING

Operation of strobe lights without the glass lens installed can result in permanent eye damage to personnel in vicinity of aircraft.

3. Transponder — ON.
4. Radar/radar stab — As Required.
5. Ignition — Continuous.
6. Landing/taxi/ice lights — As Required.
7. Compass/clock — Check.
8. Caution and warning lights — Check.
9. Controls — Free.
10. Engine instruments — Check.

CAUTION

- The temperature and duration of any engine operation during which EGT limit is exceeded should be recorded on the flight record. If limits are exceeded, shut down the engine as soon as possible. Overtemperature operation requires engine inspection.
- Should engine speed exceed the maximum overspeed limit, either with or without an overtemperature condition, shut the engine down as soon as possible. The engine shall be inspected for malfunction and possible damage when overspeed occurs.

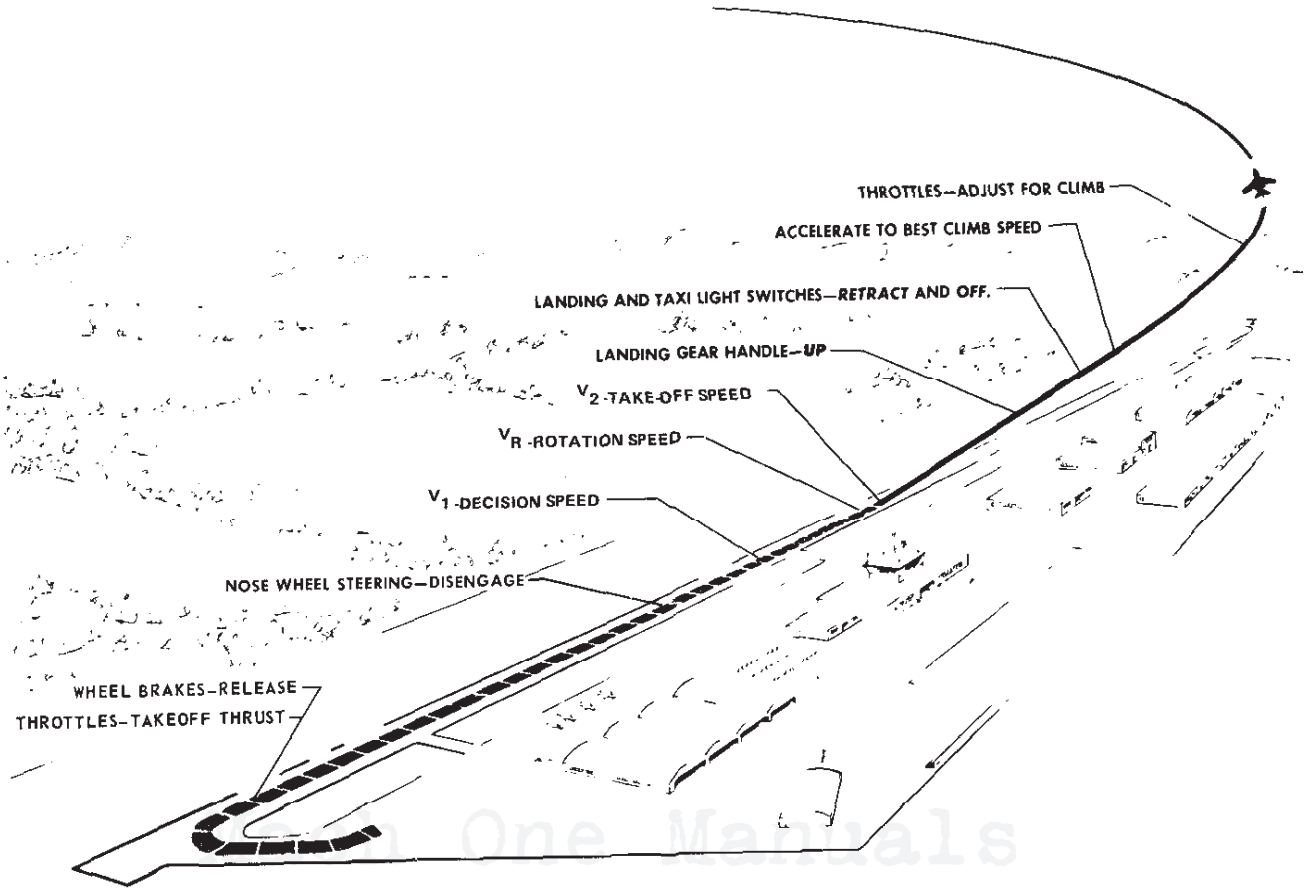
8.11 TAKEOFF

See Figure 8-4 for takeoff and initial climb.

8.11.1 Normal Takeoff

Note

- The procedures set forth will produce results given in Part XI. Refer to the appropriate charts for takeoff distance and speed.
- Nosewheel steering should be disengaged at approximately 80 KIAS.
- Ailerons and rudder may be used to effectively aid in directional control on the ground during takeoff.
- Avoid using brakes for directional control because takeoff instances will be increased.



Note

Refer to Part XI for takeoff distances and speeds.

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Figure 8-4. Takeoff and Initial Climb — Typical

With the throttles at TAKEOFF THRUST, check all engine instruments, then release the wheelbrakes. Rolling takeoffs are permitted when runway length is not critical. Once the proper engine pressure ratio has been set, no further adjustments should be made during the takeoff run. An increase in engine pressure ratio because of ram effect as airspeed increases is normal. Maintain directional control with nosewheel steering, rudder, and ailerons. During the takeoff, monitor engine instruments and observe critical engine failure speed (V_1), rotation speed (V_R), and takeoff safety speed (V_2) speeds. The nosewheel should be left on the runway until V_R is attained. At the scheduled V_R speed, rotate the aircraft smoothly to a liftoff altitude of approximately 12° and maintain a constant climb angle until climb speed is reached, then continue climb at this speed. In the event of an engine failure after V_1, initiate landing gear retraction within 3 seconds after liftoff. Above the 35-foot height, maintain V_2 speed until the altitude to be used for acceleration to the final climb speed is attained if takeoff is limited by obstacles. Accelerate to final takeoff climb speed using the appropriate takeoff flightpath procedure shown on the net takeoff flightpath chart (see Part XI).

If engine failure is recognized before attaining the decision speed, V_1, retard power levers to idle immediately, apply brakes, and use nosewheel steering as necessary.

For a normal two-engine operating takeoff, initiate rotation at V_R. If the takeoff is limited by obstacles, do not permit the maximum speed during the takeoff climb to exceed the initial climb speeds for two-engine operation shown on the takeoff airspeed charts. Maintain this speed to the altitude selected for acceleration to the final climb speed in the one-engine flightpath profile.

WARNING

A premature nosewheel liftoff or allowing airspeed to build up above recommended V_R increases ground roll considerably.

8.11.2 Crosswind Takeoff. In addition to the procedures used for a normal takeoff during a crosswind takeoff it will be necessary to keep the aircraft in a three-point attitude longer than normal. Refer to the crosswind chart in Part XI to determine the minimum nosewheel liftoff speed to be obtained before aerodynamic steering (rudder control) will be sufficient to maintain direction control. When takeoff is started, directional control is maintained with nosewheel steering. It is recommended that nosewheel steering be disengaged just before nosewheel liftoff so that rudder control is established before nosewheel liftoff and subsequent loss of nosewheel steering. Be prepared to counteract drift after breaking ground.

8.11.3 Takeoff With High Outside Air and Runway Temperatures. During takeoff with high outside air and runway temperatures, caution shall be used to prevent premature nosewheel liftoff and aircraft takeoff. High outside air temperature increases the takeoff run distance. It is imperative that the recommended takeoff speed be used. Refer to Part XI for takeoff speeds and distances.

8.11.4 Noise Abatement Climb Procedures

1. Using takeoff thrust, climb 2,500 feet above airport runway at V_2 +10 KIAS (2,500-feet AGL unless restricted by departure altitude).
2. At 2,500 feet, retard power level setting to EPR value provided in Figure 8-5, EPR setting for thrust cutback, maintaining V_2 +10 KIAS.
3. Maintain reduced thrust setting until noise abatement is no longer required.

8.12 AFTER TAKEOFF

When the aircraft is definitely airborne and there is no possibility of settling back onto the runway:

1. Landing gear — UP/Light Out.

CAUTION

The landing gear and doors should be completely up and locked before the gear-down limit speed is reached. Refer to Chapter 4 for gear-down limit speed.

Note

After takeoff, the autopilot should not be engaged prior to 1,500-feet AGL.

2. Landing and taxi light switches — RETRACT and OFF.

   When the landing and taxi lights are used for takeoff, they shall be retracted before limit airspeed is reached. Refer to Chapter 4 for the landing-taxi light extended limit speed.

3. Climb thrust — Set (approximately 800-feet AGL).

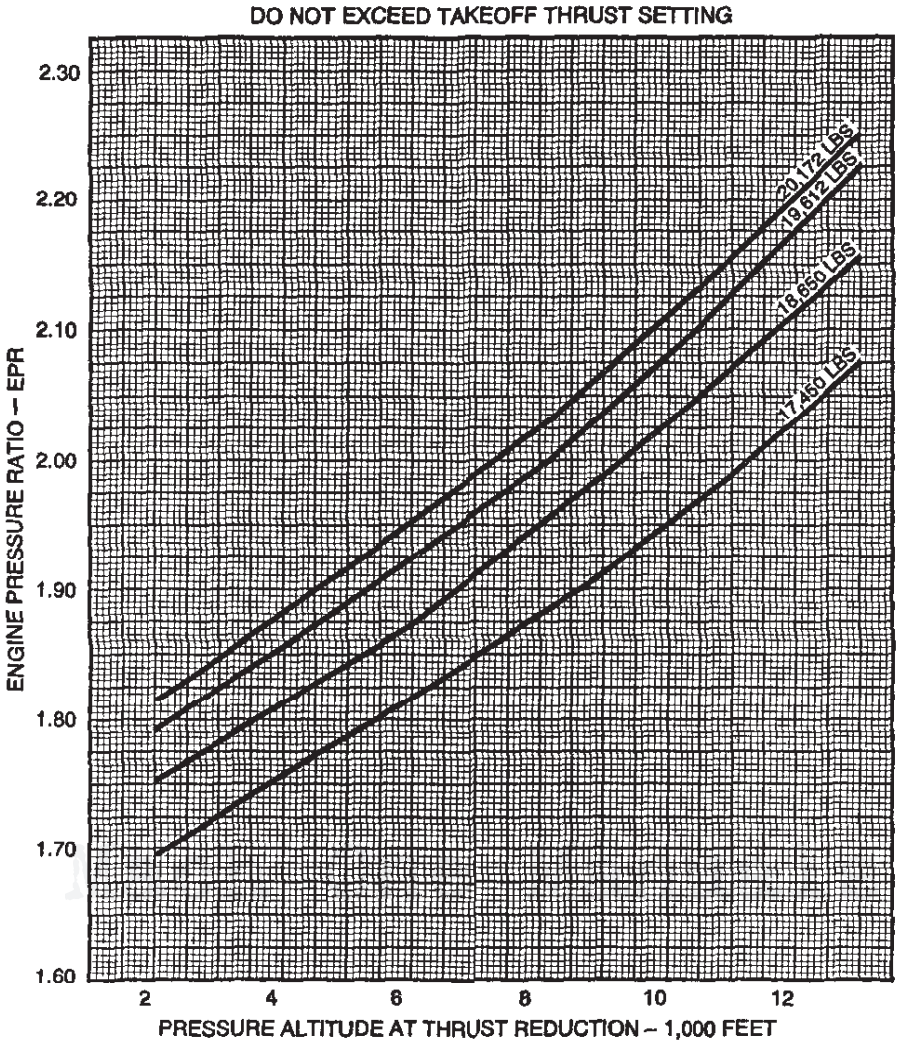
Note

Refer to paragraph 4.13, CENTER-OF-GRAVITY LIMITATIONS for limits affecting the operation of the fasten seatbelt sign.

4. Cabin pressure/temperature — Set.
5. Ignition — As Required.
6. Anti-icing — As Required.
7. Thrust reverser control switch — OFF (above 1,500-feet AGL).
8. Altimeter — Set.

8.13 CLIMB

Climbs can be made using maximum continuous thrust. When climbing at maximum continuous thrust, maintain 260 KIAS until .69 Mach is intercepted. Because of changing conditions during a climb, it is necessary to recompute and reset thrust settings periodically. As the climb progresses to higher altitudes, the possibility of overboosting the engines increases and the thrust settings shall be monitored more closely.



PRESSURE ALTITUDE	JT12-8 20,172 LBS	JT12-8 19,612 LBS	JT12-8 18,650 LBS	JT12-8 17,450 LBS	JT12A-6A ALL WGTS
2,500	1.83	1.81	1.76	1.71	1.68
3,500	1.87	1.84	1.80	1.74	1.70
4,500	1.90	1.87	1.82	1.77	1.73
5,500	1.94	1.91	1.86	1.80	1.76
6,500	1.97	1.94	1.89	1.83	1.80
7,500	2.01	1.98	1.93	1.86	1.83
8,500	2.05	2.02	1.97	1.90	1.86
9,500	2.09	2.06	2.00	1.94	1.89
10,500	2.13	2.10	2.05	1.97	1.92

EPR SETTING FOR THRUST CUTBACK SPEED IS V_2 + 10 KIAS

Figure 8-5. EPR Setting for Thrust Cutback

Note

- When a continuous climb from takeoff to a cruise altitude is being performed, auxiliary fuel feed may discontinue at approximately 25,000-feet altitude (approximately 150 pounds of fuel remaining in the auxiliary fuel tank) because of the nose-high attitude in the climb. Upon leveling off in cruise flight, auxiliary fuel feed will resume and continue until the auxiliary fuel tank is empty. (Resuming a level flight attitude restores the gravity head of this remaining portion of the auxiliary fuel.) With the FUEL QTY test switch at RH IND-RH WING LH IND-AUX, monitor auxiliary tank fuel quantity after leveling off from a continuous climb to altitudes above 25,000 feet to determine when tank is empty. Place FUEL QTY test switch at TOTAL when the auxiliary tank is empty.

- Refer to Chapter 4 for operating limits of takeoff thrust.

8.14 CRUISE

When the initial cruise altitude is reached, level off and accelerate to the desired Mach for cruise. As the desired Mach is attained, adjust thrust as necessary to maintain this Mach. The maximum EPR for any cruise condition will be the computed maximum continuous thrust for that Mach, altitude, and temperature. This maximum value will vary with Mach, altitude, and temperature and should be rechecked periodically during cruise. During level acceleration, EPR may be increased to correspond with the increasing Mach.

After reaching cruise altitude:

1. Pressurization system — Check.

   Check that cabin altitude and pressure differential are following schedule. (Refer to paragraph 2.1, AIR-CONDITIONING AND PRESSURIZATION SYSTEM.)

2. Seatbelt sign — Set.

3. Battery condition/temperature — Check (after 30 minutes of flight.)

   After the first 30 minutes and at any time the dc loadmeter readings or battery temperature readings are higher than normal, place the respective battery switch at OFF and note the change in the loadmeter reading. If this change exceeds one-half division on each loadmeter, leave the battery switch at OFF for the remainder of the flight. In cases where the battery switch is turned off, make an entry on the yellow sheet for a maintenance check of the battery. Dc loadmeter readings during the first 30 minutes of flight are not conclusive because low-charge batteries will cause higher than normal readings until charged.

   If takeoff was made with either battery isolated, battery temperature should normalize below 120 °F. Return battery switch to normal when all warning and caution lights are out and observe the generator loadmeters. Accept no more than 1-1/2 scale division rise and note load decreases within 1 to 2 minutes. If the amber light comes on, monitor temperature trend and isolate the hot battery or batteries at 140 °F. If the red 150 °F light comes on, disconnect the battery or batteries and operate the generator only.

   Note

   Certain aircraft listed in Part I have an automatic disconnect feature set at 150 °F.

4. Cabin blower — OFF (if applicable.)

5. Fuel heater — Operate As Necessary.

   The fuel heater should be used when either the FUEL FILTER BLOCK caution light comes on or the fuel temperature is at or below 0 °C (32 °F). Refer to paragraph 2.13, FUEL HEATER SYSTEM, for heater procedures.

8.15 FUEL BALANCING

During flight, it is recommended that the following fuel balancing procedure be used when the fuel imbal ance exceeds 400 pounds. Move the fuel crossfeed and tank selector switch to select the heaviest tank. This supplies both engines with fuel from the heavy tank. When the fuel level is equal in both wing tanks, return the switch to NORMAL.

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CAUTION

Do not attempt this procedure for fuel balancing if the fuel crossfeed and tank selector switch has been placed at -X-FEED because of a tank-mounted boost pump failure. This may result in shutting down the remaining operable boost pump.

Note

During fuel balancing, with the fuel crossfeed and tank selector switch at LH TANK or RH TANK position, it should be noticeable that fuel for both engines is being used from the tank selected (fuel coming back into balance). If noticeable fuel balance does not occur or if the unbalance condition increases, it is possible that the opposite tank boost pump has not been automatically deenergized. If this occurs, turn off the light wing boost pump by pulling the applicable circuit breaker. When balancing is accomplished, reset the boost pump circuit breaker and place the tank selector switch back at NORM.

Refer to Part IV for information regarding flight characteristics.

8.16 FUEL SYSTEM MANAGEMENT

Normal operation of the fuel system is from the wing tank to the respective engine. Fuel in the fuselage tank is automatically fed by gravity to the wing tanks. There is no tank selection for the fuselage tank. Fuel may become trapped in the outboard section of the wings during prolonged cruise at slow airspeeds.

1. FUEL-X-FEED & TANK SEL switch — NORM.

   Both the gravity and pressure crossfeed valves are closed. Each tank-mounted fuel boost pump will operate when its respective engine master switch is in the ON position.

2. FUEL-X-FEED & TANK SEL switch — -X-FEED.

   Both the gravity and pressure crossfeed valves are open. Both tank-mounted fuel boost pumps will operate even if one engine is shut down. The -X-FEED position should be used in the event of an engine failure or when a tank-mounted fuel boost pump has failed, indicated by a LOW FUEL PRESS caution light.

Note

- On final approach, with both engines operating, place FUEL-X-FEED & TANK SEL switch at NORM.
- During other flight conditions when required fuel flow total for both engines exceeds 2,400 pph, place FUEL-X-FEED & TANK SEL switch to NORM. With the selector switch at NORM, an engine with an inoperative tank-mounted boost pump will be on suction feed.

3. FUEL-X-FEED & TANK SEL switch — LH TANK or RH TANK.

   The gravity crossfeed valve is closed and the pressure crossfeed valve is open. Only the tank-mounted boost pump in the selected tank will operate.

8.17 DESCENT

Refer to Part XI for airspeed, time, fuel used, and distance traveled during power-off, partial thrust, or maximum range-minimum time descents.

Note

- Adjust engine thrust during descent to provide adequate pressurization. Refer to paragraph 2.1, AIR-CONDITIONING AND PRESSURIZATION SYSTEM.
- In-flight engine hangup at or near idle and slow engine acceleration can be avoided by maintaining a minimum of 50-percent rpm during all phases of the approach.

1. Anti-icing/fuel heat — As Required.

   The engine revolutions per minute shall be advanced to 1.5 EPR if icing conditions are encountered to provide sufficient compressor bleed for adequate engine anti-icing.

2. Cabin pressurization — Set to Approximately 250 Feet Above Field Elevation.

3. Fuel quantity/balance — Check.

4. Batteries — Check.

Note

When approaching terminal area and both batteries have been isolated, if a battery has cooled to 120 °F, place it on the bus for landing. If the amber light comes on, turn the battery off and land with generators only.

5. Antiskid switch — ON.

6. Seatbelt sign — ON.

8.18 APPROACH

1. Altimeter — Current Setting.

2. Radio altimeter — Low Limit.

3. Pressurization — Check.

4. Landing data — Compute.

5. Thrust reverser stops — SET (if applicable).

6. Thrust reverser control switch — EMERG STOW (if applicable).

7. Course select/NAVAIDs/POINTERS/NAV/TAC/VLF — SET.

WARNING

Ensure that the function switch NAV/TAC/VLF on the pilot instrument panel is properly selected to NAV/TAC position to ensure proper sensing of the pilot CDI.

8. Crew and passenger briefings — Completed.

8.19 BEFORE LANDING

*1. Ignition — Continuous.

*2. Seatbelt/no smoking sign — ON.

*3. Flaps — Approach.

4. Thrust reverser control switch — EMERG STOW (if applicable).

5. Landing gear handle — DOWN.

Lower the gear below gear-down limit speed.

To lower the landing gear, fully depress the release safety latch that locks the handle in the UP position. While continuing to hold this latch depressed, place the landing gear handle in the DOWN position. Then release the latch and ensure that the handle is locked down. Check for gear-down-and-locked indication.

CAUTION

Failure to keep the safety latch depressed while moving the handle downward will damage the latch mechanism. This will subsequently permit downward handle movement (independent of the latch) sufficient to lower the landing gear but not to lock the handle. An unsafe and unknown gear-down problem will then exist.

Note

If, after a normal waiting period, a safe gear-down indication is not observed, depress the landing gear downlock electric reset button to relieve the hydraulic pressure on the spring-loaded downlock pins. If a safe gear-down indication does not occur immediately, refer to landing gear emergency lowering procedure in Part V.

6. Hydraulic pressure/brakes — Check.

*7. Landing lights — As Required.

8. Yaw damper — OFF (CT-39G) (if applicable).

9. Flaps — DOWN.

10. Slats — Check.

11. Speedbrake — As required.

* During subsequent approaches/touch and go landings, it is not required to perform asterisk items.

8.20 LANDING

Although ground temperature is above freezing, extended flight at high altitudes can "cold soak" the fuel and aircraft sufficiently for icing conditions to exist until after landing. Operate the fuel heater for 1 minute in the traffic pattern to remove any ice accumulation in the fuel control.

CAUTION

If fuel heater operation is necessary before landing, its operation shall be completed prior to the final approach.

8.20.1 Normal Landing Technique. Following entry into the pattern, reduce airspeed so as to roll out on downwind at less than 180 KIAS. Lower the landing gear and wing flaps. The speedbrake may be used at the discretion of the pilot.

Note

- Do not attempt a full-stall landing because the tail could drag the ground at high angles of attack.
- The amount of wing flaps that can be used is variable; however, it is recommended that full flaps be used to decrease the landing speed. Refer to Part XI for landing distances.

Fly the base leg at 10 to 20 KIAS above the final approach speed and adjust the pattern and throttle to control the rate of descent to under 1,000 fpm. This will require above 70- to 75-percent rpm. Roll out onto final at the computed approach speed and reduce the rate of descent to a minimum value. Fly a smooth final. Adjust speed, thrust and rate of descent as necessary to arrive at the touchdown point at the desired speed. Reduce the throttle to IDLE. The rate of sink at touchdown should not exceed 6 feet per second for heavyweight landings or 10 feet per second for aircraft gross weights less than 13,000 pounds. After touchdown, lower the nosewheels to the runway. Cross arresting gear with nose tires off the ground whenever sufficient airspeed exists. At lower airspeeds, the yoke should be held full aft to lighten the nosegear load as much as possible. Aerodynamic braking is ineffectual under normal landing conditions.

CAUTION

- Do not land with brakes on, as antiskid protection will not be available on impact.
- Failure to remove weight from the nosegear when landing over E-28 arresting gear may cause nosegear damage. If split rubber tires are used to support the arresting cable, request the landing facility to spread the tires prior to landing.

8.20.2 Normal Landing Procedure (After Touchdown). The normal landing procedures are given in Figure 8-6. Extending the speedbrake after touchdown will reduce the landing ground roll distance by 6 percent.

CAUTION

If it becomes necessary to land across arresting cables, close speedbrake to prevent damage to the brake.

1. Wing flap handle — UP.
2. Thrust reversers — As Required (if applicable).
3. Wheelbrakes — As Required.

Note

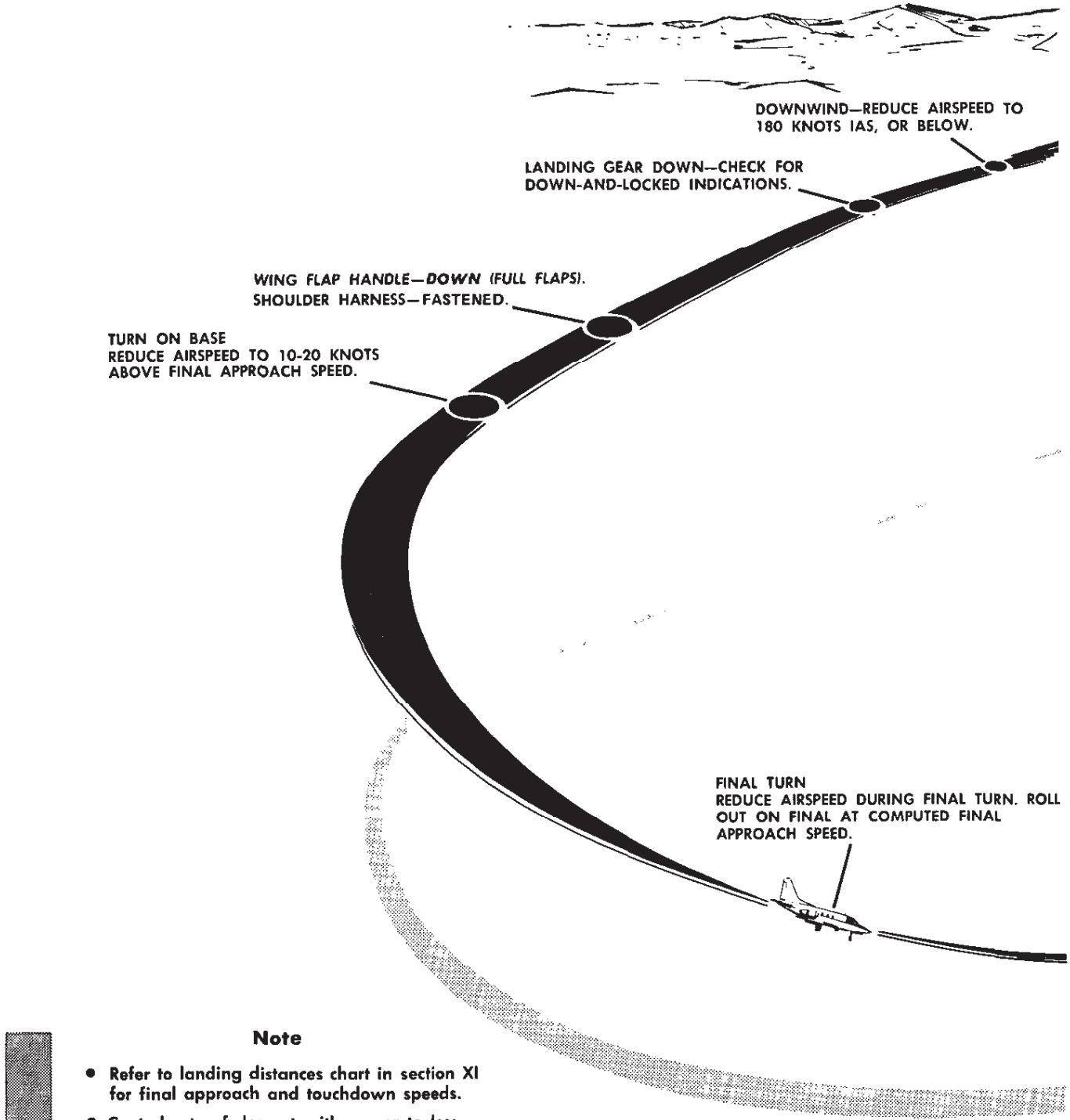
After it is ascertained that the braking system is operating normally, one engine may be secured on the landing rollout to shorten the landing rollout and reduce the amount of braking required on subsequent taxiing.

4. Nosewheel steering — Engage Prior to Flight Controls Becoming Ineffective.

Note

Ensure that rudders are approximately neutral before engaging nosewheel steering. Pilots shall be attentive towards the possibility of a nosewheel steering hard over.

5. Thrust reversers — Stow (if applicable). Initiate stow procedure prior to decelerating to 60 KIAS.



Note

- Refer to landing distances chart in section XI for final approach and touchdown speeds.
- Control rate of descent with power to less than 1000 feet per minute before flare.
- Use caution during the flare in presence of gusty winds or jet wash.

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Figure 8-6. Landing Patterns (Sheet 1 of 2)

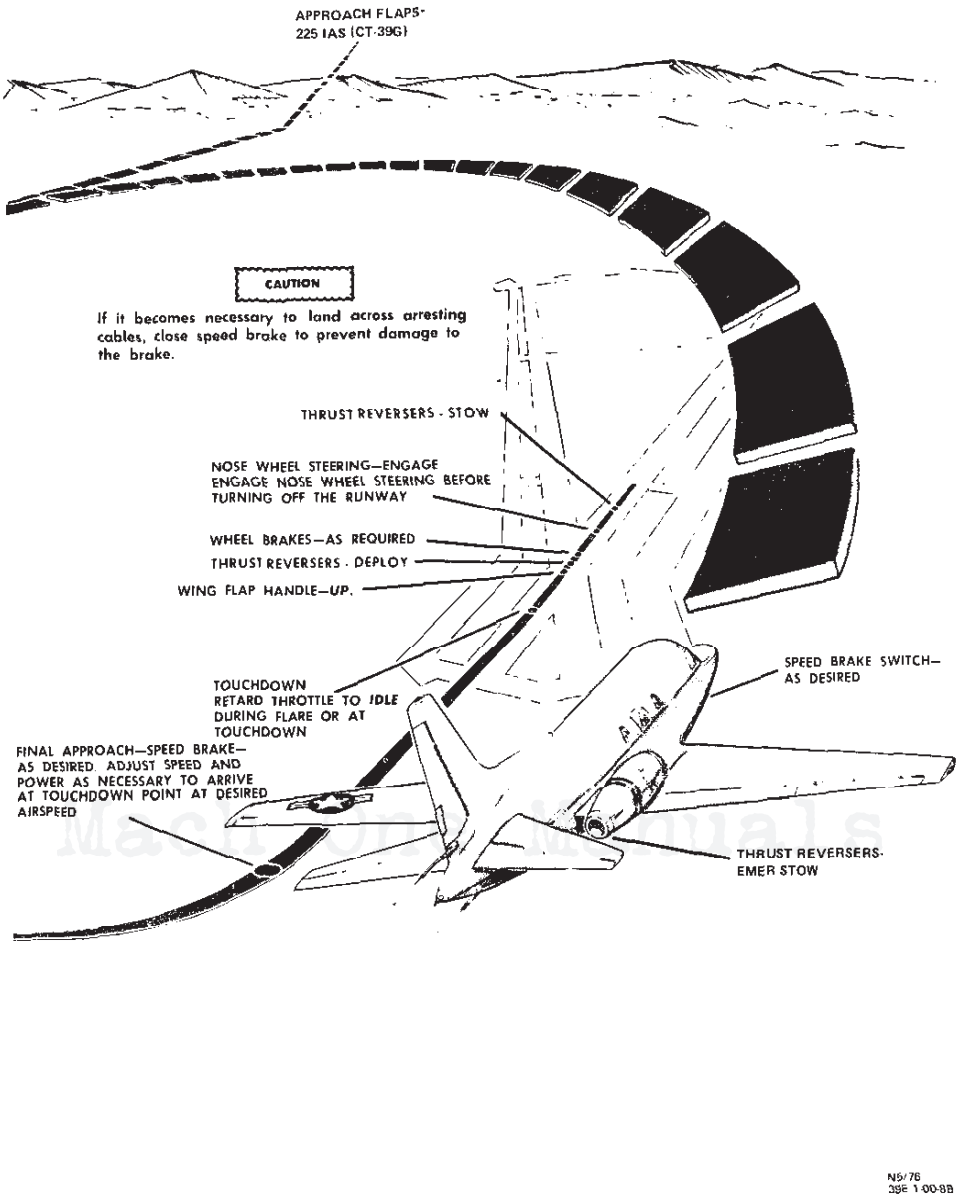

Figure 8-6. Landing Patterns (Sheet 2 of 2)

8.20.3 Heavyweight Landing. For heavyweight landings, increase the approach and touchdown speeds as listed in Part XI.

CAUTION

To avoid excessive sink rates, adjust power and altitude as necessary to maintain the proper approach and touchdown speeds for weight and configuration.

8.20.4 Minimum-Roll Landing

CAUTION

During a minimum-roll landing approach, airspeed control shall be maintained to avoid excessive sink rates. To achieve a minimum-roll landing, fly the final approach at 5 KIAS less than the recommended approach speed for weight and configuration with the speedbrake out. Do not fly approach at speeds below V_{MCA}. Short-field landings should be made utilizing a manual glideslope and rate of descent. The aircraft should touch down within the first 500 feet of the threshold.

As soon as main gear touchdown is felt, lower the nosewheel, retract the flaps immediately, deploy thrust reversers, start braking action, and engage nosewheel steering prior to flight controls becoming ineffective. On aircraft without reversers, if desired, one engine may be shut down to reduce engine thrust and aid in reducing the landing roll.

CAUTION

Use the brakes as required, applying light force on the pedals, and increase the force as the aircraft slows down.

Note

When the flaps are down, the weight on the landing gear at touchdown is about one-third of what it is when the flaps are up.

8.20.5 No-Flap Landing. No special technique is required for landing without wing flaps, other than extending the pattern. Final approach speed should be increased by 10 knots.

8.20.6 Hydroplaning. Hydroplaning, in its meaning here, is a condition where the tires of the aircraft are separated from the runway surface by a fluid. Under conditions of total hydroplaning, the hydrodynamic pressures between the tire and the runway lift the tires off the runway to the extent that wheel rotation slows and actually stops. The major factors in determining when an aircraft will hydroplane are forward speed and tire pressure. To a lesser degree, the aircraft gross weight, depth of water on the surface, texture of the surface, type of tire used, and condition of tires influence the total hydroplaning speed. Total hydroplaning in this aircraft with recommended tire pressures and one-eighth to one-fourth inch of water or slush on the runway can be expected at approximately 78 KIAS for the nosegear tires and 121 KIAS for the main gear tires. Hydroplaning is aggravated in landings with a tailwind component because of increased ground speeds. Partial hydroplaning occurs to varying degrees below these speeds.

Whenever an aircraft is subjected to hydroplaning to any degree, directional control becomes difficul Under total hydroplaning conditions, nosewheel steering is ineffective and wheel braking is nonexistent.

The adverse effects of hydroplaning can be minimized by consideration and application of the following:

1. Smooth tires tend to hydroplane with as little as one-tenth inch of water and possibly at slightly lower speeds. Ribbed tires tend to release hydrodynamic pressures and will not hydroplane until water depth is two-tenths to three-tenths inch.

2. Takeoffs with extreme crosswinds and water-covered runways should be made with caution. When liftoff speed is greater than hydroplaning speed, the aircraft is subjected to the effects of the crosswinds while hydroplaning.

3. With inoperative thrust reversers, an aborted takeoff on a wet runway initiated at or near hydroplaning speed will require considerably more runway than one aborted on a dry runway.

4. For landings without use of thrust reversers, reduce gross weight below 15,000 pounds and effect a positive main gear touchdown early to utilize available runway. Maintain directional

control with rudder to below nosegear tire hydroplaning speed. Utilize any dry spots on runway for early braking. The cautions applying to "wet to dry" braking techniques are applicable. Leave the antiskid system turned on.

8.20.7 Slippery-Runway Landing. The approach and braking technique for a slippery-runway landing is the same as for a minimum-roll landing. On a slippery runway (wet or icy), braking effectiveness varies greatly. The conditions of the runway shall be determined by ground personnel and the pilot advised accordingly. A combination of rudder, ailerons, brakes, and nosewheel steering should be used for directional control because the brakes and nosewheel steering are relatively ineffective.

CAUTION

Be alert to antiskid failure when landing on slippery runways. After approximately 3 seconds of continuous brake release, complete control of braking is returned to the pilot. Subsequent return to full braking could result in blown tires and loss of directional control.

8.20.8 Landing in Turbulence or Jet Wash. For landing in turbulence, the approach speed should be increased as needed to provide additional control margin. With a reported wind of 10 knots and gusts of 20 knots, a gust factor of 10 knots is assumed. Add one-half the gust factor to the approach speed for the aircraft landing weight.

8.20.9 Crosswind Landing. In addition to the procedures used for a normal landing, the following should be accomplished. Before landing, refer to the crosswind chart in Part XI for the headwind component in order to compute the landing distance, and for the crosswind component to determine the minimum speed for nosewheel touchdown. On the final approach, crab or drop a wing to keep lined up with the runway. However, if crabbing, the aircraft shall be aligned with the runway just before touchdown. When the direct crosswind component exceeds 30 knots, a no-flap landing should be made. At touchdown, lower the nosewheels to the runway as soon as possible. After the nosewheels are on the runway, the aircraft will assume greater directional stability, particularly if kept on the upwind side of the runway crown. Because of the close inboard location of the engines, practically no advantage is gained by using asymmetrical power settings during crosswind landings.

CAUTION

Under 90° crosswind conditions in excess of 20 knots, directional control during the landing rollout cannot be maintained with rudder alone at airspeeds of 70 KIAS or less. A combination of ailerons, rudder, and nosewheel steering will be required to maintain directional control.

8.20.10 Touch-and-Go Landing. Touch-and-go landing may be practiced, leaving the landing gear down, using the procedures outlined for a normal landing followed by a waveoff. To ensure good engine acceleration, it is desirable to maintain throttle settings at a minimum of 50-percent rpm during touch-and-go landing practice. After touchdown, proceed as follows:

CAUTION

Do not use thrust reversers during touch-and-go landing.

1. Wing flap handle — UP.
2. Speedbrake switch — IN.
3. Trim — As Required.
4. Throttles — As Required.

CAUTION

Rapid throttle advances during touch-and-go landings and waveoffs may produce undesirable yaw as a result of uneven engine acceleration. A smooth, positive throttle movement will reduce this condition to a minimum. Ninety-six-percent rpm is recommended as a maximum for touch-and-go landing.

8.20.11 Waveoff. See Figure 8-7 for complete procedure for making a waveoff.

8.21 AFTER CLEARING RUNWAY

1. ATC transponder — OFF.
2. Speedbrake — IN.
3. Takeoff trim — Reset.

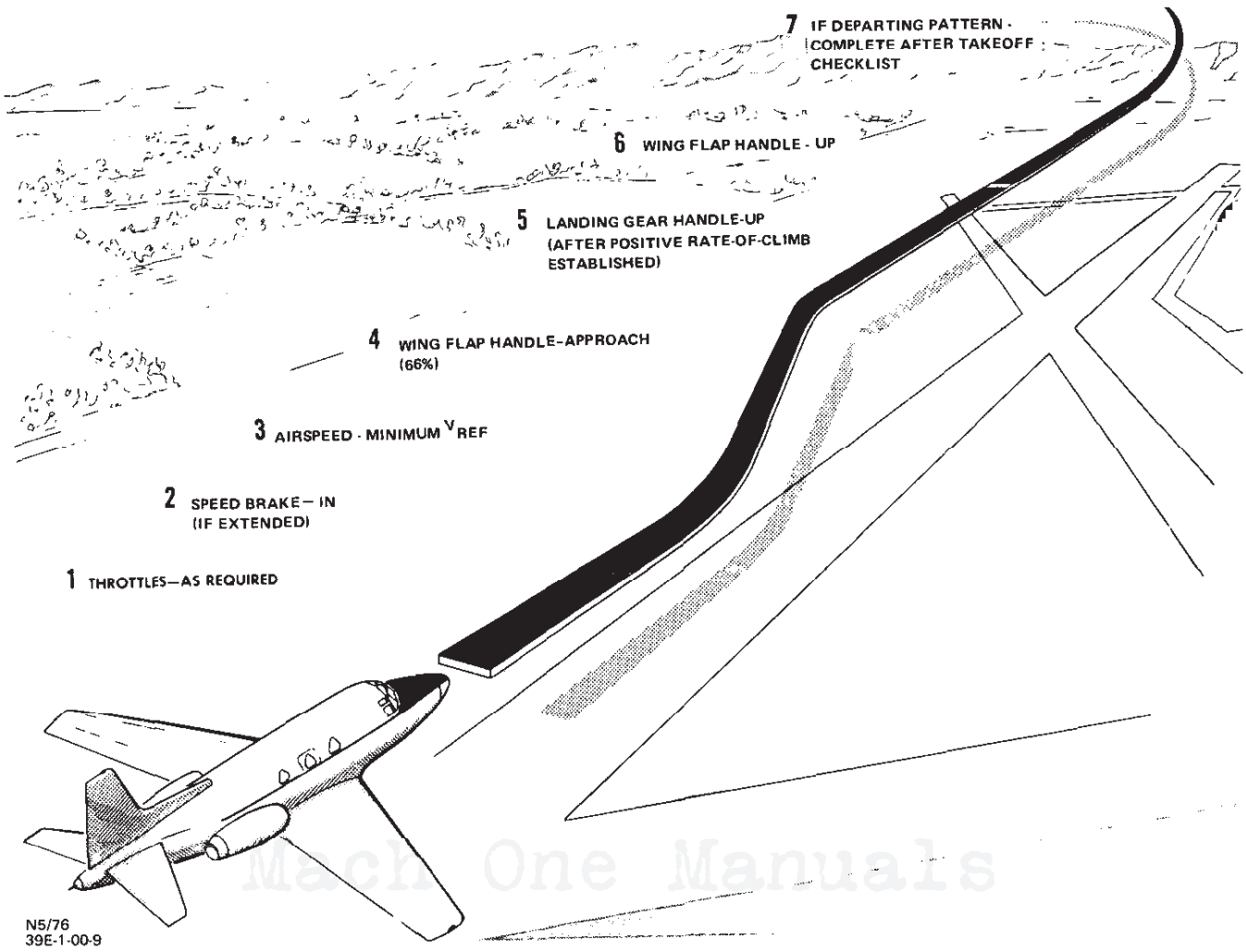

Figure 8-7. Waveoff

4. Radios/radar/IFF/autopilot — As required.
5. Windshield anti-icing switches — OFF.
6. Anti-icing control panel switches — OFF.
7. Strobe lights — OFF.

Note

If the engines have been operated at 85-percent rpm or less for 5 minutes, it is permissible to shut down one engine during taxiing, to reduce braking required, and to extend brake and tire usage.

8. Oxygen supply lever — As Required.
9. Standby attitude indicator switch — OFF.
10. Ignition — NORMAL.
11. Landing/taxi lights — As required.

8.22 ENGINE SHUTDOWN

The engine should be operated for about 5 minutes at reduced thrust (85-percent rpm or below) before shutdown to stabilize engine temperatures. Operation during approach and taxi can be considered as reduced thrust time. At the parking area, proceed as follows:

1. Landing and taxi light switches — RETRACT and OFF.
2. Strobe lights — OFF.
3. Parking brake set or chocks in place.
4. Inverter and radios — OFF.

CAUTION

Do not shut down the engine by use of the engine fire pull T-handle or engine master switch because the engine-driven fuel pump is lubricated by fuel and can be damaged by operation without fuel.

5. Throttles — OFF.

Note

Check that the engines decelerate freely, and listen for any unusual engine noises during shutdown.

6. Nosewheel steering — Disengage.

7. Cabin air selector switch — OFF.

8. Gustlock handle — Engage.

9. Cockpit light switches — OFF.

10. External lights — OFF.

11. Engine master switches — OFF.

12. Emergency light switch — Disarmed.

13. Radios, batteries, and generators — OFF.

14. Parking brake — Released (when chocked).

Note

Refuel the aircraft after landing (if possible) to prevent an out-of-balance fuel condition if the aircraft is parked overnight.

8.23 STRANGE-FIELD PROCEDURE

If the mission requires a landing at a strange field where ground personnel are not completely familiar with the aircraft or where ground personnel are not available, the pilot is responsible for the condition of the aircraft.

8.24 BATTERY CONDITION CHECK

Before engine shutdown, the batteries may be checked as follows: Turn off the left dc generator and observe the reading of the right dc loadmeter. Move the battery switch to OFF and note the decrease in the right dc loadmeter reading. If the decrease is excessive (more than one-third of the distance between increments), record the amount of decrease in the flight record and have ground maintenance performed.

8.25 NIGHT FLYING

There are no specific techniques for flying this aircraft at night that differ from those for day flight.

Note

The ice check lights are very discernible at night and may be used to supplement the anticollision, strobe, and position lights.

8.25.1 External Light Management. During night operation, the external lights should be positioned for each phase as follows:

1. On the line, before engine start: Position lights — ON.

2. When ready for taxi: Taxi lights — As Required.

3. Anticollision and strobe lights — ON (prior to taking runway).

8.25.2 Taxiing. Taxiing at night demands extra caution. Taxi slowly and use the taxi light if necessary to ensure adequate aircraft clearance. The ice check lights may be used to supplement the taxi lights.

8.25.3 Formation Flight. There are no provisions for dimming exterior lights; therefore, formation flight is not advised under normal conditions.

8.26 TURNAROUND

1. Parking brake/chocks — Set/Removed.

2. Circuit breakers — IN.

3. Seatbelt/no smoking sign — ON.

4. Electrical master/antiskid — ON.

5. Hydraulic pump/auxiliary switch — AUTO/OFF.

6. Landing/taxi lights — Retracted/Off.

7. Anticollision/external lights — Set.

8. Oxygen supply levers — ON.

9. Batteries — ON.
10. Generators — OFF.
11. Inverter switch — Check in #1/main position.
12. Essential bus circuit breaker — TEST.
13. Engines — START.
14. Battery/generator switches — Check/ON.
15. Radios/radar/IFF/autopilot — Set.
16. Emergency gyro switch — ON.
17. Fuel quantity — Balanced/Check.
18. Pilot/copilot RMI— Check/SLAVE.
19. Entry door — Closed/Locked/Light Out.
20. Gustlock/flight controls — OFF/Check.
21. Flaps/speedbrake — UP/IN.
22. Caution and emergency lights — Check/ ARMED.
23. Nosewheel steering — ENGAGED.
24. Altimeter/clock — Set.
25. Flight instrument — Check.
26. Prior to takeoff, complete Before Takeoff and Lineup Checklists.

CHAPTER 9

Functional Checkflight Procedures

9.1 GENERAL

9.1.1 Checkpilots. The most important factor in obtaining good checkflights on the aircraft is to pick experienced, conscientious checkpilots. Commanding officers will designate, in writing, those pilots within their command who are currently eligible to perform this duty.

9.1.2 Checkflights and Forms. Checkflights will be performed when directed by and in accordance with OPNAVINST 4790.2 and the directions of NAVAIRSYSCOM type commanders or other appropriate authority. FCF requirements and applicable minimums are described below. FCF checklists are promulgated separately.

9.2 REQUIREMENTS

9.2.1 Conditions Requiring Functional Checkflights. Checkflights are required under the following conditions (after the necessary ground check and prior to release of the aircraft for operational use):

A. Upon receipt of an aircraft returned from SDLM and when a reporting custodian accepts a newly assigned aircraft (all checkflight items required are prefixed A).

B. After the installation of an engine or engine fuel control (minimum required are prefixed B).

C. When fixed or movable flight surfaces or flight control system components have been installed or reinstalled, adjusted, or rerigged and improper adjustment or replacement of such components could cause an unsafe operating condition (minimum required are prefixed C).

Note

The requirements for a checkflight under circumstances other than those specified in the preceding is a determination to be made by the commanding officer. Such a determination will be based on the scope of the work accomplished and its effect on safety or reliability of operation and shall be performed for all items that cannot be checked in ground operation.

9.3 PROCEDURES

9.3.1 Functional Checkflight. The following items provide a detailed description of the functional checks, sequenced in the order in which they should be performed. In order to complete the required checks in the most efficient and logical order, a flight profile has been established for each checkflight condition and identified by the letter corresponding to the purpose for which the checkflight is being flown (i.e., A through C above). The applicable letter identifying the profile prefixes each check both in the following text and in the functional checkflight checklist. Checkflight personnel will familiarize themselves with these requirements prior to the flight. NATOPS procedures will apply during the entire checkflight unless specific deviation is required by the functional check to record data or ensure proper operation within the approved aircraft envelope. A daily inspection is required prior to the checkflight.

PROFILE

9.3.2 Pretakeoff

A 1. Brakes — Check Each With Nose Steering OFF in Normal and Emergency Operation.

A 2. Nosewheel steering.

a. Ensure that nosegear steering system can be operated from both the pilot and the copilot control switches.

b. Place the auxiliary hydraulic system switch on the hydraulic control panel to ON. Depress nosegear steering control button and check that needle on auxiliary hydraulic system pressure gauge fluctuates during steering. Green nosegear steering-on lights should be illuminated. Return auxiliary hydraulic system switch to OFF.

c. Place nosegear steering selector switch to STANDBY position. Check that main steering system fail caution light illuminates, nosegear steering is operable, and green nosegear steering-on lights are illuminated.

d. Return nosegear steering system selector switch to main position. Ensure that main steering system fail caution light goes out. Check each brake pedal for proper actuation with nosegear steering system disengaged. Reengage nosegear steering.

A 3. NAV system test.

a. Check pilot/copilot RMI.

b. Test tacan.

c. Test VOR/ILS.

d. Omega test, initialize, and enter flight plan.

e. Place radio altimeter switch on instrument panel shroud ON. Allow 90- to 180-seconds warmup, then indicator OFF flag should disappear and altitude pointer should indicate 0 to 10 feet.

AB 4. Engine runup.

a. Check for rapid rise in EPR between 79- and 81-percent rpm, indicating bleed strap operation.

b. Compute EPR.

c. From stabilized idle revolutions per minute, move throttles to takeoff thrust in one second or less. Acceleration time should not exceed 8 seconds for each engine. Maximum acceleration time differential between engines is 4 seconds.

Note

An overtemperature condition exists if EGT stabilizes at any temperature above 718 °C.

PROFILE

d. Maximum EGT limit at maximum EPR is 718 °C during the 2 minutes following acceleration, but should stabilize between 565 and 650 °C.

e. Engine revolutions per minute should be approximately 100 percent with maximum of 104.2 percent at maximum EPR.

Note

If revolutions per minute exceeds 104.2 percent, shut down engine and record an overspeed condition.

f. Fuel flow should be 2,500 to 3,300 pph.

g. Oil pressure should be 40 to 50 psi. Oil pressure fluctuation should not exceed 5 psi. Any sudden increase of 10 psi or more is cause for investigation.

h. Oil temperature within normal limits (-40 to +120 °C).

i. Check engine controls for freedom of movement through full range of travel. Throttles aligned within one-half inch of each other.

j. Check hydraulic system failure (audible warning).

k. Record any abnormal instrument readings.

9.3.3 Takeoff/Climb (FL 300: Profile A, B; 20,000 Feet: Profile C)

AB 5. EPR.

Adjust throttles to obtain maximum continuous thrust indication on EPR indicator as computed from existing runway temperature and pressure altitude conditions.

AB 6. Check altitude where throttles must be retarded to avoid exceeding maximum EPR.

AB 7. EGT — Should Not Exceed 718 °C Maximum During Takeoff, 655 °C Maximum After Setting Maximum Continuous/Climb Thrust.

AB 8. Rpm — Should Be Approximately 100 Percent; Must Not Exceed 104.2 Percent.

AB 9. Oil pressure — Should Be Between 40 and 50 PSI With Fluctuations Not More Than 5 Psi.

AB 10. Lights and vibration — Monitor All Engine Caution Lights and Note Any Excessive Engine Vibration Level.

A 11. Encoding/alerting altitude system — Check Response in Normal and Standby.

A 12. Crew compartment temperature.

PROFILE

a. With CABIN TEMP switch at AUTO and AUTO CONT rheostat rotated to COLD position, place CABIN AIR switch in turn to RH ENG, OFF, LH ENG, OFF, and BOTH ENG. Cool air should emit from the cabin and crew compartment air outlets whenever the switch is positioned at RH ENG, LH ENG, and BOTH ENG. There should be no airflow while the switch is in the OFF position.

b. With CABIN AIR switch at BOTH ENG, rotate AUTO CONT rheostat between COLD and HOT. The cabin air supply temperature should change accordingly.

c. Position CABIN TEMP switch to MAN HOT; within 30 seconds, the temperature of the cabin air supply should increase. Return CABIN TEMP switch to AUTO; temperature of the cabin air supply should return to the preset comfortable level.

d. Position CABIN TEMP switch to MAN COLD; within 30 seconds, the temperature of the cabin air supply should decrease. Return CABIN TEMP switch to AUTO. The temperature of the cabin air supply should return to the preset comfortable level.

AB

13. Anti-icing system.

a. Place ENGINE INLET switch at ON. Note a decrease of about 0.1 in EPR on each engine. Return switch to OFF.

b. Place RAM AIR INLET switch at ON. Note increased load on dc loadmeter. Return switch to OFF.

Note

The fuel heater is limited to 1 minute of operation out of every 30 minutes to prevent eventual coking of spray nozzles.

c. Place FUEL HEATER switch to ON. Note that RH FUEL HEAT ON and LH FUEL HEAT ON caution lights come on and check for an EPR drop on both RH and LH EPR gauges. Return switch to OFF. Caution lights should go off.

d. Windshield anti-icing and antifogging effectiveness should be noted during all flights. Place FWD WINDSHIELD switch at ON. Place both FWD WINDSHIELD GEN switches to ON (CT-39E). Check windshield heating by feeling the inside of the windshield panels. They should be slightly warm to the touch.

Note

Operation of the windshield heating system should not result in the WINDSH IELD HOT caution light coming on. The windshields should be kept free of outside ice or inside fogging. However, under high ambient temperature conditions near sea level and/or solar heating of the windshield panels, the WINDSHIELD HOT caution light may be on because of the 43 to 49 °C (110 to 120 °F) setting of the overheat thermoswitches. On CT-39E aircraft, if it is desirable to cycle the system and either of the ac generators drops off the line for other than ac generator overheat condition, leave switch at OFF for at least 5 minutes.

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PROFILE

A

14. Flight director/autopilot operation.

a. Press compass and vertical gyro selector switches to select No. 1 or No. 2. The respective indicator light should come on.

b. Position autopilot mode selector switch at GYRO or HEADING. All warning flags should be out of view.

c. Flight director.

(1) Set heading marker on course indicator to takeoff runway heading.

(2) Set course arrow on course indicator to desired intercept radial.

(3) On CT-39E aircraft, adjust pitch command control knob on flight director indicator to set steering bars for desired climb angle.

(4) On CT-39E aircraft, bank aircraft (maximum 35°) to align aircraft symbol with steering bars and engage autopilot.

(5) On CT-39G aircraft 160053 through 160057, manually position aircraft to desired attitude, then depress PITCH SYNC button on control wheel. Aircraft symbol should center with steering bars and aircraft maintains attitude.

d. Check VOR navigation.

(1) Press NAV portion of course selector switch on instrument panel shroud. The NAV light should come on.

(2) Tune VOR/ILS receiver for desired frequency. Check station identification and RMI No. 1 pointer.

(3) Position mode selector switch to NAV/LOC. The HDG/V/L or NL-ARM light should come on.

(4) Set course arrow on course indicator to selected VOR radial. The course pointer should reflect the same magnetic course.

(5) Set course indicator heading marker to intercept selected radial. The course arrow on the course indicator should reflect same course as No. 1 pointer on RMI.

(6) Establish intercept heading (less than 85° from takeoff runway heading). Lateral deviation bar should show deviation from course radial.

(7) When desired altitude is reached, select altitude hold mode. The ALT (CT-39E aircraft) or F/S ALT or ALT-HLD (CT-39G aircraft) light should come on.

(8) Check aircraft symbol-steering bars on flight director indicator and check that course indicator TO arrowhead is in view.

PROFILE

e. Check tacan navigation.

(1) Press TAC portion of course selector switch on instrument panel shroud. The TAC light should come on.

(2) Tune tacan receiver for desired frequency and check station identification on RMI No. 2 pointer.

(3) Check radial intercept in same manner as for VOR check in steps g.(3) through g.(8).

(4) Check range from tacan station.

f. Check ADF navigation.

(1) Set mixer switches as desired.

(2) Set function knob at ANT.

(3) Select desired frequency band and tune frequency. Check maximum deflection of tuning meter pointer.

(4) RMI single barred knob, double barred knob, or both — ADF.

(5) Function knob — ADF Check Station Bearing on RMI.

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g. Omega system test.

(1) Select NAV page 1.

(2) Check bearing and distance to station and crosscheck with tacan and VOR (a known position).

(3) Check groundspeed and true airspeed.

(4) Dial in the OMEGA COURSE to the waypoint (NAVAID), select VLF, and track inbound to the waypoint. Ensure CDI centers and crosscheck with VOR/TAC.

A

15. ATC transponder system.

a. Position function selector switch at ON. Allow 80 seconds for warmup.

b. Position mode selector as required.

c. Position code selector switches as required.

d. Position altitude reporting switch to ALT RPTG (CT-39E, CT-39G aircraft 158843) 158844, and 159361 through 159365).

PROFILE

e. Establish communications with base tower and request an identification check. Check that reply light comes on and that signals are satisfactory.

9.3.4 Level FL 300

AB 16. Engine operation.

Trim in level flight, clean configuration and adjust throttle to stabilize speed at 0.74 to 0.75 IMN. Check all flight, electrical, and engine instrument indications after 3 minutes of stabilization. Record any data out of limits.

AB 17. Engine flameout characteristics.

With aircraft flying at 180 KIAS, engines should not flame out during normal throttle adjustment made on both engines separately, nor during normal throttle adjustments with reversals at various revolutions per minute covering range from takeoff thrust to idle, performed on both engines separately.

AB 18. Acceleration time.

From stabilized idle revolutions per minute, move throttles to takeoff thrust in 1 second or less. Acceleration time should be between 3 to 12 seconds for each engine above 150 KIAS.

9.3.5 Level FL 250

A 19. Throttles — Set Throttles to 88-Percent Rpm.

A 20. Compartment pressurization.

a. Set cabin pressure controller at 8,000 feet.

b. Ensure all crewmembers are on oxygen.

c. Check the cabin altitude is 8,000 ±240 feet, then position cabin air selector to off. Cabin altitude should not exceed 23,200 feet in 3 minutes for CT-39E aircraft or 22,100 feet in 3 minutes for CT-39G aircraft.

Note

These figures are based on a leakage rate limit of 80 and 144 cfm respectively.

(1) Check pressure warning light and FASTEN SEAT BELT lights go on passing 10,000 feet cabin altitude.

(2) Check that cabin oxygen mask doors open or unlatch. USE OXYGEN MASK and NO SMOKING placards illuminate, cabin lights come on, red oxygen indicator lights come on, and oxygen warning horn sounds when altitude reaches 13,500 ±500 feet.

(3) Depress oxygen horn cutout button and check that horn is silenced.

PROFILE

(4) Pull valve pins at each passenger oxygen mask (do not pull down on mask) and check for oxygen flow.

(5) Insert valve pins and check that flow stops.

(6) Place oxygen switch at OPEN. Oxygen warning horn should sound.

(7) Depress cutout button to silence horn.

(8) Pull valve pin at one passenger mask and check for oxygen flow.

(9) Place oxygen switch at CLOSE. Check that oxygen flow at mask stops. All placards and indicator lights go off.

(10) Place oxygen switch at AUTO. Check that all lights and placards come on and warning horn sounds.

(11) Depress horn cutout button to silence horn.

(12) Select emergency pressurization, cabin pressurizes to 15,000 feet or below.

(13) Reduce engines to idle and select pressure OFF, cabin depressurizes to 25,000 feet.

(14) Engines at 88-percent rpm, set cabin pressure controller rate at 2,000 and position cabin air selector at BOTH ENG. Cabin altitude should descend rapidly to 13,500 ±1,500 feet, hold for about 4.5 to 6 minutes, then rate toward 8,000 feet. Cabin altitude should reach 8,000 ±240 feet in about 8 minutes total elapsed time. As the cabin altitude passes 13,500 feet, oxygen flow should stop and all placards and warning lights should go off.

9.3.6 Level 20,000 Feet

A C 21. Autopilot — ENGAGED; Check That Flight Controls Respond Properly to Autopilot Outputs.

A C 22. Aircraft trim change with thrust.

a. At maximum airspeed, check that flight controls respond properly.

b. Glide at idle power with the ball in the pilot attitude indicator trimmed to center.

c. In level flight, advance throttle to takeoff thrust. There should be no undesirable change in aircraft trim.

A 23. Test of fuel system.

a. Place fuel X-feed and tank selector switch to LH tank. Check that both engines use fuel from left-hand tank. Make same check on right wing. This is to test operation of relays, valves, and switches.

PROFILE

b. On CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365, test fuel jettison system for operation on both manual and automatic.

c. Balance wing fuel. Place X-feed switch to X-FEED, secure one engine, operate on single engine for about 10 minutes. Wing fuel should be approximately balanced.

AB

24. Engine secure/airstart (B profile: secure the engine requiring the B profile) — Check that airspeed is within airstart envelope. Check that fire-pull handle is in and position engine master switch to ON.

CAUTION

The ignition duty cycle is limited to 2 minutes on, 3 minutes off, 2 minutes on, 23 minutes off.

a. Place airstart switch to ON. Light-off should occur within 20 seconds. If engine revolutions per minute is below normal, the starter ignition button may be used to accelerate the start.

b. After light-off, check EGT rise and revolutions per minute increase to idle, then adjust throttle as required. Place fuel X-feed and tank selector switch to NORM and airstart switch to OFF.

9.3.7 Level 15,000 Feet

A C

25. Wing flaps/slats.

Fly 140 KIAS; select approach, then full down (note noncancellable gear warning), then retract flaps (retract 5 to 10 seconds); select full down (10 to 15 seconds), extend gear, slow to 120 KIAS, check both slats fully extended. Increase airspeed to 180 KIAS; ensure no undesirable flight characteristics because of asymmetric extension. Check slats retracted fully, raise gear and flaps, and check flaps and gear retracted fully. Continue with step 26.

A C

26. Simulated landing.

As published in current NATOPS.

A

27. Landing gear emergency extension — Maintain Aircraft Speed at 120 KIAS.

a. With landing gear handle in the UP position, pull landing gear emergency release T-handle. The force required to actuate the release mechanism should not exceed 35 pounds. The nosegear should extend in 20 seconds, and the main gear extension should not exceed 30 seconds after performing a one-half ball, wings level yaw at 125 KIAS for the respective gear.

b. Place landing gear handle in the down position and return the T-handle to stowed position. Depress landing gear electric reset button. Place gear handle in the UP position for a normal retraction of the landing gear.

PROFILE

A C

28. Stall warning system — At an Altitude of 15,000 Feet, with IGNITION Switch at NORMAL and Flaps and Landing Gear Up, Trim Aircraft for Level Flight at 120 KIAS.

    a. Record total fuel on board (to determine aircraft weight).

    b. Perform a 1g stall and record speed at which vibrator operates and Vs. Vibrator should generate at or above 107 percent of aircraft Vs as determined by aircraft weight. The red ignition lights in the engine starter buttons should come on and remain on as long as the vibrators are operating, indicating continuous ignition operation.

9.3.8 Descent

A

29. Landing gear warning signal.

    a. Descend through 12,000 ±600 feet at partial throttle with gear up and airspeed less than 140 ±10 KIAS. The landing gear warning signal should sound at 12,000 ±600 feet.

    b. Ensure that signal can be silenced by placing both throttles above 95-percent rpm.

    c. Reduce throttle and cycle gear, then silence gear warning signal by using warning horn cutout button. During gear cycle, note flashing light in landing gear handle.

    d. Check that warning signal tone level in headset is not controlled by the audio system volume control.

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9.3.9 Landing

A

30. ILS approach.

    a. Press NAV portion of course selector switch on instrument panel shroud. The NAV light should come on.

    b. Position autopilot mode selector to NAV/LOC and then APPR. The GS ARM light should come on.

    c. Set course indicator heading marker to missed approach heading.

    d. Set localizer frequency and check station identification.

    e. Center steering bars and aircraft symbol on flight director indicator. With autopilot engaged, the aircraft should automatically fly the proper glidepath. Lateral deviation bar and deviation pointer should be centered. When the glideslope is captured, check that altitude hold switch is off and ALT light (CT-39E aircraft) or F/S ALT light or ALT-HLD (CT-39G aircraft) and GS ARM light are off. The GS ENG light should be on.

    f. Set radio altimeter minimum decision altitude knob as desired.

    g. Check glideslope pointers for maximum deviation of one dot either side of center.

PROFILE

h. Monitor indicators for correct course and glideslope.

i. Monitor beacon lights to maintain altitude at checkpoints. The GS EXT light should come on with the M light.

j. Radio altimeter — From 10 to 150 feet Altitude Over Flat Terrain, Ensure That Needle Excursion Is Not Excessive.

k. Check that radio altimeter MDA or DH light comes on when preset minimum decision altitude is reached.

l. Press autopilot disengage for manual control during touchdown.

A

31. Marker beacon receiver.

a. Check that BCN monitor switch on pilot transceiver selector panel in ON.

b. Fly aircraft over an airways marker beacon. The 3,000-cycle aural tone should be audible before light comes on, letter "A" appears in white. The length of time the light will stay on is determined by the altitude and speed of the aircraft. In general, the light should stay on for a minimum of 1 mile when flying at 170 KIAS at 1,500-foot altitude.

c. Fly aircraft at normal approach altitudes along glideslope localizer course. The O light comes on (letter "O" appears in blue as approach is being started at interception of glideslope course). The 400-cycle aural tone should be heard throughout the audio system at the same time.

d. While flying on ILS approach, note the light comes on, letter "M" appears in amber when the aircraft is over inner marker. The 1,300-cycle aural tone should be heard through the audio system before the lights come on. Note that marker beacon light flashes approximately twice as aircraft passes over boundary marker.

e. Note that MARKER sensitivity switch attenuates aural tone and reduces cone of visual marker beacon acquisition.

A

32. Antiskid brake system.

a. Note operation of antiskid brake system during landing roll.

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PART IV

Flight Characteristics

Chapter 10 — Flight Characteristics

Chapter 11 — Flight Test Procedures

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CHAPTER 10

Flight Characteristics

10.1 GENERAL

Light to moderate control force is adequate for maneuvering the aircraft throughout the designed speed and altitude range. Stability and general flight characteristics are good within the operating limits of the aircraft.

10.2 STALLS

The stall characteristics of the aircraft are safe, and there is a well-defined stall indication. Some light to moderate airframe buffet is noticeable during the approach to the stall, but does not increase appreciably at the stall. A mild rolling tendency usually will be observed at stall and possibly some stick force lightening. Light wing flap buffet is also present with the flaps extended, but becomes negligible above the stall.

The aircraft can be controlled below the stall to a minimum speed, which in most conditions is limited by available longitudinal control.

The aircraft is elevator limited at stall with forward cg, landing gear and flaps extended, and idle thrust. The full stall cannot be obtained in this configuration, and the aircraft oscillates mildly in pitch if full elevator is maintained. The oscillation stops as soon as back pressure on the longitudinal control is released. Stall speeds for varying bank angles, gear and flaps up or down, and power off and on are shown in Figure 10-1.

CAUTION

Stalls or approaches to stalls shall not be conducted when any fuel imbalance exists.

10.2.1 Stall Warning (Stick Shaker). The stick shaker installation provides a warning as follows:

1. Altitude below 16,000 feet — Warning at or above 107 percent of stall speed.
2. Altitude above 42,000 feet — Warning at or above 110 percent of that speed that may cause an engine to flame out.
3. Altitude between 16,000 and 42,000 feet — Warning at or above 107 percent of stall speed, 16,000 feet to 110 percent of that speed that may cause an engine to flame out at 42,000 feet.

Note

- At all altitudes the warning signal is not less than 107 percent of stall speed and 110 percent of that speed that may cause an engine to flame out.
- With the ignition switch in the NORMAL position, the continuous ignition system is automatically energized when the stall warning system is actuated.

10.2.2 Accelerated Stalls. Accelerated stall characteristics are essentially like those for unaccelerated stalls. However, buffet warning begins considerably above stall speed, but does not increase significantly until just before stall.

10.3 STALL RECOVERY

1. Simultaneously lower nose and add power, as required.
2. Flaps to approach (if full down).
3. Gently raise nose to stop rate of descent.
4. As positive rate of climb is noted, raise gear handle.
5. Flaps up for climbout.

DATA AS OF: 6 NOVEMBER 1962
BASED ON: NAA FLIGHT TEST DATA

CT-39E

POWER OFF

GROSS WEIGHT—POUNDS	FLAPS UP				FLAPS 66% DOWN				FLAPS 100% DOWN			
ANGLE OF BANK—DEG	0	30	45	60	0	30	45	60	0	30	45	60
LOAD FACTOR—G	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0
10,000	79	85	94	112	76	81	90	107	74	79	87	104
12,000	87	93	103	123	83	89	99	117	81	87	96	114
14,000	94	101	111	132	89	96	106	126	87	94	104	123
16,000	100	108	119	141	96	103	114	135	93	100	111	132
18,000	106	114	126	150	101	109	121	143	99	106	117	140

POWER ON

GROSS WEIGHT—POUNDS	FLAPS UP				FLAPS 66% DOWN				FLAPS 100% DOWN			
ANGLE OF BANK—DEG	0	30	45	60	0	30	45	60	0	30	45	60
LOAD FACTOR—G	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0
10,000	74	80	88	105	72	77	85	101	70	75	83	99
12,000	81	87	97	115	79	84	93	111	76	82	91	108
14,000	88	95	105	124	85	91	101	120	83	89	98	117
16,000	94	101	112	133	91	97	108	128	88	95	105	125
18,000	100	107	118	141	96	103	114	136	94	101	111	132

Note
Landing gear position does not affect stall speeds.

39E 1-93-1A

Figure 10-1. Stall Speeds — KIAS (Sheet 1 of 3)

CT-39G AIRCRAFT 158843 AND 158844

DATA AS OF: 28 APRIL 1967
BASED ON: NR FLIGHT TEST DATA

POWER OFF

GROSS WEIGHT – LBS	ANGLE OF BANK-DEG	FLAPS UP				FLAPS 66% DOWN				FLAPS 100% DOWN			
		0	30	45	60	0	30	45	60	0	30	45	60
	LOAD FACTOR-G	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0
	12,000	84	92	102	121	77	85	97	116	75	82	94	113
	14,000	91	99	110	131	83	92	104	125	80	89	101	122
	16,000	98	106	118	140	88	98	111	134	86	95	108	131
	18,000	104	112	125	148	94	104	118	142	91	101	115	139
	20,000	109	118	132	156	100	110	125	150	97	106	121	146

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POWER ON

GROSS WEIGHT – LBS	ANGLE OF BANK-DEG	FLAPS UP				FLAPS 66% DOWN				FLAPS 100% DOWN			
		0	30	45	60	0	30	45	60	0	30	45	60
	LOAD FACTOR-G	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0
	12,000	82	89	100	118	74	82	94	113	71	79	92	110
	14,000	89	96	107	127	80	89	102	122	77	86	99	119
	16,000	95	103	115	136	86	95	109	131	83	92	105	127
	18,000	101	109	122	145	91	101	115	139	88	97	111	135
	20,000	106	115	128	152	97	107	121	147	93	103	117	142

Note

Landing gear position does not affect stall speeds

N3/74
39E-1-93-78

Figure 10-1. Stall Speeds — KIAS (Sheet 2 of 3)

DATA AS OF: 11 SEPT 1973
BASED ON: RI FLIGHT TEST DATA

CT-39G AIRCRAFT 159362 THROUGH
159365 AND 160053 THROUGH 160057

POWER OFF

GROSS WEIGHT — LBS	FLAPS UP				FLAPS 66% DOWN				FLAPS 100% DOWN			
ANGLE OF BANK-DEG	0	30	45	60	0	30	45	60	0	30	45	60
LOAD FACTOR-G	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0
12,000	85	91	101	119	81	87	96	114	79	85	94	111
14,000	92	98	109	129	88	94	104	123	86	92	101	120
16,000	98	105	116	138	94	100	111	131	92	98	108	128
18,000	104	112	123	146	99	107	118	139	97	104	115	137
20,000	110	118	130	154	105	112	124	147	103	110	121	144

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POWER ON

GROSS WEIGHT — LBS	FLAPS UP				FLAPS 66% DOWN				FLAPS 100% DOWN			
ANGLE OF BANK-DEG	0	30	45	60	0	30	45	60	0	30	45	60
LOAD FACTOR-G	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0	1.0	1.2	1.4	2.0
12,000	83	89	98	117	79	83	94	111	77	83	91	108
14,000	90	96	106	126	86	92	101	120	83	90	99	117
16,000	96	103	113	135	91	98	108	128	89	96	105	125
18,000	102	109	120	143	97	104	115	136	94	101	112	133
20,000	107	115	127	151	102	110	121	144	100	107	118	140

Note

Landing gear position does not affect stall speeds.

N5/76
39E-1-93-85A

Figure 10-1. Stall Speeds — KIAS (Sheet 3 of 3)

CAUTION

If landing gear or flaps are extended, do not exceed gear and flap limit speed during stall recovery; otherwise, excessive air loads may cause structural damage.

10.4 ENGINE FLAMEOUT

The engine compressor is sensitive to airflow disturbances from the wing during either the 1g or accelerated stall at altitudes above 25,000 feet. This can result in engine compressor stall and subsequent engine flameout if the condition is permitted to progress. (Refer to paragraph 4.10, HIGH-ALTITUDE FLIGHT LIMITATIONS.)

10.5 SPINS AND RECOVERY

WARNING

Intentional spins in this aircraft are prohibited.

A spin test program is not required for this aircraft. However, wind tunnel tests and analytical studies indicate an optimum procedure for recovering from a spin. If a spin is inadvertently entered, proceed as follows:

1. Apply recovery controls, full rudder against the spin, control wheel and control column neutral, and trim aircraft nose down.
2. Move throttles to IDLE.
3. When rotation stops, neutralize controls and recover from the stalled condition.

WARNING

Do not hold recovery controls after rotation stops; otherwise, the aircraft may enter a spin in the opposite direction.

10.6 FLIGHT CONTROL EFFECTIVENESS

10.6.1 Horizontal Stabilizer and Elevators. Longitudinal control friction is very low, resulting in light elevator breakout force. Although static longitudinal stability is positive, stick forces can become very light in cruise and power approach configurations at aft cg's. Dynamic longitudinal stability is good in all configurations.

Light to moderate trim and control force changes will be noted after prolonged climbs or after prolonged low-speed level flight in an abnormal nose-high attitude (exceeding 8-1/2°) with one-half to three-quarters wing fuel. These changes occur with fuel shifting outboard and aft in the wing tanks during a prolonged nose-high attitude with partial fuel, resulting in a moderate aft cg change. (Refer to paragraph 2.14.18, Fuel Availability During Flight.)

Longitudinal trim changes induced by moderate changes in thrust are easily controllable and forces are light. Changes in longitudinal trim caused by raising the gear are negligible. When raising the flaps, changes are moderate. When the aircraft is trimmed for the approach configuration of gear and flaps down, a large trim change occurs if the gear or flaps are raised and the throttles are advanced to TAKEOFF THRUST. Stick forces are high in order to counteract the trim change. In order to avoid encountering the high stick forces, thrust and configuration changes should not be made simultaneously and the aircraft should be trimmed continuously throughout the thrust and configuration changes to keep stick forces low.

The elevators provide excellent pitch control throughout the speed and altitude range of the aircraft.

10.6.1.1 Horizontal Stabilizer Trim. The horizontal stabilizer trim system is adequate throughout the speed and altitude range of the aircraft. However, it is possible to stall the stabilizer trim motors if an attempt is made to trim out a push force in excess of approximately 50 pounds at high airspeeds. Reducing airspeed or reducing the push force while actuating trim eliminates the stalled condition of the trim motors. When the landing gear is retracted, trimming is accomplished at a low-speed rate; with the gear extended, the trimming rate is high.

10.6.2 Ailerons. The ailerons provide adequate lateral control in cruise and high-speed flight. Lateral control also is adequate in the approach and landing configurations. The ailerons are effective in steering the aircraft on the ground at speeds down to about 50 KIAS. The landing gear geometry is such that the change in lift between the wing panels when aileron is applied causes the aircraft to tilt and turn toward the down wing (application to left aileron causes the right wing to rise, depressing the left main gear strut and causing the aircraft to tend to turn left). At speeds above about 50

KIAS during takeoff run or landing roll, the ailerons should be used to augment directional control rather than asymmetric braking or asymmetric application of thrust.

10.6.2.1 Aileron Trim. The aileron trim is adequate for all normal flight conditions. Any allowable asymmetric fuel condition that develops between the wing fuel tanks can be trimmed out.

10.6.3 Rudder. The rudder effectiveness is adequate under all flight conditions, including single-engine operation. Dihedral effect (roll because of yaw) is positive. Consequently, the rudder is most effective in picking up a low wing at low airspeeds. The rudder is effective into the stall.

10.6.3.1 Rudder Trim. There is sufficient rudder trim available to trim out all yaw at all speeds during two-engine operation. At recommended cruise speed during single-engine operation, the mild yaw because of asymmetric thrust can be trimmed out easily. However, at low airspeeds, a 5° bank into the operating engine assists in maintaining directional control.

10.6.4 Flaps. The wing flaps provide reduced approach and landing speeds. Light buffet is encountered when the flaps are lowered full down.

10.6.5 Speedbrake. The speedbrake can be fully extended at any airspeed and is very effective as a drag device. No buffet occurs when the speedbrake is extended (gear and flaps up). A slight nose-down trim change occurs, and cockpit and cabin noise increases noticeably when the speedbrake is extended.

10.6.6 Slats. The wing leading edge slats improve aircraft stability at high angles of attack, decrease aircraft drag in maneuvering flight, and delay the onset of buffet. They are designed to operate at both low and high Mach numbers. Slat operation is automatic and depends on the airspeed and angle of attack. The slats begin to open at approximately 180 KIAS.

10.7 LEVEL-FLIGHT CHARACTERISTICS

10.7.1 Low Speed. Recommended speeds for takeoff, approach, and landing are given in the appropriate charts in Part XI. At low speeds, flight characteristics are conventional with normal effectiveness and aircraft response. (Refer to paragraph 10.8, LATERAL-DIRECTIONAL CHARACTERISTICS for additional information.)

Note

During low-speed flight when power is changed from idle power to full power, as in a waveoff, there is a noseup tendency. Trim should be used as power is applied to assist in maintaining desired attitude.

10.7.2 Cruise Speed. Longitudinal and lateral control forces are light in the cruise speed region at high altitude and very little pilot effort is required to effect a change in aircraft attitude. Cruise conditions can be maintained quite easily.

10.8 LATERAL-DIRECTIONAL CHARACTERISTICS

The static lateral-directional stability of the aircraft is positive for all airspeeds and configurations. Holding constant neutral rudder and releasing all pilot pressures (neutral control) from the ailerons dampens out yawing oscillations in turbulent or gusty conditions.

10.9 MANEUVERING FLIGHT

Maneuvering flight elevator force gradients are essentially linear with g when the cg is forward. When the cg is aft, the gradient flattens and less change in elevator force is required for a given change in g. Increasing altitude at constant Mach number results in maneuvering forces becoming somewhat lighter.

10.10 DIVES

During maximum-rate descents, stability and control characteristics are good. Control forces are at a comfortable level and no adverse aircraft characteristics prevail.

10.11 THRUST REVERSER CHARACTERISTICS

Thrust reversers are provided as an additional deceleration device. Normal operation of the thrust reversers does not cause any tendency to deviate from the runway centerline. In case of engine failure when single-engine thrust reverser operation is required, small rudder inputs initially and at approximately 60 KIAS nosegear steering will be sufficient to maintain runway centerline.

10.12 ANTISKID CHARACTERISTICS

If brakes are applied until antiskid cycling occurs (maximum performance landing), the aircraft may drift left or right of the runway centerline as airspeed decreases because of asymmetric antiskid cycling. The drift is controllable with nosegear steering, and control becomes relatively easy if the drift is anticipated. When antiskid cycling is noticed, the brakes should be released slightly, then reapplied, attempting to remain below the pressure that results in cycling.

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CHAPTER 11

Flight Test Procedures

11.1 FLIGHT TEST PROCEDURES

Pilots assigned for flight testing shall be designated in writing by the commanding officer. Fuel loading and crew requirements shall be determined by the commanding officer or his designated representative and will take into account safety of flight and the purpose for which test flight is conducted. Flight test procedures shall be in accordance with the current edition of the CT-39E Periodic Maintenance Requirements Manual, NAVAIR 01-60GBE-6, and NAVAIR Instruction 4700.2.

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PART V

Emergency Procedures

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CHAPTER 12

Emergency Procedures Introduction

12.1 INTRODUCTION

The procedures contained in this part are considered the best for coping with the various emergencies that may be encountered during operation of this aircraft. Only single failures are considered; however, each failure presents a different problem. Items indicated with an asterisk (*) are considered memory items and shall be completed without reference to the checklist. A pilot with a thorough knowledge of emergency procedures and their asterisked items will be better able to cope with the problems encountered. Even though the procedures are considered the best possible, sound judgment must be used when multiple emergencies, adverse weather, terrain clearance, etc., are encountered.

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12.2 PILOT RESPONSIBILITY

If an emergency arises, the crew should be kept informed and the nearest ground station informed as to the type of emergency and planned action. If the emergency requires action by the crew, the pilot will notify them of the emergency and action required.

12.3 FIRE EQUIPMENT AND WARNINGS

Each engine pod is equipped with a fire and overheat detector circuit and a pressure-type fire extinguisher system. Two portable hand fire extinguishers for use in case of a cabin fire are located in the aircraft.

WARNING

- Whenever a fire warning light comes on or a cabin fire is suspected, it is recommended that the crew and passengers be alerted, oxygen masks be used, and that the pilot select 100-percent oxygen.
- Do not attempt to restart the engine if the extinguisher has been actuated.
- The extinguisher agent (bromotrifluoromethane) in the engine fire extinguishing system can produce toxic effects if inhaled.
- Use discretion when the second container is to be selected as only two extinguishing agent containers are available.

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CHAPTER 13

Ground Emergencies

13.1 EMERGENCY ENTRANCE AND ESCAPE EXIT

For emergency entrance and escape exits, see Figure 13-1.

13.2 ABNORMAL ENGINE STARTS

During the engine start sequence, if there are indications of a hung/hot start (i.e., excessive fuel flow, rapidly rising EGT on light-off, no oil pressure, etc.) or other abnormal indications, abort the start, proceeding as follows:

*1. Throttle — OFF.

*2. Engine — Windmill (if required).

Allow the engine to windmill for 30 seconds, firmly holding starter button in.

3. If a lineman signals fire or there are cockpit indications of a fire, proceed to ENGINE FIRE ON DECK procedures in paragraph 13.3.

WARNING

Do not attempt to restart engine until the cause of the fire or overheat condition has been determined.

CAUTION

If EGT rises rapidly to 475 °C, anticipate a hot start. Attempting a start with greater than 10 knots of tailwind component increases probability of a hot start.

13.3 ENGINE FIRE ON DECK

If a fire light illuminates or an outside observer reports a POD fire, proceed as follows:

*1. Stop the aircraft.

*2. Throttle (affected engine) — OFF.

*3. Fire pull handle (affected engine) — PULL.

*4. Fire extinguisher switch — EXT NO. 1.

5. Fire extinguisher switch — EXT NO. 2 (if necessary).

6. Notify ground control/tower.

7. Engine master switch (affected engine) — OFF.

8. Shut down other engine.

9. Engine master switch (other engine) — OFF.

Note

Allow at least 2 seconds before completing step 10.

10. Electrical master switch — OFF.

11. Evacuate aircraft — As Required.

13.4 HYDRAULIC FAILURE AUDIBLE WARNING ON TAXI

*1. Auxiliary hydraulic — ON.

*2. Stop aircraft.

3. Investigate problem before further taxi.

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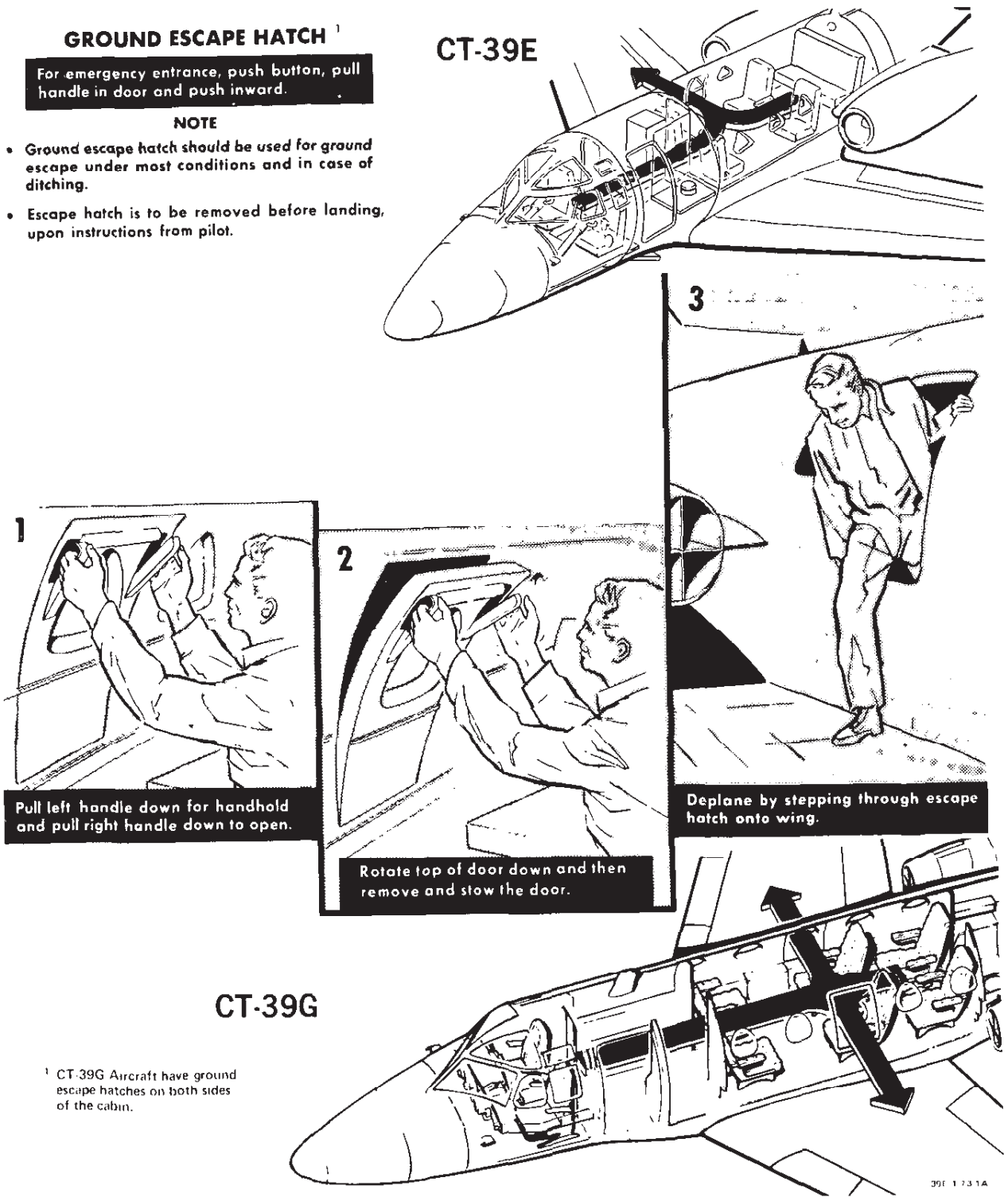
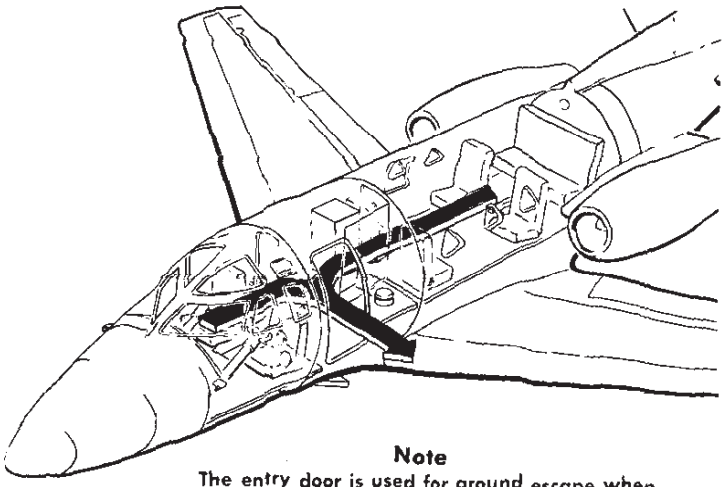


Figure 13-1. Emergency Entrance and Escape Exits (Sheet 1 of 2)

ENTRY DOOR

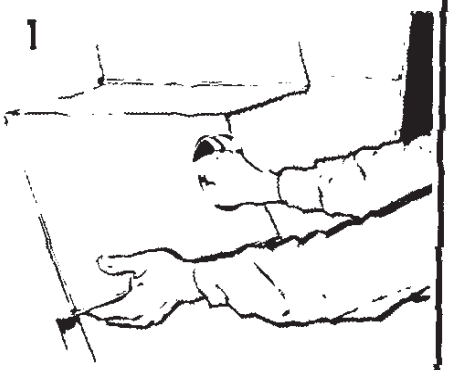
For emergency entrance, turn handle in door and push inward at the lower edge of the door, rotating the top outward and down. (Refer to *"Entry Door Operation" in Part III.)*



Note

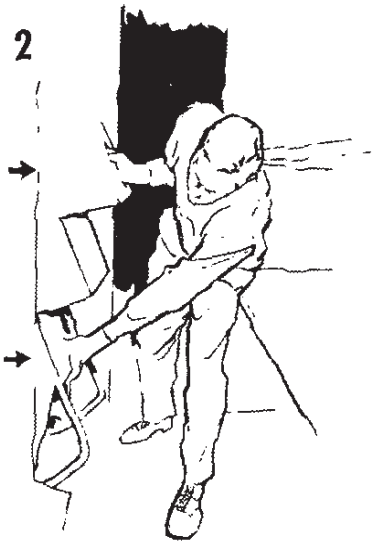
The entry door is used for ground escape when fire is present on the right side of the aircraft or the emergency condition does not warrant the use of the ground escape hatch.

1



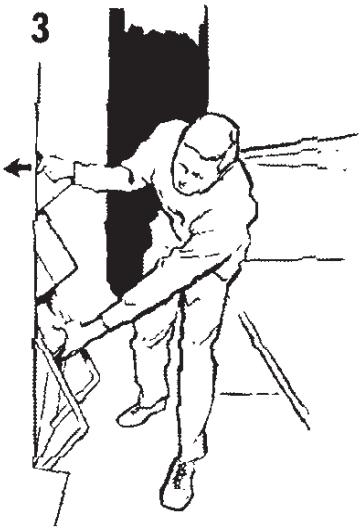
With door in place, turn rotary locking handle one-quarter turn clockwise to unlock door.

2



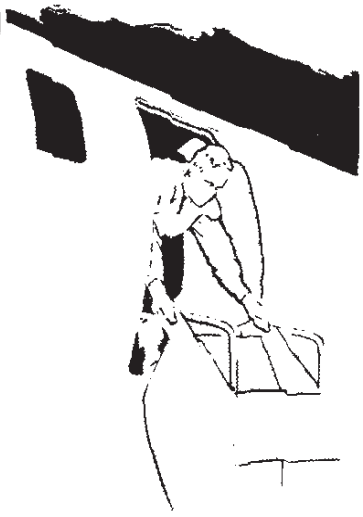
Pull entire door inward first, using an even pressure on the upper and lower sections.

3



Rotate top of door outward.

4



Grasp the handles and allow the door to rotate downward until door is in place, or it is resting on the ground, *depending on condition of aircraft.*

Figure 13-1. Emergency Entrance and Escape Exits (Sheet 2 of 2)

13.5 WHEELBRAKE FAILURE

An emergency braking system is provided for use in case of a hydraulic system failure. To initiate emergency braking, pull the emergency brake control handle and apply pressure to the top of the rudder pedals. Five applications may be required to obtain emergency braking.

WARNING

Brake pedals are interconnected. Either pilot holding pedals depressed will prohibit the other pilot from making brake applications. In the case of wheelbrake failure, the copilot should not attempt braking action unless directed by the pilot.

Brake pressure should be built up and maintained after touchdown before rudder effectiveness is lost and during whatever taxiing is necessary to clear the runway. When committed to emergency braking, do not revert to normal braking until the failure has been corrected by maintenance. After the malfunction has been corrected, the emergency brake control handle must be repositioned (pushed in) before normal power braking is available.

CAUTION

- When the emergency brake control handle is pulled, check that it is locked in the extended position. If the handle fails to lock, the handle may creep in, resulting in the loss of emergency braking.
- Do not attempt to taxi on the emergency brake system other than to clear the runway. When operating on emergency brake system, do not release pedals completely as brake pressure will be lost.
- The antiskid system is inoperative when emergency braking is used.

13.6 HOT BRAKES

Hot brakes are indicated by fading, mushy brakes or smoke from brakes.

*1. Stop aircraft — Clear Runway If Possible.

2. Notify tower.

3. Brakes — Release (when chocked).

4. Perform aircraft shutdown.

5. Evacuate aircraft toward front — If Required.

13.7 STANDBY NOSEWHEEL STEERING FAILURE

If, while operating in standby with the selector switch at STANDBY, a malfunction of the system occurs and is jeopardizing directional control of the aircraft, nosewheel steering disengage can be accomplished.

*1. Stop aircraft.

2. Electrical master switch — OFF.

Note

When the electrical master switch is placed at OFF, all systems including the dc generators will be inoperative.

3. Nosewheel steering selector switch — MAIN.

4. Electrical master switch — ON.

Note

- Further operation of the nosewheel steering system is not recommended until the system has been checked by maintenance.
- Do not attempt to reengage nosewheel steering using the yoke button.

13.8 NOSEWHEEL STEERING SYSTEM FAILURE

*1. Stop aircraft.

2. Select STANDBY (if required).

3. Taxi to ramp.

4. Investigate cause.

WARNING

Do not takeoff with a known main steering failure.

CHAPTER 14

Takeoff Emergencies

14.1 ABORT

If takeoff must be aborted for any reason prior to decision speed, proceed as follows:

*1. Throttles — Idle.

*2. Speedbrake — As Required.

*3. Wheelbrakes — Apply Maximum Braking for Existing Conditions.

*4. Thrust reverser switch — Armed (as required).

*5. Thrust reversers — As Required.

*6. Throttles — Off (if required).

7. Engine master switches — Off (if required).

14.2 ARRESTMENTS

The aircraft has no arresting hook capability. A preliminary study indicated that barrier engagements with a standard Air Force-type runway overrun barrier are unlikely. The study revealed that the nosegear doors may contact the barrier webbing, resulting in a nosegear engagement. Should this occur at low speed, the aircraft may not be controllable; if this occurs at speeds in excess of 50 knots, the nosegear may fail.

If engagement with an overrun barrier is unavoidable, retract speedbrake and aim nosewheel between the vertical straps of barrier webbing.

14.3 TIRE FAILURE

Tire failure on takeoff may present more problems than a tire failure on landing. Directional control is more difficult at the higher takeoff gross weights during an abort; therefore, if speed is at or near takeoff speed, continue takeoff and burn fuel down or jettison[1], before landing.

CAUTION

If takeoff is continued, the landing gear should not be retracted. If tire failure is suspected, have the tire visually checked for fire by a report from another aircraft or the tower and land as soon as practical. Landing should be made in accordance with the instructions in this chapter.

14.3.1 Nosegear Tire Failure on Takeoff. Nosegear tire failure is serious if either tire fails on the takeoff or landing roll. If one tire has failed or lost pressure, the remaining tire is definitely overloaded and it is much more likely to fail, especially at high gross weights. In case of complete nosegear tire failure on the takeoff run and if speed is too slow to continue takeoff, the takeoff should be aborted.

Note

- If nosegear tire failure occurs at or near nosewheel liftoff speed, the pilot may elect to continue the takeoff in order to reduce the gross weight of the aircraft. Control on the ground is much easier at lighter gross weights.

- Even though heavy braking increases the load on the nosegear, it is considered more important that the aircraft be stopped as quickly as possible rather than to attempt to lighten nosewheel loading at the expense of a longer roll. However, holding the control wheel full back during braking may reduce some of the load from the nosewheels.

[1] CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365.

14.3.2 Nosegear Tire Failure on Landing. When landing is to made with a flat nosegear tire, proceed as follows:

1. Normal approach and landing.

2. Nosewheels — Hold Off Maintaining Directional Control With Rudder and Ailerons.

   Hold the nosewheels off as long as possible. The nosewheels can be held off longer by retracting the wing flaps, maintaining attitude, and full noseup trim.

3. Thrust reversers — As Required.

4. Nosewheel steering — Engage.

CAUTION

Avoid extreme rudder pedal deflections when nosewheel steering is engaged since this may cause nosewheels to skid or skip sideways and steering effectiveness will be lost.

5. Brakes — As Required.

14.3.3 Main Gear Tire Failure on Takeoff. Tire failure on takeoff may present more problems than tire failure on landing. Directional control is more difficult and braking efficiency is greatly reduced at higher gross weights with failure of one or both main gear tires. Therefore, under certain conditions, the takeoff should be continued rather than aborted.

14.3.4 Main Gear Tire Failure on Landing. When landing with a flat main gear tire, proceed as follows:

1. Fuel — Jettison[1] (if necessary).

2. Normal approach.

   Land on side of runway that is away from flat tire. This will reduce the need for differential braking if the aircraft pulls toward the low tire.

3. Normal touchdown.

4. Thrust reversers — As Required.

5. Nosewheel steering — Engage.

6. Wheelbrakes — Maintain Directional Control.

   Use maximum brake away from swerve. Deliberately blowing the good tire may or may not be helpful in keeping the aircraft on the runway. The aircraft will tend to roll straighter with both main gear tires blown, but it is very difficult to turn under this condition. Braking efficiency is greatly reduced with both main gear tires blown.

14.4 NOSEWHEEL STEERING LIGHT ON AFTER TAKEOFF

If the nosewheel steering light remains on after takeoff, proceed as follows:

CAUTION

- Do not retract landing gear with the nosewheel steering light illuminated.
- Do not exceed 180 KIAS when landing gear is extended.

1. Steering button — Depress and Release.

If light remains on:

2. Position nose steering selector switch to STANDBY and back to MAIN. If light remains on, pull the main nosewheel steering circuit breaker.

Note

On CT-39G aircraft 160053 through 160057, the autopilot pitch-sync system will be inoperative when the main nosegear steering circuit breaker is pulled.

3. If light remains on after accomplishing step 2, pull standby steering circuit breaker. If light remains on, do not retract landing gear and land aircraft. Neutralize rudder pedals prior to nosewheel touchdown.

[1] CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365.

CAUTION

- Do not land aircraft with nosewheel steering selector switch in STANDBY.

If light goes out after accomplishing step 2:

4. Retract landing gear and leave the main nosewheel steering circuit breaker out for the remainder of the flight.

When steering is required during landing rollout:

5. Neutralize rudder pedals and select standby steering.

14.5 ENGINE FAILURE

Failures of jet engines, as a rule, result from improper fuel scheduling caused by malfunction of the fuel control system or incorrect operating technique during critical flight conditions. If engine failure is because of improper operating technique, an airstart can usually be made to restore engine operation. In case of obvious mechanical failure, an airstart should not be attempted. Except for higher rates of descent, the aircraft has normal flight characteristics with both engines dead. The single-engine handling characteristics are excellent. Unbalanced engine thrust does have a slight tendency to yaw the aircraft toward the dead engine that must be neutralized with aileron and rudder or trim. The aircraft can be trimmed hands-off, and all normal flight maneuvers may be performed as long as the recommended single-engine airspeed is maintained. (See Figure 14-1.)

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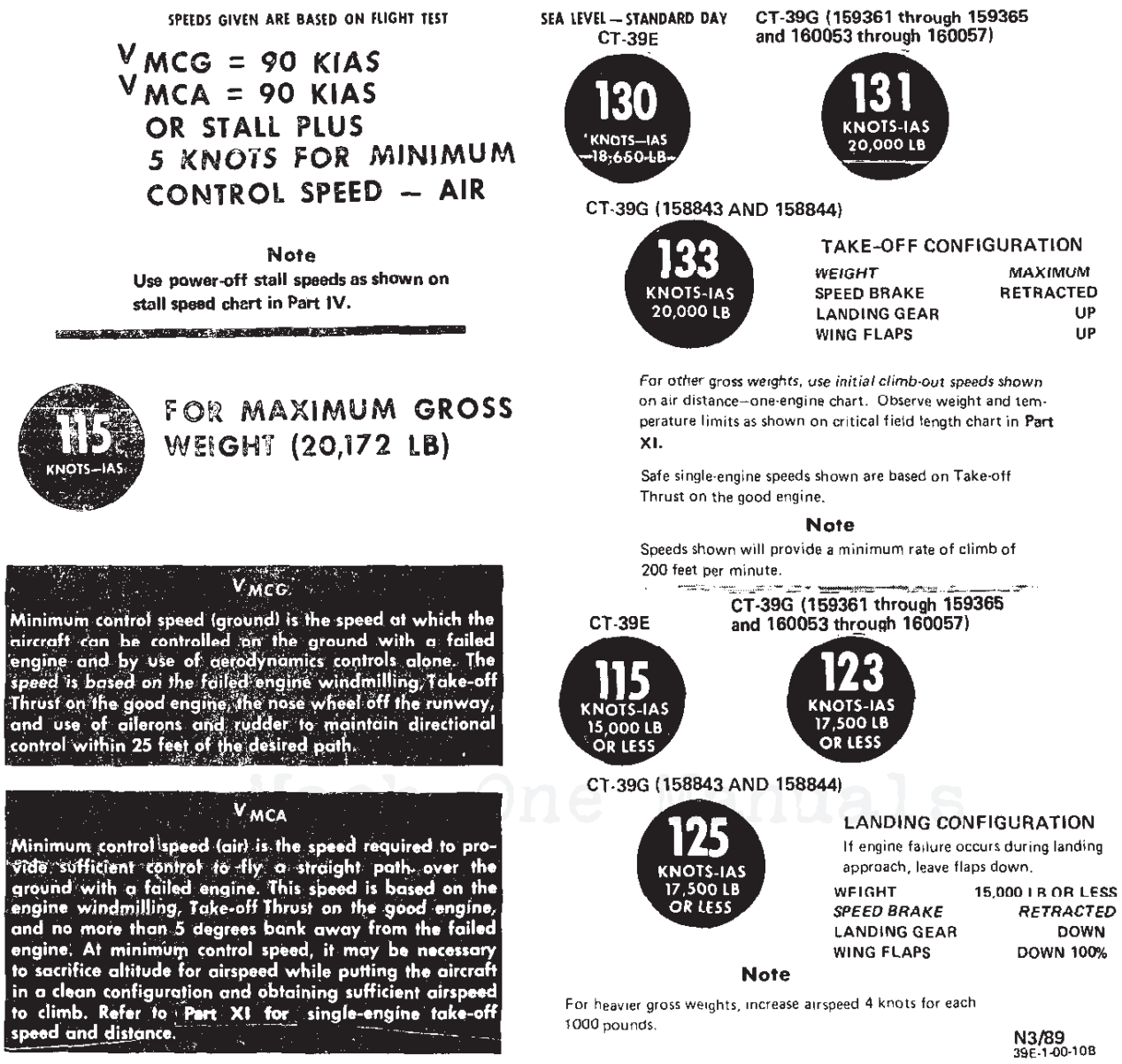

Figure 14-1. Minimum Control Speed and Safe Single-Engine Speed

CHAPTER 15

In-Flight Emergencies

15.1 ENGINE FAILURE IN FLIGHT

Note

For single-engine climb speed, refer to Part XI.

If an engine failure should occur after passing decision speed, the takeoff shall be continued. When the aircraft is safely airborne, move the landing gear handle to up, maintain positive aircraft control, and establish a positive climb rate at or above V_2. Determine which engine has failed (the aircraft will tend to yaw into the failed engine) and maintain directional control.

*1. Climb to safe altitude — 800-Feet AGL Minimum.

Before commencing the Engine Fire Checklist, continue to climb to safe altitude.

WARNING

Before proceeding with the remainder of the checklist, positive identification of the failed engine is critical.

Note

Refer to paragraph 15.3, ENGINE AIR-START, if a restart is contemplated.

2. Throttle (affected engine) — Idle.

3. Throttle (affected engine) — OFF.

4. Engine master switch (affected engine) — OFF.

5. Fuel crossfeed — X-FEED.

Note

If extended operation is expected before a landing is possible and fuel balance cannot be maintained within limits with the fuel selector in the crossfeed position, LH or RH tank position may be utilized to balance fuel.

6. Dc generator switch (affected engine) — OFF.

7. Ac generator switch (E model, affected engine) — OFF.

8. Cabin air selector — Operating Engine.

9. Fuel — Jettison[1] (if necessary).

10. Land as soon as possible.

15.2 ENGINE FIRE IN FLIGHT

Note

For single-engine climb speed, refer to Part XI.

If an engine fire should occur after passing decision speed, the takeoff shall be continued. When the aircraft is safely airborne, move the landing gear handle to up, maintain positive control, and establish a positive climb rate at or above V_2.

*1. Climb to safe altitude — 800-Feet AGL Minimum.

Before commencing the Engine Fire Checklist, continue to climb to safe altitude.

[1]CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365.

WARNING

Before proceeding with the remainder of the checklist, positive identification of the affected engine is critical.

*2. Throttle (affected engine) — Idle.

If the fire warning light goes out within 10 seconds, continue operation at reduced thrust and land as soon as possible. If warning light remains on, continue with the remaining steps.

*3. Throttle (affected engine) — OFF.

4. Fire pull handle — Pull.

5. Thrust reverser control switch — OFF.

6. Fire extinguisher — EXT. NO. 1.

7. Engine master switch (affected engine) — OFF.

If fire continues:

8. Fire extinguisher — EXT. NO. 2.

9. Fuel crossfeed — X-feed.

10. Dc generator switch (affected engine) — OFF.

11. Ac generator switch (E model, affected engine) — OFF.

12. Cabin air selector — Operating Engine.

CAUTION

Jettisoning fuel is not recommended with a continuing engine fire.

13. Land as soon as possible.

15.3 ENGINE AIRSTART

15.3.1 Immediate Restart. When an immediate restart is required, proceed as follows:

CAUTION

Unless corrective action is taken immediately, a relight by this procedure is unlikely.

*1. Throttle (affected engine) — Idle.

*2. AIR START switch (affected engine) — ON.

3. AIR START switch — OFF after relight is obtained.

Steps 4 and 5 must be performed if restart is not obtained.

4. Throttle — OFF.

5. AIR START switch — OFF.

Further airstart attempts should be made using the airstart procedure in the following paragraph.

15.3.2 Airstart. Successful airstarts can be made at altitudes of 29,000 feet and below within a wide range of airspeeds. (See Figure 15-1.) Before attempting an airstart, an effort should be made to determine the cause of the engine failure. If the failure is caused by an obvious mechanical breakdown, as indicated by the engine instruments or by excessive vibration, a restart should not be attempted.

1. Airspeed and altitude — Check.

Airspeed and altitude should be within airstart range.

2. Fuel crossfeed and tank selector switch — X-FEED.

Opening the fuel crossfeed ensures adequate fuel pressure.

3. Fire pull handle — Check In.

4. Engine master switch — ON.

5. Forward windshield anti-ice switch — OFF.

6. AIR START switch (affected engine) — ON.

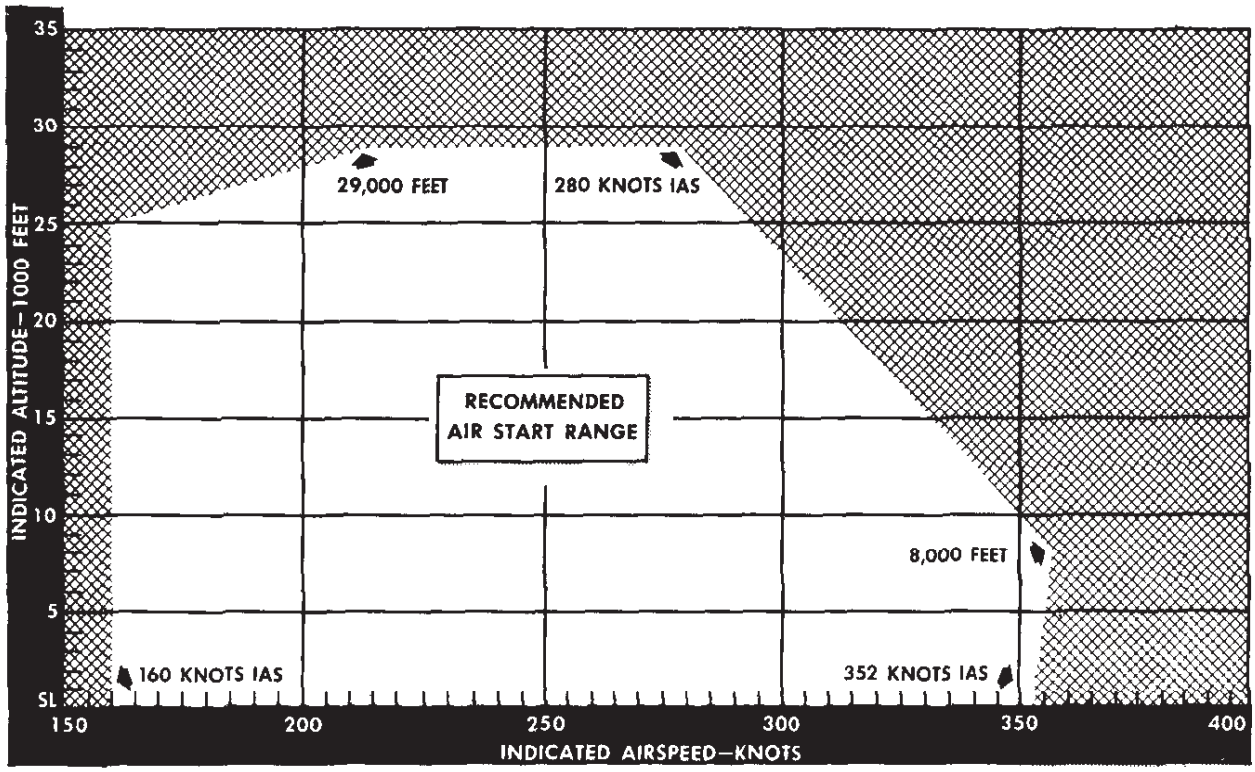

Figure 15-1. Airspeed and Altitude for Engine Airstarts

CAUTION

The ignition duty cycle is limited to 2 minutes ON, 3 minutes OFF, 2 minutes ON, and 23 minutes OFF.

Note

- If engine revolutions per minute is below normal, the starter button may be used to accelerate the start. However, the AIR START switch must first be in the OFF (center) position to allow the starter button to engage the starter.
- If engine revolutions per minute drops below generator cutout speed, it will be necessary to reset the generator after the start is completed.

7. Throttle — IDLE, Then As Required.

   Check exhaust temperature rise and revolutions per minute increase to idle revolutions per minute, then advance throttle as required. If there is no indication of light-up within 20 seconds after the throttle has been moved to IDLE or if engine fails to accelerate to idle revolutions per minute within 45 seconds after light-up, retard throttle to off and repeat start attempt.

8. Fuel crossfeed and tank selector switch — NORM.

   Move the fuel crossfeed switch to NORM unless a fuel pressure caution light is on.

9. AIR START switch — OFF.

Note

The AIR START switch has to be turned off before the generator can be brought back on the line.

10. Forward windshield anti-ice switch — ON.

11. Cabin air selector switch — Both.

15.3.3 Engine Secure Checklist. If the engine fails to start or the decision has been made not to attempt an airstart, proceed as follows:

1. Throttle — OFF.
2. AIR START switch — OFF.
3. Engine master switch — OFF.
4. Fuel crossfeed and tank selector switch — X-FEED.
5. Fuel — Jettison[1].
6. Cabin air selector switch — Operating Engine.
7. Land as soon as possible.

   Depending upon what has caused the engine failure, determine the type of pattern to be flown. If the failure may affect the good engine, it is recommended that the applicable portion of the precautionary approach pattern be used. (See Figure 5-4.) If the failure is isolated and will not affect the operation of other systems, it is recommended that a normal landing be made.

15.4 DOOR-OPEN-IN-FLIGHT EMERGENCY

Illumination of the door-open caution light indicates that the entry door is not fully locked closed. If this condition exists, pressurization air to the door seals will be shut off and depressurization will occur, illuminating the cabin pressure failed warning light when the cabin altitude goes above 10,000 feet. If depressurization does not occur, the door-open caution light indication is probably false and a faulty light circuit should be suspected. If the door-open caution light comes on in flight, follow this procedure.

*1. Fasten seatbelts sign — ON.

   All occupants fasten seatbelts.

*2. Reduce airspeed and altitude.

WARNING

Maintain balanced flight while reducing airspeed and altitude to prevent side loading on door. Sideloads may cause door to open. Damage to the aircraft or injury to the occupants can result if this door is opened in flight.

3. Investigate cause of caution light.
4. Fuel — Jettison[1] (if necessary).
5. Land as soon as practicable (if necessary).

15.5 ELECTRICAL POWER SYSTEM FAILURE

15.5.1 Total Electrical Failure. If all five 28-vdc essential bus circuit breakers in the cockpit or cabin become disengaged, all dc and ac essential bus power will be lost. All electrical equipment, instruments, lighting, and pressurization powered by this bus will be inoperative. The emergency lighting and standby attitude gyro will be operative. In the event this occurs, proceed as follows:

*1. Crew oxygen masks — ON, 100% OXYGEN.

*2. Passenger oxygen switches — EMERGENCY OPEN.

*3. Initiate descent to a safe altitude — 10,000 Feet If Possible.

4. Electrical master switch — OFF.
5. All electrical equipment — OFF.

Note

Electrical loads must be reduced prior to resetting circuit breakers.

6. Five cockpit essential bus circuit breakers — Engaged.
7. Five cabin essential bus circuit breakers — Engaged.

[1]CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365.

Note

Use oxygen walkaround bottle if necessary when checking cabin circuit breakers.

8. Electrical master switch — ON.

9. Essential electrical equipment — ON.

10. Monitor loadmeters.

11. Maintain flight in VFR conditions if possible and land as soon as practical.

15.5.2 Essential Bus Circuit Breaker Out. Illumination of the ESS BUS C/B OUT caution light indicates that one or more of the five essential bus circuit breakers located in the cabin behind the rear bench seat have become disengaged. If the caution light illuminates during flight, proceed as follows:

1. All nonessential electrical equipment — OFF.

2. Twenty-eight vdc circuit breakers (cockpit and aft cabin) — Check IN. Note and Reset Any Disengaged Circuit Breakers.

3. Maintain flight with all nonessential equipment turned OFF.

4. Monitor loadmeters.

5. Maintain flight in VFR conditions if possible and land as soon as practical.

15.5.3 Dc Generator Failure. If the dc generator-off caution light comes on, check the dc loadmeters and voltmeters to determine which dc generator has failed. With one generator off, only one 2,500-VA inverter will be operational; also, on CT-39G aircraft, the 750-VA instrument inverter will be activated to power windshield heat for one forward windshield at half power. If the engine is operating, attempt to bring the generator back into the circuit as follows:

1. FWD WSHIELD rotary switch — OFF (CT-39E).

CAUTION

FWD WSHIELD rotary switch must be turned OFF to prevent thermal shock damage to the windshield when the generator is reset.

2. Generator circuit breakers — Check.

   If any of the generator circuit breakers are popped, attempt to reset them. If any pop again, do not reset.

3. Appropriate (LH/RH) generator voltmeter button — Push, Reference Output To Ascertain Generator Serviceability.

4. Generator switch — RESET, Check Affected Generator Voltage; If Normal, Switch to ON.

   If the caution light remains out and the loadmeter shows normal system loads, the failure was temporary. If the caution light comes on again, turn generator switch to OFF. Pull affected dc generator control circuit breaker OUT.

5. FWD WSHIELD rotary switch — ON (CT-39E); CYCLE, OFF-ON (CT-39G).

15.5.4 Generator Hot. If either ac or dc generator hot caution light comes on in flight, proceed as follows:

*1. Affected generator switch — OFF.

*2. Caution light — Check out.

   If light does not go out, the failure is mechanical and not electrical. Proceed with step 3.

*3. Throttle — OFF (affected engine).

   Reduce generator revolutions per minute to minimize generator heating.

4. FUEL X-FEED & TANK SEL switch — X-FEED.

15.5.5 Failure of Both Dc Generators (CT-39E Aircraft and CT-39G Aircraft 158843 and 158844). If the dc generator-off caution light comes on and the loadmeters and voltmeters indicate both generators have failed, the paralleling bus and the dc secondary bus will be automatically disconnected and the essential bus emergency relay energized closed, providing an automatic connection between the battery bus and the dc essential bus.

Note

Both main and standby inverters are lost with both dc generators off.

Note

The hydraulic pump will only operate while the aircraft is on the ground.

Attempt to bring generators back into the circuit as follows.

Note

- If the automatic connection between the battery bus and the essential bus fails, the pilot can manually energize the essential bus emergency relay by moving the battery switch to ESS.
- The battery switch should be placed at ESS only after it has been determined that the essential bus emergency relay has failed to transfer power operations.
- All dc buses except the dc essential bus are automatically disconnected from the battery bus so that the remaining battery power will be conserved for essential purposes only.

1. FWD WSHIELD rotary switch — OFF (CT-39E).

CAUTION

The FWD WSHIELD rotary switch must be OFF to prevent possible thermal shock damage to the windshield when generators are reset.

2. Generator circuit breakers — Check.
3. Generator switch — RESET, Check Affected Generator Voltage; If Normal, Switch to ON.
4. Dc loadmeter — Check.

If dc loadmeters show normal system load:

5. FWD WSHIELD rotary switch — ON (CT-39E).

If generators fail to reset after several attempts:

6. Generator switches — OFF.

Note

With both dc generators failed in flight, the hydraulic pump motor will be inoperative. Refer to paragraph 15.18, HYDRAULIC POWER SYSTEM FAILURE.

7. Battery switch/switches — ESS/EMERG (if necessary).

   The length of time that usable battery power is available for continuous operation is approximately 45 minutes. Battery output may be decreased, however, by a number of variable factors including low state of battery charge.

8. All nonessential electrical equipment — OFF.

   Turn off all nonessential equipment and any essential equipment that flight conditions permit.

9. Fuel — Jettison[1] (if necessary).
10. Land as soon as possible.

15.5.6 Failure of Both Dc Generators (CT-39G Aircraft 159362 Through 159365 and 160053 Through 160057). Failure of both dc generators will be indicated by the illumination of the dc generator-off caution light, no indicated load on either loadmeter, and battery voltage to the dc essential bus will be indicated on the voltmeter. If this condition should occur, the battery bus is automatically connected to the paralleling bus and the essential bus (electrical master and battery switches at ON). If the automatic bus transfer fails, the pilot can manually energize the paralleling and essential bus by placing the battery switches at EMERG. Attempt to bring the generators back into the circuit as follows:

1. Generator circuit breakers — Check.
2. Generator switch — RESET, Check Affected Generator Voltage; If Normal, Switch to ON.
3. Dc loadmeter — Check.

If generators fail to reset after several attempts:

4. Generator switches — OFF.
5. Battery switches — EMERG (if necessary).

[1] CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365.

The length of time that usable battery power is available for continuous operation is approximately 45 minutes. Battery output may be decreased, however, by a number of variable factors such as low state of battery charge and electrical equipment in operation.

6. All nonessential electrical equipment — OFF.

   Turn off all nonessential equipment and any essential equipment that flight conditions permit, including the hydraulic pump.

7. Fuel — Jettison[1] (if necessary).

8. Land as soon as possible.

15.5.7 Inverter Failure (CT-39E). If the main inverter fails, the main inverter failure caution light comes on and ac power requirements automatically transfer to the standby inverter. Should this automatic transfer fail (evidenced by the standby inverter failure caution light illuminating), proceed as follows:

1. Inverter switch — STBY.

2. STANDBY INV FAIL caution light — Check Out.

   If the STANDBY INV FAIL caution light stays illuminated, the standby inverter has failed.

   If the standby inverter fails and the standby inverter failure caution light illuminates, proceed as follows.

3. Inverter switch — MAIN.

Note

The inverter switch must be in MAIN for automatic transfer to the instrument inverter.

If the instrument inverter fails, the ac instrument power caution light comes on indicating all ac electrical power is lost.

15.5.8 Inverter Failure (CT-39G). If the No. 1 inverter fails, the amber #1 INV OFF caution light comes on and ac power requirements automatically transfer to the No. 2 inverter. Should this automatic transfer fail (evidenced by the #2 INV OFF caution light illuminating), proceed as follows:

1. Inverter switch — No. 2.

2. #2 INV OFF caution light — Check out.

If both the Nos. 1 and 2 inverters fail, as indicated by the amber #1 INV OFF and #2 INV OFF caution lights coming on, flight instrument power only is automatically supplied by the instrument inverter.

Note

The inverter switch must be at No. 1 for this transfer to take place.

If the instrument inverter fails, the AC INST PWR OFF caution light comes on indicating all ac electrical power is OFF.

15.6 ELECTRICAL FIRE AND ISOLATION

Circuit breakers protect the circuits and tend to eliminate electrical fires. However, if an electrical fire occurs, proceed as follows to isolate the cause of the fire.

*1. Crew oxygen masks — ON; 100% OXYGEN.

*2. Passenger oxygen normal and emergency switches — OPEN (if necessary).

3. All nonessential electrical equipment — OFF.

4. All circuit breakers — Check for Indication of Defective Circuit.

5. Portable fire extinguisher — If Needed.

   Electrical fires will continue to emit fumes and smoke after the source of fire has been removed. Allow sufficient time for the indication or recurrence of fire when restoring power to equipment that had been turned off.

6. Individually restore all equipment previously turned OFF.

7. Refer to paragraph 15.10, SMOKE AND FUME ELIMINATION.

[1] CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365.

15.7 CABIN FIRE (NONELECTRICAL)

If a fire is discovered in the cabin, proceed as follows:

*1. Crew oxygen masks — ON; 100% OXYGEN.

*2. Passenger oxygen normal and emergency switches — OPEN (if necessary).

3. Use portable fire extinguisher (if needed).

4. Refer to paragraph 15.10, SMOKE AND FUME ELIMINATION.

15.8 AFT FUSELAGE HOT

Illumination of the aft fuselage hot caution light indicates excessive temperature in the vicinity of the pressurization ducts in the aft fuselage. While a duct failure is the most probable cause, the possibility of a fire must not be overlooked. In case of an aft fuselage hot light, proceed as follows:

*1. Crew oxygen masks — ON; 100% OXYGEN.

*2. Passenger oxygen normal and emergency switches — OPEN (if necessary).

*3. Cabin air selector — Below 15,000 Feet RAM; Above 15,000 Feet EMER PRESS.

4. Aft fuselage hot caution light — Check Out.

If light remains on:

5. Battery temperature — Check.

If battery is hot:

6. Battery — Isolate.

Note

If the aft fuselage hot light goes out and there is no indication of fire, landing may be delayed until practical.

If aft fuselage hot light remains on:

7. Land as soon as possible.

15.9 BATTERY HOT

If the amber 120 °F caution light comes on, proceed as follows:

1. Monitor the temperature trend.

If temperature rises over 150 °F, the red warning, BATT HOT annunciator, and master caution lights will come on. (In certain aircraft listed in Part I, automatic battery isolation occurs.)

At 140 °F, proceed as follows:

2. Hot battery or batteries — Isolate.

If temperature continues to rise to full scale:

3. Land as soon as possible.

If temperature normalizes below 120 °F:

4. Battery switch — Normal.

15.10 SMOKE AND FUME ELIMINATION

To eliminate smoke and fumes, proceed as follows:

*1. Crew oxygen masks — ON; 100% OXYGEN.

*2. Passenger oxygen normal and emergency switches — OPEN (if necessary).

*3. Cabin air selector switch — Check LH or RH.

Note

If smoke and fumes are detected early enough or are not strong enough to require immediate action, the contaminated air may be isolated by moving the cabin air selector switch to LH ENG. If this does not eliminate the contaminated air, move the cabin air selector switch to RH ENG.

4. Descend to safe altitude (10,000 feet), if possible.

5. Portable fire extinguisher — As Required.

If contaminated air persists:

6. Cabin air selector switch — RAM & DUMP.

7. Pilot sliding window — OPEN (slow to 150 KIAS before opening).

8. Land as soon as possible.

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15.11 CABIN AIR HOT

If the cabin-air-hot caution light comes on, proceed as follows:

1. Cabin temperature selector switch — MAN COLD.

   Momentarily hold switch at MAN COLD as often as necessary and release to OFF.

If high temperature persists and it is desirable to lower cabin temperature:

2. Cabin air selector switch — EMER PRESS.

   Govern cabin temperature/pressurization with No. 2 (RH) throttle. If unable to maintain satisfactory cabin temperature/pressure at flight level, descend to altitude not requiring pressurization.

3. Cabin air selector switch — RAM.

15.12 DEPRESSURIZATION

If depressurization is necessary, proceed as follows:

1. Crew oxygen masks — ON; 100% OXYGEN.
2. Pilot oxygen diluter levers — 100% OXYGEN.
3. Cabin occupant oxygen masks — Check On.
4. Cabin air selector switch — RAM & DUMP.

Note

Because of the controlled rate of dump, cabin pressure will require about 15 seconds to dump, depending on the pressure differential.

5. Oxygen warning horn cutout button — Push.
6. Descend to 25,000 feet or below.

15.13 PRESSURIZATION SYSTEM FAILURE

Illumination of the cabin pressurization failure warning light indicates a malfunction of either the normal pressurization source or the development of a leak in the cabin or door seal. If the malfunction is in the system, the use of emergency pressurization will alleviate the condition and pressurized flight may be continued if the resultant cabin pressurization is satisfactory. If emergency pressurization is used with the right-hand engine thrust setting high enough to provide an air temperature which exceeds 265 °F (129 °C) at the point of entering the cabin distribution ducting, the cabin-air-hot caution light will illuminate. If the cabin-air-hot caution light illuminates, retard the right-hand throttle until the light extinguishes. When using emergency pressurization, the cabin temperature control is bypassed and the cabin temperature may rise. If the malfunction is because of a leak in the cabin or door seals, the use of emergency pressurization may not alleviate the condition. If a known cabin or door seal leak exists and a 15,000-foot minimum cabin altitude cannot be maintained on emergency pressurization, the cabin air selector switch should be moved from EMER PRESS to BOTH ENG for maximum pressurization.

If it is not possible to maintain satisfactory cabin pressurization, proceed as follows:

*1. Crew oxygen masks — ON; 100% OXYGEN.

*2. Cabin air selector switch — EMER PRESS or BOTH ENG. As Required.

*3. Descend as rapidly as possible without exceeding Mach/airspeed limit.

If emergency descent is necessary, the following is recommended:

a. Throttles — IDLE.

b. Roll — 60° to 90° ANGLE OF BANK. Allow the Nose of the Aircraft to Fall 30° Below the Horizon.

c. Speedbrake — OUT.

   As airspeed approaches 270 KIAS, reduce nosedown attitude to approximately 20° and reduce angle of bank. Control airspeed with angle of bank and control forces. Maintain 350 KIAS. Do not exceed V_{MO}.

When below 15,000 feet:

4. Cabin air selector switch — RAM or RAM & DUMP, As Desired.

   If it is desired to open the pilot window for smoke and fume removal, slow to 150 KIAS before opening.

5. Passenger oxygen NORMAL switch — OPEN (if necessary).

6. Oxygen warning horn cutout button — Push.

7. Throttles — As Desired.

Note

If maximum range is a consideration, descend to 25,000 feet and use an indicated Mach of 0.55. If maximum range is not a consideration, descend to below 13,000 feet.

15.14 ENGINE OIL SYSTEM FAILURE

15.14.1 Oil Pressure (Above 50 Psi). If oil pressure indicates above 50 psi during flight, proceed as follows:

1. Throttle — Reduce Thrust (as required to maintain pressure within limits).

2. Oil pressure — Monitor.

If oil pressure remains above 50 psi or increases again after throttle reduction:

3. Shut down engine.

4. Fuel — Jettison[1] (if necessary).

5. Land as soon as practicable.

15.14.2 Oil Pressure (35 to 40 Psi). If during flight, the oil pressure drops to the 35- to 40-psi range and is steady, reduce thrust on the affected engine to below 90-percent revolutions per minute and proceed with the flight. The condition shall be corrected before the next flight. If the oil pressure fluctuates in the 35- to 40-psi range, proceed as follows:

1. Monitor engine instruments.

2. Secure engine — If Required.

3. Fuel — Jettison[1] (if necessary).

4. Land as soon as practicable.

15.14.3 Oil Pressure (Below 35 Psi). If oil pressure drops to or fluctuates below 35 psi, proceed as follows:

*1. Monitor engine instruments.

*2. Secure engine — If Required.

3. Fuel — Jettison[1] (if required).

4. Cabin air selector switch — Operating Engine.

5. Land as soon as practicable.

15.14.4 Oil Overheat (Constant Thrust)

1. Oil Pressure — Monitor.

2. Throttle — Reduce Thrust.

3. Oil temperature gauge — Monitor.

If temperature remains high:

4. Shut down engine and land as soon as practicable.

15.14.5 Oil Overheat (After Thrust Reduction)

1. Oil Pressure — Monitor.

2. Advance throttle and/or reduce altitude.

3. Oil temperature gauge — Monitor.

If temperature remains high:

4. Shut down engine and land as soon as practicable.

WARNING

If the oil temperature does not drop within acceptable limits within a reasonable time after the throttle is advanced, the oil pressure gauge should be closely monitored as an engine oil system failure and subsequent engine failure could occur.

15.15 FUEL SYSTEM FAILURE

Failure of a tank-mounted boost pump or engine-driven centrifugal fuel pump causes the low fuel pressure caution lights to come on. If the caution lights come on, proceed as follows to preclude a possible engine flameout.

1. Fuel crossfeed and tank selector switch — X-FEED.

[1]CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365.

WARNING

When operating on crossfeed with only one boost pump operating, the last 30 gallons (200 pounds) of fuel in the tank with the inoperative boost pump will not be available. If both boost pumps are inoperative, a total of 60 gallons (400 pounds) will not be available.

2. Low fuel pressure caution light — Check.

   If failure was because of an inoperative tank-mounted boost pump, the low fuel pressure caution light will go out. However, if the engine-driven pump failed, the caution light will remain on.

If caution light goes out:

3. Circuit breaker of failed boost pump — Pull.

If caution light remains on:

4. Fuel crossfeed and tank selector switch — Normal.

Note

- With the fuel crossfeed and tank selector switch at X-FEED, one tank-mounted boost pump will maintain normal two-engine operation at all altitudes (provided the remaining boost pump is receiving fuel); however, the wing fuel may become out of balance. If this condition occurs, a steady-state constant-heading sideslip of one-half to one ball deflection with the heavy wing high will correct the condition. Fuel will transfer at a rate of approximately 100 pounds per minute using one-half ball deflection.

- When operating with one tank-mounted boost pump inoperative, do not use the LH TANK or RH TANK positions of the fuel crossfeed and tank selector switch for fuel balancing. Selection of the heavy tank would deenergize the operable tank-mounted boost pump, placing both engines on suction feed.

At high engine thrust settings and low fuel level conditions, the remaining boost pump may not reliably support both engines. Therefore, under the following conditions, place the fuel crossfeed and tank selector switch at the NORM position:

1. On final approach with both engines operating

2. During other flight conditions when the total required fuel flow for both engines exceeds 2,400 pph

3. If, during single boost pump operation, fuel transfer between the right and left wing tanks is insufficient to limit the fuel imbalance between them to 200 pounds maximum.

Note

With the fuel crossfeed and tank selector switch at NORM, the engine with the inoperative fuel boost pump will be on suction feed. The indicated fuel imbalance will not be accurate if the aircraft is flying in the steady-state sideslip.

The engine-driven centrifugal fuel pump has suction-feed capabilities in case of tank-mounted boost pump failure; however, this capability is limited, dependent upon fuel temperature and altitude. The engines will operate on suction feed (tank-mounted boost pump inoperative) at higher altitudes with "cold soaked" fuel than they will with warmer fuel.

The aircraft has demonstrated suction-feed capabilities up to 15,000 feet altitude regardless of fuel temperature. With "cold-soaked" fuel, engines have operated on suction feed up to 29,000 feet.

Note

Takeoff shall not be attempted if a boost pump is inoperative.

15.16 HUNG DECELERATION OR HIGH FLIGHT IDLE

At low fuel temperature, small quantities of water in the fuel control end of the compressor discharge line may freeze. This can cause a shift in fuel scheduling, resulting in high flight idles or hung decelerations.

For hung deceleration or high flight idle, proceed as follows:

1. Fuel heater switch — ON.

If condition is corrected, leave fuel heater on unless fuel temperature exceeds +5 °C (41 °F). If condition persists:

2. Place the fuel heater switch at OFF after:

   a. Five minutes of fuel heater operation

   b. Fuel temperature is above +5 °C (41 °F).

3. Shut down affected engine (if necessary).

15.17 AUTOMATIC FUEL JETTISONING FAILURE (CT-39E, CT-39G Aircraft 158443, 158844, and 159362 Through 159365)

During automatic fuel jettisoning, a malfunction of a cell-mounted float switch could cause the respective fuel jettison shutoff valve to remain open. If the valve remains open, all fuel in the respective wing tank will be jettisoned. An open jettison valve is indicated by the fuel jettison open caution light remaining on and a continuation of fuel jettisoning (as indicated by abnormal decrease of fuel quantity). If this occurs, proceed as follows:

1. Fuel jettison switch — OFF.

   Automatic shutoff should occur between approximately 1,400 and 2,200 total pounds of fuel remaining.

2. Fuel jettison open caution light — Check Out.

3. Fuel quantity gauges — Check.

   Check fuel quantity gauges to determine whether fuel jettisoning has stopped. If fuel continues to jettison, proceed to step 4.

4. Fuel jettison circuit breakers — Pull.

5. Land as soon as possible.

15.18 HYDRAULIC POWER SYSTEM FAILURE

1. Check hydraulic pressure/hydraulic fluid low light.

   If hydraulic fluid low light is on, turn hydraulic pump switch OFF and pull the hydraulic pump control circuit breaker. Proceed to step 5.

2. Check hydraulic pump control circuit breaker is in.

If pump operation is restored, monitor system.

3. Hydraulic pump switch — MANUAL.

   If MANUAL position activates hydraulic pump, do not exceed 1-minute operation, monitor/maintain hydraulic pressure with manual switch. Do not reset.

4. Hydraulic pump switch — OFF/RESET, then ON.

   If pump is not restored, turn hydraulic pump switch OFF and pull pump control circuit breaker. Proceed to step 5.

5. Speedbrake switch — Centered/OFF and Pull Speedbrake Control Circuit Breaker.

   Placing the speedbrake switch at off (center) may prevent loss of hydraulic fluid if a line failure has occurred beyond the speedbrake selector valve. The speedbrake may be extended using accumulator pressure; this should be done only if essential to the mission and then only for one extension. (Retract an extended speedbrake using the speedbrake dump switch.) Place the auxiliary hydraulic power switch and the speedbrake switch at OFF as soon as pressure is no longer needed.

6. Thrust reverser control switch — OFF/As required (if available).

Note

If there is a leak in the hydraulic system and the switch is in the emergency stow position, turning the switch off will isolate a possible source of the leak.

Before landing:

7. Lower landing gear by emergency method.

8. Emergency brake control handle — As Required.

9. Anti-skid — As Required.

10. Hydraulic auxiliary power switch — As Required.

After landing:

11. Thrust reverser control switch — ON/As Required (if available)

CAUTION

If there is a leak in the hydraulic line to one of the thrust reversers, selecting the on position will put hydraulic fluid under pressure into the engine area, greatly increasing the chance of a fire. If a leak can be confirmed as the cause of the hydraulic malfunction, arm and use the thrust reversers only if absolutely necessary.

12. Thrust reversers — Deploy (if available).

13. Clear runway — Shut Down Engines.

CAUTION

Do not attempt to taxi the aircraft with less than 1,700 psi auxiliary system pressure. As soon as practical, secure engines and have aircraft towed to line area.

Note

Do not set the parking brake.

15.19 SPEEDBRAKE SYSTEM FAILURE

If the speedbrake will not close by using the speedbrake switch, proceed as follows:

1. Speedbrake switch — OFF (Center) Position.

2. Speedbrake power circuit breaker — Pull.

3. Speedbrake emergency dump switch — Dump.

15.20 AUTOPILOT SYSTEM MALFUNCTION

In the event of erratic or uncontrollable autopilot maneuvers, proceed as follows:

1. Control wheel button — Press (DISENGAGE light on).

   Manual control of aircraft is assumed. If autopilot fails to disengage, proceed to step 2.

2. A/P EMERGENCY DISCONNECT or A/P DISCONNECT lever — DISENGAGE.

   Manual control of the aircraft is assumed.

3. A.P. POWER switch (CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365) — OFF (DISENGAGE light out).

15.21 TRIM FAILURE

If any of the three trim systems should fail in either extreme-travel position, the force required to neutralize the controls or to move the control surface to the opposite extreme is not beyond physical capabilities. If a failure occurs in the rudder trim switch or the normal trim switch on the control wheel, proceed as follows:

1. Trim circuit breakers — Check In.

2. Trim control selector switch — ALT.

3. Longitudinal, lateral, and rudder alternate trim switches — Use As Necessary.

Note

With autopilot engaged, stabilizer trim may be available.

15.21.1 Horizontal Stabilizer Trim Inoperative

15.21.1.1 Cruise Trim. In the event of an inoperative horizontal stabilizer during flight, proceed as follows:

1. Adjust airspeed to provide a comfortable control wheel force.

Note

If the horizontal stabilizer becomes inoperative while trimmed in a normal cruise attitude, approximately 235 knots will provide a control wheel force that requires the least pilot effort.

2. If a landing is to be made with excessive nose down trim, make a no-flap approach.

Note

A no-flap landing will provide a lower yoke force.

15.21.1.2 Landing Trim. In the event that the stabilizer becomes inoperative in the landing position and a go-around is required, proceed as follows:

1. Throttles — Flight Takeoff Thrust EPR.

2. Flaps — Leave DOWN.

3. Landing gear — Leave DOWN.

4. When sufficient altitude is reached, reduce thrust to prevent exceeding 130 knots. Yoke force will be approximately 20 pounds.

Note

If the GA is made in the approach configuration (66-percent flap), maintain flap setting and use a speed of 140 knots.

15.21.2 Runaway Stabilizer

1. Actuate primary flight controls to counteract the trim condition.

2. TRIM DIS button — Depress.

3. Autopilot — Disengage.

4. Trim system selector switch — OFF and Then ALT.

If an uninitiated trimming action starts:

5. Trim system selector switch — OFF.

If trimming action continues:

6. TRIM DIS button — Depress.

Note

Continue flight without use of trim.

15.21.3 Trim Disconnect System Malfunction (CT-39G Aircraft 160053 Through 160057 and Aircraft Changed by Service Bulletin 74-6)

If an inadvertent trim disconnection occurs:

1. Normal and alternate trim system circuit breaker — PULL (open).

2. Continue flight without use of trim.

15.22 DAMAGED WINDSHIELD

In the event a crack occurs in the forward windshield outer ply and there is a danger that the airstream could further deteriorate the outer ply (possible engine ingestion), the following procedure should be observed. Distortion or bubbling is not considered for this procedure.

1. Reduce airspeed as much as practicable.

2. Forward windshield anti-ice switch — OFF.

Note

Defogging and birdproofing are not available.

3. Ignition switch — Continuous.

4. Airspeed — 250 KIAS or Below at Altitudes Below 10,000 Feet.

5. Maintain altitude above 10,000 feet (flight condition permitting).

6. Land as soon as practicable.

15.23 THRUST REVERSER DEPLOYED IN FLIGHT

If a thrust reverser deployment is indicated in flight, proceed as follows:

*1. Throttle (affected engine) — Retard to Idle.

*2. Thrust reverser control switch — EMERG STOW.

3. Check respective LH or RH T/R lights — OUT.

If thrust reverser stows, proceed as follows:

4. Leave thrust reverser control switch in EMERG STOW.

5. Land as soon as practicable.

If thrust reverser does not stow:

6. Secure affected engine.

7. Land as soon as practicable.

Note

An inadvertent thrust reverser deployment in flight will result in heavy buffet that decreases as the respective engine thrust is decreased. The aircraft is readily controllable; however, altitude cannot be maintained until thrust is reduced on the engine with the deployed thrust reverser.

15.24 THRUST REVERSER UNSAFE

If the thrust reverser UNSAFE caution light comes on in flight, proceed as follows:

1. Thrust reverser control switch — EMERG STOW.
2. Land as soon as practicable.

CAUTION

Caution should be exercised when using the thrust reversers on landing because the reverser that initiated the automatic stow condition will not deploy and an asymmetric reverser thrust condition will occur.

15.25 LOST PLANE PROCEDURE

1. Conserve fuel.
2. Climb — Maintain Highest Feasible Altitude, As Situation Permits, To Facilitate Radar Pickup.
3. Attempt communication with all available equipment.
4. Squawk emergency (7700).
5. Attempt navigation by all available means.
6. Be alert for aircraft vectored to join in.

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CHAPTER 16

Landing Emergencies

16.1 DITCHING

Experience indicates that the aircraft has good static buoyancy qualities, floating near level or slightly tail-low. Exit should be through the ground escape hatch. The crew and pilot should be familiar with their stations and duties. The success of any ditching depends on the ability of the crew to perform their duties thoroughly and quickly. As much fuel as possible should be jettisoned[1] To lighten the aircraft and improve flotation. Ditch while power is available if possible so that the most desirable approach can be made.

In case of a ditching, each individual will be expected to perform the following.

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16.1.1 Pilot in Command

1. Alert crew and passengers, "Prepare for ditching."
2. Jettison[1] fuel as necessary.
3. Tighten safety belt and shoulder harness.
4. Cabin air selector switch — RAM & DUMP (below 10,000 feet).
5. Landing gear handle — Check UP.
6. Wing flap handle — Down.

Note

Wing flaps will collapse on impact.

7. Speedbrake switch — IN.
8. Direct crewman, "Remove and stow ground escape hatch."
9. Observe sea state and wind.

   Plan approach to parallel any uniform swell pattern and attempt to touch down along a wave crest or just after crest passes. Wind observations below 1,000 feet will be accurate. If the wind is as high as 25 knots or the surface is rough and irregular, ditch into the wind on the falling side of a wave.

10. Normal approach — Aircraft Attitude Should Be Wings-Level and Slightly Nose-High.
11. Signal crew and passengers, "Brace for ditching."
12. Throttles — OFF (at or just before touchdown).
13. Electrical master switch — OFF.

When aircraft has definitely stopped:

14. Assist copilot and passengers as necessary.
15. Leave aircraft after copilot and inflate LPP when clear.

16.1.2 Copilot

1. Acknowledge, "Prepare for ditching."
2. Initiate radio distress procedures.
3. Tighten safety belt and shoulder harness.
4. On signal, "Brace for ditching," shield face.

When aircraft has definitely stopped:

5. Assist pilot, crewman, and passengers as necessary.

[1] CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365.

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6. Leave aircraft after cabin occupants and inflate LPP when clear.

16.1.3 Crewman

1. Acknowledge, "Prepare for ditching."

2. Distribute and assist passengers in donning LPPs.

3. Stow folding tables and check security of passengers.

4. Remove ground escape hatch and stow.

The hatch is to be removed on signal from the pilot.

5. Assume flight station and tighten safety belt.

6. On signal, "Brace for ditching," hold head down and shield face and upper torso with hands and forearms.

When aircraft has definitely stopped:

7. Assist crewmembers and passengers as necessary.

8. Remove raft from rack.

9. Leave aircraft and inflate LPP/raft when clear.

16.2 FORCED LANDING

If both engines are flamed out and a prepared surface is not available, land with the landing gear down. For forced landing procedures, see Figure 16-1.

16.3 PRECAUTIONARY APPROACH

Whenever the possibility of a dual flameout situation arises, it is recommended that a precautionary approach be made. (See Figure 16-1.)

16.4 MAXIMUM GLIDE

For maximum glide distance (refer to Chapter 26) with both engines failed (windmilling or frozen), the best gliding speed is 170 KIAS for a clean aircraft with landing gear and flaps up and speedbrake in. (See Figure 16-2.) When speed is maintained at 170 KIAS, the glide ratio is about 12.5 to 1. Thus, for every 10,000 feet of altitude, the aircraft will glide about 21 nm. The glide ratio and glide distance of the aircraft with the landing gear down are about half of those obtainable with the landing gear up.

16.5 SINGLE-ENGINE LANDING

If a single-engine landing is required, it is the pilot decision as to the type of pattern and degree of flaps to be used. If the failure occurs when committed to landing and will not affect operation of the good engine, a normal landing may be made utilizing 100-percent flaps and maintaining a normal approach speed plus 10 knots. (See Figure 16-1.)

16.5.1 Planned Single-Engine Landing. If the cause of the engine failure is not definitely confined to one engine or may affect additional aircraft systems, applicable portions of the forced landing procedure should be used. Single-engine normal approach speed will be normal approach speed plus 10 knots.

16.6 SINGLE-ENGINE WAVEOFF

For single-engine waveoff, see Figure 16-3.

16.6.1 Engine Idle Hangup During Waveoff. If engine idle hangup should occur during a waveoff, follow single-engine waveoff procedure.

*1. Airstart switch (affected engine) — ON.

After waveoff:

2. Airstart switch — OFF.

3. Dc generator — Check and Reset If Required.

16.7 LANDING GEAR EMERGENCY OPERATION

16.7.1 Landing Gear Emergency Retraction on the Ground

1. Depress landing gear override button.

2. Gear handle — UP.

CAUTION

Use of the landing gear downlock override button is not recommended because of unsymmetrical gear retraction that would cause a more severe condition than originally existed.

SPEEDS GIVEN ARE APPLICABLE FOR ANY WEIGHT CONDITION

PROCEDURE TO USE WHEN:
Engines are windmilling or seized or *whenever an emergency requires.*

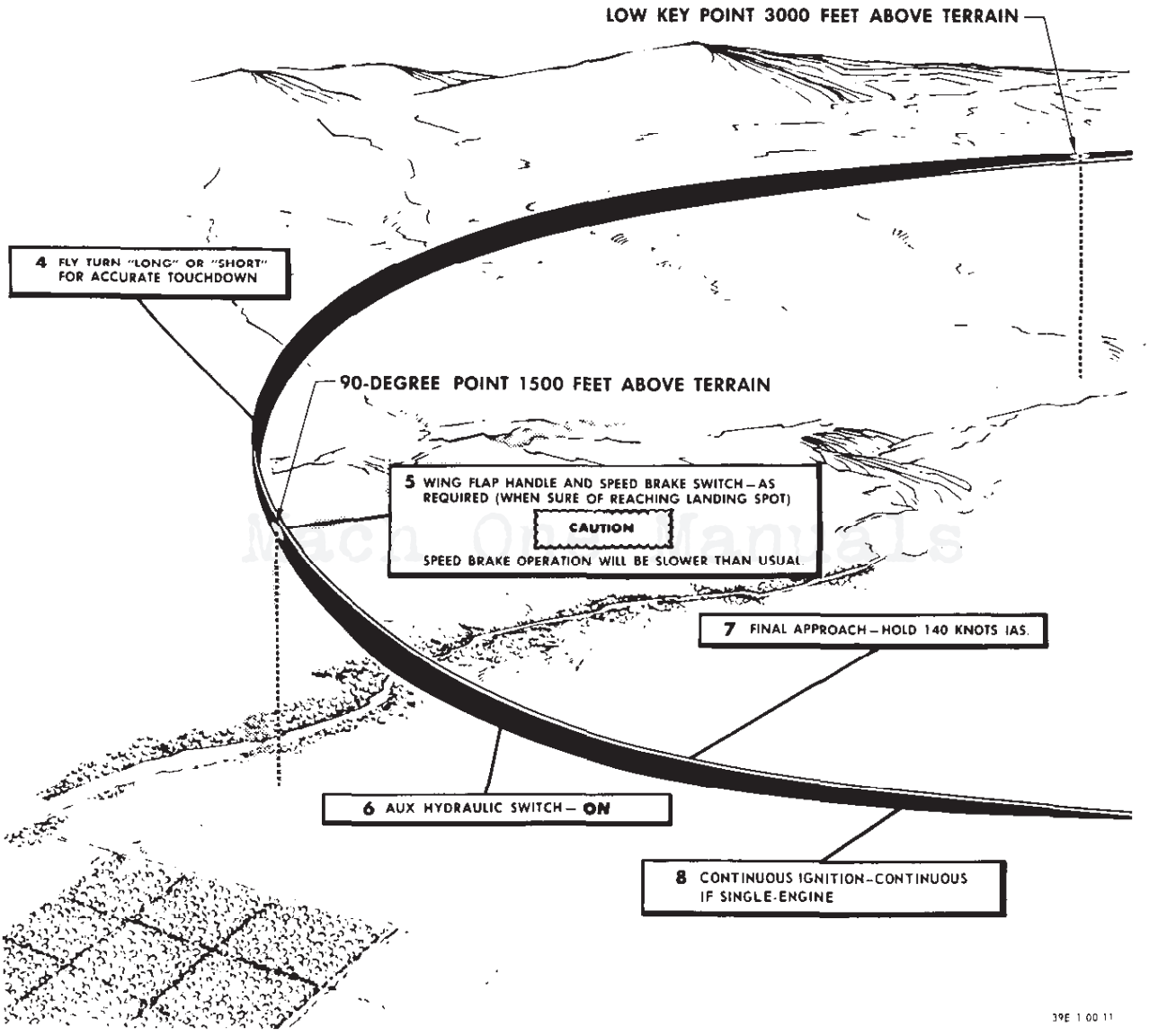

Figure 16-1. Precautionary Approach (Sheet 1 of 2)

16.7.2 Unsafe Indication Following Retraction (One or More Landing Gear Green Position Lights Illuminated Following Retraction). If one or more landing gear green position lights are on following retraction with the gear handle up or if the landing gear warning light is out with the handle up and one or more landing gear green position lights is on, resolve these gear and any other gear abnormalities or suspected abnormalities not covered in Part V of the NATOPS flight manual by the following method:

1. Maintain airspeed below gear-down limit speed.
2. Landing gear handle — DOWN.

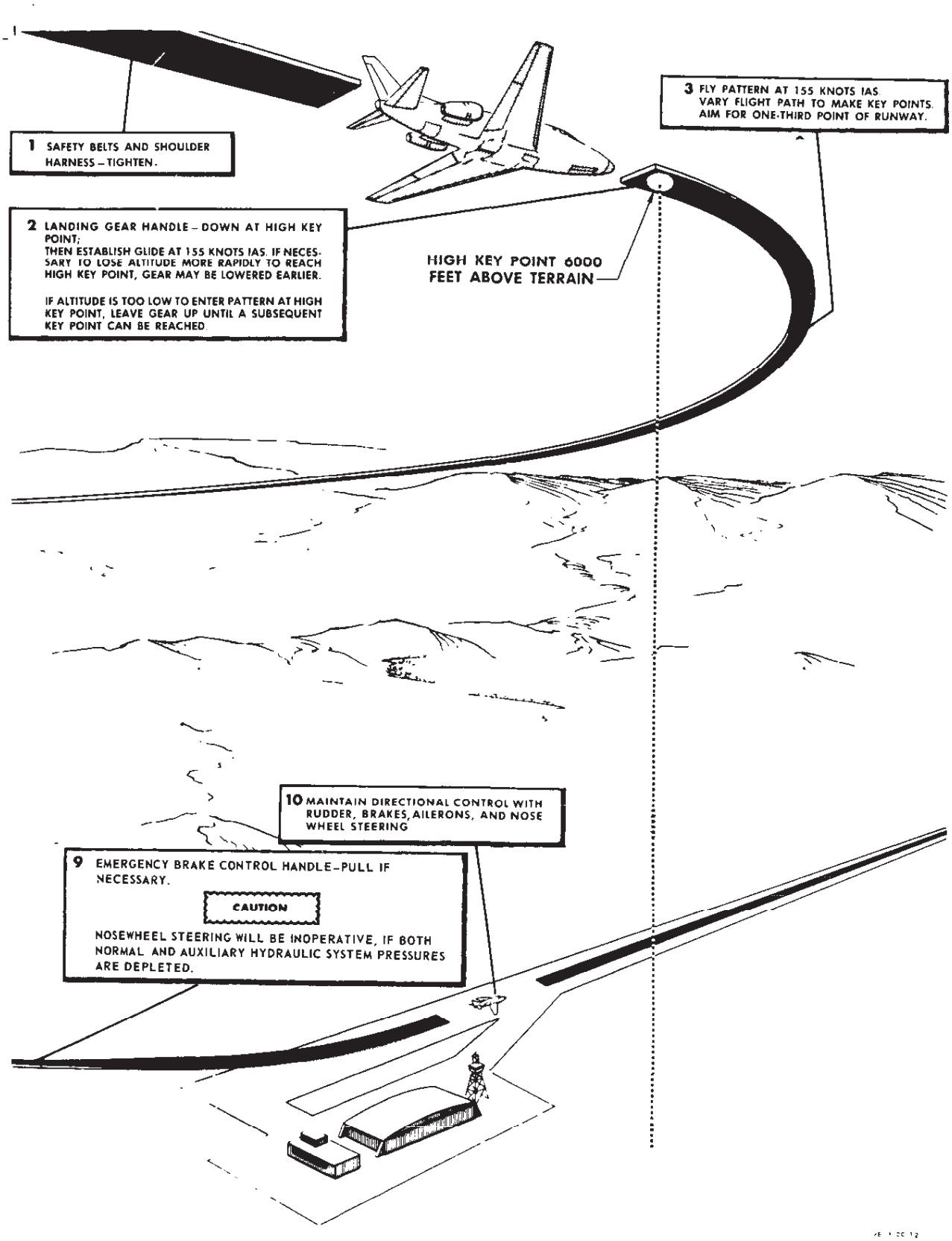

Figure 16-1. Precautionary Approach (Sheet 2 of 2)

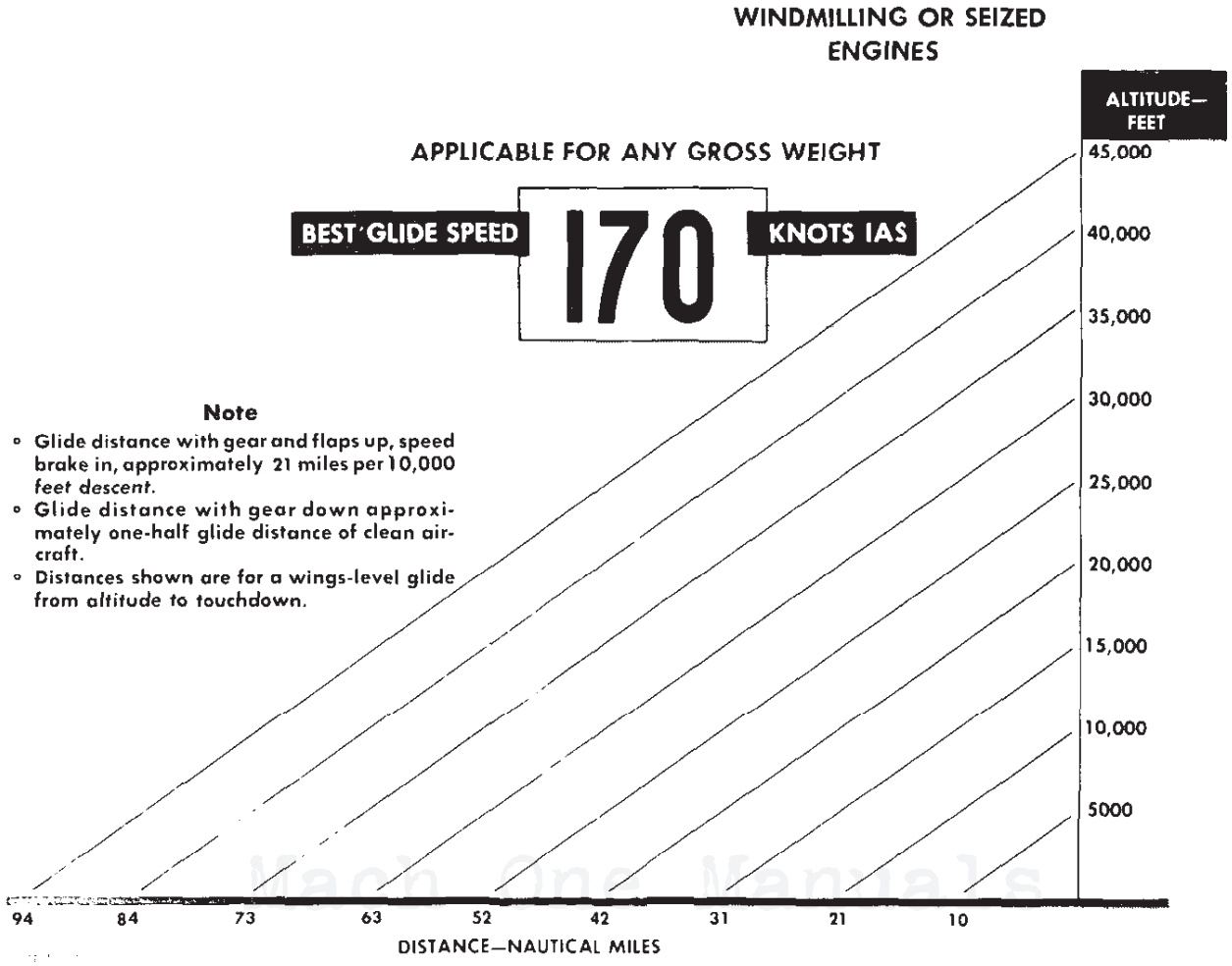

Figure 16-2. Maximum Glide Distances

3. Lower gear using Landing Gear Emergency Lowering procedure in paragraph 16.7.5.

4. Obtain visual check of gear position (if possible).

5. Land as soon as practicable.

16.7.3 Unsafe Indication Following Retraction (Steady Light in Landing Gear Handle). If the landing gear warning light (steady light in the landing gear handle knob) remains on after the gear is retracted, proceed as follows:

1. Maintain airspeed below gear-down limit speed.

2. Obtain visual check of gear position (if possible).

If no visual check is available or visual check confirms an abnormal condition, proceed with step 3.

3. Landing gear handle — DOWN.

Check landing gear position indicators for safe indication.

4. Land as soon as practicable.

If visual check indicates all gear up and door closed, *proceed with step 5.*

5. Landing gear handle — DOWN.

6. Landing gear ELEC RESET button — PRESS.

7. Landing gear handle — UP.

If the landing gear fails to retract or the landing gear warning light remains on, proceed with step 8.

8. Landing gear handle — DOWN.

9. Land as soon as practicable.

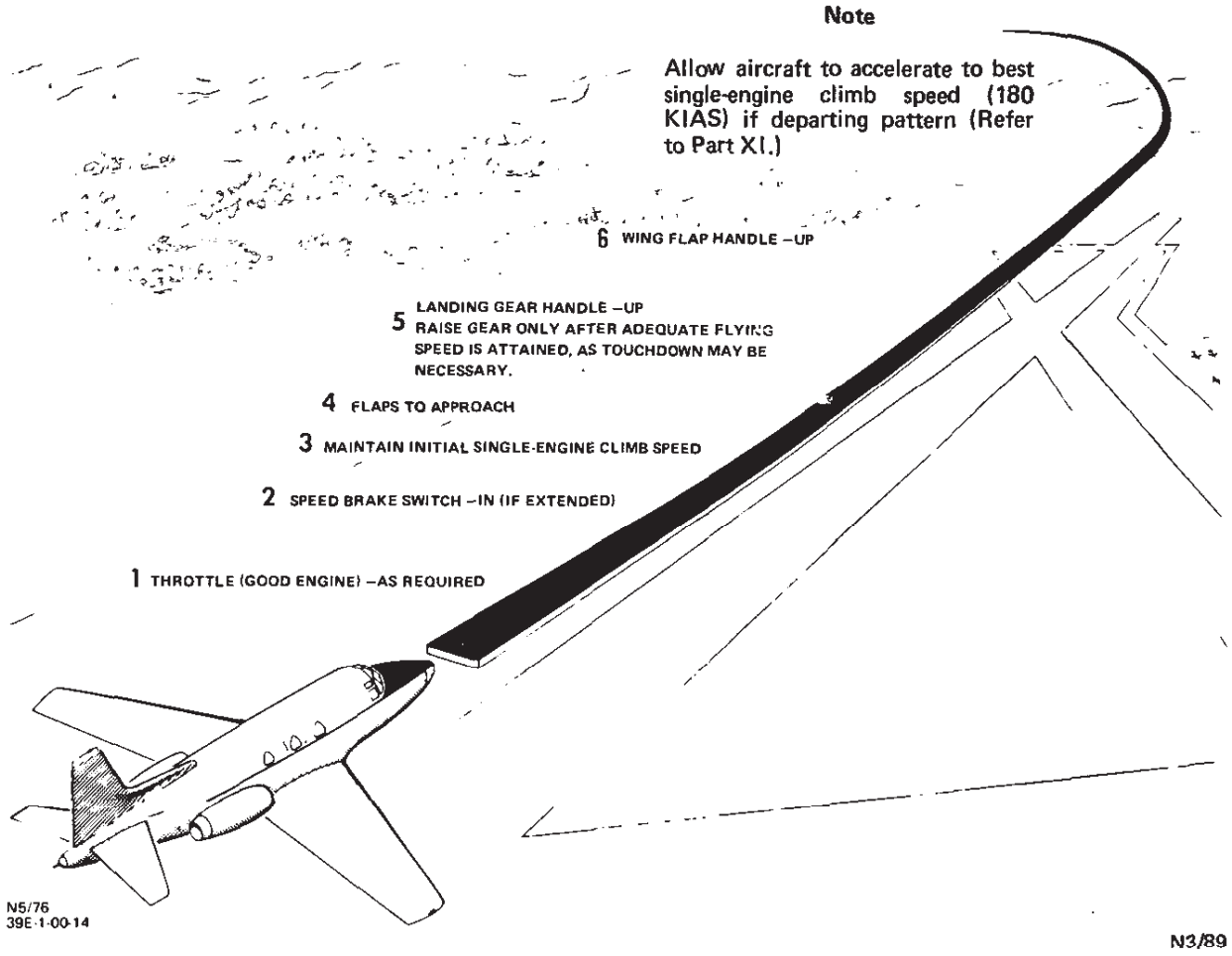

Figure 16-3. Single-Engine Waveoff

16.7.4 Unsafe Indication Following Extension

1. Caution light — Test.

Check landing gear position indicators to confirm unsafe gear.

2. Landing gear circuit breakers — IN.

3. Landing gear downlock ELEC RESET button — PRESS.

4. Perform landing gear emergency lowering.

16.7.5 Landing Gear Emergency Lowering

1. Flaps — UP.

2. Airspeed — Reduce to Below 180 KIAS.

Lowering may be accomplished by any speed below 180 KIAS; however, 120 KIAS is recommended for fastest emergency lowering.

3. Landing gear emergency release handle — Pull and Place Restraining Hook in Hole Provided in Left Console.

CAUTION

- Do not stow the landing gear emergency release handle. Advise maintenance personnel to restow the handle after the malfunction has been corrected.

- When the landing gear is extended by the landing gear emergency release T-handle, the main gear doors will remain extended.

4. Landing gear handle — DOWN (if possible).

5. Landing gear position lights — Check for Safe Indication.

6. Obtain visual check if possible.

Note

- A smooth and gradual increase of aileron and rudder force that results in a yaw angle of a single-ball-width deflection should be held until the respective main gear locks down. Repeat single-ball-width deflection for the other main gear. However, if the main gear still fails to lock, the yaw angle may be increased as much as required as long as gear limit speed is not exceeded.

- All landing gear should lock down in approximately 30 seconds; this time may be longer during cold-weather operation.

16.7.6 Landing Gear Unsafe After Emergency Lowering. If an unsafe main gear indication is observed in the cockpit and hydraulic pressure is available, proceed as follows:

1. Gear handle — Down.

2. Emergency release T-handle — Stow.

3. Gear electric reset button — Depress.

Note

Should the MAIN GR MAINT SW be inadvertently left in the OPEN position, this check is invalid.

4. Obtain visual check.

Closing of main gear doors may be detected audibly or visually by a slight drop in hydraulic pressure or observation from tower or chase plane. If gear appears down and gear doors are closed, downlocks are in place and normal landing should be made. If gear appears down and gear doors are open, downlocks may not be in place or positioned to exert sufficient pressure to keep the gear down; a normal landing is recommended, keeping sideloads to a minimum. If this is not possible, consideration should be given to a belly landing.

After landing roll, stop aircraft, check all downlocks, and pin gear. Tow the aircraft only after inspection.

WARNING

The maintenance safety locks will not prevent main gear from retracting when towing.

16.8 LANDING WITH NOSEGEAR UP

1. If the nosegear fails to extend, retract the main gear and execute a belly landing. If unable, proceed as follows.

2. Loose equipment — Secure.

3. Fuel — Jettison[1] (if necessary).

4. Alert crew and passengers.

The aft passenger seats and/or couch should be occupied whenever possible to aid holding the aircraft in a nose-high attitude after touchdown.

5. Ground escape hatch — Remove and Stow.

6. Position, anticollision, and strobe light switches — OFF.

7. Safety belts and shoulder harness — Tighten.

8. Speedbrake switch — OUT, Then Off (center).

9. Minimum-roll landing approach.

10. Wing flap handle — UP Immediately After Touchdown.

11. Throttles — As Required.

12. Nose — Hold Off.

Apply full noseup trim and hold control wheel aft as necessary. Lower nose gently before losing elevator control.

13. Thrust reversers — As Required.

Stow thrust reversers when nose is close to runway before losing elevator control to minimize

[1]CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365.

force of nose contact with runway that might be caused by use of the thrust reversers.

14. Maintain directional control with rudder, aileron, and differential braking.

15. Engine and electrical master switches — OFF.

16. Braking — As Required After Lowering Nose.

WARNING

Arresting cables should be removed before employing this procedure.

16.9 LANDING WITH EITHER MAIN GEAR UP

1. If the main gear fails to extend, retract the gear and execute a belly landing. If unable to retract the gear, proceed as follows.

2. Loose equipment — Secure.

3. Fuel — Jettison[1] (if necessary).

4. Alert crew and passengers.

5. Ground escape hatch — Remove and Stow.

6. Position, strobe, and anticollision light switches — OFF.

7. Safety belts and shoulder harness — Tighten.

8. Normal approach.

9. Wing flap handle — Down.

10. Speedbrake switch — In.

11. Hold unsafe gear off as long as possible.

12. Throttles — OFF.

13. Engine and electrical master switches — OFF.

16.10 BELLY LANDING

A skid on the lower fuselage (wing joint) is provided to facilitate a belly landing and to lessen aircraft damage. Before a belly landing, jettison[1] as much fuel as possible so that fuel will be confined to center wing tank only. If the equipment is available and time and conditions permit, the runway should be foamed to eliminate fire hazard and reduce aircraft damage.

1. Loose equipment — Secure.

2. Fuel — Jettison[1] (if necessary).

3. Alert crew and passengers.

4. Ground escape hatch — Remove and Stow.

5. Position, strobe, and anticollision light switches — OFF.

6. Safety belts and shoulder harness — Tighten.

7. Normal approach.

8. Wing flap handle — Down.

9. Throttles — As Required To Maintain Minimum Sink Rate In Level Attitude.

10. Speedbrake switch — IN.

11. Throttles — OFF (at or just before touchdown).

CAUTION

Avoid touching down in a nose-high attitude to prevent the aircraft from slapping down on the runway at touchdown.

12. Engine and electrical master switches — OFF.

16.11 LANDING WITH ONE OR BOTH SLATS INOPERATIVE

If one or both slats should fail to extend, the final approach airspeed should be increased by 15 knots.

[1] CT-39E, CT-39G aircraft 158843, 158844, and 159362 through 159365.

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PART VI

All-Weather Operation

Chapter 17 — All-Weather Operation

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CHAPTER 17

All-Weather Operation

17.1 INTRODUCTION

The procedures set forth in this part may be repetitious or may be different from or in addition to the normal operating procedures in parts I and III. Particular emphasis should be placed upon the recommendations shown in the instrument flight procedures of this part, because these steps and procedures are the minimum requirements whenever operating in IFR conditions.

17.2 INSTRUMENT FLIGHT PROCEDURES

This aircraft has all the basic flight instruments and radio navigation equipment for IFR flight as well as the UHF and VHF command radios and the SSB HF radio required to control the flight. Integrated flight, autopilot, and weather radar systems aid the pilot during instrument flight.

Note

Although it is desirable to avoid flight through areas of turbulence and thunderstorms, it is not always possible. This is particularly true when flying under instrument flight conditions, especially at night. Clouds containing high moisture content are radar reflective and, therefore, can be detected and avoided by use of the weather radar. It is recommended that the aircraft be steered through the areas of least radar return to avoid the most turbulent areas.

17.2.1 Instrument Takeoff

1. Line up visually with centerline of runway.
2. Course indicator — Rotate Heading Marker to Runway Azimuth or Departure Heading, As Appropriate.
3. IFS — Adjust Indices to 10° Climb Attitude.
4. Radio altimeter — Set As Desired.
5. Hold brakes and advance throttle to TAKEOFF THRUST.
6. Check engine instruments and recheck flight instruments.
7. Anti-icing — Climatic. If icing conditions are expected, turn on engine inlet and wing and ram-air inlet anti-icing systems.
8. Release brakes.
9. Maintain directional control. Use whatever outside reference is available and use course indicator for heading reference.
10. Initiate nosewheel liftoff at normal VFR nosewheel liftoff speed.
11. Rotate aircraft to establish 12° pitch attitude on flight director indicator.
12. After takeoff, maintain heading and attitude by reference to flight instruments.
13. Landing gear handle — UP After Definite Climb Is Established.

Note

There is an inherent lag in the altimeter.

14. Maintain 12° pitch attitude until climb schedule is established.

17.2.2 Instrument Climb

1. Do not turn or bank aircraft below 500 feet.
2. Limit angle of bank to 30°.
3. If low-altitude maneuvering is required, do not exceed climb speed.

CAUTION

Do not attempt to use split reference by looking for ground references while flying at low level during instrument conditions. Avoid disorientation by not allowing attention to be diverted from the cockpit instruments.

17.2.3 Instrument Cruising Flight. Use normal techniques and procedures for instrument cruising flight.

17.2.4 Radio Navigation Equipment. Tacan, ADF, and VOR are provided for en route navigation and provide adequate reception.

17.2.5 Holding. An airspeed of 200 KIAS is recommended for holding at all gross weights. This speed provides good handling characteristics with only a slight increase of fuel consumption over that required for maximum endurance.

17.2.6 Instrument Letdowns

17.2.6.1 Jet Penetrations and Low Approaches. Jet penetrations provide high-speed and high rate-of-descent letdown from altitudes to a point where a VFR approach or an instrument approach can be made. Before descent, the pilot will obtain the latest weather information at the destination and at the alternate destination, if required. Before commencing the penetration, the pilot will make the following checks:

1. Reset the altimeter (as required).
2. Select anti-ice as required.
3. Squawk ATC transponder mode as directed by approach control.
4. Check minimums for type approach and missed-approach procedure; also check course, distance, and fuel required to proceed to alternate destination.
5. For a tacan approach, crosscheck the azimuth pointer against the compass heading or by using the ADF homer (if one is located at the destination).

At penetration fix, reduce thrust to 75-percent rpm and establish penetration airspeed. Start penetration turn after descending to published altitude and limit bank angle to 30°. After level-off at minimum penetration altitude, reduce airspeed to 175 KIAS and use speedbrake as required. (See Figures 17-1, 17-2, and 17-3 for typical penetrations and approaches.)

CAUTION

Fuel indication may vary and be in error as much as 200 pounds because of changes in the aircraft attitude during approach.

17.2.7 Missed Approach

1. Throttles — As Required.
2. Speedbrake — IN (if extended).
3. Airspeed — Minimum of V_{REF}.
4. Wing flap handle — Approach (66 percent).
5. Landing gear handle — UP (after positive rate of climb established).
6. Wing flap handle — UP.

If departing pattern:

7. Complete After Takeoff Checklist.

17.3 ICE AND RAIN

Rain without icing conditions presents no particular problem other than restricted visibility. The aircraft is equipped with windshield wipers for use during ground operation, takeoff, or landing.

Note

In heavy precipitation or turbulence, place IGNITION switch at CONTINUOUS.

The aircraft is not certified for flight into known icing conditions. Flightcrews should not accept clearance that place the aircraft in icing for prolonged periods. If it becomes obvious that an ATC clearance is placing the aircraft in icing conditions, every consideration should be given to declaring an emergency. If icing conditions are encountered, the aircraft shall not be operated in such conditions except to transit or exit the icing conditions, and then only for the minimum time necessary.

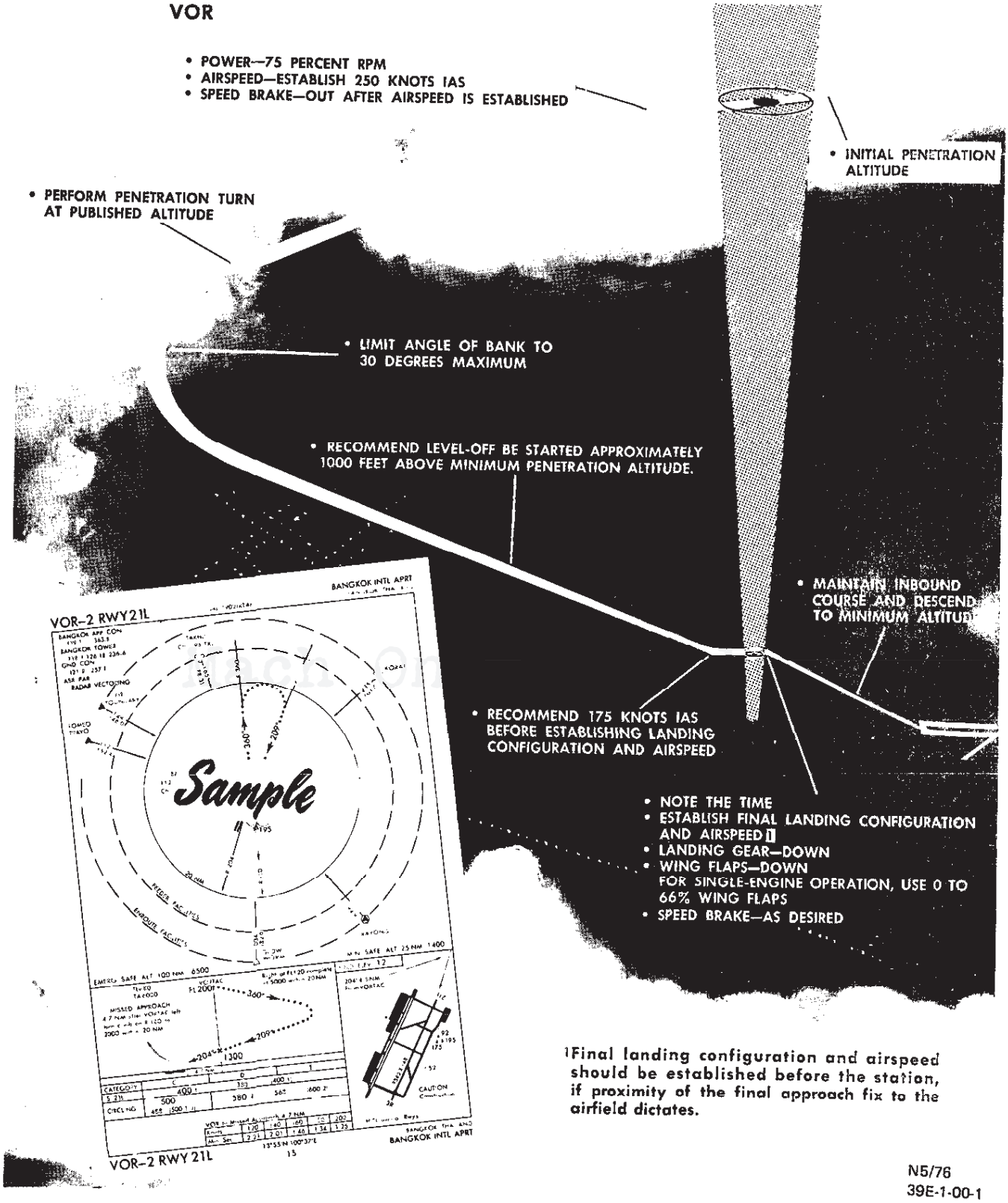
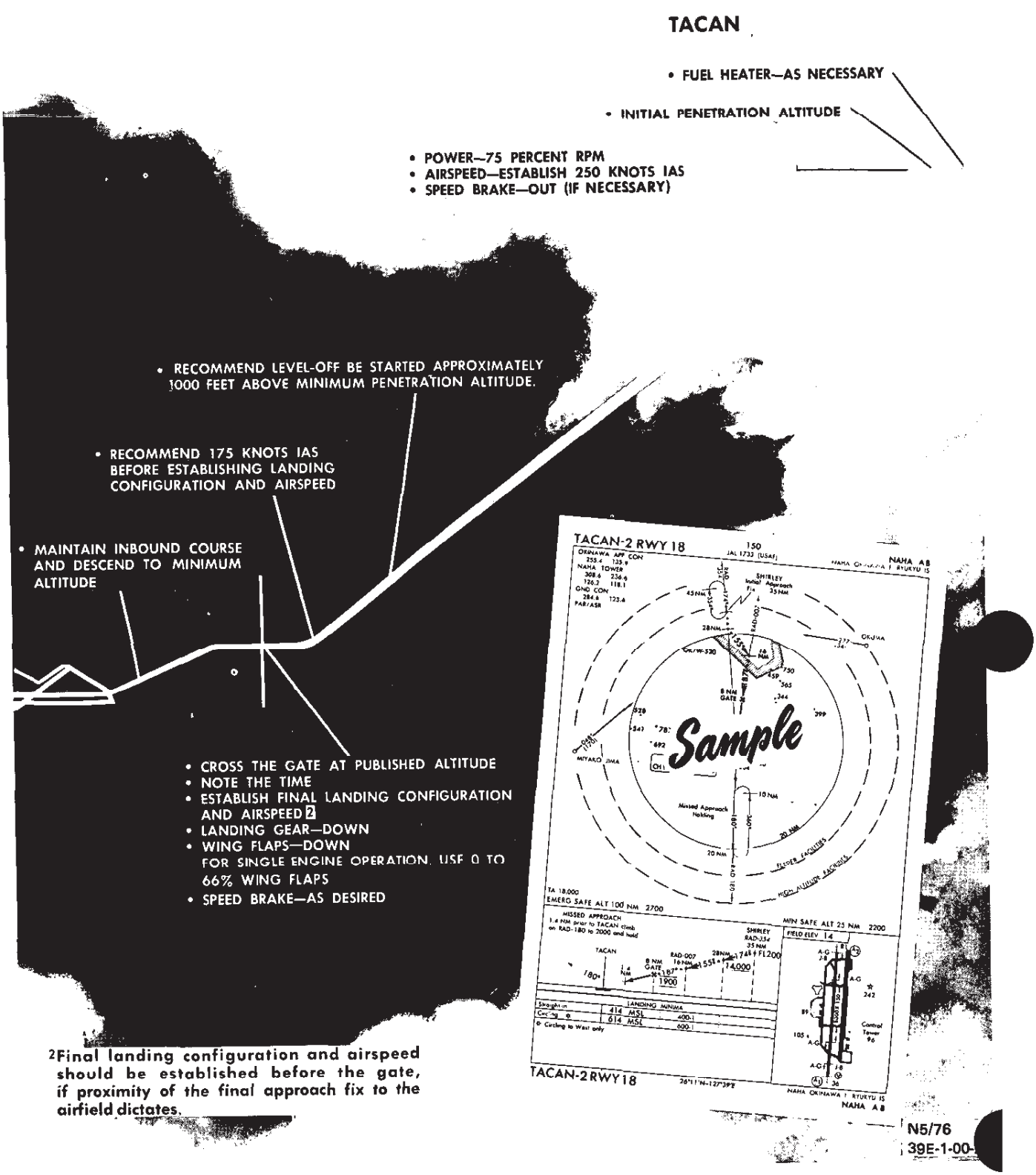


Figure 17-1. Jet Penetration and Low Approach (Sheet 1 of 2)



2Final landing configuration and airspeed should be established before the gate, if proximity of the final approach fix to the airfield dictates.

Figure 17-1. Jet Penetration and Low Approach (Sheet 2 of 2)

Note

All turns are standard rate but not exceeding 30 degrees.

(TYPICAL)

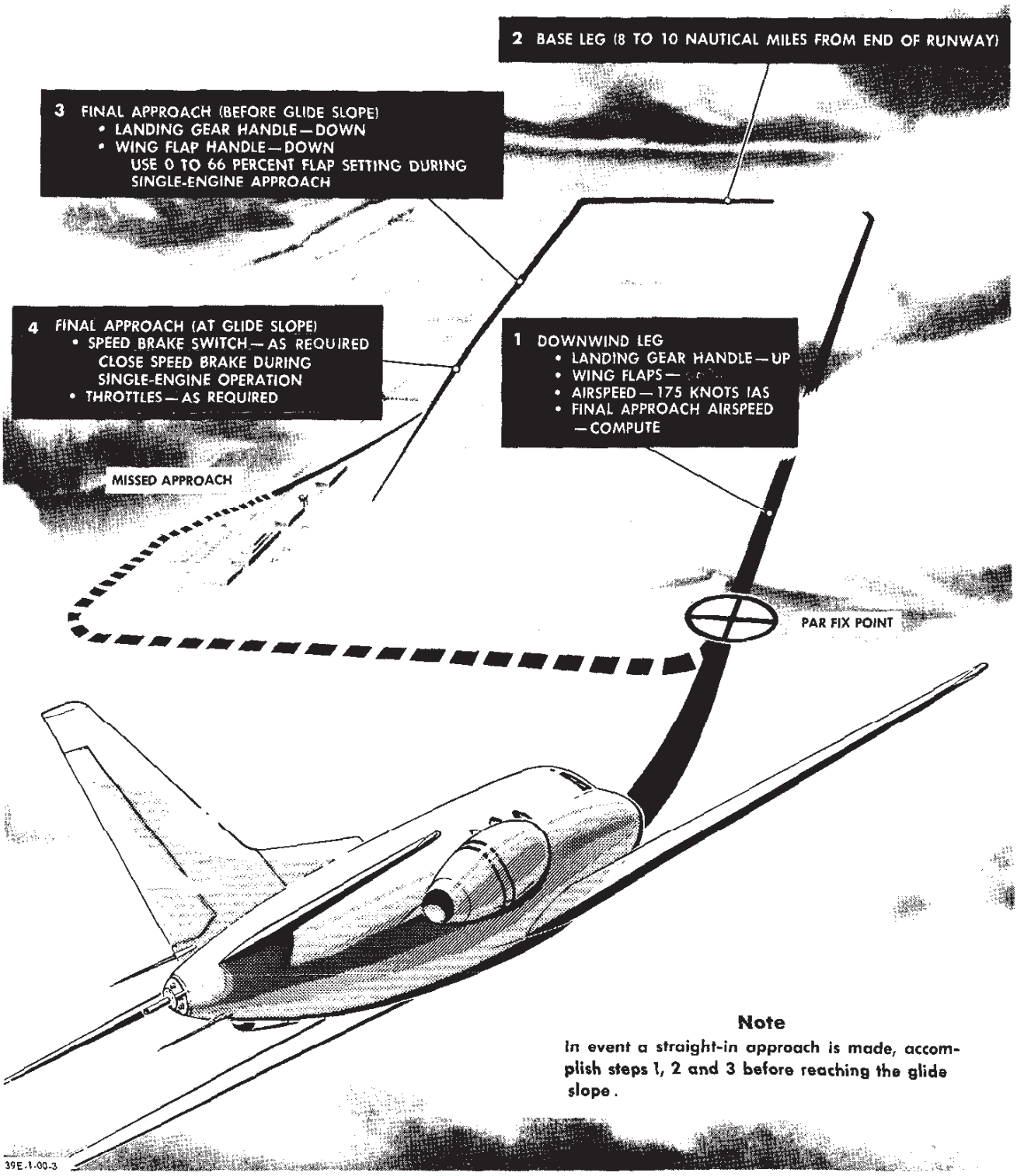


Figure 17-2. Precision Approach Radar — Typical

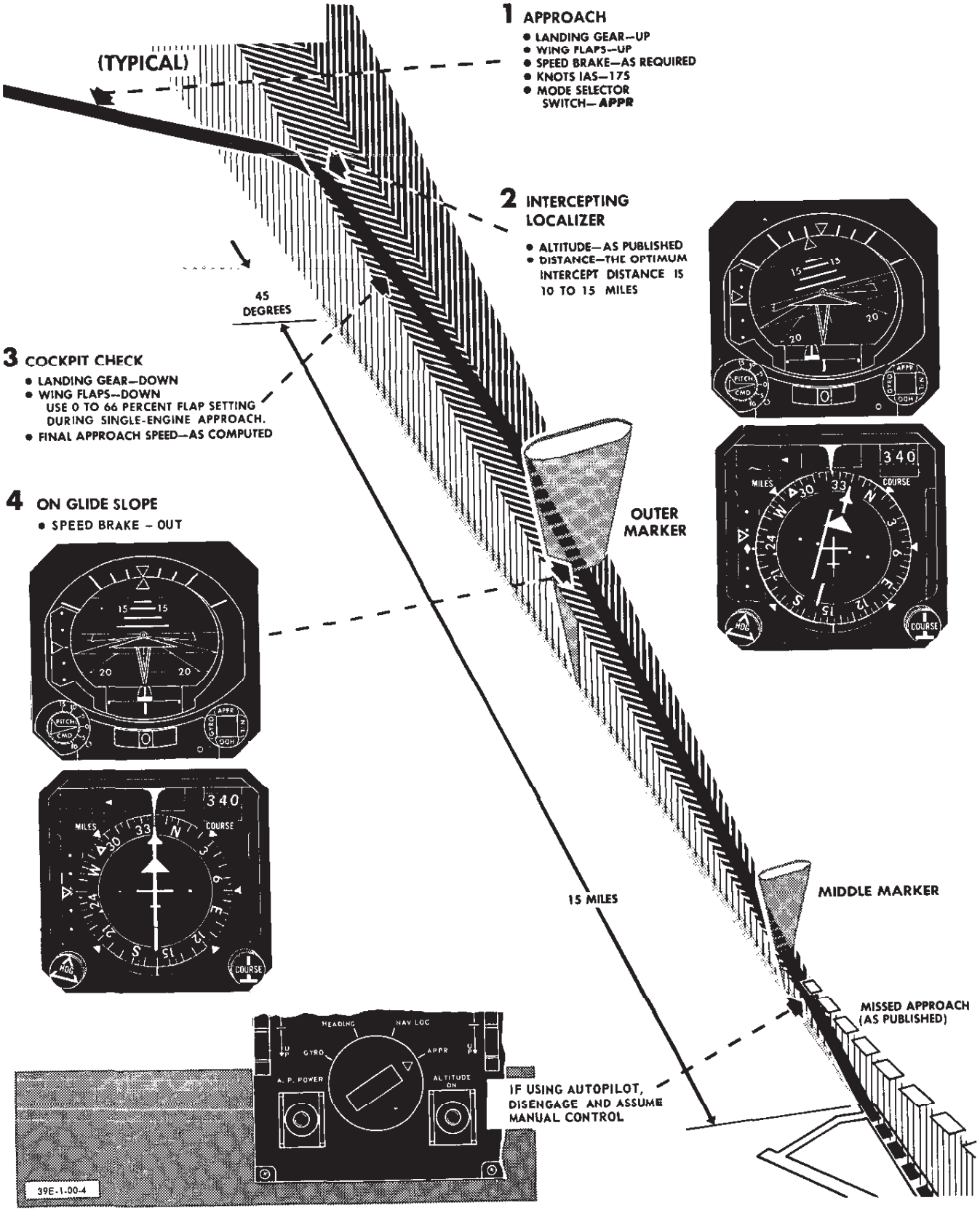


Figure 17-3. ILS Approach — Typical

Note

Pilot reports of light to moderate icing for other aircraft may actually be moderate to severe for the CT-39.

Icing of the engine inlet area is possible during operations in weather with temperatures near the freezing point. The aircraft engine inlet duct and guide vane anti-ice systems should be used whenever the ambient air temperature is in the range of +5 °C (41 °F) to -20 °C (-4 °F) with visible moisture or if ice is noted. It is recommended that the ram-air inlet anti-ice system be used only when visible moisture is present and the ambient air temperature is between +5 °C (41 °F) and -5 °C (23 °F). Fuel heater operation should be initiated if fuel temperature is 0 °C (32 °F) or below, or if the fuel block caution light is illuminated.

When at a constant throttle setting with no mechanical difficulties, a decrease in revolutions per minute with drop in EPR can indicate engine icing. EGT can be expected to increase if ice build-up is allowed to become heavy, increased EGT can precede the onset of compressor stall. Should ice actually form at the air inlet to the engine because the anti-icing system was not turned on soon enough, the throttle on the affected engine should be reduced as much as possible, but not below 75-percent rpm, as soon as the anti-icing system has been turned on. This will minimize the danger of internal damage to the engine from ingested ice.

During takeoff into fog or low clouds when temperatures are at or near freezing, the engines will be subject to icing. If fog is present or the dewpoint is within 4 °C (7 °F) of the ambient air temperature, engine icing can occur without wing icing. If possible, takeoffs should be avoided whenever the temperature is between +5 °C (41 °F) and -10 °C (14 °F). If takeoff must be made under these conditions, the engine anti-icing system should be turned on immediately before takeoff and an EPR or 1.5 percent maintained.

Fuselage or engine inlet ice may be dislodged and ingested by the engines, causing damage or complete loss of the associated engine. If ice has accumulated on the aircraft, the crew should be prepared for a sudden and catastrophic engine failure at any time.

The aircraft can be operated in areas of ice crystals sometimes encountered as high altitude cirrus clouds, without encountering any engine problems. However, the pitot system may malfunction from ice crystal ingestion and give erroneous airspeed indications. If this situation is suspected, fly thrust settings, configurations, and altitude until above-freezing temperatures are encountered and indications return to normal.

If icing conditions are encountered at freezing atmospheric temperatures, immediate action should be taken as follows:

1. Change altitude rapidly by climbing or descending in layered clouds, or change course as appropriate to exit or avoid areas of icing.

2. Maintain indicated airspeed/Mach as high as practical to minimize time/airframe exposure to icing conditions. Airspeed should not be reduced to less than slat opening speed to prevent the slats from freezing in the open position. A minimum of 1.5-percent EPR shall be maintained and, if necessary, the landing configuration should be achieved prior to descending through icing to avoid retarding throttles below 1.5-percent EPR in such conditions. Airspeed should be increased during flight approach, and landing under icing conditions, and angle of bank should be minimized because of the associated loss of lift. When feasible, a straight-in approach should be made.

WARNING

- Any ice accumulation can increase the aircraft stall speed as much as 30 percent.

- Any ice accumulation will decrease the stall angle of attack and render the stall warning system unreliable. Stall may occur before the onset of stick-shaker.

3. Monitor EGT closely, reducing engine revolutions per minute as necessary to prevent excessive exhaust temperatures. For additional information, refer to paragraph 17.4.1, Engine Surge and Flameout Caused by Adverse Weather Conditions.

17.4 TURBULENCE AND THUNDERSTORMS

The following procedures are recommended for severe turbulent air penetration.

CAUTION

The potential danger of thunderstorms is statistical history. Never enter them by choice.

Flight through the areas of severe turbulence associated with thunderstorm activity should be avoided. Whenever marginal conditions are indicated or doubt exists as to the severity of unknown weather conditions, thorough preplanning with the weather forecaster shall be accomplished. Careful planning and good judgement shall be exercised and capabilities realized before entering or circumnavigating areas of weather activity. Use the weather radar to aid in circumnavigating cumulus buildup and areas of heavy precipitation in frontal and storm areas.

Should flight through an area of thunderstorm activity be necessary, the recommended procedures for flight in severe turbulence are:

1. Airspeed (Figure 17-4).

   A gust penetration speed of 225 KIAS or 70 indicated Mach number, whichever is less, should be established. Trim the aircraft for level flight at this speed. Severe turbulence will cause large and often rapid variations in indicated airspeed. Do not chase the airspeed.

2. Attitude.

   The attitude of the aircraft becomes the primary reference in areas of extreme turbulence. After establishing the aircraft attitude and trim setting for the target airspeed of 225 KIAS, use the attitude indicator as the primary instrument. Do not change the trim settings after the proper attitude has been established. Fly this attitude during turbulent conditions. Maintain controls as near neutral as possible to avoid exaggerating the flight conditions by overcontrolling. Do not use sudden or extreme control inputs. Extreme gusts will cause large attitude changes, but smooth and moderate use of the elevator will reestablish the desired attitude. Be very conscious of tendencies to overcontrol with the rudder at this time. Attempt to maintain neutral rudder at all times.

3. Altitude.

   Severe vertical gusts may cause appreciable altitude variations. Allow altitude to vary. Sacrifice altitude in order to maintain the desired attitude and airspeed. Do not chase the altimeter.

4. Thrust.

   Establish and maintain the thrust setting consistent with the desired penetration speed and attitude. Change thrust only in case of extreme airspeed variation.

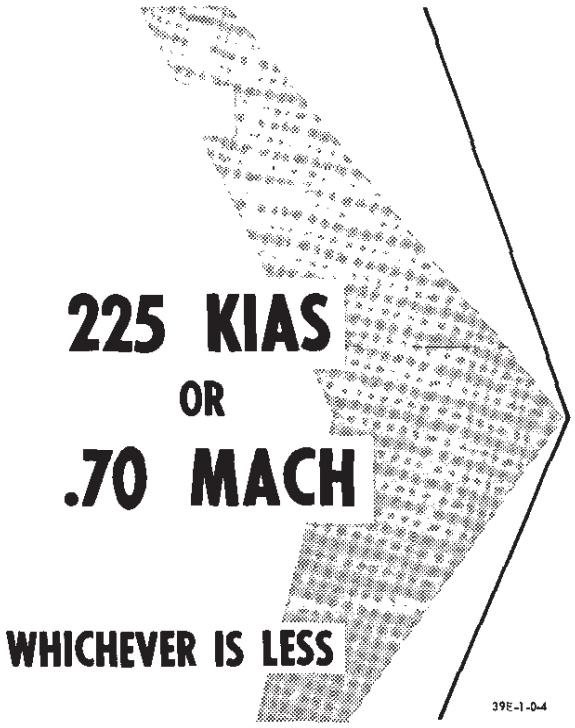


Figure 17-4. Best Penetration Speed

5. Engine ignition.

   Place the IGNITION switch in the CONTINUOUS position.

6. Autopilot.

   Disengage for severe turbulence. If the autopilot is used during mild turbulence, the altitude hold mode should be disengaged.

17.4.1 Engine Surge and Flameout Caused by Adverse Weather Conditions. The following factors, singly or in combination, can cause engine flameout:

1. Penetration of cumulus buildups with associated high liquid content.

2. Engine inlet and guide vane icing.

3. Turbulence associated with penetration can result in excessive noseup angles of attack, causing marginal engine performance.

Areas of turbulent air, hailstones, or thunderstorms should be avoided whenever possible because of the increased danger of engine flameout.

CAUTION

Flying in turbulence or hail may increase inlet duct distortion. At higher altitudes, this distortion can result in engine surge and possible flameout. However, normal airstarts may be accomplished as outlined in Part V.

Exhaust temperature and EPR gauges should be monitored continuously during weather penetration. Exhaust temperature indication alone may be too late to allow timely corrective action. The engine guide vane anti-icing system prevents the formation of ice and is not a deicer.

CAUTION

The engine guide vane anti-icing system is to be used as an anti-icing system, not a de-icing system.

When possible, icing conditions should be anticipated in advance and the engine guide vane anti-icing system should be turned on to warm up the engine air inlet. If ice has already begun to build up before the engine guide vane anti-icing system is turned on, the throttle setting should be reduced to minimize danger of internal engine damage until all ice has been broken off and ingested by the engines. When the presence of ice is no longer evident, the engines should be checked at idle before the throttles are advanced to the desired setting.

17.5 COLD-WEATHER PROCEDURES

The majority of cold-weather operating difficulties are encountered on the ground. The following instructions are intended to supplement the normal operating instructions and should be followed, when applicable, to ensure satisfactory operation of the aircraft and its systems when arctic-type weather is encountered. Extreme diligence on the part of both the ground crews and flightcrews is the answer to successful arctic operation.

17.5.1 Before Entering Aircraft

1. Have all protective covers and duct plugs removed.

2. Perform exterior inspection as outlined in Chapter 8.

3. Check to ascertain that entire aircraft is free of snow, frost, and ice. Brush off light snow and frost. Remove all ice and encrusted snow.

WARNING

Remove all snow and ice from the wings, fuselage, and tail before flight. Depending on the weight and distribution of the snow and ice, takeoff distances and climbout performance can be adversely affected. The roughness, pattern, and location of the snow and ice can affect stall speeds and handling characteristics to a dangerous degree. In-flight structural damage may also result because of the vibrations induced by unbalanced loads of accumulated ice and snow. Also, remove all snow and ice from all landing gear switches using a deicing fluid if available. False gear indications in the cockpit may result from the snow and ice buildup.

CAUTION

Do not chip or scrape ice from aircraft surface as this may cause damage.

4. Check to see that engines are free of internal ice and rotate freely by hand.

5. Ensure that hydraulic reservoir has been serviced properly. Refer to paragraph 3.1.3, Hydraulic System.

6. Remove all dirt and ice from landing gear shock struts, actuating cylinder pistons, and all limit switches. Clean struts and pistons with a rag soaked in hydraulic fluid to avoid damaging packings and seals.

7. Inspect area behind aircraft to ensure that loose snow and ice will not be blown on personnel, other aircraft, or equipment during engine start.

17.5.2 Starting Engine

1. At temperatures below 0 °F (-18 °C), external dc electrical power should always be available to ensure starting. Aircraft battery power is not adequate to provide 10-percent rpm required for normal starting after the aircraft has been

exposed to below-zero temperatures for a 24-hour period or longer.

2. Successful battery starts can be made at temperatures as low as -20 °F (-29 °C) when the engines have been shut down for periods up to 4 hours.

WARNING

If the aircraft has been "cold soaking" overnight at -40 °F (-40 °C) or below, engine starts should not be attempted without first preheating the fuel control for 10 to 15 minutes with a portable ground heater. This procedure will prevent serious fuel flow surges during engine acceleration to IDLE and eliminates the dangers of a hot start or excessive engine accelerations. JP-5 fuel has a freeze point of -55 °F (-48 °C) and is not to be used at the freezing point or lower temperatures. However, to ensure proper engine performance, operation at temperatures below -45 °F (-43 °C) should be avoided when using JP-5 fuel.

CAUTION

It is possible to obtain successful engine starts at low temperatures by placing the throttle in IDLE at 9-percent rpm. Extreme caution shall be exercised and starts should not be attempted at less than 9-percent rpm. Fuel flow and EGT should be closely monitored and the engine should be secured if the EGT or the fuel flow limits are exceeded.

Note

If starting revolutions per minute cannot be reached on the first start attempt, subsequent start attempts may produce higher revolution per minute. A minimum of 2 minutes should be observed between start attempts.

17.5.3 Warmup and Ground Check. Normally, engine warmup is unnecessary; oil pressure may be as high as 60 psi, but will return to normal after approximately 5 minutes. As soon as the engine stabilizes at idle revolutions per minute with normal gauge reading, the throttle may be advanced to TAKEOFF THRUST. However, if the engine has been "cold soaked" at temperatures of -31 °F (-35 °C) or below, a 2-minute warmup at IDLE will be required.

WARNING

Make sure all instruments have warmed up sufficiently to ensure normal operation. Check flight instruments for sluggishness during taxiing.

17.5.4 Taxiing

1. Taxi at slow speed when taxiing over rough, snow-packed surface. (Check that idle revolutions per minute is properly set as this will reduce taxi speed.)

CAUTION

The tubeless nosewheel tires may go flat while taxiing at -20 °F (-29 °C) or below. This is caused by the tires "working" in the ruts of the snow-packed taxiway and pulling loose around the tire bead seal. To prevent this, heat should be applied to the nosegear tires for 15 to 20 minutes before taxiing or tubes should be installed for continuous low-temperature operation.

2. When the nosewheel is centered, the steering system is designed to provide a less rigid or sudden steering control. This is to provide the optimum steering on dry, smooth surfaces and to prevent small rudder deflections while taxiing or during takeoff from causing abrupt directional changes. However, when taxiing on icy, rutted, or rough snow-packed surfaces, the lack of this positive, hard control may be noticeable and extra precaution shall be exercised to maintain good steering control during these conditions.

3. By taxiing on the side of the taxiway off the crown, good steering can be maintained with the nosewheel out of the centered position.

4. Increase distance between aircraft while taxiing at subfreezing temperatures to ensure safe stopping distance and to prevent icing of aircraft surfaces by melted snow and ice picked up by the jet blast of a preceding aircraft.

17.5.5 Before Takeoff. Make normal full-power engine check. However, if field conditions make this

impossible, final checks shall be made during the first part of the takeoff roll.

CAUTION

Takeoff from a runway covered with a considerable accumulation of wet snow or slush is not recommended. Accumulation of heavy slush and subsequent freezing in the right wheelwell may cause a partial obstruction of aileron movement.

17.5.6 Takeoff. During low-temperature takeoff, the aircraft acceleration and performance are improved, and the landing gear handle shall be placed in the UP position as soon as there is definite climb established to avoid exceeding the gear-down limit speed.

17.5.7 After Takeoff. After takeoff from a wet snow-covered or slush-covered field, operate landing gear through several complete cycles to prevent gear freezing in retracted position. (Considerably slower operation of landing gear may occur in cold weather because of stiffening of all lubricants.) Also, cycle wing slats by varying airspeed or apply g to prevent their freezing.

17.5.8 Landing. Use slippery-runway landing procedures in Chapter 8.

Note

- Aerodynamic braking is particularly ineffective during cold temperatures when the idle thrust from the engines is high. Braking should be commenced as soon as the nosewheel is on the runway since braking is good at high speed when landing on snow-packed runways.
- The landing roll can be substantially reduced by shutting one engine down after the aircraft is in the three-point attitude on the runway.
- During ground operation of the thrust reversers, as during operational checks, one-half of the engine thrust is being deflected downward. If the ramp, taxiway, or runway is covered with rain, slush, snow, or ice, the engine exhaust (while in reverse thrust) will blow the slush forward and up against the wing and wheelwell area.

17.5.9 Before Leaving Aircraft

1. Whenever possible, leave the aircraft parked with full fuel tanks.
2. Check that battery is removed when aircraft is parked outside at temperatures below -20 °F (-29 °C) for more than 4 hours.
3. Leave pilot sliding window partially open to allow air circulation within aircraft to decrease windshield frosting.
4. Close entry door.
5. Check that proper protective covers are installed on aircraft.

17.6 HOT-WEATHER AND DESERT PROCEDURES

Hot-weather and desert procedures differ from normal procedures when high temperatures coupled with blowing sand and dust are encountered. Extreme precaution shall be exercised by both the ground and flightcrews to prevent damage to the aircraft systems during desert operations. Proper protection and inspection of the aircraft while it is on the ground and observance of the precautions covered in this part will ensure the most successful operation.

17.6.1 Before Entering Aircraft

1. Cool pressurized compartments with a portable air conditioner if one is available.
2. Remove all protective covers and dust plugs.
3. Inspect tires and shock struts for proper inflation.
4. Clean dust and sand from struts, hydraulic pistons, and limit switches.
5. Inspect door and ground escape hatch seals for deformation and damage because of high temperatures.
6. Check wing slats for proper movement.
7. Always place aircraft in a position to avoid sandblasting the equipment and personnel during engine starts and runup.
8. Check intake ducts and remove any accumulation of dust and sand.

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17.6.2 On Entering Aircraft

1. Check instruments and electrical equipment for excessive moisture from high humidity and groundheat them if necessary to dry them.

2. Check cockpit for excessive accumulations of dust or sand.

Note

If the aircraft is exposed to direct sun or high temperatures, the windshield overheat caution light may come on.

17.6.3 Ground Operation

1. Expect engines to accelerate to idle more slowly than on a normal or cold day.

2. Minimize duration of engine ground operation.

3. Limit use of brakes during taxiing to prevent overheating.

4. Keep sufficient distance between aircraft during taxiing to prevent sand and dust from being blown into engines.

17.6.4 Takeoff. Delay rotation if the takeoff roll is not critical until reaching takeoff speed to provide positive control and a higher initial rate of climb.

WARNING

Takeoff planning is of prime importance when operating from a marginal-length runway with high ambient temperatures. (Refer to appropriate charts in Part XI.)

1. Delay nosewheel liftoff if takeoff roll is not critical until reaching takeoff speed to provide positive control and a higher initial rate of climb.

2. Be prepared for sudden gusts of wind during takeoff.

17.6.5 Approach and Landing

1. Calculate recommended approach speed carefully and maintain recommended indicated approach and touchdown speeds. Groundspeed will be higher than normal because of the high outside air temperature.

2. Anticipate a long landing roll.

3. Use all available runway for stopping to avoid overheating wheelbrakes.

17.6.6 Before Leaving Aircraft

1. Make sure that protective covers are installed on pitot heads and intake and exhaust ducts.

2. Close entry door and cockpit window.

Note

No unusual problems should develop if the aircraft is kept closed during high-temperature periods.

3. Tie the aircraft down if a possibility of windstorms exists.

Note

Tiedown fittings are stored in the baggage compartment.

PART VII

Communication Procedures

Chapter 18 — Communication Procedures

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CHAPTER 18

Communication Procedures

Not applicable to this aircraft.

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PART VIII

Weapon Systems

Chapter 19 — Weapon Systems

Mach One Manuals

Mach One Manuals

CHAPTER 19

Weapon Systems

Not applicable to this aircraft.

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PART IX

Flightcrew Coordination

Chapter 20 — Flightcrew Coordination

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CHAPTER 20

Flightcrew Coordination

20.1 INTRODUCTION

The duties of the pilot/crew team are necessarily integrated and each shall support and contribute to the performance of the other. In this chapter, specific responsibilities are delineated; however, in case of aircraft system malfunction, emergency, or unfamiliar circumstances where assistance is desired, cooperation and initiative become paramount. The PIC is responsible for the successful completion of any mission assigned to his aircraft.

20.1.1 Flightcrew Requirements

20.1.1.1 Basic Crew, Transport Mission

1. PIC
2. Copilot (third pilot or above)
3. Crewman.

Note

- The requirement for a crewman may be waived by the commanding officer. This authority may not be delegated.
- PIC/copilot designations are independent of PAC and PNAC.

20.1.1.2 Functional Checkflight Crew

1. PIC
2. Copilot (T3P or above for profiles A and B; pilot trainee or above for other profiles)
3. Crewman (as appropriate).

20.1.1.3 Training Crew

1. PIC
2. Copilot/trainee.

Note

At the commanding officer's discretion, a T2P may be assigned as PIC on flights not carrying noncrewmembers when accompanied by a T3P or higher qualified copilot.

20.2 DESCRIPTION OF AIRCREW POSITIONS

20.2.1 Pilot in Command. The PIC shall be in command of the aircraft and is responsible for the safe and orderly conduct of the flight from preflight planning through postflight inspection.

The authority and responsibility of the PIC for the flight are independent of the presence of other persons senior to him in the crew or among the passengers except as stated in Article 0823, U.S. Navy Regulations.

The PIC will be thoroughly familiar with this manual, squadron, SOP, and all other pertinent directives from higher authority. At the discretion of the PIC, the crew duties and responsibilities outlined in the following paragraphs may be reassigned as appropriate within the crew. The PIC has the authority to delay or discontinue a flight when, in his opinion, conditions are unsafe for starting or continuing a flight. The PIC shall use discretion in each case of adverse weather or emergencies encountered when deciding if the copilot is capable of executing a safe departure or approach.

20.2.2 Copilot. The copilot is second in command and is responsible for assisting the PIC in the performance of his duties and such other duties as may be assigned. For purposes of all that follows, the AC is responsible for all copilot duties when not in physical control of the aircraft.

In the event of disability of the PIC during flight, the copilot with the highest designation shall take command of the flight and assume the authority, duties, and responsibilities of the AC to the next en route station or to a closer alternate as the situation warrants.

The PNAC shall constantly monitor all maneuvers being performed by the pilot, calling to attention any deviations from normal operations. When, in the PNAC's opinion, the flight is bordering on unsafe conditions, it is essential that the PAC be prompted regardless of designation or rank.

20.2.3 Crewman. The crewman is the individual most directly associated with the aircraft load, passengers, and material status. The crewman is charged with the safety and welfare of all noncrew personnel on board and is responsible to ensure the aircraft is properly secured and ready for the next leg of the mission.

20.2.4 Specific Aircrew Responsibilities by Flight Phase

20.2.4.1 Flight Planning

1. PIC — The PIC shall be responsible for ensuring that all phases of flight planning are completed prior to the brief. This includes forecast weather, NOTAMs, en route fuel logs, and any other preflight planning appropriate to the flight. He will review and brief minimum landing ground roll distances when operating to fields with less than 8,000 feet of runway.

2. Copilot — The copilot shall prepare all required flight planning documents for presentation, review, and approval by the PIC prior to flight.

3. Crewman — The crewman need not be involved in flight planning except as required in the preparation of the aircraft for flight. He should, however, be present at the mission planning brief.

20.2.4.2 Mission Planning. Mission planning shall be accomplished and briefed by the crew during the flight planning phase of the flight.

20.2.4.3 Preflight

1. PIC — The PIC shall ensure that a thorough preflight inspection has been accomplished. He should personally conduct a visual inspection even if a preflight has been completed by the copilot or crewman. The PIC shall confer with the crewman over the proposed aircraft load.

2. Crewman — Preflight responsibilities when aircrew are assigned are as follows:

   a. Verify the fuel load and obtain any special instructions from the PIC before flight.

   b. Check for previous discrepancies, noting the corrective action taken.

   c. Preflight the aircraft in accordance with current instructions.

   d. Provision the aircraft in accordance with current instructions/squadron SOP.

   e. Perform the loading and securing of cargo.

20.2.4.4 Start. Engine start shall be conducted in accordance with the normal procedure section of this manual as amplified by individual SOP.

1. PIC/Copilot — Both pilots will adhere to published starting procedures and, whenever possible, utilize a ground power unit. When not available and a battery start is required, use of off/override (if installed) procedures are recommended when starting the second engine (see immediate second engine battery start procedures). Starting either engine while taxiing is not recommended.

2. Crewman — The crewman should monitor all start evolutions and assist the pilots as directed. A crewman should be stationed as a fire watch at the start of the first engine if ground support personnel are not available.

20.2.4.5 Taxi/Takeoff. This phase is critical to flight safety and demands the highest standards of crew coordination. There should be no superfluous conversation or activity in the cockpit at this time. Each member of the crew must be thoroughly briefed, with no question as to duties. Takeoff briefs must be tailored to existing conditions including weather, hazards, runways, equipment, elevation, departure procedures, performance standards, and emergency procedures. Specific briefings are not included here but are suggested for inclusion in squadron SOP/Stan notes, including all the foregoing data. Briefs must be concise and thorough. Specific crew duties for this phase are as follows.

1. PIC — The PIC will monitor or control the taxi/takeoff evolution regardless of who is at the controls. The PIC should not accept clearance onto the runway until all appropriate checklists are completed and the crew is ready. If, at anytime, a crewmember should become rushed, behind the aircraft or unready for release, the PIC

must slow the pace until all crewmembers are ready for takeoff.

2. Copilot — The copilot shall assist or conduct the takeoff evolution as briefed by the PIC. In either case, the pilot providing the backup shall ensure that all appropriate checklists are completed prior to the pilot advancing power for takeoff. Specific responsibilities for backup of the pilot may be delineated in squadron SOP/Stan notes. The following should be accomplished as a minimum:

   a. When power levers are set to takeoff thrust, observe throttle position, computed EPR, revolutions per minute, and other engine instruments for proper indications before commencing roll.

   b. Observe proper operation of bleedstraps.

   c. Observe normal operation of all aircraft/engine components during initial takeoff roll.

   d. Ensure nosewheel steering properly deselected at briefed airspeed. Monitor all engine instruments and caution lights for faulty indications or warnings. Be prepared to call "Abort" prior to decision speed if required.

   e. Call "Go" at computed decision speed (V_1), ensuring that the pilot places both hands on the yoke indicating that the aircraft is committed to flight and beyond abort capability. The copilot may then readjust the throttles for the pilot in case power settings require fine tuning .

   f. Call "rotate" at the computed speed. Ensure the pilot applies a positive rotation to the aircraft.

   g. Call "V_2" at the computed speed. Ensure the landing gear are retracting and that all aircraft/engine parameters are set for the climbout phase.

   h. Manage communication and navigation once the takeoff evolution is safely accomplished.

   i. Maintain constant vigilance for other aircraft.

3. Crewman — The crewman shall ensure all passengers receive a takeoff safety brief, are properly strapped in, and are positioned for center-of-gravity considerations. During the takeoff evolution, the crewman shall remain strapped into a seat until the aircraft has climbed through a safe altitude (at least 800-feet AGL) on departure. Ensure security of personnel/cargo and report to PIC. At that time, the crewman may come forward and sit in the jump seat to assist in forward crew vigilance if directed by PIC.

20.2.4.6 Departure

1. PIC — The PIC shall ensure that the departure is flown in accordance with Part III procedures. Engine performance and adherence to departure instructions will be monitored. Autopilot may be used at the pilot's discretion when above 1,500-feet AGL. The PAC, in addition to monitoring aircraft performance, should be looking outside for traffic. The PAC should not be selecting radio frequencies, altitudes, searching for publications, or performing other duties delegable to the other pilot.

2. Copilot — The copilot, when not actually in control of the aircraft, is charged with navigation, communication, monitoring the flight plan or assigned clearance, and selecting the proper publications and charts required for the departure. It should be briefed that one pilot should maintain forward lookout at all times.

3. Crewman — The crewman shall assist the flightcrew as directed by the PIC in the departure. The most useful position is in the jump seat.

20.2.4.7 En Route

1. PIC — The PIC shall ensure that the en route phase of the flight is conducted with maximum focus on the comfort and safety of the passengers. The PIC selecting any change in aircraft configuration or systems should notify the PNAC before effecting the change.

2. Copilot — The copilot, when not in physical control of the aircraft (PNAC), is responsible for the following during the en route stage:

   a. Safe navigation of the aircraft

   b. Proper communication with controlling facilities

   c. Backing up the PAC as required

   d. Ensure the PAC complies with ATC instructions.

e. Maintain fuel logs and other flight planning data as directed by the PIC.

f. Monitor the basic airwork being accomplished by the PAC and provide warning when the aircraft is about to exceed assigned flight parameters.

3. Crewman — The crewman is responsible for periodically monitoring cockpit instrumentation, using lookout doctrine procedures, and assisting the flightcrew as required. Additionally, he shall ensure the safety, comfort, and welfare of the passengers during all evolutions of the flight.

20.2.4.8 Descent, Approach, and Landing. This phase of flight, especially approach to landing, is the most critical and demanding of all phases. Absolute adherence to the principles of good crew coordination is essential.

1. PIC — Whether performing pilot or copilot duties, the PIC will ensure that the entire descent, approach, and landing phase is conducted professionally and accurately. The PIC must maintain the "sterile cockpit" atmosphere where only essential communication is conducted within the cockpit. The PIC will ensure that a thorough descent and approach briefing is delivered by the PAC. He must supervise the entire evolution to ensure that confusion with controlling facilities is minimized and that the aircraft is flown as depicted on the approach procedure or as instructed by ATC. The PIC must recognize minor deviations and ensure they are corrected before becoming hazardous and ensure that the entire crew and passengers are ready for landing. The landing rollout is perhaps the most critical phase of flight in the T-39. The PIC, as such, must be wary for difficulties encountered after landing, such as a wet runway, braking problems, adverse winds, arresting gear, improper approach technique, and any assortment of other problem(s) that have historically caused landing mishaps in the T-39E/G models. There should be no surprises during this critical phase of flight. Each action by the PAC should be fully briefed and completely understood by the PNAC. There must be complete backup to each segment of this phase. A "standard NATOPS backup" followed by "call field in sight" is not a satisfactory instrument approach briefing and the PIC should guard against complacency in this regard.

2. Copilot — The copilot is constantly in training for PIC and should try to project into this role for each decision made in the course of the flight. This is especially true during the final stage of the flight. The copilot duties during this phase of the mission are to assist the PIC as directed in order to effect a safe and accurate instrument approach procedure and landing. However, this does not preclude independent thinking on the part of the copilot. Comments and suggestions as well as questions are central to the "multipiloted concept" and professional crew coordination.

3. Crewman — The assigned aircrew are stationed as directed by the PIC during the instrument approach and landing. After ensuring the passengers are prepared for landing, an air crewman may, at the option of the PIC, be assigned as "another set of eyes" to the jumpseat during the en route descent. Prior to landing, the air crewman shall be strapped into a seat.

20.2.4.9 Postlanding, Taxi, Shutdown, Post-flight

1. PIC — The PIC must ensure the crew avoids complacency during the postlanding phase. There have been countless mishaps involving the safe accomplishment of a flight followed by a mishap involving aircrew inattention after landing. The PIC must avoid this by maintaining his vigil until engines are shut down with all checklists and postflight procedures completed.

2. Copilot — The copilot must make the same effort to avoid postflight complacency. The copilot must remain oriented and in complete communication with the control tower as well as his PIC. Even on the ground, there should always be at least one set of eyes outside the aircraft.

3. Crewman — The crewman should assist the PIC as directed and ensure that all passengers remain strapped into their seats until the aircraft comes to a final stop. The crewman should then direct disembarkation of all passengers and offloading baggage.

20.2.4.10 Special Considerations

1. FCF — Refer to Chapter 9 for detailed description of FCF procedures. Crew coordination while accomplishing a postmaintenance checkflight

flight is exactly the same as that required for other flights. No special crew concept other than that already listed is required. Emphasis on briefing before, during, and after the flight must be made because of the special nature of the FCF flight.

2. Training flights — Training syllabus requirements and procedures are listed in Parts II and III of this manual. Crew coordination principles already discussed apply to the training mission as well as other types of missions. There are, however, some technique items to emphasize that can make the T-39 training evolution go smoother and safer.

   a. Briefings — When briefing a nondesignated T-39 pilot (newly assigned pilots should have completed contract T-39 simulator training at an accredited training school), all phases of the training evolution should be thoroughly covered to preclude misunderstanding on the part of the trainee. Syllabus sheets/checkoff lists are recommended.

   b. Start, takeoff, and flight evolution — The crew for any training flight involving a non-NATOPS qualified pilot must include an instructor pilot. Nonrequired personnel should not be onboard during training evolutions (exception: additional trainees). Adequate time should be allocated for nonqualified trainees to ensure that the flight is not rushed. It is essential that the instructor pilot emphasize the principles of crew coordination already addressed to impress upon the trainee that the T-39 is a multipiloted aircraft.

3. Emergencies and crew coordination — Reacting correctly and promptly during emergency situations can only be guaranteed if the proven concepts of crew coordination are followed routinely. The advantage of having two qualified pilots and an air crewman must never be taken for granted. Successful handling of emergency procedures is accomplished by a flightcrew, not a single pilot.

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PART X

NATOPS Evaluation

Chapter 21 — NATOPS Evaluation

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CHAPTER 21

NATOPS Evaluation

21.1 CONCEPT

The standard operating procedures prescribed in this manual represent the optimum method of operating CT-39E/G aircraft. The NATOPS evaluation is intended to evaluate compliance with NATOPS procedures by observing and grading individuals and units. This evaluation is tailored for compatibility with various operational commitments and missions of both Navy and Marine Corps units. The prime objective of the NATOPS evaluation program is to assist the unit commanding officer in improving unit readiness and safety through constructive comment. Maximum benefit from the NATOPS program is achieved only through vigorous support of the program by commanding officers as well as flight crewmembers.

21.2 APPLICABILITY

The NATOPS evaluation shall be administered annually to pilots.

21.3 DEFINITIONS

The following terms used throughout this chapter are defined as to their specific meaning within the NATOPS program.

1. NATOPS evaluation — A periodic evaluation of individual flight crewmember standardization consisting of an open-book examination, a closed-book examination, an oral examination, and a flight evaluation.

2. NATOPS reevaluation — A partial NATOPS evaluation administered to a flight crewmember who has been placed in an Unqualified status by receiving an Unqualified grade for any of his ground examinations or the flight evaluations. Only those areas in which an unsatisfactory level was noted need be observed during a reevaluation.

3. Minor discrepancies and/or omissions — Minor discrepancies and/or omissions, as may be referred to in the grading criteria, are defined as those that would not adversely affect the successful completion of the mission or jeopardize the safety of the crew and/or equipment.

4. Momentary deviations — Very brief deviations from the tolerances set forth in the grading criteria, that will not be considered in marking, provided the evaluee is alert in applying corrective action and the deviation does not jeopardize the safety of the aircraft or crew and does not exceed the limitations prescribed for a Conditionally Qualified grade. Cumulative momentary deviations, however, will result in downgrading.

5. Qualified (Q) — That degree of standardization demonstrated by a very reliable flight crewmember who has a good knowledge of standard operating procedures and a thorough understanding of aircraft capabilities and limitations.

6. Conditionally Qualified (CQ) — That degree of standardization demonstrated by a flight crewmember who meets the minimum acceptable standards. He is considered safe enough to fly as a pilot in command or to perform normal duties without supervision, but more practice is needed to become Qualified.

7. Unqualified (U) — That degree of standardization demonstrated by a flight crewmember who fails to meet minimum acceptable criteria. He should receive supervised instruction until he has achieved a grade of Qualified or Conditionally Qualified.

8. Area — A routine of preflight, flight, or postflight.

9. Subarea — A performance subdivision within an area that is observed and evaluated during an evaluation flight.

10. Critical area — Any area or subarea that covers items of significant importance to the overall

mission requirements, the marginal performance of which would jeopardize safe conduct of the flight.

11. Emergency — An aircraft component system failure or condition that requires instantaneous recognition, analysis, and proper action.

12. Malfunction — An aircraft component or system failure or condition that requires recognition and analysis, but which permits more deliberate action than that required for an emergency.

21.4 IMPLEMENTATION

The NATOPS evaluation program shall be carried out in every unit operating naval aircraft. The various categories of flight crewmembers desiring to attain/retain qualifications in the CT-39E shall be evaluated in accordance with OPNAVINST 3710.7. Individual and unit NATOPS evaluations will be conducted periodically; however, instruction in and observation of adherence to NATOPS procedures shall be on a daily basis within each unit to obtain maximum benefits from the program. The NATOPS coordinators, evaluators, and instructors shall administer the program as outlined in OPNAVINST 3710.7. Evaluees who receive a grade of Unqualified on a ground or flight evaluation shall be allowed 30 days in which to complete a reevaluation. A maximum of 60 days may elapse between the date the initial ground evaluation was commenced and the date the flight evaluation is satisfactorily completed.

21.5 GROUND EVALUATION

The ground evaluation consists of two parts: phase I, the open- and closed-book examinations; and phase II, an oral examination. The ground evaluation shall be completed with a grade of Qualified before commencement of phase III, the flight evaluation.

21.5.1 Open-Book Examination. Questions used may be taken from the pilot question bank. The number of questions on the examination will not exceed 75 or be less than 25. The purpose of the open-book examination portion of the written examination is to evaluate pilot knowledge of appropriate publications and the aircraft. The time for this examination should not exceed 3 hours.

21.5.2 Closed-Book Examination. Questions used for the closed-book examination may be taken from the pilot question bank and shall include questions concerning normal procedures, emergency procedures, and aircraft limitation. The number of questions on the examination will not exceed 75 or be less than 25. Questions designated critical will be so marked. An incorrect answer to any question in the critical category will result in a grade of Unqualified being assigned to the examination.

21.5.3 Oral Examination. The questions may be taken from this manual and drawn from the experience of the instructor/evaluator. Such questions should be direct and positive and should in no way be opinionated.

21.6 GRADING INSTRUCTIONS

21.6.1 Written Examination. Examination grades shall be computed on a 4.0 scale and converted to an adjective grade of Qualified or Unqualified.

21.6.2 Open-Book Examination. To obtain a grade of Qualified, an evaluee must obtain a minimum score of 3.5.

21.6.3 Closed-Book Examination. To obtain a grade of Qualified, an evaluee must obtain a minimum score of 3.3.

21.6.4 Oral Examination. A grade of Qualified or Unqualified shall be assigned by the instructor/evaluator using the following criteria:

Qualified (Q) — Demonstrated thorough knowledge of proper cockpit procedures, both normal and emergency; was thoroughly familiar with aircraft systems.

Unqualified (U) — Indicated an obvious lack of understanding of proper cockpit procedures, both normal and emergency; was unfamiliar with aircraft systems.

21.7 FLIGHT EVALUATION

The flight evaluation is designed to measure with maximum objectivity the degree of standardization demonstrated by pilot and crewmembers. It is not intended to measure the proficiency and/or ability of those evaluated beyond a point necessary to ensure safety of flight. Within reasonable limits, any individual evaluated should be able to attain a Qualified grade based upon demonstrated knowledge without regard to special proficiency or ability.

The process uses three integrated parts: the flight evaluation worksheet OPNAV Form 3510/11, flight evaluation criteria, and the evaluation report OPNAV Form 3710/7 (Figure 21-1).

NATOPS EVALUATION REPORT
OPNAV 3710/7 (4-90) S/N 0107-LF-009-8000 **REPORT SYMBOL OPNAV 3710-21**

NAME (Last, first, initial)	GRADE	SERVICE NUMBER
SQUADRON/UNIT	AIRCRAFT MODEL	CREW POSITION
TOTAL PILOT/FLIGHT HOURS	TOTAL HOURS IN MODEL	DATE OF LAST EVALUATION

NATOPS EVALUATION

REQUIREMENT	DATE COMPLETED	GRADE		
		Q	CQ	U
OPEN BOOK EXAMINATION				
CLOSED BOOK EXAMINATION				
ORAL EXAMINATION				
*EVALUATION FLIGHT				
FLIGHT DURATION	AIRCRAFT BUNO	OVERALL FINAL GRADE		

REMARKS OF EVALUATOR/INSTRUCTOR

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☐ CHECK IF CONTINUED ON REVERSE SIDE

GRADE, NAME OF EVALUATOR/INSTRUCTOR	SIGNATURE	DATE
GRADE, NAME OF EVALUEE	SIGNATURE	DATE

REMARKS OF UNIT COMMANDER

RANK, NAME OF UNIT COMMANDER	SIGNATURE	DATE

*WST, OFT, COT, or cockpit check in accordance with OPNAVINST 3710.7 (effective edition)

Figure 21-1. NATOPS Evaluation Report

The flight evaluation worksheet is designed to enable the instructor/evaluator to record pertinent information for the purpose of a comprehensive postflight reconstruction and critique. Following completion of the evaluation, data from the worksheet is compared with the grading criteria to determine the numerical/adjectival equivalent for entry on the evaluation report form.

21.8 FLIGHT EVALUATION GRADING CRITERIA

Only those subareas observed or required will be graded. The grade assigned for a subarea shall be determined by comparing the degree of adherence to standard operating procedures with adjectival ratings listed below. Momentary deviations from standard operating procedures should not be considered as unqualifying, provided such deviations do not jeopardize flight safety and the evaluee applies prompt corrective action.

Note

All areas and subareas are critical.

21.8.1 Critique. The critique is the terminal point in the NATOPS evaluation and will be given by the instructor/evaluator administering the check. The critique involves processing data collected and oral presentation of the evaluation worksheet. Deviations from standard operating procedures will be covered in detail, using all collected data and the evaluation worksheet.

21.9 AIRCRAFT COMMANDER, SECOND PILOT, THIRD PILOT LOCAL/ INSTRUMENT NATOPS EVALUATION CRITERIA

1. Predeparture

   a. *Takeoff performance computations*

      Qualified — Accurately utilized governing charts to ascertain EPR, V_1, V_R, and V_2.

      Conditionally Qualified — Computed takeoff performance with minor errors not serious enough to affect safety of flight.

      Unqualified — Mistakes if followed would jeopardize safety of flight.

   b. Fuel planning

      Qualified — Complied with and was familiar with governing directives concerning required fuel and fuel computations.

      Conditionally Qualified — Omitted minor items affecting safety of flight.

      Unqualified — Not familiar with governing directives and/or made erroneous computations that could affect safety of flight.

   c. Weight and balance

      Qualified — Thoroughly familiar with weight limitations outlined in Part I of the manual.

      Conditionally Qualified — Minor errors in knowledge of limitations that would not affect the safe completion of the mission.

      Unqualified — Did not meet the criteria of Conditionally Qualified.

   d. Preflight inspection

      Qualified — Completed thorough aircraft inspection. Inspected data sheets of 10 previous discrepancies and preflight inspection form.

      Conditionally Qualified — Completed aircraft inspection with omissions in minor areas that did not affect the safety of the proposed flight. Did not check previous discrepancy sheets of preflight inspection form.

      Unqualified — Failed to conduct aircraft inspection properly and omitted important items that would affect the safety of the proposed flight.

2. Runup and Taxi

   a. Starting Procedures

      Qualified — Executed checklist properly. Demonstrated a thorough knowledge of all items. Fire watch properly utilized. Started engine using the proper technique and checked engine instruments normal after start.

Conditionally Qualified — Difficulty in starting engines. Did not check engine instrument following start, fire watch not properly posted.

Unqualified — Failed to obtain start because of technique. Did not utilize fire watch. Any other item not meeting requirements of Conditionally Qualified or affecting safety of operations.

b. Taxiing

Qualified — Complied with procedures required by the manual. Taxi signal men were utilized, power and speeds regulated closely, watches were properly posted when required, all lights and taxi aids were utilized as necessary. Checklists were accomplished without omission or discrepancy. Smooth use of throttle and brakes. Proper compliance with tower instructions was demonstrated.

Conditionally Qualified — Procedures required by the manual accomplished with omissions, deviations, or discrepancies that did not either adversely affect successful completion of the mission, jeopardize safety, or cause undue delay.

Unqualified — Any omission or discrepancy that either precluded successful completion of the mission, jeopardized safety, or caused excessive delay. Taxied in rough and hazardous manner, improper use of checklist.

c. ATC clearance

Qualified — Demonstrated ability to take and read back ATC clearance correctly.

Conditionally Qualified — Utilized nonstandard radio voice procedures, but understood ATC clearance.

Unqualified — Unable to copy or understand ATC clearance.

d. Radio management

Qualified — Set up radios in accordance with those facilities needed for initial climbout and on-course navigation. Checked all communication/navigation equipment in accordance with the best means available. Operated equipment in accordance with manual and appropriate directives. Fully utilized available radios and NAVAIDs.

Conditionally Qualified — Tuned, but did not identify radio signals or use radio NAVAIDs for backup.

Unqualified — Radio equipment needed for flight not properly tuned or set prior to takeoff.

e. Crew briefing

Qualified — Thoroughly briefed crew on normal and emergency cockpit procedures, reviewed ATC clearance, assured radios were tuned to facilities associated with initial climbout and/or emergency return, aircraft performance (i.e., speeds (including V_2 and takeoff speeds)), and fuel dumping.

Conditionally Qualified — Same as Qualified but omitted some minor items in the crew briefing.

Unqualified — Omitted one or more major items from the crew briefing.

3. Takeoff (the criteria below apply to normal, short-field, and crosswind takeoffs)

Qualified — Received and acknowledged takeoff clearance. Properly positioned aircraft and completed final checks. Maintained good directional control on takeoff run. Utilized takeoff performance data. Made smooth transition to climb attitude.

Conditionally Qualified — Same as Qualified except for minor deviations in procedure and technique not detrimental to flight safety.

Unqualified — Did not receive and acknowledge takeoff clearance. Exhibited poor or unsafe technique in either directional control, liftoff, transition, or in establishing climb.

a. Emergency return (includes abort)

Qualified — Maintained positive control of the aircraft, completed emergency checklist in accordance with standard procedures, and effected smooth transition to final approach and landing. Ensured adequate briefing was given crew, passengers, and traffic controller.

Conditionally Qualified — Carried out items listed in Qualified, but with some delay in executing checks.

Unqualified — Jeopardized flight safety by failing to cope with emergency in timely fashion or used improper sequence of checks, causing confusion in cockpit and compounding the emergency.

4. Climb to level-off

a. ATC clearance execution

Qualified — Successfully accomplished the approval instrument departure or, if radar-vectored, complied with all altitudes and heading assignments. Navigated by aircraft instruments to arrive over designated fixes on ETA and assigned altitude.

Conditionally Qualified — Adhered to clearance limitations, but required repeat radio transmission. Slow to react to instructions/clearances and had difficulty establishing and maintaining desired track.

Unqualified — Proceeded without being certain of clearance or broke clearance limitations.

b. Climbout procedures

Qualified — Proper use of Climb Checklist. Aircraft checks completed and crew reports received. Maintained correct climb thrust setting by using correct EPR charts. Climb airspeed maintained within ±10 knots. Prescribed headings within ±5°.

Conditionally Qualified — Aircraft checks completed, but reports not received. Established but did not maintain correct climb power setting. Climb airspeed maintained within ±20 knots. Prescribed headings within ±10°.

Unqualified — Failed to complete Climb Checklist. Reports not received. Did not establish correct climb power. Failure to satisfy requirements listed under Conditionally Qualified.

5. Cruise Procedures

Qualified — Aircraft checks were completed. Headings were maintained within ±5°. Cruise thrust was properly set.

Conditionally Qualified — Cruise airspeed maintained within ±10 knots, heading within ±10°.

Unqualified — Failed to satisfy requirements listed under Conditionally Qualified.

6. Airwork

a. Normal and steep turns

Qualified — Demonstrated coordinated turns with smooth reversal at 260 knots at bank up to 45° utilized for 180° and 360° turns. Altitude variation should be less than ±100 feet. Angle of bank should be within 5° of that prescribed.

Conditionally Qualified — Maintained altitude within 200 feet and angle of bank within 10°.

Unqualified — Unable to meet above criteria.

b. Partial panel

Qualified — Maintained altitude within 100 feet, airspeed within 10 knots, and heading within 10 percent of amount of turn. Executed timed turns with an error of less than 10 percent of amount of turn.

Conditionally Qualified — Maintained altitude within 200 feet, airspeed within 15 knots, and heading within 15°. Executed timed turns with an error of more than 10 percent, but less than 15 percent of amount of turn.

Unqualified — Unable to meet above criteria.

c. Unusual attitudes and recovery

Qualified — Demonstrated ability to effect a successful recovery expeditiously and with a minimum loss of altitude, observing all aircraft operating limitations.

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Conditionally Qualified — Somewhat slow to comprehend aircraft position in relation to the horizon resulting in a more severe aircraft attitude before recovery was effected.

Unqualified — Unable to recover in a safe flight attitude within aircraft limitations.

d. Approach to stalls

Qualified — Initiated proper recovery technique and recovered with less than 200 feet loss of altitude.

Conditionally Qualified — Minor deviations in entering and recovery techniques. Recovered with less than 300 feet loss of altitude.

Unqualified — Improper technique and/or loss of more than 300 feet of altitude.

7. Holding

Qualified — Demonstrated a thorough knowledge of holding procedures by entering the pattern properly, maintaining correct airspeed, making the proper correction for winds, using the proper voice procedures, and adjusting the pattern to depart properly and on time.

Conditionally Qualified — Did not correct properly for winds, causing greater adjustment requirements in the turns. Used nonstandard voice procedure. Did not adjust the pattern properly to depart on time.

Unqualified — Entered the pattern by making a wrong initial or subsequent turn. Failed to maintain proper altitude or airspeed control. Did not make required voice reports.

8. MDF problems (not applicable)

9. Penetration (not applicable)

10. Approaches (the criteria below apply to PAR/ASR, ILS, ADF, VOR, and tacan approaches)

Qualified — Followed procedures as published, complied with approach control instructions. Maintained airspeed within 10 knots. Maintained assigned altitude and heading. Did not descend below minimum altitudes, as published. Demonstrated good tracking technique. Minor deviations in glideslope and alignment. Indicated time over airport within 20 seconds. Maintained field altitude minimums.

Conditionally Qualified — Followed procedures as published and complied with approach control instructions. Maintained airspeed within 10 knots. Tracking erratic and no apparent correction for wind was applied. Large deviations in glideslope and alignment, but could have completed a successful approach. Maintained field altitude minimums within +150 feet.

Unqualified — Unsafe procedures and/or failed to comply with approach control instructions. Failed to meet airspeed, altitude, and time criteria specified above. Failed to maintain field altitude minimums within the above criteria. Safety of crew and aircraft were jeopardized, landing highly improbable.

a. Missed approach

Qualified — Prepared for missed approach (to the extent that reference to published procedure was not required or did not interrupt necessary acts) timely. Correct thrust applications, raised gear and flaps properly following power, did not descend below minimums. No deviation from published procedures.

Conditionally Qualified — Not adequately prepared for missed approach (requiring excessive references to published procedure). Incorrect power application, improper sequence of power, gear, and flaps, did not descend below minimums. Satisfactorily carried out published procedures.

Unqualified — Descended below minimum, erratic use of power. Unsatisfactory completion of published procedures.

b. Low visibility

Qualified — Properly executed manual procedures for low-visibility approach, keeping within the minimums of the instrument approach procedure used to arrive over the landing area. Positive control of the aircraft maintained while maneuvering at these low altitudes with extreme caution being used not to jeopardize safety of flight by excessive angles of bank.

Conditionally Qualified — Some deviations from established procedures evidenced, but no unsafe conditions noted.

Unqualified — Failed to meet the criteria of Conditionally Qualified.

c. Transition to landing

Qualified — Entered and flew traffic pattern in accordance with the appropriate governing directives. Proper execution of Approach and Before Landing Checklists. Flew the traffic pattern within altitude and airspeed range.

Conditionally Qualified — Entered and flew traffic pattern in accordance with appropriate directives, but with deviations that did not interfere with traffic or jeopardize safety. Late calling for or completing checklist. Flew traffic pattern with minor deviations in altitude and/or airspeed range.

Unqualified — Deviated from prescribed pattern and interfered with other traffic or jeopardized safety. Failed to call for or complete Before Landing Checklist. Exceeded limitations for Conditionally Qualified.

11. VFR landing pattern

a. Normal

Qualified — Maintained positive control of speed, power, and rate of descent. Aircraft aligned with runway throughout final approach. Maintained directional control during rollout. Smooth, effective, use of brakes and nosewheel steering.

Conditionally Qualified — Had minor difficulty with transition. Handled aircraft roughly and used poor technique in either flare, landing, or rollout. Left excessive power on or made flat touchdown in first third of runway.

Unqualified — Had serious difficulty with transition. Failed to maintain positive control of aircraft during either flare, landing, or rollout. Did not touch down in first third of runway. Landed hard, jeopardized safety.

b. Crosswind

Qualified — Properly executed crosswind technique, maintaining positive directional control throughout the flare, touchdown, and rollout. Braking was used to best advantage to slow aircraft while transition to nosewheel steering gave assurances of continued positive directional control.

Conditionally Qualified — Insufficient measures were taken to correct drift on final approach or touchdown, improper technique used to maintain directional control, or aircraft landed in slight skid.

Unqualified — Safety of flight endangered by poor judgment of crosswind situation or dangerous drift existent on touchdown.

c. No-flap

Qualified — Maintained correct attitude, airspeed, and properly coordinated power settings.

Conditionally Qualified — Attitude, airspeed, and power settings correct except for minor deviations.

Unqualified — Unsatisfactory control of airspeed, attitude, and power settings resulting in a go-around.

d. Short field

Qualified — The approach was made according to the recommended procedures as outlined in this manual. Brakes were applied immediately and controlled in order to maintain directional control and minimum roll commensurate with the length of the runway. Maintained complete control of the aircraft at all times.

Conditionally Qualified — Minor deviations from above criteria. Safe approach and landing. Landed long, but within limits to make a safe stop.

Unqualified — Approach and landing deviated from the recommendations as outlined in

this manual. Touchdown was short or so long that stopping within limits was doubtful. Control during applications of reverse pitch and brakes caused erratic landing roll. Overall handling of landing tended to compromise safety.

e. Landing with emergency (engine, hydraulic, etc.)

Qualified — Emergency analyzed correctly and corrective action taken in order to accomplish safe approach and landing.

Conditionally Qualified — Minor discrepancies in handling emergency situation, but safe approach and landing accomplished.

Unqualified — Incorrect technique in handling emergency. Unsafe approach or landing.

12. Shutdown and postflight procedures

Qualified — Completed appropriate checklists without omissions, visual exterior inspection conducted, completed necessary logs and forms. Assured that flight plan was closed out.

Conditionally Qualified — Omitted checklist items and exterior inspection. Failed to note obvious maintenance discrepancies, none of which were considered unsafe.

Unqualified — Major checklist items were omitted, improper knowledge and use of postflight forms. No postflight inspection.

13. Emergencies (The criteria below apply to all emergencies: brake fire and failure; engine malfunctions, shutdown; fuel system management, after engine shutdown; pressurization malfunction; instrument failure; compartment fire; electrical fire; emergency descent; smoke elimination; hydraulic failure; engine-out waveoff; rollout reversing with engine emergencies.)

Qualified — Demonstrated proper technique, coordination, and knowledge of aircraft emergency and distress procedures. Good knowledge of bailout/ditching procedures.

Conditionally Qualified — Same as Qualified except for minor deviations.

Unqualified — Inadequate knowledge or poor technique in executing emergency or distress procedures.

14. Crew coordination

Qualified — Crew was well coordinated and pilot was aware of the duties and responsibilities of each. Crew performance as a team reflected pilot emphasis on conformance to standard procedures.

Conditionally Qualified — Crew was coordinated in actions and pilot was generally aware of crew duties and responsibilities. Performance as a team did not jeopardize safety or crew ability to complete mission. Did not require crew to perform tasks in accepted manner.

Unqualified — Crew performance showed complete lack of coordination by the pilot. Tasks were performed in violation of accepted publications.

15. Use of checklist

Qualified — Timely calling for and proper execution of checklist.

Conditionally Qualified — Late calling for or completing checklist. Minor deviations or omissions noted.

Unqualified — Failed to call for or complete checklists.

16. In-flight refueling (not applicable)

17. Aerial delivery (not applicable)

21.10 CT-39E/G FLIGHTCREW PLANE CAPTAIN NATOPS EVALUATION CRITERIA

1. Flight planning and preparation

a. Professional equipment

Qualified — Had all necessary personal and professional equipment. Displayed positive knowledge of the proper use, care, and operation of such equipment.

Conditionally Qualified — Marginal knowledge of the proper use, care, and operation of personal and professional equipment.

Unqualified — Did not have required personal or professional equipment. The lack of this gear and proper knowledge of its use could have caused compromise of safe operation. Successful survival in an emergency is doubtful.

b. Preflight inspection duties

Qualified — Reported sufficiently in advance of flight to complete preflight inspection/duties, precluding delay of flight. Completed thorough preflight inspection in compliance with standard forms. Ascertain that all required is aboard and that coffee and water have been procured.

Conditionally Qualified — Completed aircraft inspection/duties with omission in minor areas that did not cause delay or affect the safety of the proposed flight.

Unqualified — Failed to conduct proper aircraft inspection and/or omitted important items that would affect the safety of flight. Failed to have all flight attendant equipment, food, coffee, and water aboard.

c. Preflight inspection paperwork

Qualified — Accomplished all necessary paperwork.

Conditionally Qualified — Completed necessary paperwork, but with minor deviations from established procedures.

Unqualified — Did not accomplish required paperwork.

2. Preflight

a. Aircraft servicing

Qualified — Displayed positive knowledge of requirements and proficiency in the servicing of engine oil, fuel, oxygen, and hydraulic system components as outlined in approved manual and directives. All safety precautions were heeded.

Conditionally Qualified — Servicing of oxygen, engine oil, fuel, and hydraulic system components were accomplished with minor deviations of approved manuals and directives. Deviations did not compromise safety.

Unqualified — Servicing of engine oil, fuel, oxygen, and hydraulic system components was not accomplished or not done in an approved manner. Safety precautions were partially or completely disregarded, thereby endangering aircraft and/or personnel.

b. Ensure all survival and flight attendant equipment aboard.

Qualified — Inspected aircraft to ensure adequate survival and flight attendant equipment aboard to accomplish mission. Knew procedures to procure additional equipment.

Conditionally Qualified — Completed inspection with minor omissions not affecting safety of flight.

Unqualified — Failed to ensure adequate survival equipment was aboard to accomplish mission.

c. Knowledge of weight and balance

Qualified — Checked proper computations of weight and balance to ensure aircraft was within allowable operating limits.

Conditionally Qualified — Followed proper procedure in computing weight and balance, but made minor errors that did not result in improper loading or exceed allowable limits of the aircraft.

Unqualified — Made major errors in computations, resulting in improper cabin load or exceeding allowable limits of aircraft.

d. Baggage handling

Qualified — Loaded passenger baggage in proper sequence according to offloading points. Assured proper stowage and securing of baggage in designated compartments.

Conditionally Qualified — Loaded baggage within allowable limits, but in improper sequence.

Unqualified — Exceeded weight of red-line limits of compartments. Unnecessarily rough in handling.

e. Briefing

Qualified — Briefed the passengers completely concerning the use of seatbelts, the observance of the no smoking and seatbelt signs, the availability of coffee and water aboard, safety regulations, and ditching procedures.

Conditionally Qualified — Same as Qualified, but omitted some items or incompletely briefed on any one item.

Unqualified — Gave incomplete briefing. Omitted most items that should have been covered.

3. Flight

a. Care of passengers

Qualified — Displayed a constant, courteous impression in meeting the needs and requests of passengers. Conformed to high standards of personal service and passenger comfort at all times. Anticipated needs and took initiative to convenience. Ensured that all requirements and requests were handled expeditiously and correctly. Was consistently polite and attentive in dealing with passengers, assisting them whenever possible.

Conditionally Qualified — Possessed all the attributes of Qualified, but with occasional minor oversights.

Unqualified — Impolite in dealing with passengers. Lacked initiative in assisting passengers and failed to comply with requests. Disregarded standards of comfort desired and required for passengers.

b. In-flight emergency procedures

Qualified — Exhibited complete understanding of the corrective actions required for various in-flight emergencies. Knew the emergency checklists thoroughly. On drills, conducted corrective measures in a highly professional manner.

Conditionally Qualified — Basic familiarity with emergency procedures and the corrective actions required for various in-flight emergencies. Familiar with the checklists. Minor deviations, not considered unsafe.

Unqualified — Unsatisfactory knowledge of emergency procedures or corrective actions required. Attempted to perform an unsafe action during emergency drill conducted by the aircraft commander.

c. Ditching

Qualified — Demonstrated a thorough knowledge of ditching procedures. Explained in detail all actions required for ditching in the proper sequence. Was able to execute the ditching expeditiously, safely, and correctly, requiring a minimum of supervision.

Conditionally Qualified — Demonstrated a satisfactory knowledge of ditching procedures. Was able to explain the actions required in ditching. Minor errors were noted, not detrimental to his own devices.

Unqualified — Did not know ditching procedures or left doubt that he would survive ditching or bailout if left to his own devices.

d. Care and use of equipment

Qualified — Demonstrated correct use of all equipment applicable to his duties. Took care not to abuse or misuse equipment.

Conditionally Qualified — Same as Qualified, but could have been more careful in handling of equipment.

Unqualified — Inadequate knowledge or improper utilization of equipment.

e. Location and use of survival equipment

Qualified — Demonstrated a complete knowledge of all survival equipment normally carried in the aircraft, its use, care, and stowage locations. Thorough knowledge of how and when to launch rafts, properly inflate them, board them, etc.

Conditionally Qualified — Demonstrated a satisfactory knowledge of survival equipment, its use, care, stowage. Minor errors or

deviations observed, but not considered to be unsafe.

Unqualified — Demonstrated lack of knowledge of survival equipment.

f. Customs/agriculture procedures

Qualified — Ensured that all necessary forms were completed and aboard aircraft and that aircraft was properly disinsected where required. Was familiar with custom regulations.

Conditionally Qualified — Complied with agriculture requirements, but committed minor errors in preparation of forms.

Unqualified — Failed to properly comply with customs or agriculture regulations. Unfamiliar with prohibited items.

g. Forms use

Qualified — Completed all required forms expeditiously and accurately. Specifically checked passenger/cargo manifests and completed flight summary for given leg of the flight. All required forms for specific stations completed prior to arrival.

Conditionally Qualified — Completed all required forms, but with minor omissions that did not adversely affect the flight.

Unqualified — Failed to satisfactorily complete necessary forms.

h. Aircraft cleanliness

Qualified — Maintained a maximum standard of cleanliness neatness at all times. Ensured that the cabin is cleaned thoroughly at regular intervals and that all waste receptacles were serviced at each stop en route. Displayed initiative in maintaining a high degree of aircraft cleanliness at all times.

Conditionally Qualified — Same as Qualified, but with minor deviations.

Unqualified — Neglected to empty waste receptacles at en route stops. Lacked concern and effort to keep the cabin spaces neat and clean. Failed to inspect the aircraft for cleanliness or to correct discrepancies that existed.

i. Fuel system

Qualified — Possessed a thorough knowledge of the fuel system and its operation. Was able to explain in detail the entire system, its operation and limits.

Conditionally Qualified — Possessed a working knowledge of the fuel system and its operation. Was able to explain the system, its operation and limits. Minor errors were noted, though not detrimental to safety or operation.

Unqualified — Did not know the system or was unable to operate safely.

j. Electrical system

Qualified — Demonstrated a thorough knowledge of the system, its operation and limitations. Thoroughly familiar with all switches and controls. Had a thorough knowledge of both ac and dc circuits.

Conditionally Qualified — Demonstrated a good knowledge of the system. Minor errors were noted, not detrimental to the electrical system or safety. Further instruction needed.

Unqualified — Did not know the electrical system. Showed a marked need for further instruction.

k. Hydraulic system

Qualified — Possessed a thorough knowledge of the hydraulic system, its operation, and limits. Was able to explain in detail the system and its components.

Conditionally Qualified — Possessed a good knowledge of the system. Minor errors in explaining system and components. Further instruction needed.

Unqualified — Did not know hydraulic system. Showed a marked need for further instruction.

4. Postflight

a. Servicing and securing

Qualified — Displayed professional knowledge of requirements and proficiency in

refueling and securing aircraft. All requirements, as outlined in approved manuals, and directives were accomplished in a professional manner. All safety precautions were heeded.

Conditionally Qualified — Refueling and securing were accomplished in accordance with approved manual and directives, but with minor deviations. Deviations did not compromise safety. Aircraft considered not completely secured, but not endangered.

Unqualified — Refueling and securing were not accomplished or were not accomplished in approved manner. Safety precautions were partially or completely disregarded, thereby endangering aircraft and/or personnel. Required supervision and/or further instruction.

b. Inspections

Qualified — Extremely thorough and competent in accomplishing required inspections. No deviations from manuals or other directives. Advised pilot on aircraft discrepancies so that a complete and accurate aircraft status was determined.

Conditionally Qualified — Completed the required inspections as prescribed in appropriate manuals and documents with minor deviations. Advised pilot of discrepancies, but was slightly vague. Was doubtful of the true status of the aircraft, but not of items that would place the aircraft in a down status. Overall performance displayed room for improvement.

Unqualified — Did not or was unable to perform daily/turnaround inspections. Failed to advise pilot of aircraft discrepancies; consequently, the true status of the aircraft was unknown. Needed supervision and/or further instruction.

c. Offloading procedure

Qualified — Observed safe practices in cargo handling. Ensured compliance with load limits of aircraft and cargo handling equipment. Provided proper safeguards where appropriate.

Conditionally Qualified — Same as Qualified except for minor deviations.

Unqualified — Used unsafe procedure in offloading. Exceeded allowable limits of aircraft or equipment. Failed to provide proper safeguards.

d. Aircraft secured as required

Qualified — Ensured aircraft entry doors and hatches were secured.

Conditionally Qualified — Same as Qualified except for minor deviations.

Unqualified — Made little effort to ensure aircraft was properly secured.

e. Security compliance

Qualified — Ensured compliance with security regulations concerning special mail and/or cargo.

Conditionally Qualified — Same as Qualified except for minor deviations.

Unqualified — Failed to provide proper security as required.

5. General

a. Crew coordination

Qualified — Coordinated action/requirements smoothly and effectively with all crewmembers. Anticipated demands upon crew position. The overall performance of the crew was enhanced by his critical decisions to the needs of others in the performance of their assigned duties.

Conditionally Qualified — Attempted to coordinate action/requirements, but lacked the desired effect. Intercommunication discipline at times interfered with crew. Responses to checklist were made with only minor deviations. Did not jeopardize mission.

Unqualified — Displayed undesirable traits related to crew coordination. Intercommunication discipline was nonexistent. Responses to checklist were vague or improper. Performance impaired the smooth accomplishment of the mission.

b. Location and use of survival equipment

Qualified — Demonstrated a complete knowledge of all survival equipment normally carried in the aircraft, its use, care, and stowing positions. Thorough knowledge of how and when to launch rafts, properly inflate them, board them, etc.

Conditionally Qualified — Demonstrated satisfactory knowledge of survival equipment, its use, care, and stowage. Minor errors or deviations observed, but not considered to be disqualifying.

Unqualified — Demonstrated lack of knowledge of survival equipment.

c. Knowledge of aircraft

Qualified — Possessed a complete knowledge of the equipment and systems of the aircraft related to the performance of duties and execution of emergency procedures. Demonstrated the ability to properly operate all related equipment in an efficient manner. Knew the various aircraft configurations and understood the limitations and procedures involved in each case.

Conditionally Qualified — Displayed a general knowledge of equipment and systems related to performance of duties. Needed instruction and training in different configurations and related equipment.

Unqualified — Unfamiliar with necessary systems and equipment. Lacked knowledge of limitations or operation of equipment essential to performance of duties and/or safety.

d. Military courtesy

Qualified — Rendered proper military courtesy to crew, passengers, and ground personnel. Polite and cheerful in dealing with all passengers.

Conditionally Qualified — Occasionally neglected to render proper courtesies.

Unqualified — Consistently lacking in military courtesy. Disagreeable or rude to officers, crew, passengers, or ground personnel.

e. Knowledge of first aid

Qualified — Possessed an excellent knowledge of the basic fundamentals of first aid. Knew the location and use of all first-aid equipment aboard.

Conditionally Qualified — Same as Qualified, but with minor discrepancies.

Unqualified — Little or no knowledge of the basic fundamentals of first aid. Did not know the location or use of the subject equipment.

f. Sidearm procedures

Qualified — Understood regulations concerning sidearms as regards to couriers and passengers. Properly complied with same.

Conditionally Qualified — Same as Qualified with minor deviations.

Unqualified — Unfamiliar with sidearm procedures or regulations. Improper use of sidearms.

g. Supervisory ability

Qualified — Demonstrated good leadership qualities, got the best results from men under him.

Conditionally Qualified — Sometimes a little lax, but generally got good results from subordinates.

Unqualified — Very lax, could not get good results from subordinates.

21.11 PILOT QUESTION BANK

Note

Questions on emergency procedures are designated by an asterisk (*).

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*1. If a main failure occurs after decision speed, continue takeoff. TRUE/FALSE

2. When operating on one boost pump and wing fuel becomes unbalanced, the following procedure should be used to balance fuel. _____

*3. Maximum allowable IAS/Mach limitations are: _____

4. Maximum allowable airspeed with the landing/taxi lights extended is: _____ knots.

*5. Maximum allowable exhaust temperature during start is: _____ °C.

6. Maximum exhaust temperature at takeoff thrust and during engine accelerations is _____ °C.

7. Maximum allowable engine revolutions per minute is _____ percent.

*8. The starter can be used to aid an airstart. TRUE/FALSE

9. The gyrocompass is fast slaved by _____.

*10. To provide sufficient compressor bleed air for engine anti-icing, a minimum of _____ EPR shall be maintained.

*11. The engine fire extinguishing system can be activated by pulling the _____ and selecting _____.

*12. Describe the procedure if low oil pressure is encountered during flight.
35-40 psi: _____.
Less than 35 psi: _____.

13. The air precharge on the auxiliary hydraulic accumulator is _____ psi (standard day).

*14. What portions of the hydraulic systems may be operated by the auxiliary hydraulic accumulator?
_____

*15. Best turbulent air penetration speed is _____ knots.

*16. Procedure for engine failure/fire before decision speed is: _____
_____.

*17. Procedure for engine failure/fire after decision speed is: _____
_____.

*18. What procedure should be followed if the nosewheel steering light is on after takeoff? _____
_____.

*19. Describe procedure for an immediate airstart. _____
_____.

20. Maximum zero fuel weight is _____ pounds.

21. Maximum ramp weight (before taxi) is _____ pounds.

*22. An emergency signal is produced by the ATC transponder when set to mode _____, code _____.

23. During normal cruising flight, oil pressure fluctuation should not exceed _____ psi.

24. If both dc generators are lost, the main inverter will also be lost. TRUE/FALSE

25. Pressing the fire and overheat detection circuit test button will illuminate two _____, two _____, and the aft fuselage _____ light if the circuits are continuous and operating. There is a restriction on how often you may use this press-to-test feature. TRUE/FALSE

26. If the contents of either or both of the engine fire extinguishers have been discharged into the engine compartments or overboard, this will be indicated by a yellow or red disk located in the fuselage skin _____. It is not mandatory that these indicators be checked before each flight. TRUE/FALSE

27. Aircraft power is supplied by two _____ gas turbine engines.

28. Rated sea-level static thrust of each of these engines is _____ pounds.

29. Anti-icing for the engines is provided by bleeding air from the _____ stage of compression.

30. Turning either engine master switch to ON opens the respective engine _____ valve and _____ bus power is made available for subsequent starter operation.

31. After engine shutdown, the engine master switch should be placed at _____ before the electrical master or battery switch is placed at OFF. This allows the _____ shutoff valve to close.

32. Engine starts may be made by use of the aircraft _____ or _____ power or by _____ starting.

33. By insertion of the external power plug, the _____ are disconnected from the starting bus.

34. The starting cycle may be interrupted at anytime by _____ the starter button _____

35. The flight controls are _____ operated.

36. There are _____ separate fuel cells in the aircraft.

37. Fuel from the fuselage tank feeds by _____ feed to the center wing section.

38. Fuel ejectors are in the forward center wing section of the main fuel cell to ensure adequate fuel supply to the respective _____.

39. Total fuel in the fuselage tank may be determined by selecting the position _____. This position will also give the _____ fuel in the _____ wing tank.

40. The low fuel level caution light will come on when either fuel quantity gauge indicates approximately _____ pounds remaining.

41. Fuel may be jettisoned at the rate of 180 pounds per minute, depending upon _____, temperature, and _____. Jettison automatically stops when total fuel remaining is _____ to _____ pounds if the fuel jettison switch is positioned at _____.

42. Additional fuel may be jettisoned by using the _____ position of the fuel jettison switch. Fuel *may/may* not be jettisoned when the weight of the aircraft is on the main landing gear. (Circle one)

43. The auxiliary hydraulic system will not extend or retract the landing gear. TRUE/FALSE.

44. All of the caution lights are _____ in color, while the warning lights are _____ . The pilot _____ lights rheostat provides dimming control for the caution and warning light panel.

45. The Mach/airspeed audible warning system is actuated whenever the aircraft is below feet and airspeed of _____ knots is exceeded.

46. The landing gear is _____ actuated and _____ operated.

47. The landing gear _____ shall be used after the gear has been lowered by the emergency method to permit subsequent gear retraction.

48. The landing gear warning system consists of a _____ and a _____ or _____ red light. The warning system is actuated anytime the aircraft is below _____ feet, the airspeed is less than _____ knots, either throttle position is less than _____ -percent rpm, and the landing gear is not down.

49. The nosewheel steering system is _____ powered, _____ actuated, and _____ controlled.

50. With the nosewheel steering selector switch at MAIN, if the main system fails, transfer to the standby system will be automatic. TRUE/FALSE

51. Emergency braking cannot be accomplished if both pilots apply brakes simultaneously. TRUE/FALSE

52. The speedbrake switch should normally be _____ in flight. The speedbrake caution light will come on anytime the brake is partially down and the throttles are at _____ -percent rpm or more.

53. If each fuel gauge reads _____ pounds or greater, this indicates that there is fuel in the aft fuselage tank. Takeoffs and landings are not recommended with a wing fuel imbalance in excess of _____ pounds.

54. Throttle should be advanced to idle during a start at _____ to _____ ____ -percent rpm.

55. The starter is limited to _____ minutes of operation during any _____ minute period.

56. If a fuel flow of over 600 pounds per hour is noted during a start, _____ immediately to prevent engine _____ .

57. Extended operation at high altitude may "cold-soak" fuel. Operate fuel heater for minute before entering _____ approach.

58. Aerodynamic braking is _____ under normal landing conditions.

59. The engines should be operated for about _____ minutes at reduced thrust (_____ percent or below) before shutdown.

60. When the landing gear is _____, trim is accomplished at a low speed rate; with the gear _____, trim rate is high.

61. In the landing configuration, the wing slats will start out at about _____ knots and be fully extended at about _____ knots.

62. Continuous ignition should be selected whenever flying in turbulence or heavy _____.

63. Windshield heat is provided by heating coils powered by two separate _____.

64. The autopilot should be _____ when encountering turbulence to preclude undue _____ on the airframe.

65. The entry door should be closed during towing of the aircraft to prevent bouncing of the door against its stops. TRUE/FALSE

66. The radio altimeter system measures terrain clearance from an altitude of _____ feet to _____ during landing approach.

67. An _____ controlled, hydraulically operated antiskid braking system prevents excessive skidding. If a malfunction develops during operation, the entire system is electrically rendered inoperative and the _____ caution light illuminates.

68. The antiskid system may be reset by _____.

69. If a landing is being made using the _____ system, it is recommended that the antiskid system be turned _____ to _____ hydraulic accumulator pressure.

*70. The two-position emergency speedbrake dump switch permits _____ of the speedbrake in flight should _____ failure occur.

71. The landing gear door ground control switch is located in the _____ landing gear well. The switch is labeled _____ and _____ and should be in the _____ position before flight.

72. The autopilot may be engaged only in the _____ and _____ modes of the integrated flight system.

73. The autopilot will operate only in conjunction with the _____ instruments.

74. If the trim system selector switch is in any position other than NORMAL during autopilot operation, the autopilot stabilizer _____ will be inoperative as indicated by illumination of the _____ caution light.

*75. If nosegear tire failure should occur at or near nosewheel liftoff speed, it is recommended to continue takeoff and burn-down or jettison fuel in order to reduce gross weight for the ensuing landing. TRUE/FALSE

76. Minimum oil pressure at idle revolutions per minute is _____ psi; normal oil pressure operating range is _____ to _____ psi.

*77. If the NOSE STEER ON light remains on after takeoff, what procedure should be followed?
_____

78. While performing touch-and-go landings, maintain throttle settings at a minimum of _____-percent rpm to ensure good engine acceleration and do not add power in excess of _____ -percent rpm for the go-around.

79. Avoid sustained engine speeds between _____ and _____ -percent rpm since this is the range in which the interstage bleed valves are opened or closed by the engine fuel control.

80. Fuel indications may vary by as much as _____ pounds because of changes in aircraft attitude during penetration or approach.

*81. Engine flameout may occur above _____ feet because of turbulent airflow induced by high angle of attack and low airspeed. Above _____ feet, engine flameout may occur below 180 KIAS because of fuel control/engine operational characteristics.

*82. When using the emergency braking system, either pilot holding the brake pedals depressed will prohibit the other pilot from making brake applications since the brake pedals are _____ _____.

83. A maximum-range descent from flight level 400 will consume _____ pounds of fuel, take _____ minutes, and cover _____ miles if the descent is carried to sea level.

84. Landing on runway 29 at NAS North Island, the wind is reported as "350°, 30 knots with gusts to 40 knots." Crosswind" component is _____ knots. Landing is *Recommended/Not Recommended.* Headwind component is _____ knots.

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85. The VOR system is provided azimuth information from the _____ gyro.

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PART XI

Performance Data

Chapter 22 — Performance Data Introduction

Chapter 23 — Takeoff

Chapter 24 — Climb

Chapter 25 — Range

Chapter 26 — Descent

Chapter 27 — Landing

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CHAPTER 22

Performance Data Introduction

22.1 SCOPE AND ARRANGEMENT

The flight performance charts in this part provide data for accurate preflight planning. Using these charts is the most effective method of determining optimum aircraft performance. The charts are arranged in the following sequence:

Chapter 22 — Introduction
Chapter 23 — Takeoff
Chapter 24 — Climb
Chapter 25 — Range
Chapter 26 — Descent
Chapter 27 — Landing

Two types of charts are included: (1) profile-type charts for operational and maximum range and (2) graphical charts for takeoff, climb, descent, and landing. The profile-type charts are provided for "quickie mission planning." These charts are optimized for no-wind conditions and are based on specific Mach numbers to aid flight planning by reducing the number of computations that shall be made. A discussion and explanation of the use of the charts is provided.

22.2 DESCRIPTION OF PERFORMANCE PRESENTATION

Basically mission planning with these charts involves nothing more than the determination of each leg of the mission and the summarizing of these legs to determine the time, fuel, distance, etc.

22.3 PERFORMANCE DATA BASIS

Flight planning information shown is based on flight test and estimated data developed from flight tests of earlier series aircraft of a similar model. All charts are based on what is believed to be normal engine performance characteristics, under ICAO standard day atmospheric conditions; however, corrections for nonstandard atmospheric conditions have been included on the charts.

22.4 PERFORMANCE DATA DEFINITIONS

1. V_1 (critical engine failure speed) — The speed above which the takeoff is continued and below which the takeoff is abandoned in the event of an engine failure.

2. Refusal speed — Used in conjunction with critical field length chart to determine speed at which an aircraft can either accelerate to refusal speed and then stop or accelerate to refusal speed and continue takeoff while one engine is failed at refusal speed. All available remaining runway will be required to get the aircraft airborne.

3. V_2 (takeoff safety speed) — Must be not less than 1.1 X V_{MC} or 1.2 V_S and must be attained before reaching a height of 35 feet.

4. V_R (rotation speed) — The speed at which the aircraft is rotated to the takeoff attitude. V_R must not be less than V_1 or less than 1.05 X V_{MC}. It must be also high enough to allow V_2 to be attained before aircraft reaches a height of 35 feet.

5. V_S — The stalling speed or the minimum steady flight speed at which the aircraft is controllable.

6. V_{MCA} — Minimum control speed (air) is the speed required to provide sufficient control to fly a straight path over the ground with a failed engine. This speed is based on the engine windmilling, takeoff thrust on the good engine, and no more than 5° bank away from the failed engine. At minimum control speed, it may be necessary to sacrifice altitude for airspeed while putting the aircraft in a clean configuration and obtaining sufficient airspeed to climb.

7. V_{MCG} — Minimum control speed (ground) is the speed at which the aircraft can be controlled on the ground with a failed engine and by use of aerodynamic controls alone. The speed is based on the failed engine windmilling, takeoff thrust

on the good engine, the nosewheel off the runway, and use of ailerons and rudder to maintain directional control within 25 feet of the desired path.

8. V_B — The design speed for maximum gust intensity.

9. V_F — The design flap speed.

10. V_{MO} — The maximum operating limit speed.

11. V_{SO} — Stall speed or the minimum steady flight speed in the landing configuration.

12. V_{REF} — Final approach speed at the 50 feet threshold height in a normal landing configuration. V_{REF} is 1.3 X V_{SO}.

13. Balance field length — The distance within which the aircraft can either:

    a. Accelerate to V_1 and then stop

    b. Accelerate to V_1 and continue to a height of 35 feet with one engine cut at V_1.

14. Critical field length — The distance within which the aircraft can either:

    a. Accelerate to refusal speed and then stop

    b. Accelerate to refusal speed and continue takeoff while one engine is failed at refusal speed. All available remaining runway will be required to get the aircraft airborne.

22.5 AIRSPEED AND ALTITUDE CORRECTIONS

22.5.1 Definitions

1. IAS — Indicated airspeed is the reading taken from the airspeed indicator.

2. CAS — Calibrated airspeed is indicated airspeed corrected for installation effects.

3. EAS — Equivalent airspeed is calibrated airspeed corrected for compressibility effects.

4. TAS — True airspeed is equivalent airspeed corrected for atmospheric density.

5. GS — Groundspeed is true airspeed corrected for wind.

6. Indicated pressure altitude — Reading of the pressure altimeter when set to barometric pressure of 29.92-inches H_g.

7. Calibrated pressure altitude — The indicated altitude corrected for altimeter error.

8. Altimeter position correction — That figure (in feet) that when applied to the indicated altitude, gives calibrated pressure altitude.

22.5.2 Airspeed Correction. Because of the position of the airspeed static pressure source, a correction shall be applied to the CAS to obtain IAS. For CT-39G aircraft, this correction can be obtained from Figures 22-1 and 22-2. For CT-39E aircraft, the airspeed correction can be obtained as follows: For approach airspeeds and configurations (full or partial flaps), add 1 knot to CAS to obtain IAS. For other airspeeds and configurations, add 2 knots to CAS to obtain IAS. As the aircraft approaches a stall, regardless of configuration, add 3 knots to CAS to obtain IAS.

22.5.3 Altitude Correction Because of the position of the altimeter static-pressure source, a correction shall be applied to the indicated pressure altitude to obtain calibrated pressure altitude. This correction can be obtained from Figures 22-3 through 22-5. Instrument error is the only difference between calibrated pressure altitude and true pressure altitude. Since this chart does not consider instrument error, true pressure altitude may be considered the same as calibrated pressure altitude. Altitude correction shall not be applied to the encoding altimeter indication when functioning in the NORM mode.

22.5.4 Ambient Air Temperature (T_{amb}). Ambient air temperature is obtained either from ground meteorological sources or while in flight from the cockpit free air temperature gauge corrected by the appropriate correction factor (Figure 22-6).

Certain computers enable computing the error of temperature indicating systems when the recovery coefficient for the system is known. With these computers, first determine temperature rise, using Mach number, indicated free air temperature, and .8 as recovery coefficient (.8 will usually be shown; however, another value may be substituted). This temperature rise shall then be divided by .8 (or substitute value) and multiplied by the recovery coefficient for this aircraft, which is .68. The T_{amb} is colder than indicated by the result.

22.5.5 Total Air Temperature (T_{t2}). For aircraft with a production temperature system, the values from the cockpit free air temperature gauge must be corrected

by the correction factor determined from Figure 22-7. For aircraft with a Rosemount total temperature system, the free air temperature gauge provides T_{t2} without any corrections required.

22.5.6 Mach Number Correction. To convert true Mach number to indicated Mach number, add .005.

22.5.7 Airspeed Conversion. An airspeed conversion chart (Figure 22-8) is used to convert CAS to TAS or true Mach number. IAS shall be converted to CAS before entering the chart.

22.6 NONSTANDARD DAY TEMPERATURE

When the free air temperature is higher than standard, the aircraft performs as if it weighs more than it actually does. When the free air temperature is lower than standard because of the flat-rated engines, the aircraft performance is essentially the same as on a standard day. Where cruise is performed at the cruise ceiling (limited by maximum continuous thrust) at temperatures above standard, the cruise altitude is decreased. If cruise is performed below the cruise ceiling, the maximum cruise thrust is not being used. In this case, an excess of thrust is available on a standard day.

The thrust loss because of an increase in ambient temperature has only the effect of requiring a higher cruise engine setting (EPR) and a slightly higher specific fuel consumption.

Use of an equivalent weight permits utilization of the standard day charts under nonstandard temperature conditions. Equivalent weight is obtained by applying a weight increment to actual aircraft gross weight. The equivalent weight is used only to determine the nonstandard climb performance and cruise ceiling. For each 10 °C (18 °F) rise in air temperature above standard day conditions, increase actual gross weight, as shown on the individual chart, before entering the climb chart. For temperatures below standard day conditions, use actual gross weight, as shown on the individual chart, before entering the climb chart. Cruise performance is determined for the actual aircraft gross weight.

Standard day temperature throughout the altitude range of the aircraft can be determined from the standard day temperature chart in Figure 22-9.

22.7 FUEL SPECIFIC WEIGHT

All data given in this part is based on a specific fuel consumption in pounds. Therefore, the charts may be used when either JP-4 or JP-5 fuel is used. The basic weight of the fuels will vary with refining process as well as with fuel temperature. A chart is included (Figure 22-10) that shows the variations in fuel specific weight with fuel temperature. To use the chart, determine the basic specific weight for the particular refining process and assume a variation with fuel temperature parallel to that shown on the chart for the fuel being used.

Note

Since the specific fuel consumption of JP-5 is the same as that of JP-4, a slight increase in range (for the same gallons of fuel) will be noted when JP-5 is used because of the increased weight per gallon.

22.8 STANDARD ATMOSPHERE TABLE

The standard atmosphere table (Figure 22-11) provides the standard day conditions at various altitudes for density ratio, $1\sqrt{\sigma}$ temperature, speed of sound in knots, and pressure in inches of mercury and pressure ratio. To find the value of any item on a standard day, read the figure to the right of the altitude in the appropriate column.

DATA AS OF: 28 APRIL 1967
BASED ON: NR FLIGHT TEST

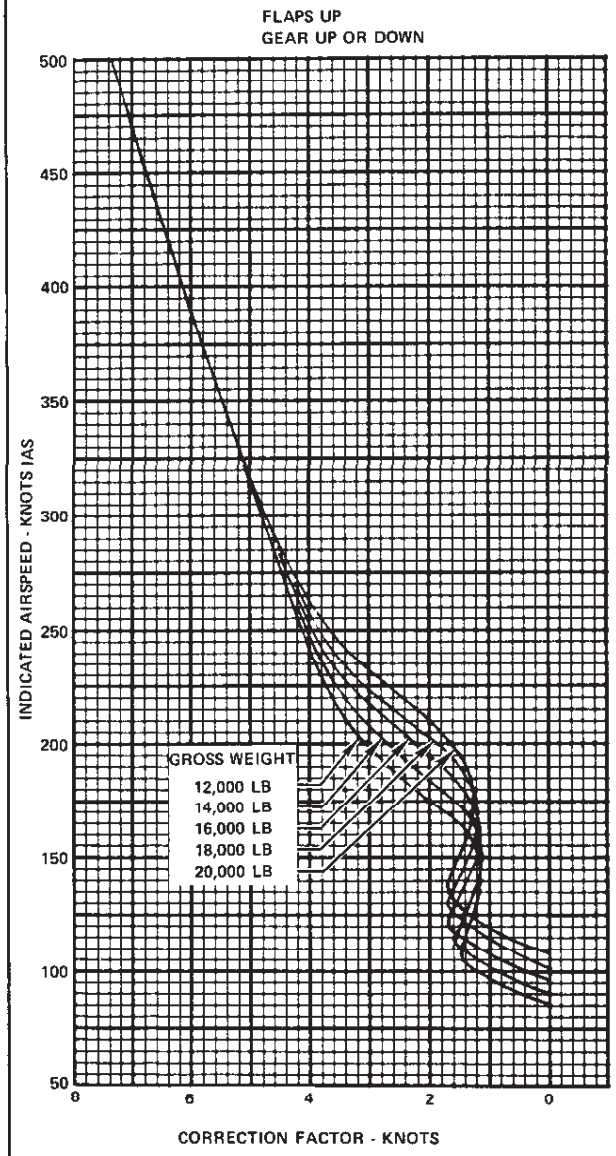
AIRSPEED POSITION CORRECTION

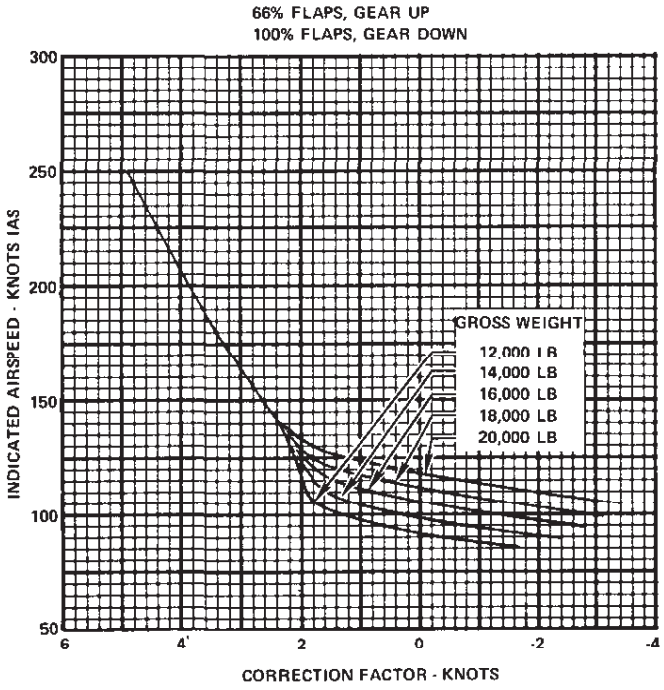
PRIMARY SYSTEM

MODEL: CT-39G 158843
ENGINES (2): JT12A-8

Note

Calibrated airspeed equals indicated airspeed minus correction factor.





N3/74
39E-1-93-79

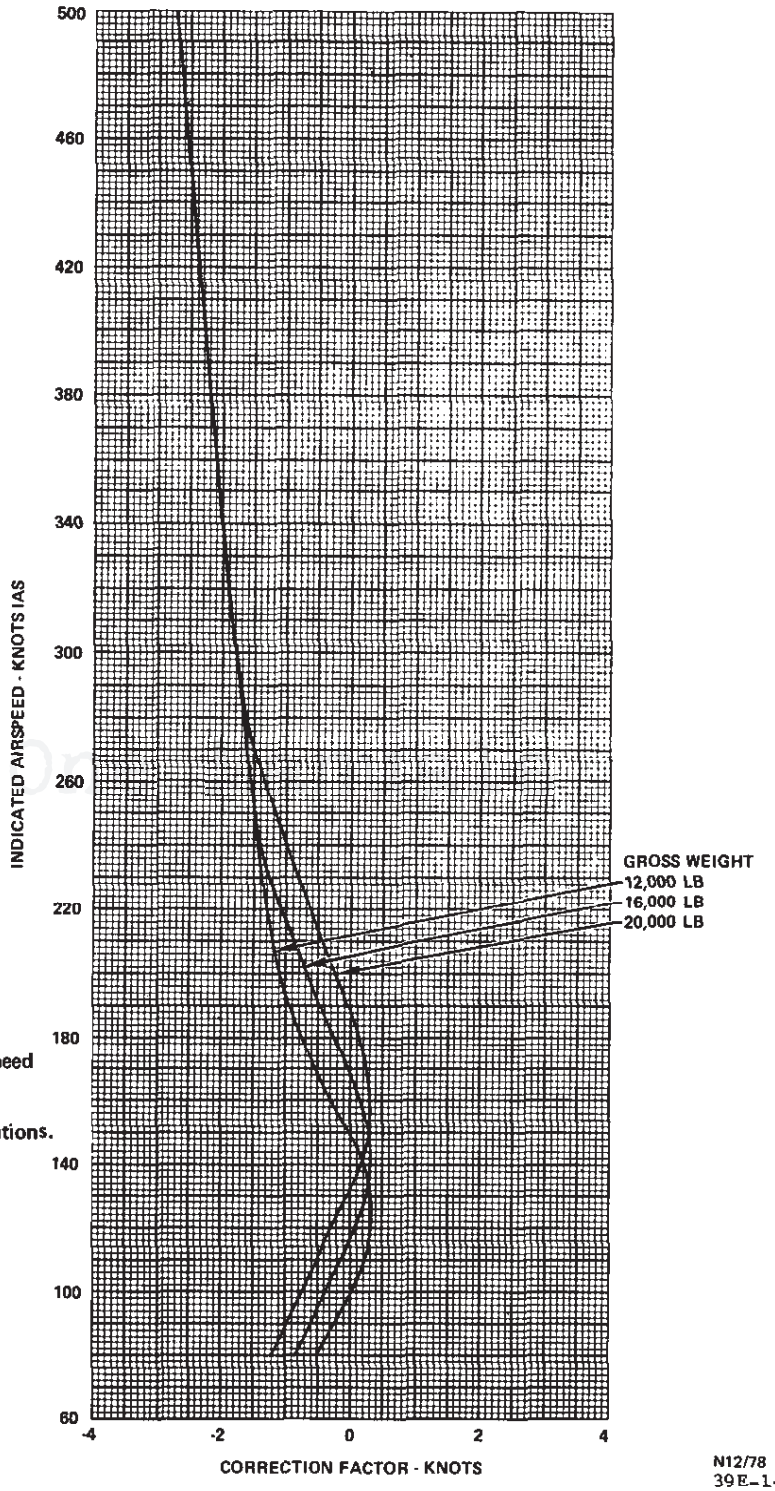
Figure 22-1. Airspeed Position Correction — CT-39G 158843

DATA AS OF: 14 SEPT 1973
BASED ON: Rockwell International FLIGHT TEST

AIRSPEED POSITION CORRECTION

MODEL: CT-39G 158844, 159362 THRU 159365 AND 160053 THRU 160057
ENGINES (2): JT12A-8

FLAPS UP
GEAR UP OR DOWN



Note

- Calibrated airspeed equals indicated airspeed plus correction factor.
- Applicable for all flap and gear configurations.

N12/78
39E-1-93-86A

Figure 22-2. Airspeed Position Correction — CT-39G 158844, 159362 Through 159365, and 160053 Through 160057

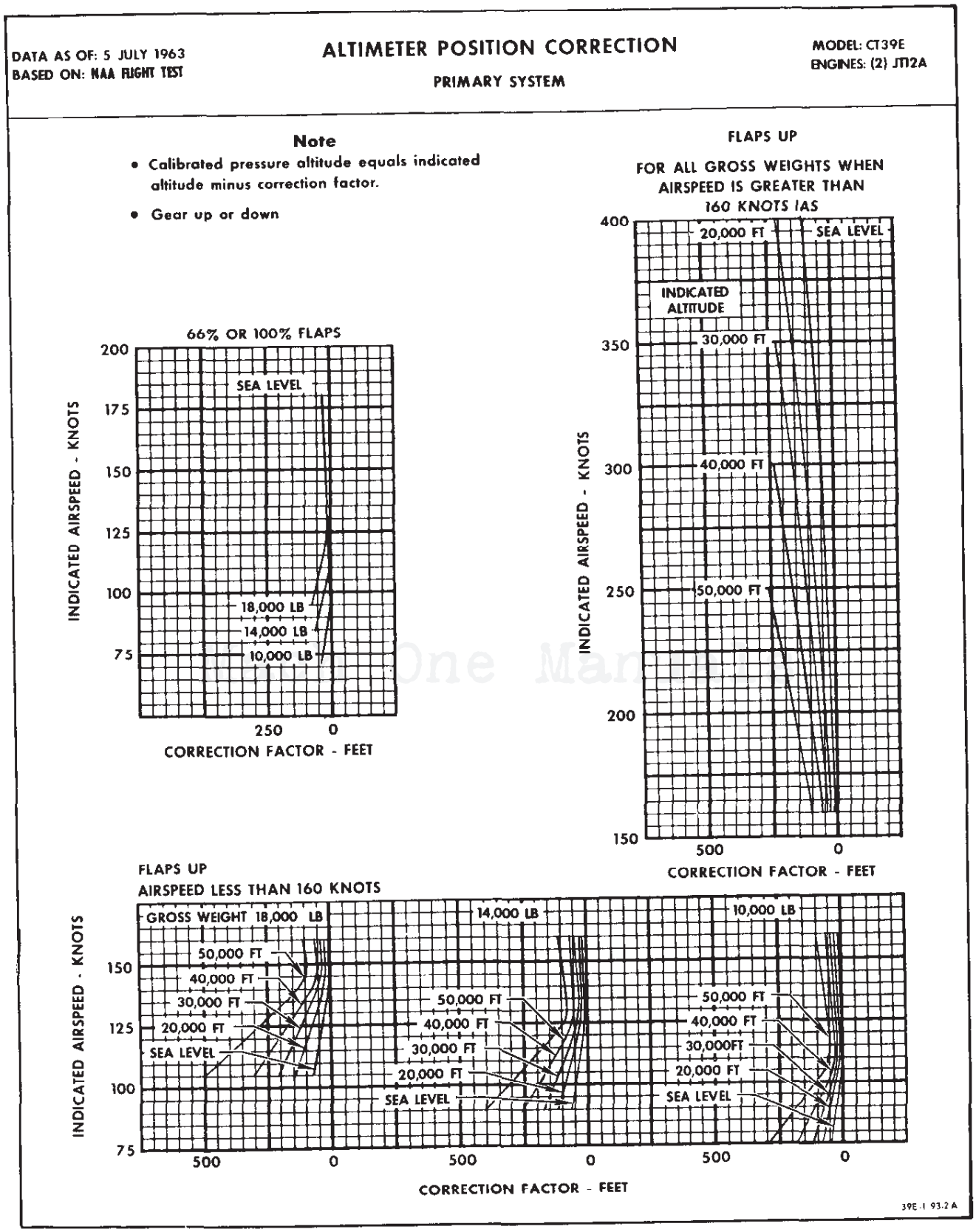


Figure 22-3. Altimeter Position Correction — CT-39E

DATA AS OF: 28 APRIL 1967
BASED ON: NR FLIGHT TEST

ALTIMETER POSITION CORRECTION

PRIMARY SYSTEM

MODEL: CT-39G 158843
ENGINES (2): JT12A-8

Note

Calibrated pressure altitude equals indicated altitude minus correction factor.

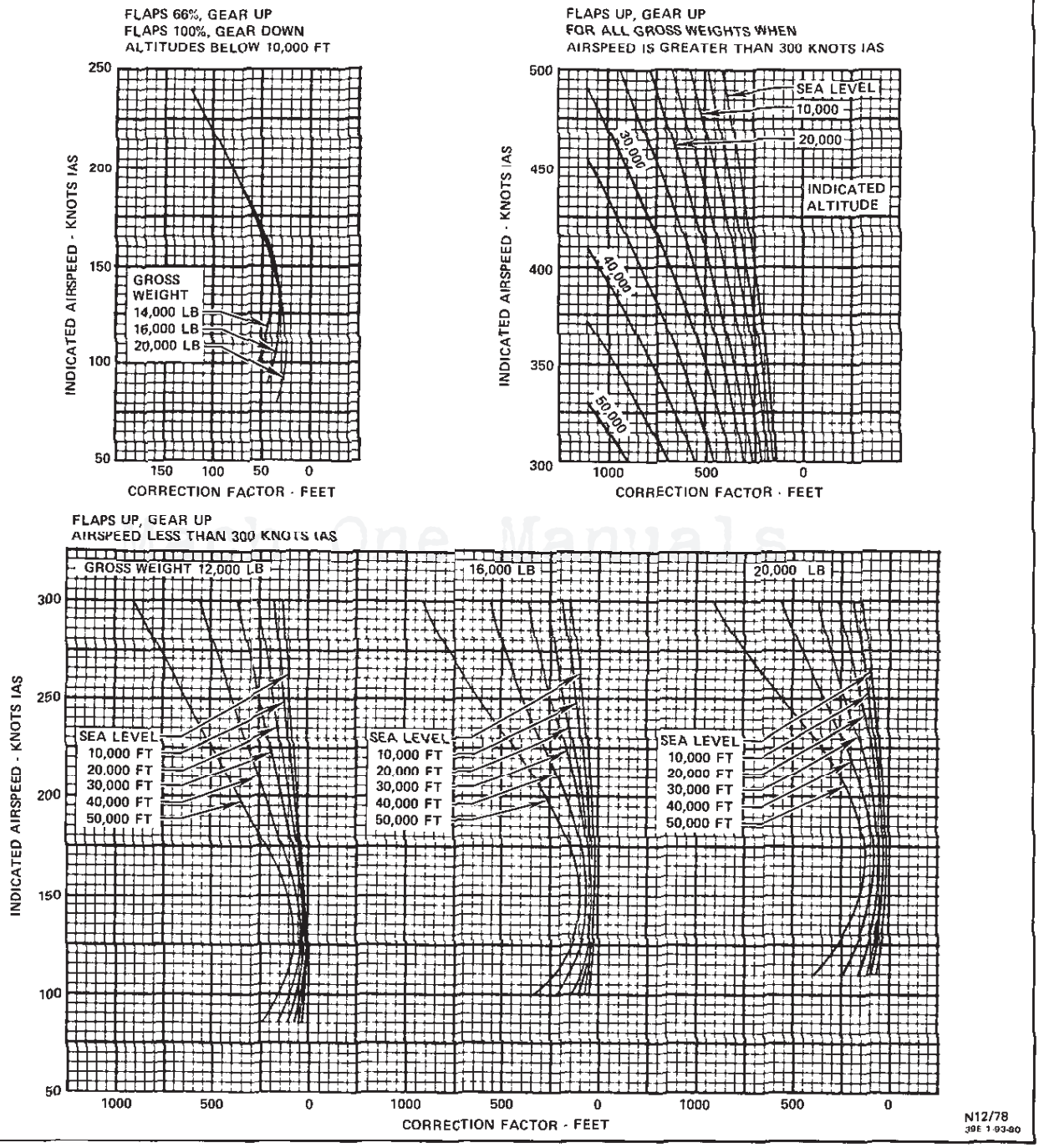


Figure 22-4. Altimeter Position Correction — CT-39G 158843

DATA AS OF: 14 SEP 1973
BASED ON: Rockwell International FLIGHT TEST

ALTIMETER POSITION CORRECTION

MODEL: CT-39G 158844, 159362 THRU 159365 AND 160053 THRU 160057
ENGINES (2): JT12A-8

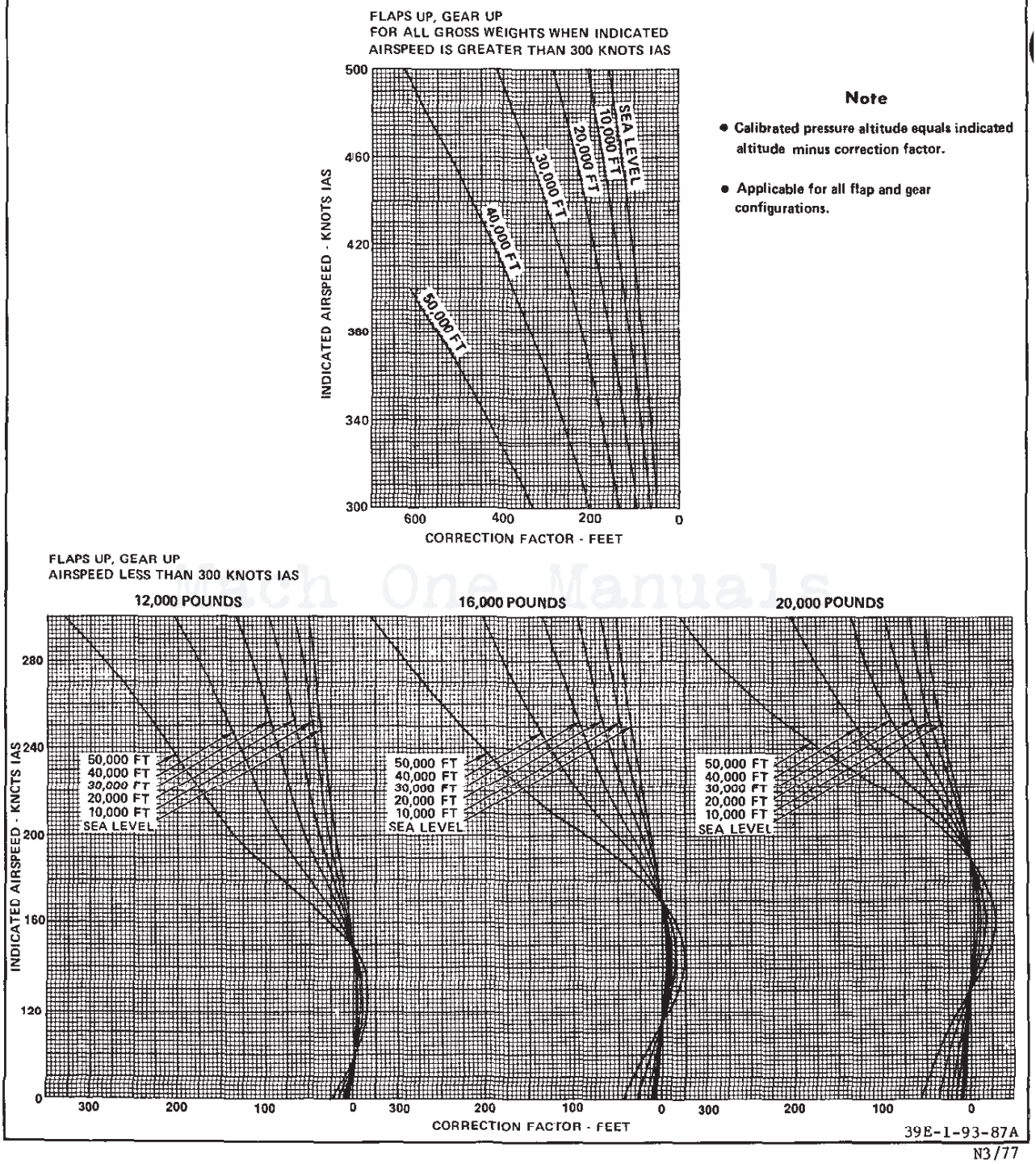


Figure 22-5. Altimeter Position Correction — CT-39G 158844, 159362 Through 159365, and 160053 Through 160057

NOTE

SUBTRACT THE AMBIENT AIR TEMPERATURE CORRECTION FROM THE INDICATED FREE AIR TEMPERATURE TO OBTAIN AMBIENT AIR TEMPERATURE.

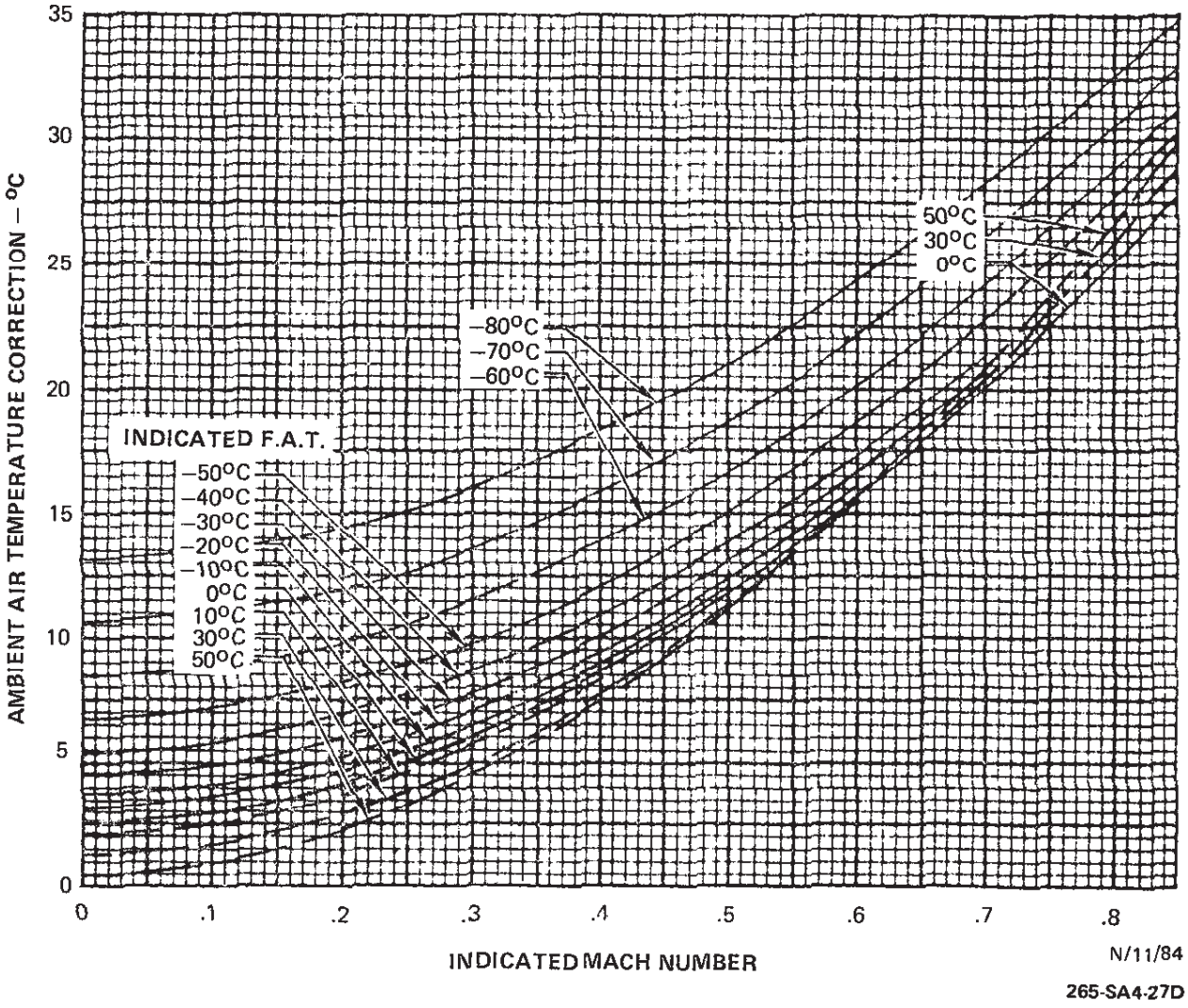


Figure 22-6. Ambient Air Temperature Correction

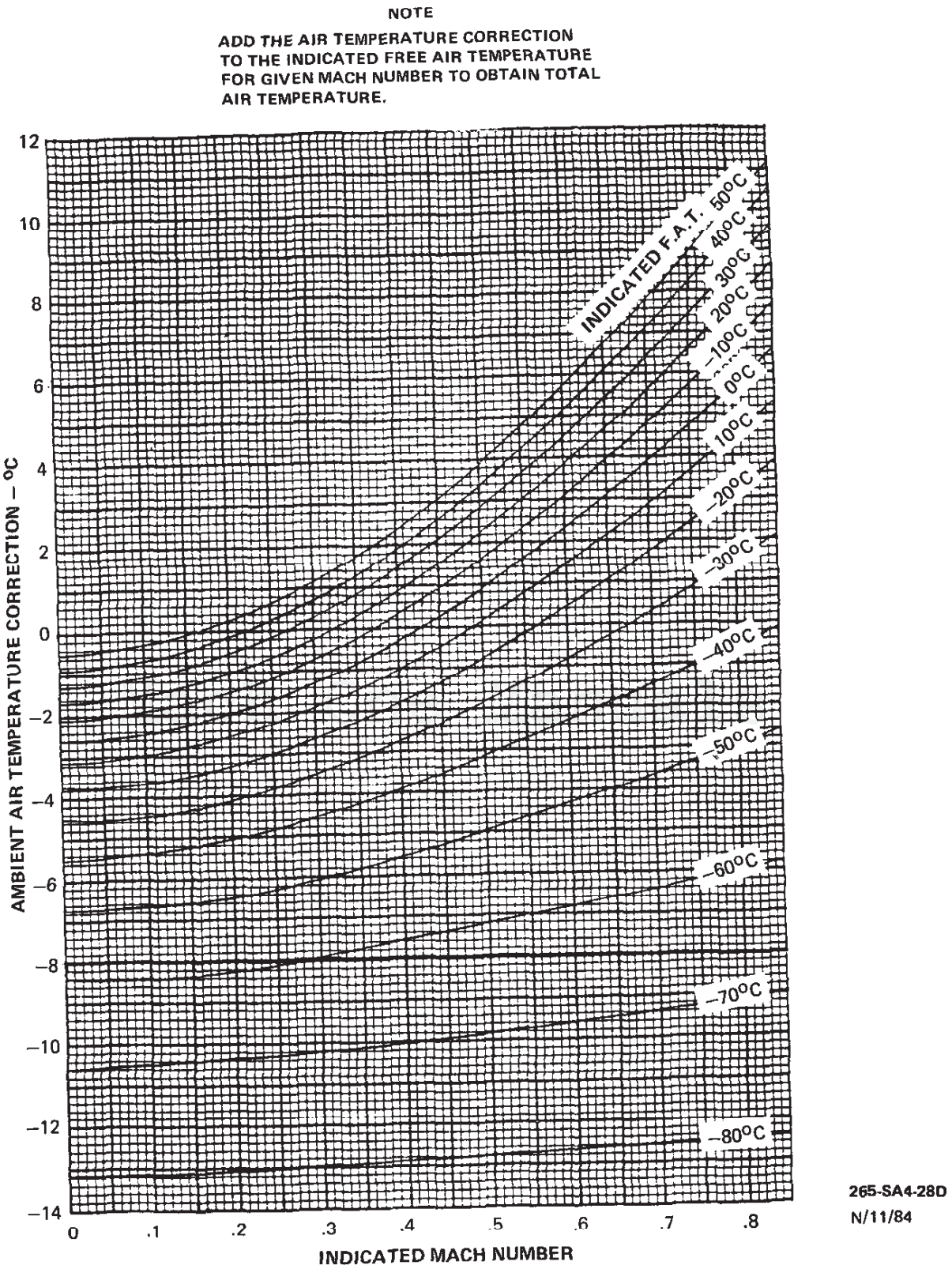
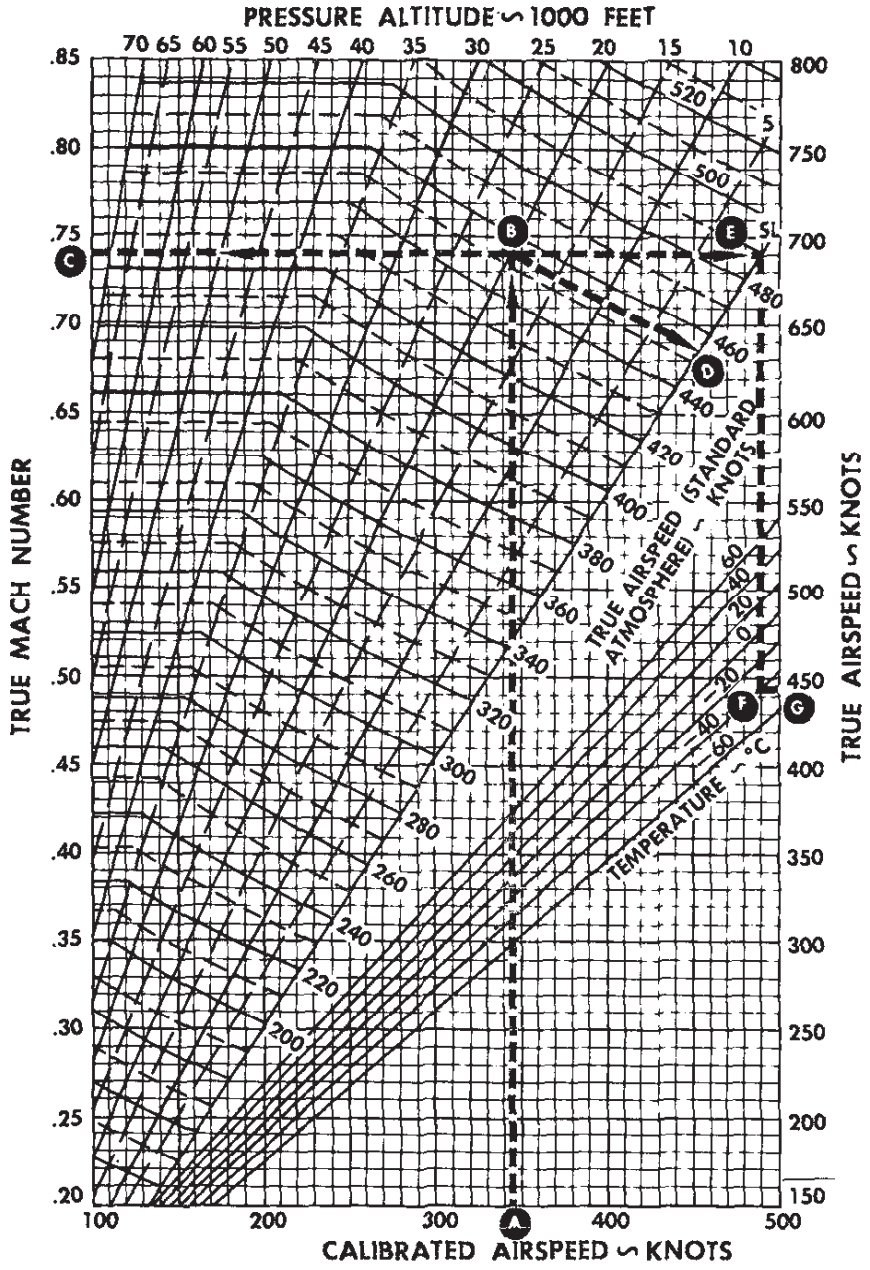


Figure 22-7. Total Air Temperature Correction

AIRSPEED CONVERSION DATA



HOW TO USE CHART

Enter with CAS (A), move vertically to altitude (B) to find true Mach number (C) and Standard Day true airspeed (D), move horizontally along true Mach number to sea-level altitude (E), and drop vertically to actual temperature (F) to find non-Standard Day true airspeed (G)

EXAMPLE

345 knots CAS at 20,000 feet is .74 true Mach number and 454 knots TAS on an ARDC Standard Day or 444 knots on a −40°C day.

T39D-1-93-5C

Figure 22-8. Airspeed Conversion Data

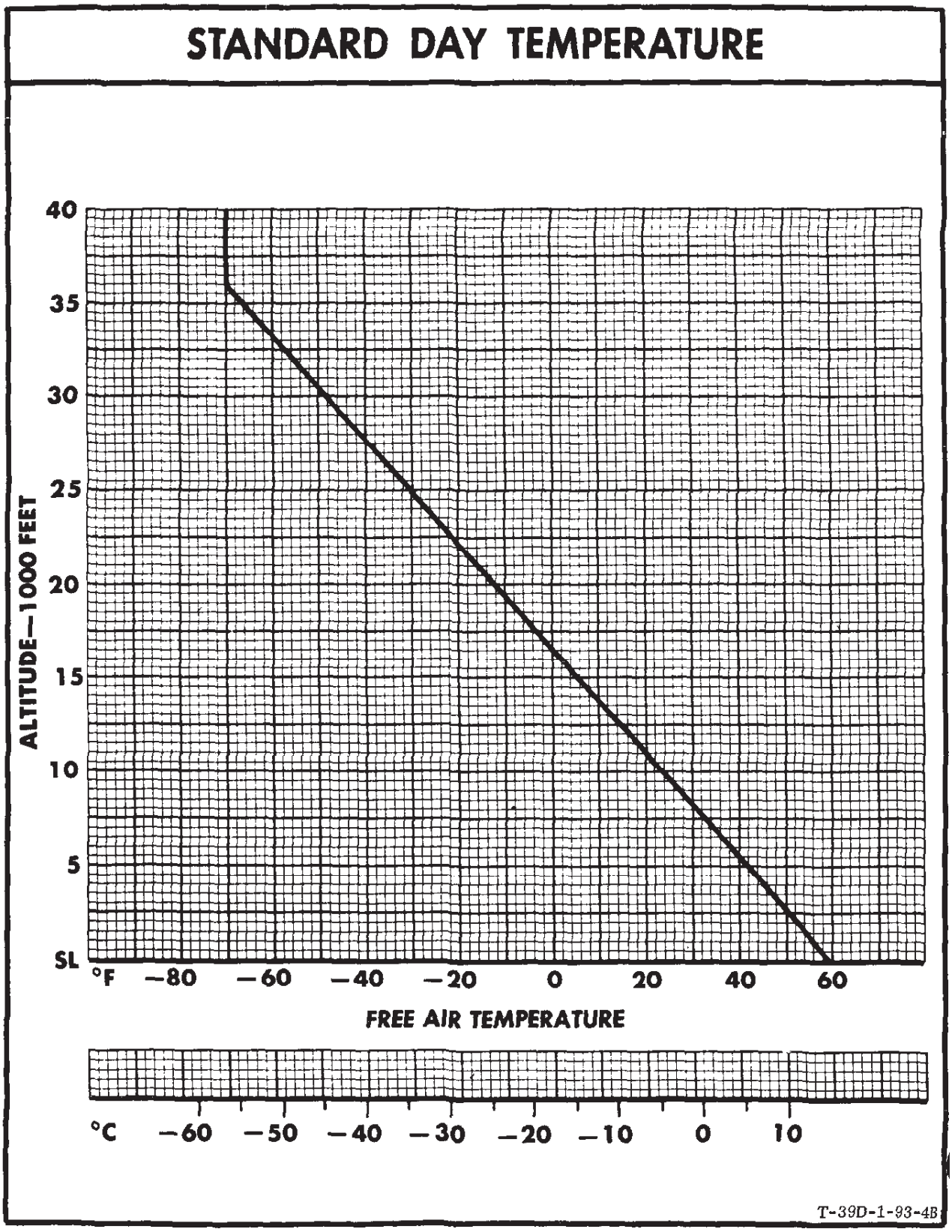


Figure 22-9. Standard Day Temperature

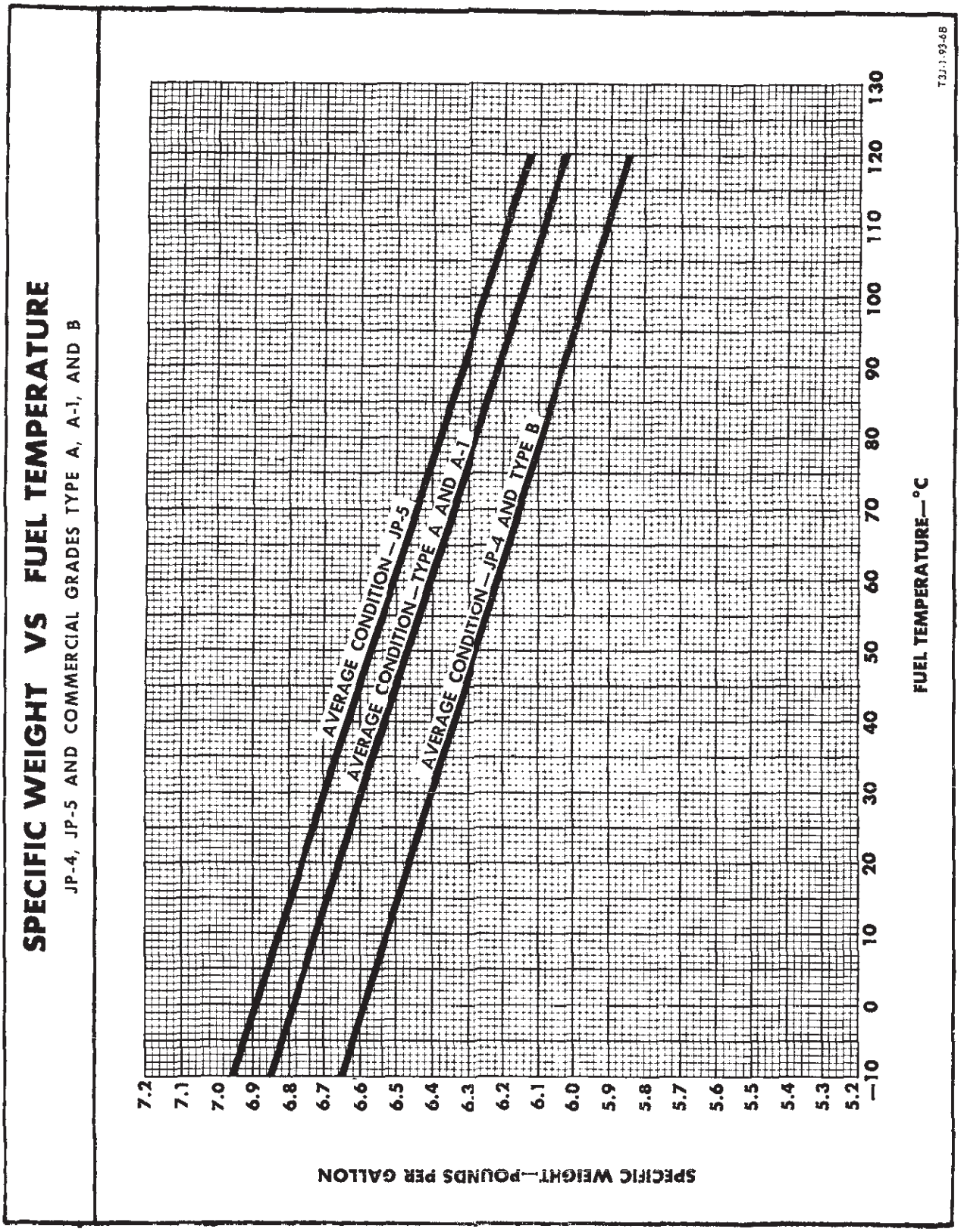


Figure 22-10. Specific Weight Versus Fuel Temperature

STANDARD ATMOSPHERE TABLE

STANDARD S L CONDITIONS:

TEMPERATURE 15°C (59°F)
PRESSURE 29.921 IN. Hg 2116.216 LB/SQ FT
DENSITY .0023769 SLUGS/CU FT
SPEED OF SOUND 1116.89 FT/SEC 661.7 KNOTS

CONVERSION FACTORS:

1 IN. Hg 70.727 LB/SQ FT
1 IN. Hg 0.49116 LB/SQ IN.
1 KNOT 1.151 MPH
1 KNOT 1.688 FT/SEC

ALTITUDE FEET	DENSITY RATIO σ	$\frac{1}{\sqrt{\sigma}}$	TEMPERATURE °C	TEMPERATURE °F	SPEED OF SOUND (KNOTS)	PRESSURE IN. Hg	PRESSURE RATIO P/PO
0	1.0000	1.000	15.0	59.0	661.7	29.92	1.0000
1,000	.9711	1.015	13.0	55.4	659.5	28.86	.9644
2,000	.9428	1.030	11.0	51.9	657.2	27.82	.9298
3,000	.9151	1.045	9.0	48.3	654.9	26.82	.8963
4,000	.8881	1.061	7.1	44.7	652.6	25.84	.8637
5,000	.8617	1.077	5.1	41.2	650.3	24.90	.8321
6,000	.8359	1.094	3.1	37.6	648.7	23.98	.8014
7,000	.8107	1.111	1.1	34.1	645.6	23.09	.7717
8,000	.7860	1.128	−0.8	30.5	643.3	22.23	.7429
9,000	.7620	1.146	−2.8	26.9	640.9	21.39	.7149
10,000	.7385	1.164	−4.8	23.4	638.6	20.58	.6878
11,000	.7156	1.182	−6.8	19.8	636.2	19.80	.6616
12,000	.6933	1.201	−8.8	16.3	633.9	19.04	.6362
13,000	.6715	1.220	−10.8	12.7	631.5	18.30	.6115
14,000	.6502	1.240	−12.7	9.1	629.0	17.58	.5877
15,000	.6294	1.260	−14.7	5.6	626.6	16.89	.5646
16,000	.6092	1.281	−16.7	2.0	624.2	16.23	.5423
17,000	.5894	1.303	−18.6	−1.5	621.8	15.58	.5206
18,000	.5702	1.324	−20.6	−5.1	619.4	14.95	.4997
19,000	.5514	1.347	−22.6	−8.7	617.0	14.35	.4795
20,000	.5331	1.370	−24.6	−12.2	614.6	13.76	.4599
21,000	.5153	1.393	−26.6	−15.8	612.1	13.20	.4410
22,000	.4980	1.417	−28.5	−19.3	609.6	12.65	.4228
23,000	.4811	1.442	−30.5	−22.9	607.1	12.12	.4051
24,000	.4646	1.467	−32.5	−26.5	604.6	11.61	.3881
25,000	.4486	1.493	−34.5	−30.0	602.1	11.12	.3716
26,000	.4330	1.520	−36.5	−33.6	599.6	10.64	.3557
27,000	.4178	1.547	−38.4	−37.1	597.1	10.18	.3404
28,000	.4031	1.575	−40.4	−40.7	594.6	9.742	.3256
29,000	.3887	1.604	−42.4	−44.3	592.1	9.315	.3113
30,000	.3747	1.634	−44.4	−47.8	589.5	8.904	.2976
31,000	.3612	1.664	−46.4	−51.4	586.9	8.507	.2843
32,000	.3480	1.695	−48.3	−54.9	584.4	8.125	.2715
33,000	.3351	1.727	−50.3	−58.5	581.8	7.757	.2592
34,000	.3227	1.760	−52.4	−62.1	579.2	7.402	.2474
35,000	.3106	1.794	−54.3	−65.6	576.6	7.061	.2360
36,000	.2989	1.829	−56.2	−69.2	574.0	7.733	.2250
37,000	.2853	1.872	−56.5	−69.7	573.7	6.418	.2145
38,000	.2719	1.918	−56.5	−69.7	573.7	6.118	.2045
39,000	.2592	1.964	−56.5	−69.7	573.7	5.832	.1949
40,000	.2471	2.012	−56.5	−69.7	573.7	5.559	.1858
41,000	.2356	2.019	−56.5	−69.7	573.7	5.299	.1771
42,000	.2245	2.110	−56.5	−69.7	573.7	5.051	.1681
43,000	.2140	2.161	−56.5	−69.7	573.7	4.815	.1609
44,000	.2040	2.214	−56.5	−69.7	573.7	4.590	.1534
45,000	.1945	2.267	−56.5	−69.7	573.7	4.376	.1462
46,000	.1854	2.322	−56.5	−69.7	573.7	4.171	.1394
47,000	.1768	2.379	−56.5	−69.7	573.7	3.976	.1329
48,000	.1685	2.436	−56.5	−69.7	573.7	3.791	.1267
49,000	.1606	2.495	−56.5	−69.7	573.7	3.614	.1208

T3J-1-93-88

Mach One Manuals

Figure 22-11. Standard Atmosphere

CHAPTER 23

Takeoff

23.1 TAKEOFF PLANNING

This chapter covers the information and charts to be used to obtain takeoff speeds and distances. The terms used in the planning procedure are defined in the following paragraphs. Takeoff 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. Increases in any of these variables, except wind, tend to increase takeoff ground roll to a point where, on a takeoff in which normal techniques are used, the takeoff may not be successfully completed in the available runway length. The use of the takeoff planning charts is explained in the following paragraphs, and chase-through examples are provided where necessary. See Figure 23-1 for takeoff profile description.

23.2 RUNWAY DISTANCE MARKING SYSTEM

The numbering and placement of runway distance markers reflects the distance remaining to the end of the runway in 1,000-foot increments. These markers are placed alongside the runway, and the appropriate markers become the takeoff ground roll distance marker. In accordance with the marker system on a runway length in excess of the 1,000-foot interval (10,500), one-half the length over the exact thousands in feet (1/2 (10,500 - 10,000)) shall be added to the distance above on the markers to determine the actual distance remaining (i.e., at marker No. 6, the distance remaining would be 6,250 feet).

23.3 TAKEOFF RUNWAY REQUIREMENTS — ONE ENGINE

The single-engine takeoff runway requirement charts (FO-15 and FO-16) provide the FAA-required runway length. FAA runway required is based on the balanced field concept; that is, the distance from brake release to the 35-foot obstacle is equal to the distance from brake release to V_1 and stop (in each case assuming an engine failure at V_1 speed). The runway required distances shown on these charts assume an engine failure at V_1 and acceleration on one engine through V_R speed to liftoff. An engine failure at a speed greater than V_1 will provide a shorter ground roll distance. With an engine failure at any speed less than V_1, abort takeoff.

If the existing runway is not of sufficient length for the conditions (ambient temperature, field elevation, or gross weight) reduce the gross weight or wait for a lower temperature prior to takeoff.

The necessary takeoff speeds used in conjunction with the runway requirements are found on the V_1, V_R, and V_2 speed charts in this chapter. The takeoff runway requirement charts are designed for a normal aircraft configuration; however, correction factors are included on the charts for engine inlets anti-ice. The charts are based on using a dry, hard-surface runway. The charts are also based on using a tire inflation pressure of 185 psi. When tire inflation pressure of 214 psi is used, the takeoff and landing distance charts must be adjusted by adding 200 feet to the computed data.

It will be noted that the 13,000- and 14,000-pound gross weight curves are broken at the lower end. This abrupt change in the curve is imposed by the V_{MCG} limitation.

23.4 V_1, V_R, AND V_2 SPEEDS (TAKEOFF SPEEDS)

The takeoff speed charts (FO-17, FO-18, and FO-19) provide V_1, V_R, and V_2 speeds for particular takeoff condition (airport air temperature, airport pressure altitude, gross weight at brake release, and engine inlets anti-ice on or off).

Enter chart with proper airport air temperature, extend horizontally to right, and intercept airport pressure altitude line. The sharp break in the airport pressure altitude curves is caused by the engine flat rate limit. Descend to gross weight at brake release curve and then proceed horizontally to right to intercept V_1 and V_R weight curves. The V_1 speed weight curves have a sharp break at the top: this is to ensure

the V_1 speed is always greater than V_{MCG}. At point of interception, read down for V_1 and V_R indicated airspeeds. V_2 speeds are provided on the gross weight at brake release curves.

EXAMPLE (Figure FO-18):

1. Airport air temperature (40 °F)
2. Airport pressure altitude (5,000 feet)
3. Cabin pressure on
4. Anti-ice ram air inlets
5. Gross weight at brake release (17,000 pounds)
6. V_1 speed (110 KIAS)
7. V_R speed (116 KIAS)
8. V_2 speed (122 KIAS).

23.5 V_1 SPEED AS LIMITED BY BRAKE KINETIC ENERGY

The maximum allowable takeoff weight may be restricted by the energy absorption capability of the wheelbrakes in the event of a rejected takeoff. The V1 speed as limited by brake kinetic energy (F0-20, FO-21, and FO-22) is the maximum speed on the ground from which a stop can be accomplished within the demonstrated energy capabilities of the brakes. Determine from V_1 speed, as limited by brake kinetic energy chart, that limiting V_1 speed is equal to or higher than V_1 speed, as shown on applicable V_1, V_R, and V_2 speed charts. Normally, the maximum V_1 will be greater than the normal V_1 except for the conditions of a downhill runway slope and a tailwind at the higher altitudes and temperatures. Under these conditions, the guidelines for wind and slope on the runway requirements chart will give the correct runway required.

EXAMPLE (Figure FO-17):

A. Takeoff gross weight — 14,580 Pounds

B. Indicated airspeed — 142.4 Knots

C. Altitude baseline — Sea Level

D. Altitude — 2,400 Feet

E. Air temperature baseline — 59 °F

F. Air temperature — 84 °F

G. Slope baseline — 0 Percent

H. Runway slope — 1 Percent Uphill

I. Wind baseline — 0 Knots

J. Headwind — 4 Knots

K. Maximum V_1 (1-percent slope, 4-knot headwind)-139 Knots.

23.6 ACCELERATION CHECK

The acceleration check chart gives an indicated airspeed at any desired checkpoint distance and, conversely, the ground run distance required to accelerate to any desired indicated airspeed. (See Figures 23-2 and 23-3.) An allowable airspeed tolerance of 5 KIAS is the maximum speed deficiency that is acceptable at any checkpoint distance. If acceleration is deficient to the extent that the speed tolerance is exceeded at any checkpoint distance, a serious malfunction of the aircraft is indicated and the takeoff should be aborted even if excess runway is available. Use of the chart is explained by a chase-through sample problem.

23.7 CLIMB-TAKEOFF — GEAR UP (Second Segment)

The climb-takeoff — gear up chart (FO-23 and FO-24) is used for the climb segment starting when the landing gear and wing flaps are retracted to a point at least 800 feet above the airport pressure altitude. This segment is defined as the second-segment climb and is flown at an airspeed equal to V_2.

During second-segment climb, the aircraft must be capable of maintaining a climb gradient of at least 2.4 percent with one engine inoperative and the other engine at takeoff thrust.

Third-segment climb is that segment that may be used for level-flight acceleration at 800 feet above the airport pressure altitude to final takeoff-climb speed.

23.8 SECOND-SEGMENT LIMITING WEIGHT

The second-segment limiting weight charts (Figures 23-4 through 23-7) are provided so that it can be determined if second-segment climb gradients may be met with one engine inoperative. These only become critical at the heavier weight and at the higher airport pressure altitudes. Enter chart with airport air

temperature, proceed horizontally to airport pressure altitude curve, and extend vertically to maximum allowable weight at brake release readout. This gross weight (at brake release) is the maximum gross weight allowable for the particular takeoff condition to maintain the required second-segment climb gradient with one engine inoperative.

23.9 TAKEOFF RUNWAY REQUIREMENTS — TWO-ENGINE OPERATION

The normal (two-engine) takeoff distance charts (FO-25 and FO-26) provide total distance and ground roll for a normal two-engine takeoff.

Enter chart at left with airport air temperature and proceed to right to obtain total distance from brake release to an altitude of 35 feet. This performance is obtained by using takeoff thrust on both engines, flaps up, engine inlets anti-ice on or off, and using the V_R speed from the applicable V_1, V_R, and V_2 speed chart in this chapter. For flight planning purposes, to determine if runway is of sufficient length to cope with an engine failure, use applicable takeoff runway requirement chart for single-engine takeoff.

23.10 STATIC TAKEOFF THRUST ENGINE PRESSURE SETTING

The static takeoff thrust EPR setting charts (Figures 23-8 through 23-10) show the EPR required for takeoff for any given ambient temperature and pressure altitudes. The temperature gauge in the cockpit should not be used for takeoff; the tower reported temperature should be used.

23.11 CROSSWIND CHART

The crosswind chart (Figure 23-11) is used to determine the headwind or tailwind component in order to compute the takeoff and landing distances. It is also used to determine the crosswind component and whether a safe takeoff or landing can be made. The chart permits determining the minimum nosewheel liftoff and touchdown speeds when limited by crosswind.

Note

Figure 23-11 contains a sample problem for determining headwind or tailwind and crosswind components and the nosewheel liftoff and touchdown speeds.

For the purpose of determining the crosswind component and whether a safe takeoff or landing can be made, the chart is divided into two areas (recommended and not recommended). The crosswind baseline is based on the use of aerodynamic steering (rudder) only and does not include the effects of nosewheel steering or differential braking.

Rudder effectiveness varies directly with speed. Considering this fact, assume full rudder is required to maintain aircraft heading aligned with runway heading at rotation. It can be seen, then, that at aircraft liftoff speed there will be virtually no additional rudder available to obtain a crab condition necessary to maintain flightpath in line with the runway. Also, if full rudder is required in a sideslip approach to landing, full rudder will not be adequate to maintain directional control at the lower speeds encountered during flare and touchdown. The chart considers these requirements and therefore includes minimum nosewheel liftoff and nosewheel touchdown speeds that have to be observed when limited by crosswind.

The nosewheel liftoff speed used should not be less than the normal precomputed liftoff speed as obtained from the takeoff ground roll chart.

WARNING

- As the chart shows, certain crosswind conditions result in a component that will not permit safe takeoff or landing, no matter what the increase in speed.
- Increases in nosewheel liftoff or touchdown speed will obviously mean takeoff or landing rolls in excess of normal. If speed increase required is excessive, the available runway length could be exceeded; consequently, takeoff or landing should not be attempted.

23.12 CRITICAL FIELD LENGTH

Critical field length is the minimum permissible runway length to be available for takeoff for any particular gross weight, temperature, and pressure altitude. Any engine failure at refusal speed on a critical length runway would require all remaining runway for abort or completion of single-engine takeoff with no specific allowance for height at the end of the available runway.

The critical field length chart is used to determine the minimum field length required for takeoff. (Figures 23-12 and 23-14). In addition, the chart provides

corrections for wind and runway slope and runway condition. The "chase through" indicates critical field length for the sample conditions.

The takeoff runway requirement chart is substantially different than the critical field length chart and is more restrictive because of a different data base criteria. The takeoff runway requirement chart is based on attaining a height over the end of the runway of at least 35 feet on single engine. The critical field length chart makes no allowance for height at the end of the runway, assuming single-engine failure at or above refusal speed.

WARNING

If critical field length exceeds the runway available, the takeoff shall not be attempted at that gross weight.

The maximum gross weight limited by critical field length may be found by reading into the chart with given atmospheric data and runway length.

Note

When takeoffs are made with a combination of high airport pressure altitude, high temperature, and high gross weight, 200-fpm climb gradients may not be obtained if an engine failure is experienced after takeoff. The gross weight limiting factor for a one-engine-out condition is listed on the critical field length chart. When the outside air temperature is less than 60 °F, there is no gross weight limiting factor.

The takeoff speeds shown on the critical field length chart are for single-engine takeoffs. These speeds differ slightly from two-engine takeoff speeds because aircraft acceleration is faster with two-engine operation. It is necessary to adhere to this speed in order to obtain the performance shown on the chart.

Nosewheel liftoff speed, as obtained from the takeoff ground roll chart for two-engine operation, remains the same for single-engine takeoff. When using critical field length charts, the refusal speed should be derived from the refusal speed chart (Figures 23-13 and 23-15). Under certain conditions, the refusal speed is a lower speed than V_1 as derived from the takeoff runway requirement charts.

23.13 REFUSAL SPEED

This chart shows refusal speed based on the added criteria of runway length and condition. This chart should be used in conjunction with the critical field length chart to determine actual refusal speed. Under conditions where refusal speed is greater than V_R, then V_R should be used as the decision/refusal speed.

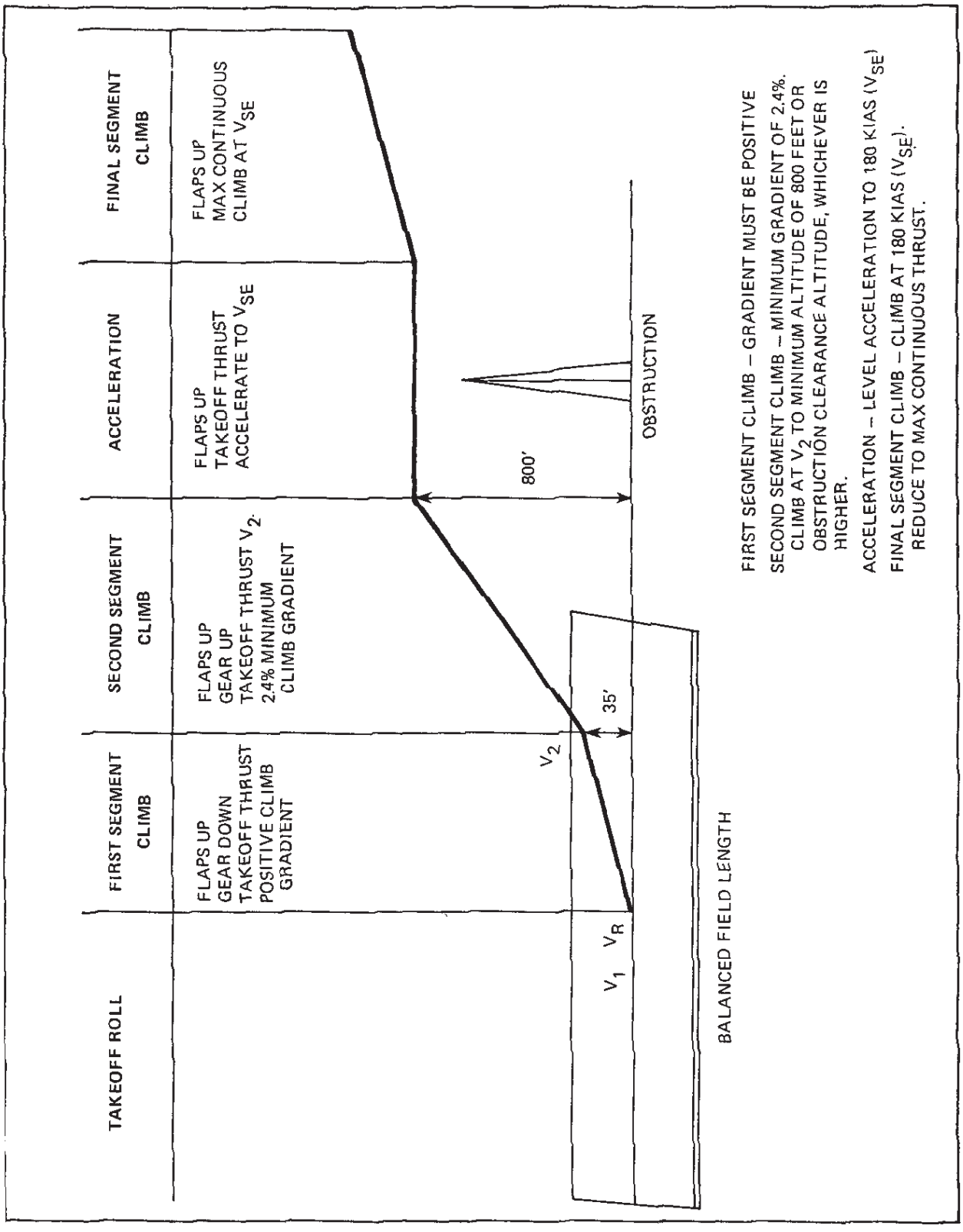


Figure 23-1. Single-Engine Failure at V_1 — Takeoff Continued

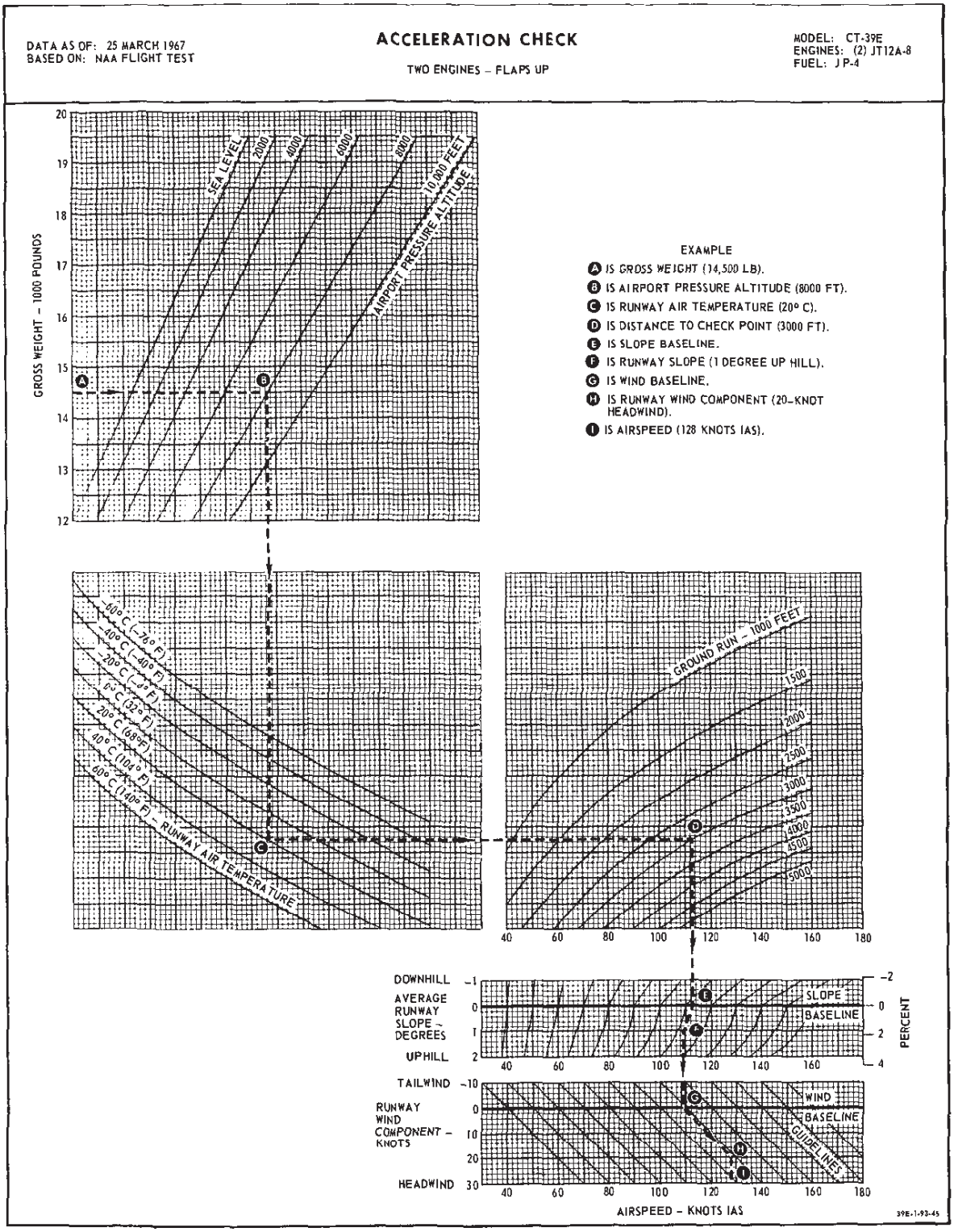


Figure 23-2. Acceleration Check — CT-39E

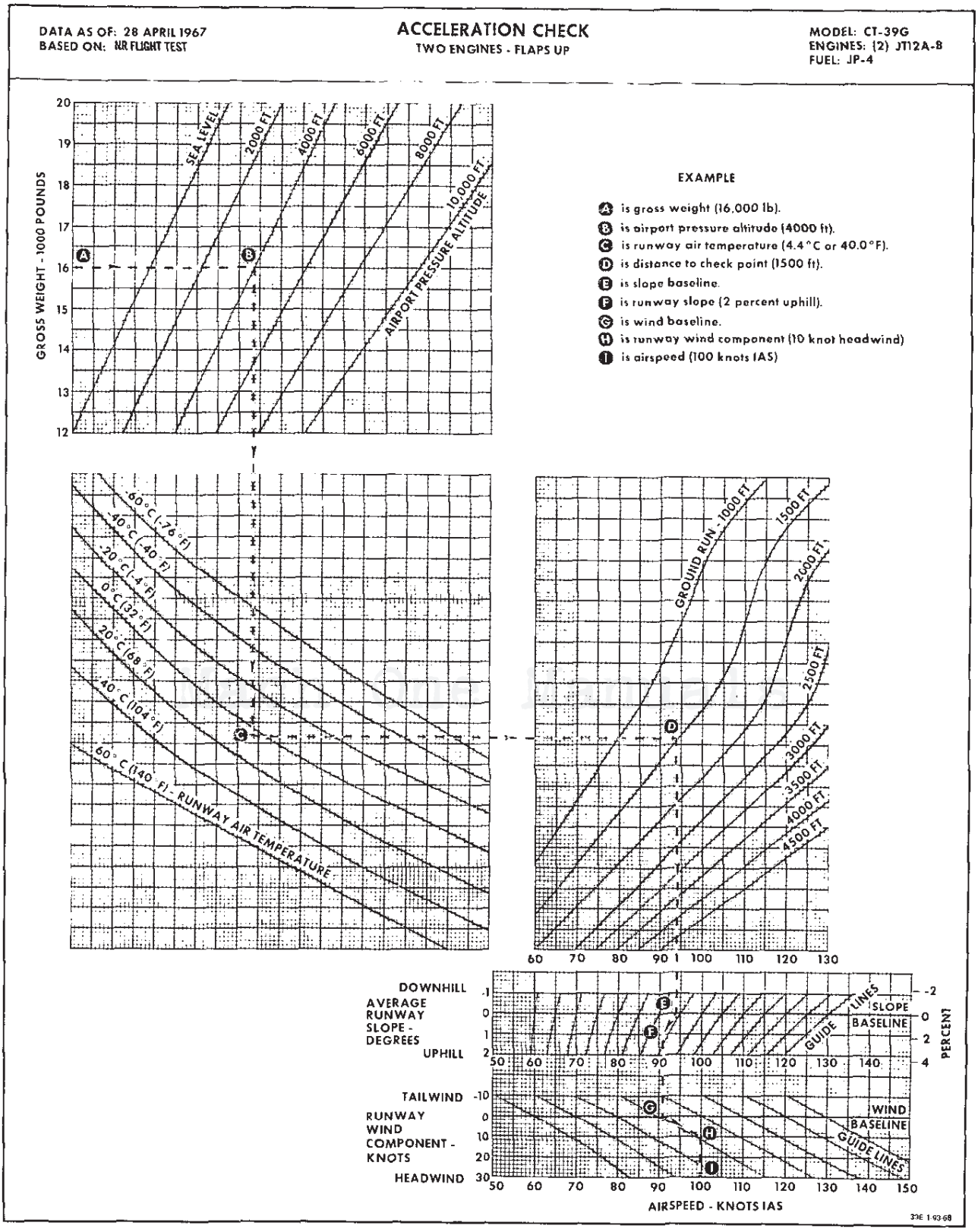


Figure 23-3. Acceleration Check — CT-39G

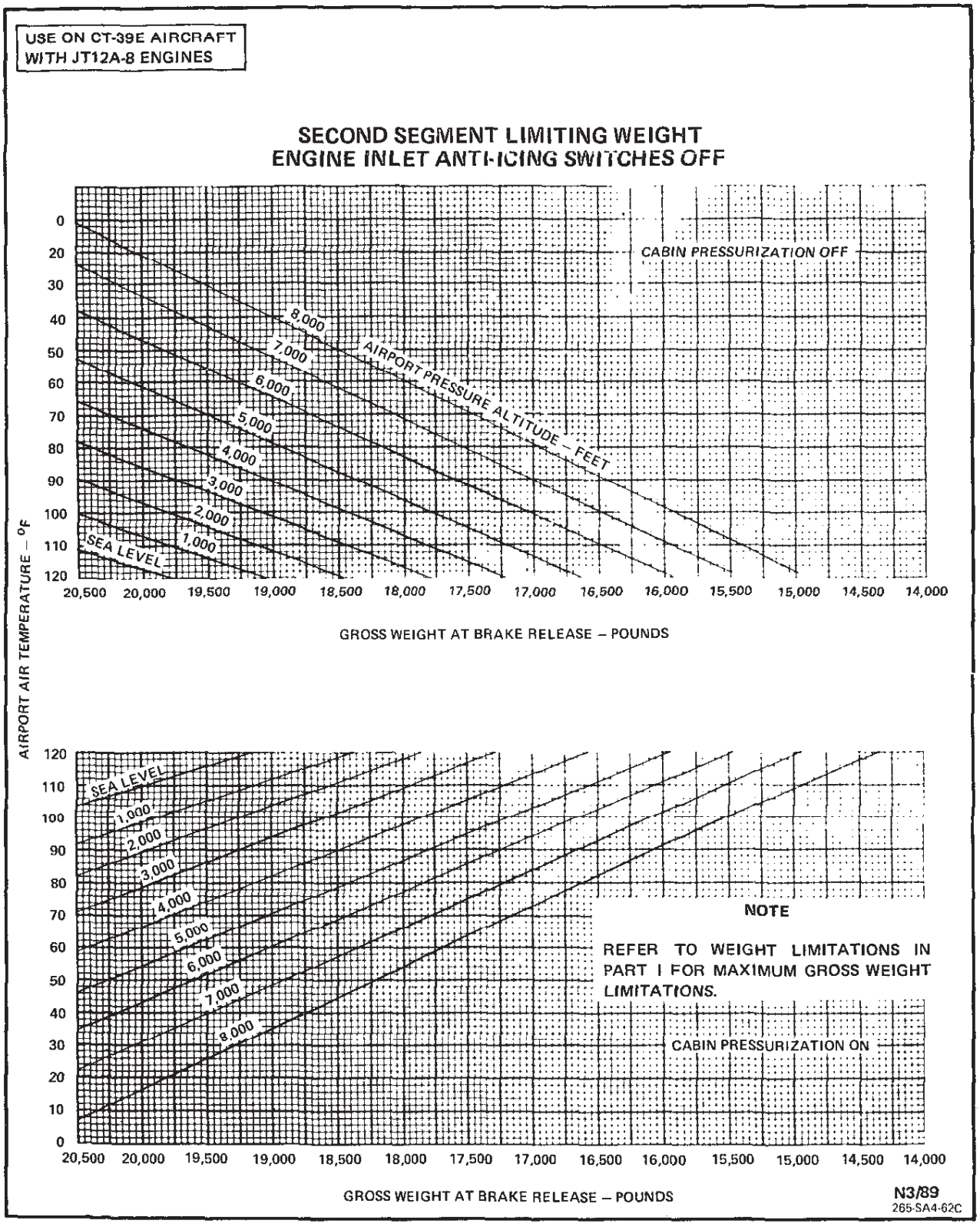


Figure 23-4. Second Segment Limiting Weight-Engine Inlet Anti-Icing Switches OFF — CT-39E

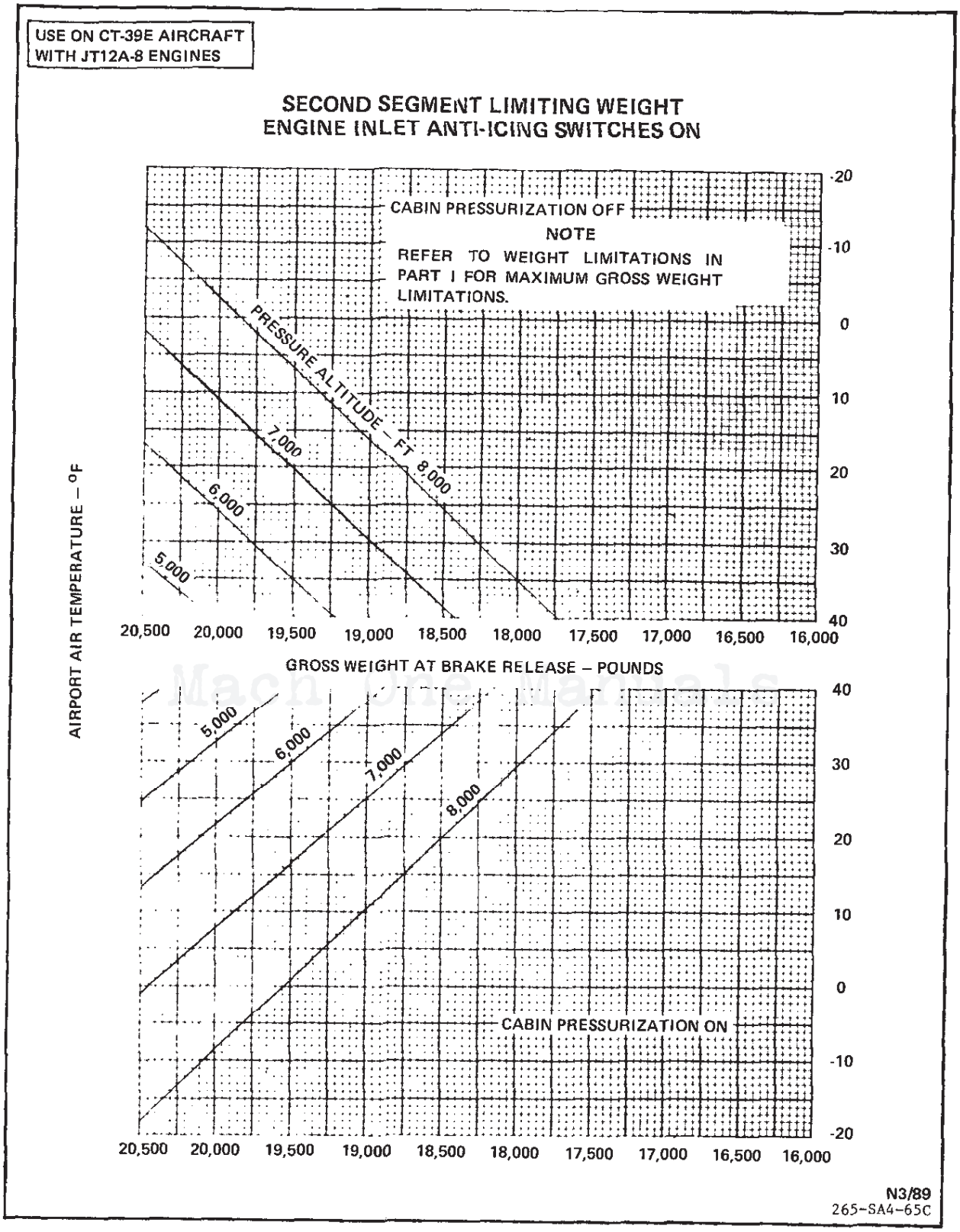


Figure 23-5. Second Segment Limiting Weight-Engine Inlet Anti-Icing Switches ON — CT-39E

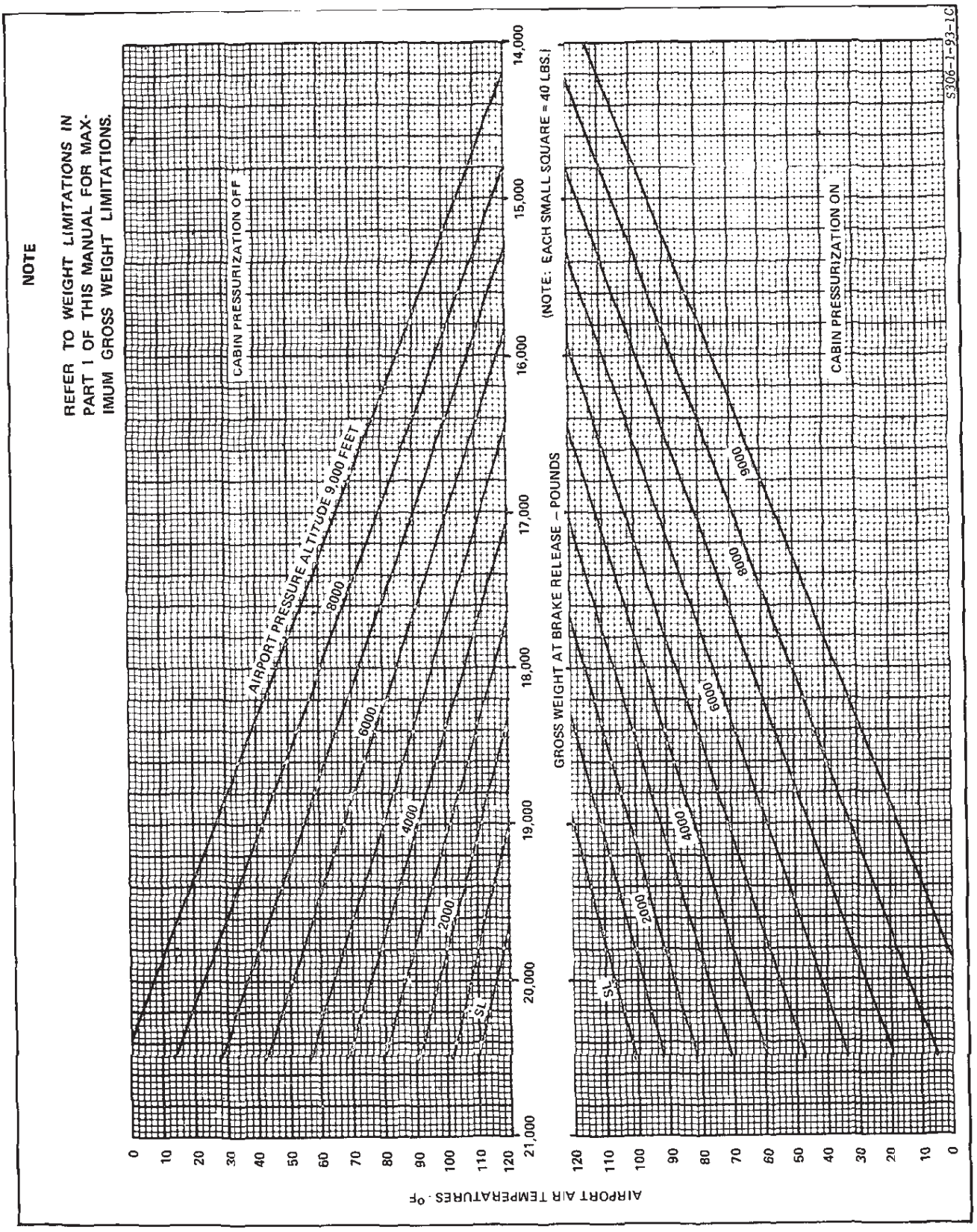


Figure 23-6. Second Segment Limiting Weight-Engine Inlet Anti-Icing Switches OFF — CT-39G

NOTE

REFER TO WEIGHT LIMITATIONS IN PART I OF THIS MANUAL FOR MAXIMUM GROSS WEIGHT LIMITATIONS.

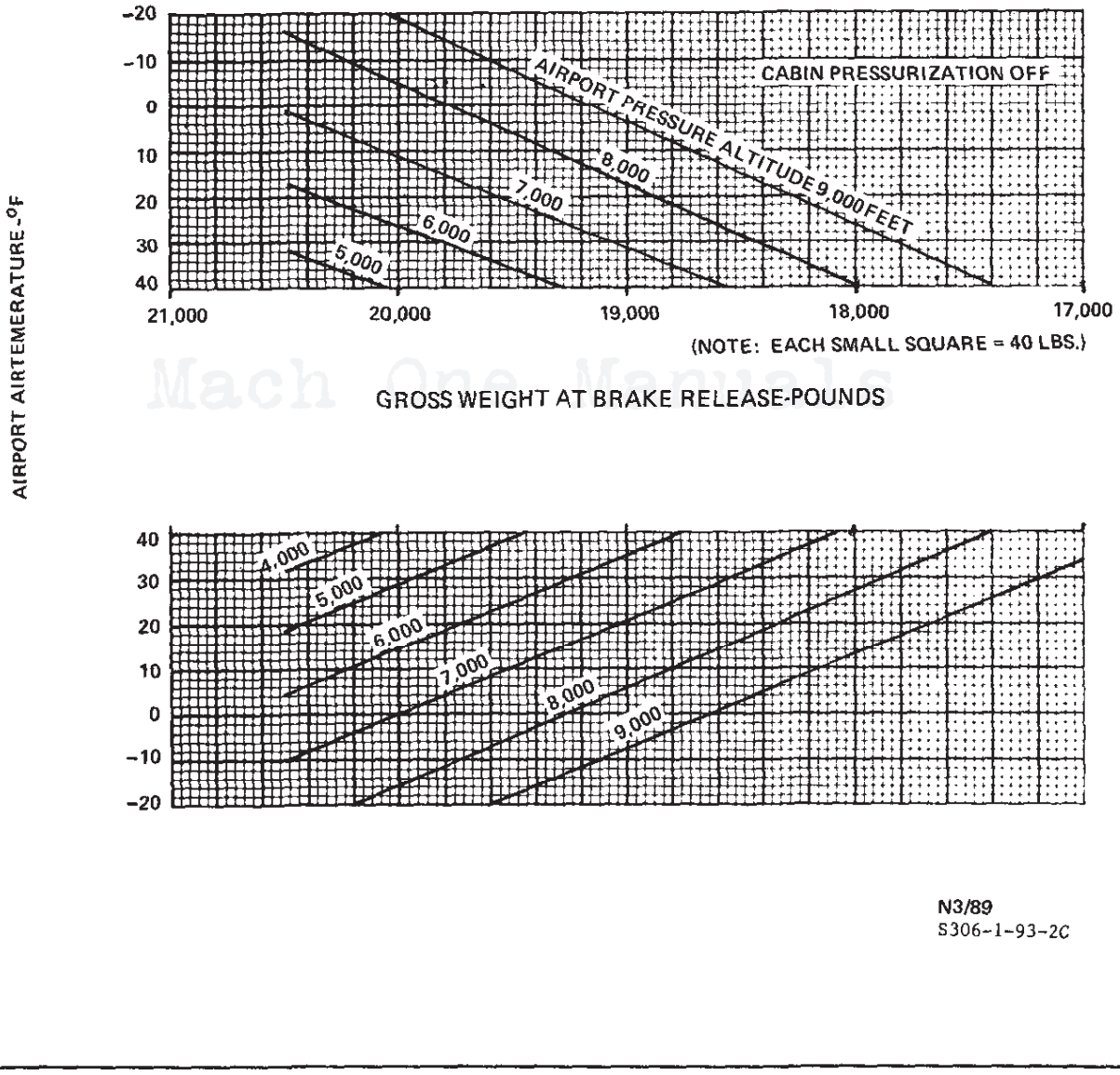


Figure 23-7. Second Segment Limiting Weight-Engine Inlet Anti-Icing Switches ON — CT-39G

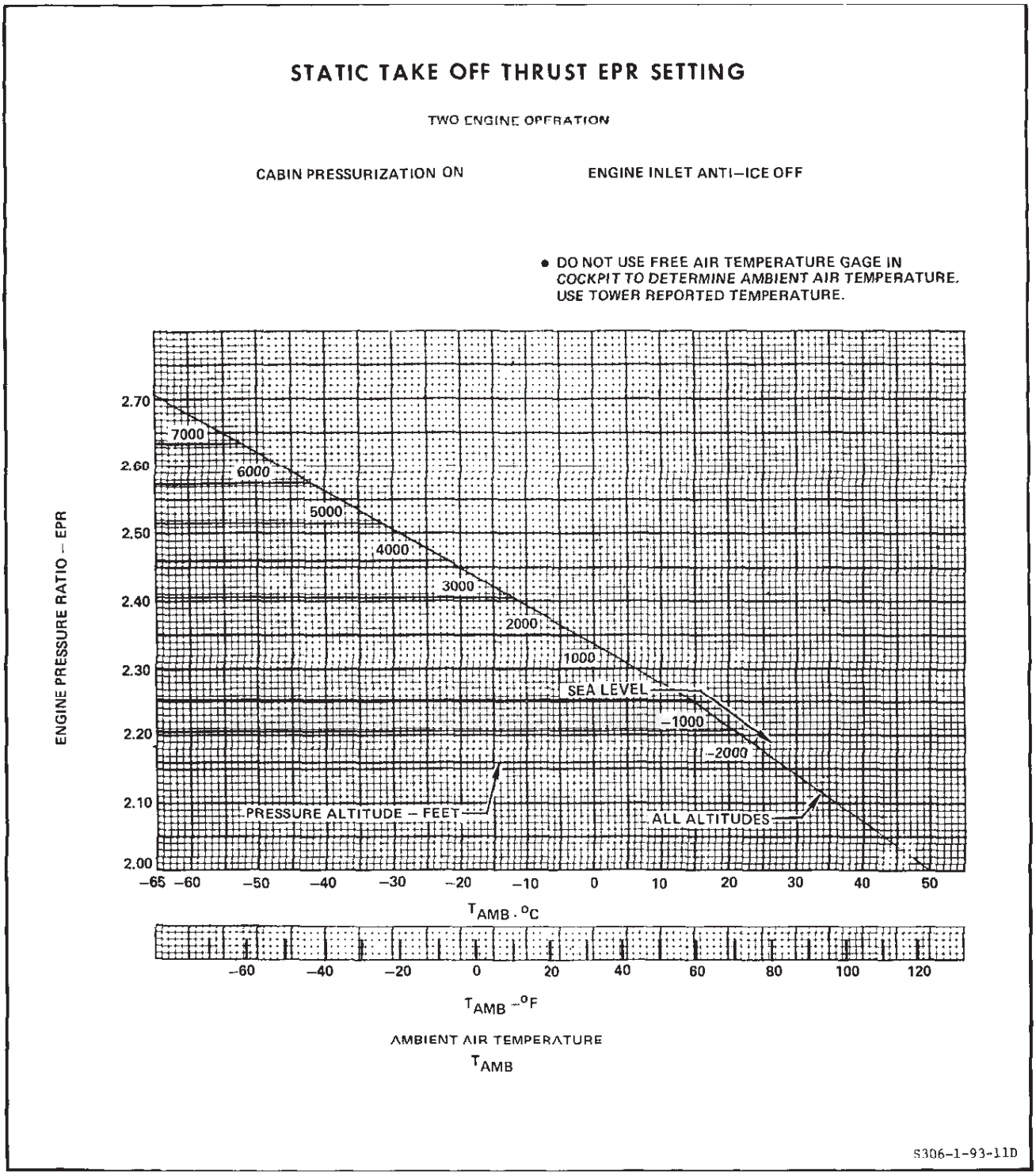


Figure 23-8. Static Takeoff Thrust EPR Setting-Cabin Pressurization ON — Anti-Ice OFF

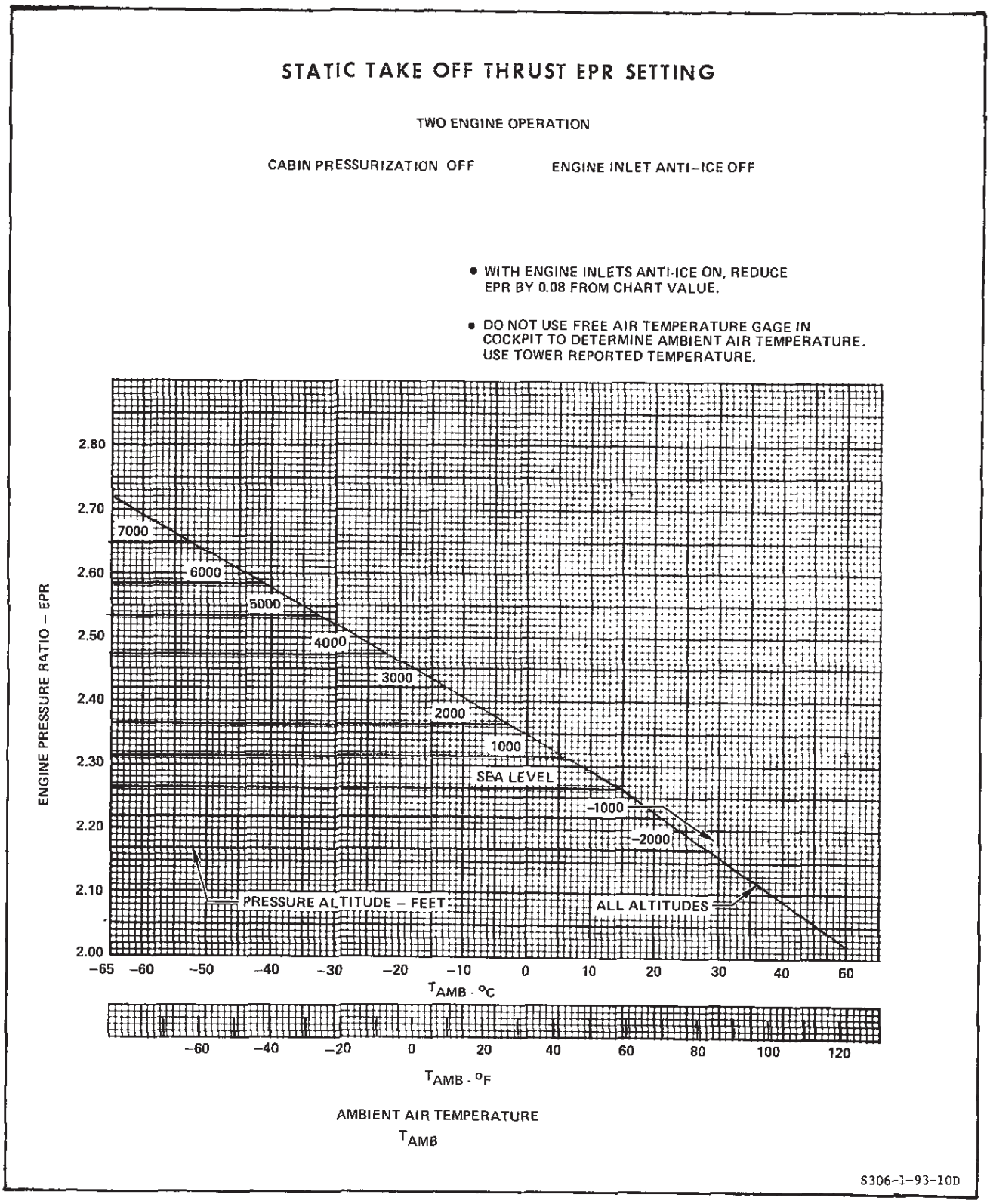


Figure 23-9. Static Takeoff Thrust EPR Setting-Cabin Pressurization OFF — Anti-Ice OFF

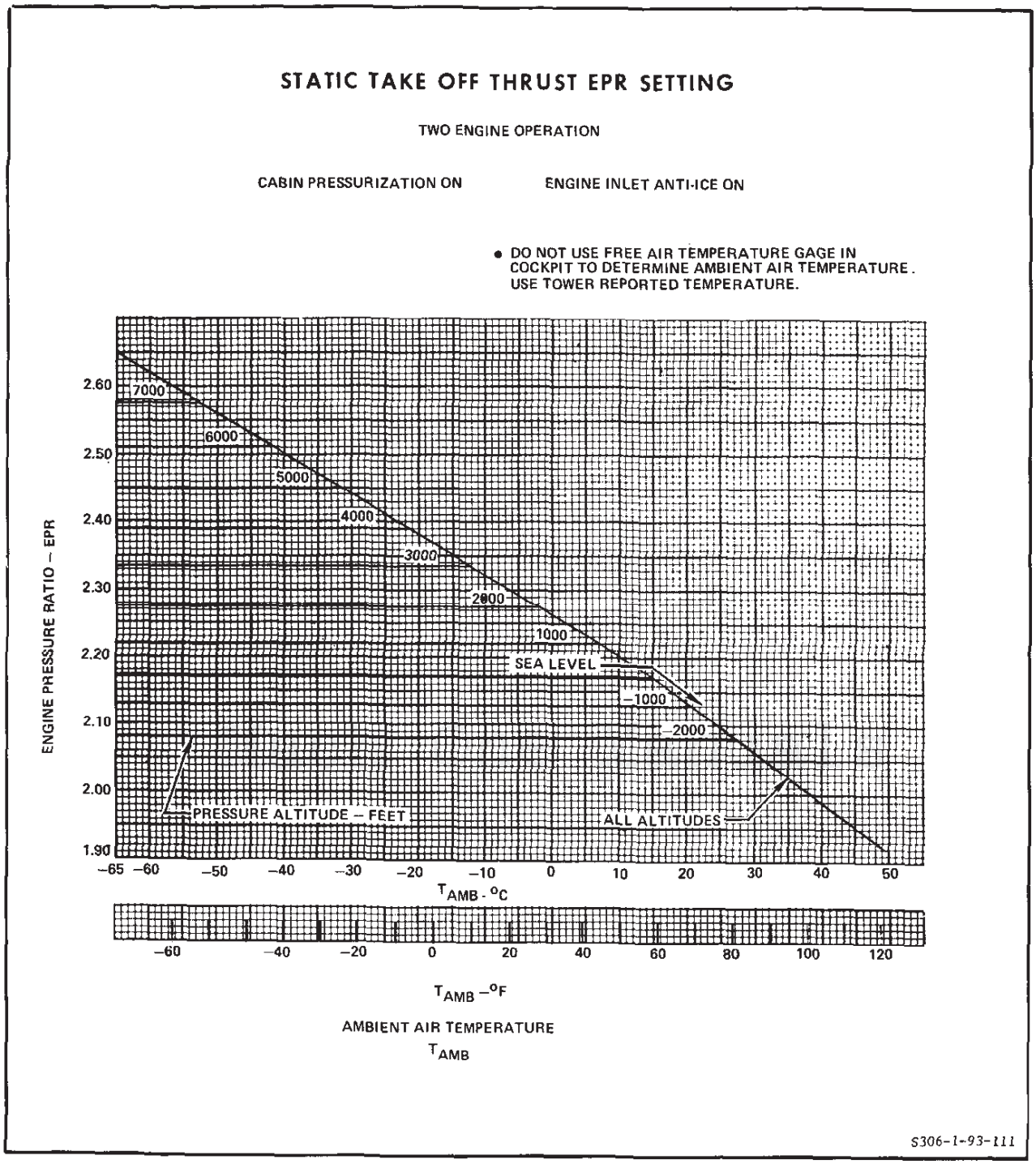
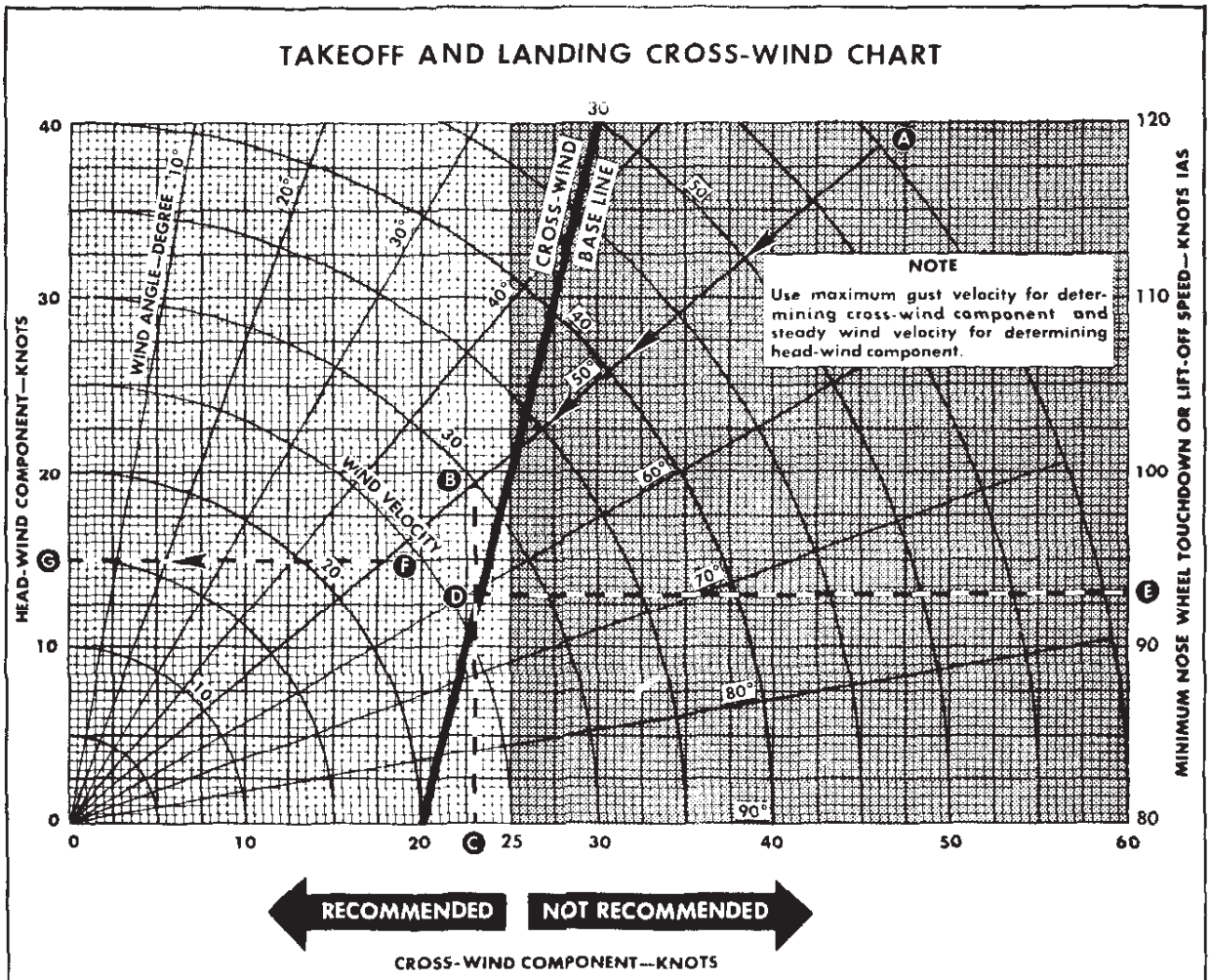


Figure 23-10. Static Takeoff Thrust EPR Setting-Cabin Pressurization ON — Anti-Ice ON



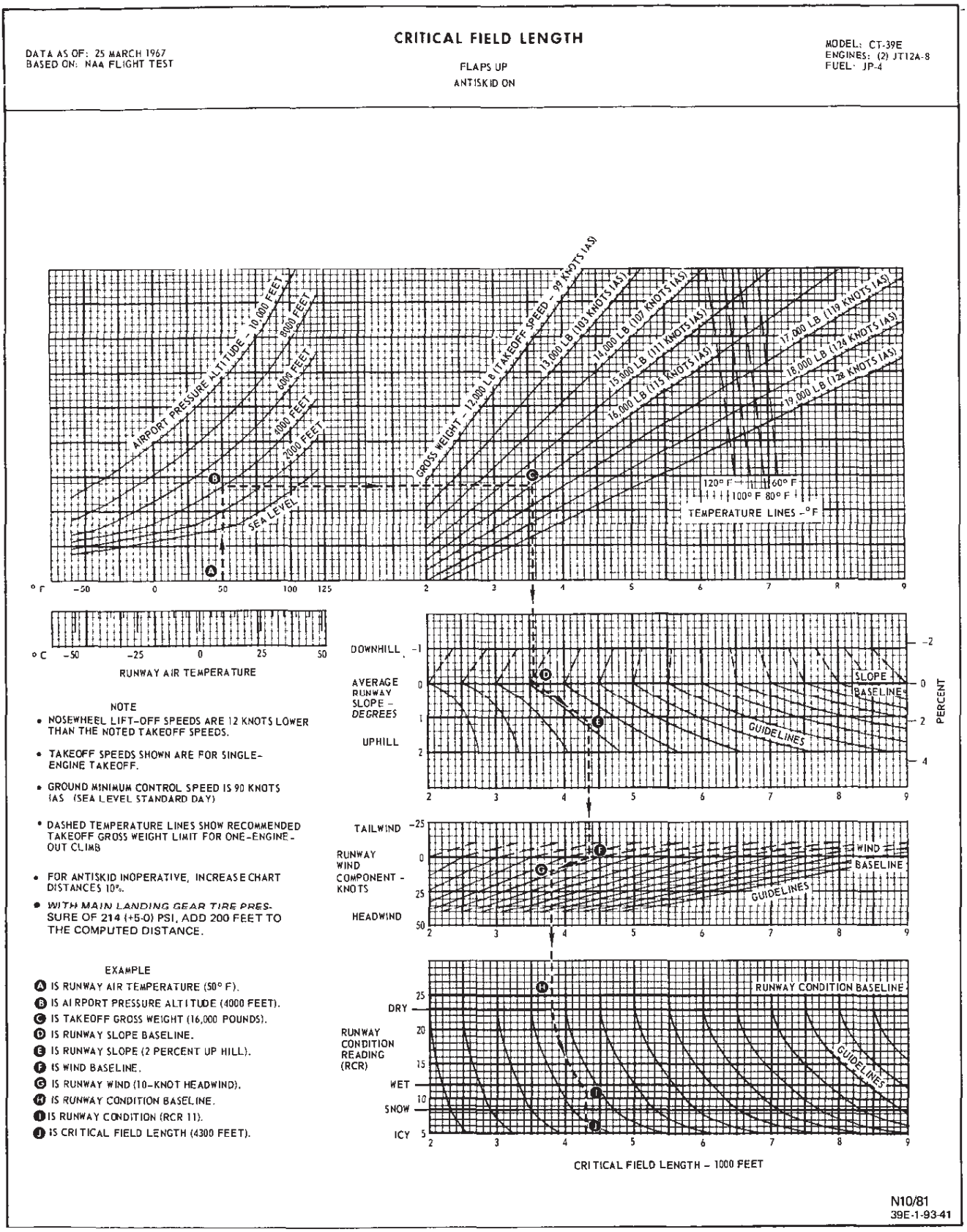
NOTE

- This information does not apply to single-engine operation.
- Minimum speed for rudder effectiveness is 60 knots.
- To obtain tail-wind component, enter chart with direction of wind taken from reciprocal of runway heading.
- The cross-wind base line is based on the use of aerodynamic steering only and does not include the effects of nose wheel steering.
- Recommended maximum cross-wind component is 25 knots.

EXAMPLE

- (A) is cross-wind direction from runway heading (50 degrees).
- (B) is maximum gust velocity (30 knots).
- (C) is cross-wind component (23 knots).
- (D) Proceed vertically until cross-wind base line is contacted.
- (E) Proceed horizontally to right to determine minimum nose wheel lift-off or touchdown speed (93 knots IAS).
- (F) is steady wind velocity (23 knots).
- (G) is head-wind component (15 knots).

Figure 23-11. Takeoff and Landing Crosswind Chart



NOTE

- NOSEWHEEL LIFT-OFF SPEEDS ARE 12 KNOTS LOWER THAN THE NOTED TAKEOFF SPEEDS.
- TAKEOFF SPEEDS SHOWN ARE FOR SINGLE-ENGINE TAKEOFF.
- GROUND MINIMUM CONTROL SPEED IS 90 KNOTS IAS (SEA LEVEL STANDARD DAY)
- DASHED TEMPERATURE LINES SHOW RECOMMENDED TAKEOFF GROSS WEIGHT LIMIT FOR ONE-ENGINE-OUT CLIMB
- FOR ANTISKID INOPERATIVE, INCREASE CHART DISTANCES 10%.
- *WITH MAIN LANDING GEAR TIRE PRESSURE OF 214 (+5-0) PSI, ADD 200 FEET TO THE COMPUTED DISTANCE.*

EXAMPLE

- (A) IS RUNWAY AIR TEMPERATURE (50° F).
- (B) IS AIRPORT PRESSURE ALTITUDE (4000 FEET).
- (C) IS TAKEOFF GROSS WEIGHT (16,000 POUNDS).
- (D) IS RUNWAY SLOPE BASELINE.
- (E) IS RUNWAY SLOPE (2 PERCENT UP HILL).
- (F) IS WIND BASELINE.
- (G) IS RUNWAY WIND (10-KNOT HEADWIND).
- (H) IS RUNWAY CONDITION BASELINE.
- (I) IS RUNWAY CONDITION (RCR 11).
- (J) IS CRITICAL FIELD LENGTH (4300 FEET).

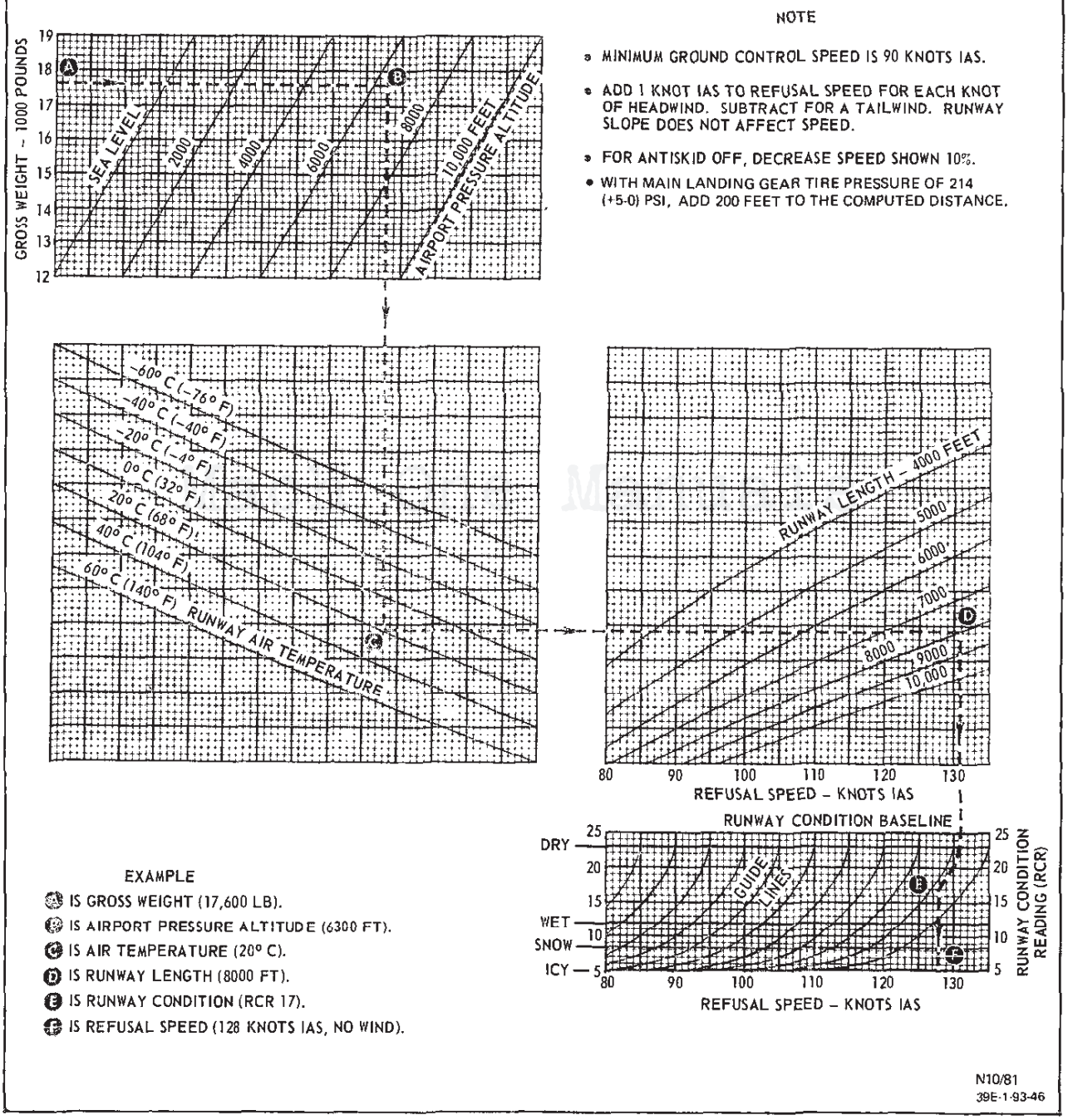
Figure 23-12. Critical Field Length — JT12A-8 Engines

DATA AS OF: 25 MARCH 1967
BASED ON: NAA FLIGHT TEST

REFUSAL SPEEDS

TAKEOFF THRUST – FLAPS UP

MODEL: CT-39E
ENGINES: JT12A-8
FUEL: JP-4



NOTE

- MINIMUM GROUND CONTROL SPEED IS 90 KNOTS IAS.
- ADD 1 KNOT IAS TO REFUSAL SPEED FOR EACH KNOT OF HEADWIND. SUBTRACT FOR A TAILWIND. RUNWAY SLOPE DOES NOT AFFECT SPEED.
- FOR ANTISKID OFF, DECREASE SPEED SHOWN 10%.
- WITH MAIN LANDING GEAR TIRE PRESSURE OF 214 (+5-0) PSI, ADD 200 FEET TO THE COMPUTED DISTANCE.

EXAMPLE

- (A) IS GROSS WEIGHT (17,600 LB).
- (B) IS AIRPORT PRESSURE ALTITUDE (6300 FT).
- (C) IS AIR TEMPERATURE (20° C).
- (D) IS RUNWAY LENGTH (8000 FT).
- (E) IS RUNWAY CONDITION (RCR 17).
- (F) IS REFUSAL SPEED (128 KNOTS IAS, NO WIND).

N10/81
39E-1-93-46

Figure 23-13. Refusal Speeds — CT-39E

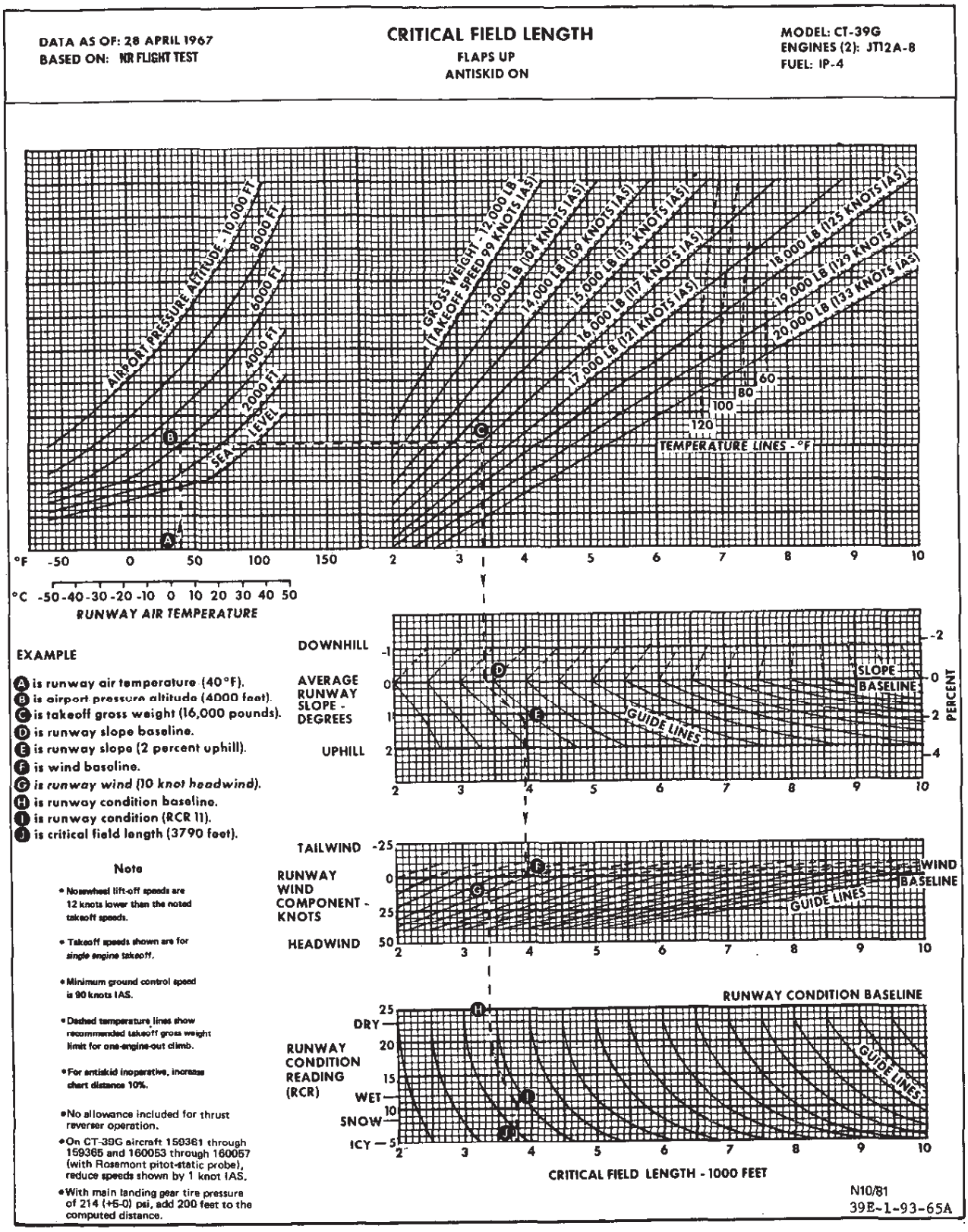


Figure 23-14. Critical Field Length — CT-39G

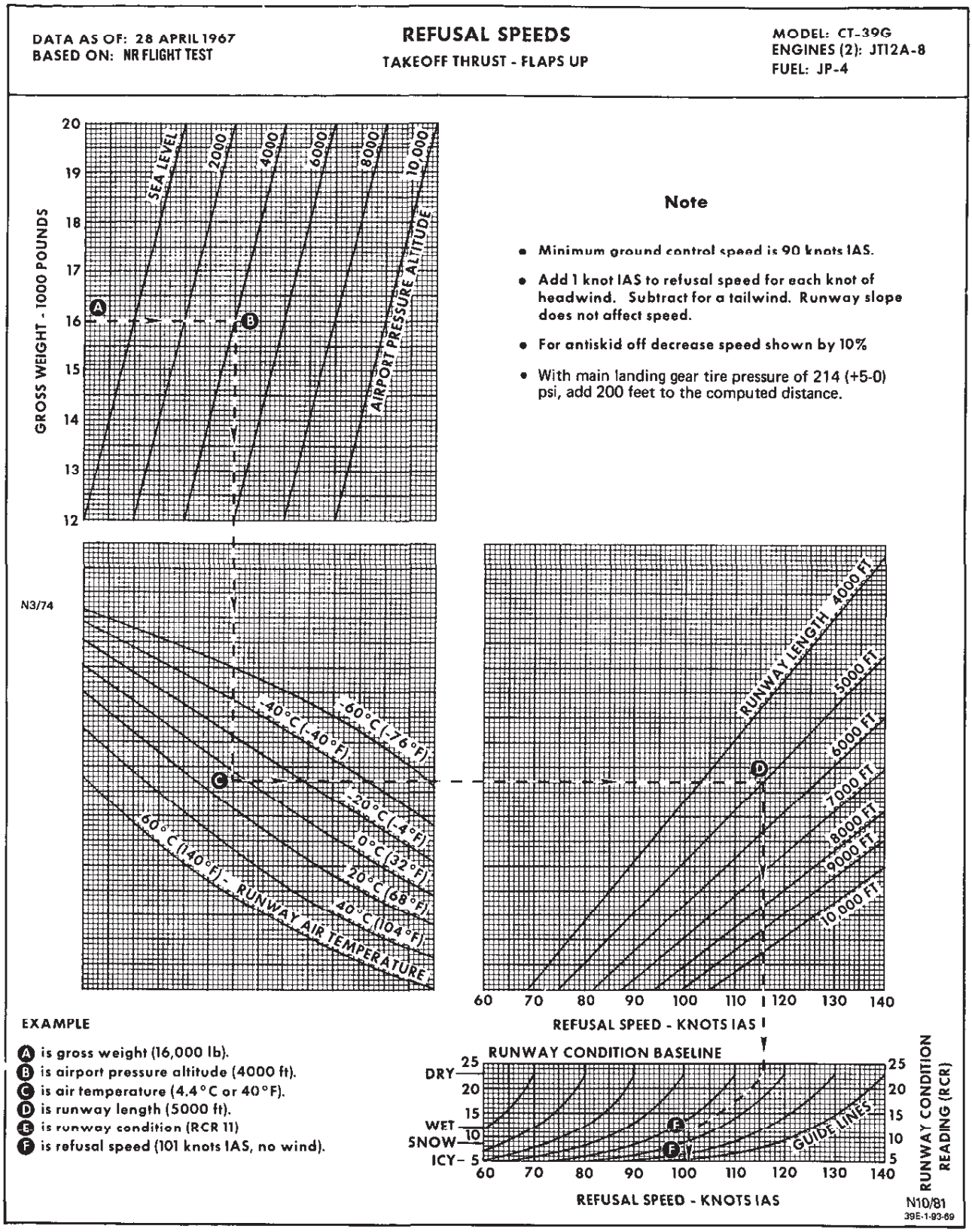


Figure 23-15. Refusal Speeds — CT-39G

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CHAPTER 24

Climb

24.1 DESCRIPTION

Climb charts for maximum continuous thrust operations, based on constant-speed climb, are shown in Figures 24-1 through 24-4. Time and distance are plotted against gross weight with guidelines to show the reduction in gross weight during climb because of the fuel used. Cruise (service) ceiling (100 fpm) is superimposed on the graph.

To approximate gross weight at start of climb, consider that 380 pounds of fuel will be used for start, taxi, takeoff, and acceleration to climb speed.

To obtain the climb data desired, enter the applicable 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 guidelines 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 the 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, shall still be determined by the difference in gross weights. The effect of temperature on time, fuel, and distance to climb is accounted for by using a corrected gross weight at start of climb (increased at temperatures above standard, not changed at temperatures below standard). To make the temperature correction more realistic, a mean deviation from standard temperature should be used rather than the sea-level deviation. This can be obtained by noting the nonstandard day temperatures throughout the climb, obtaining the difference from the corresponding standard day temperatures, and using the average of these obtained differences as the temperature to be used when applying the correction.

Note

Adjustments to gross weight for temperatures other than standard are minor and need only be applied for long-range or precise mission planning.

24.2 MAXIMUM CONTINUOUS THRUST ENGINE PRESSURE RATIO SETTING

The maximum continuous thrust EPR setting charts show the EPR required for maximum continuous thrust operation for any ram-air temperature and pressure altitude. (See Figures 24-5 through 24-8.) Separate charts are presented for operation with cabin pressurization on or cabin pressurization off. Correction factors are given on the charts for operation with engine inlet anti-icing on. Maximum continuous thrust EPR is for use in climb and acceleration and should not be used for normal operations.

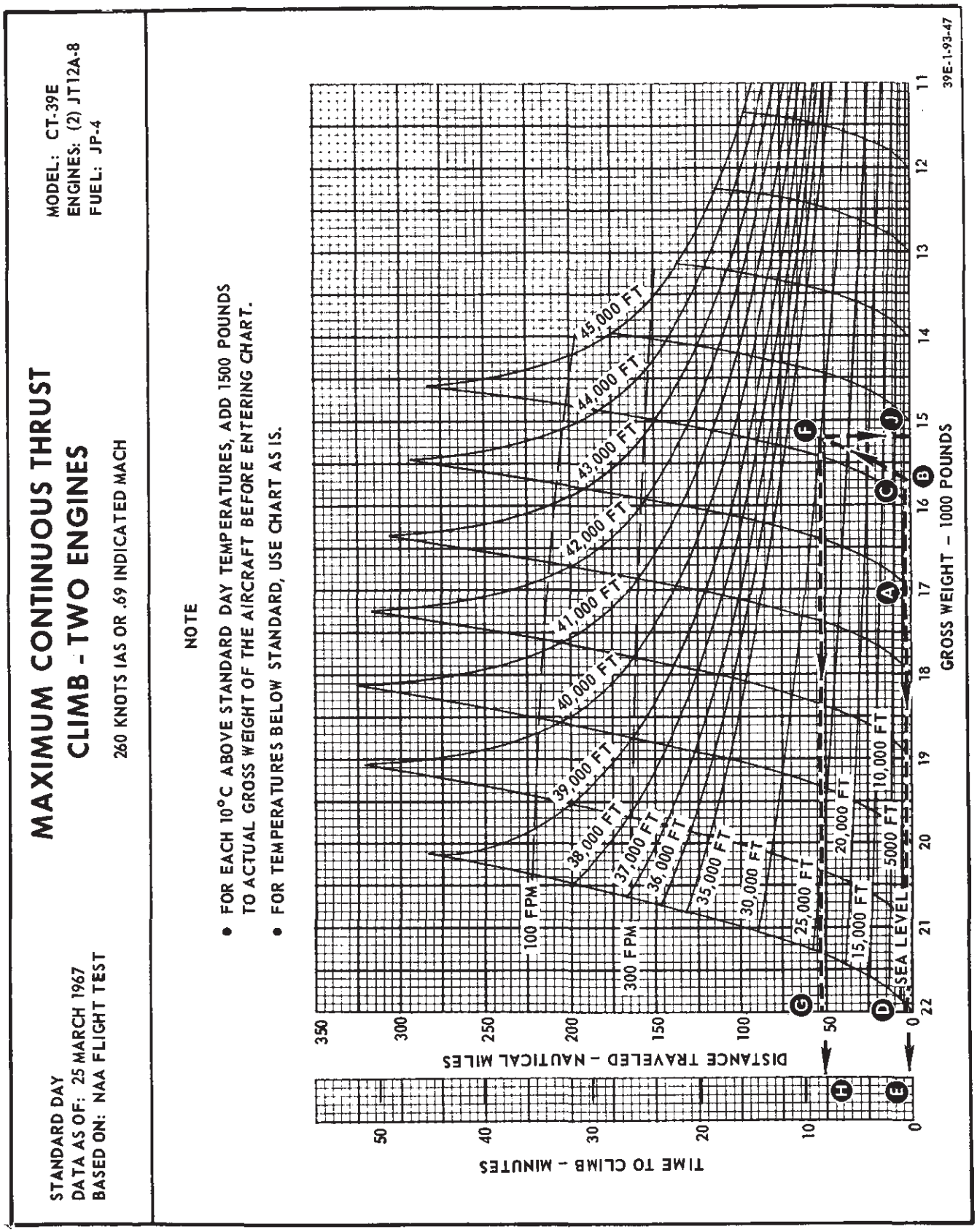

Figure 24-1. Maximum Continuous Thrust Climb (Two Engines) — CT-39E

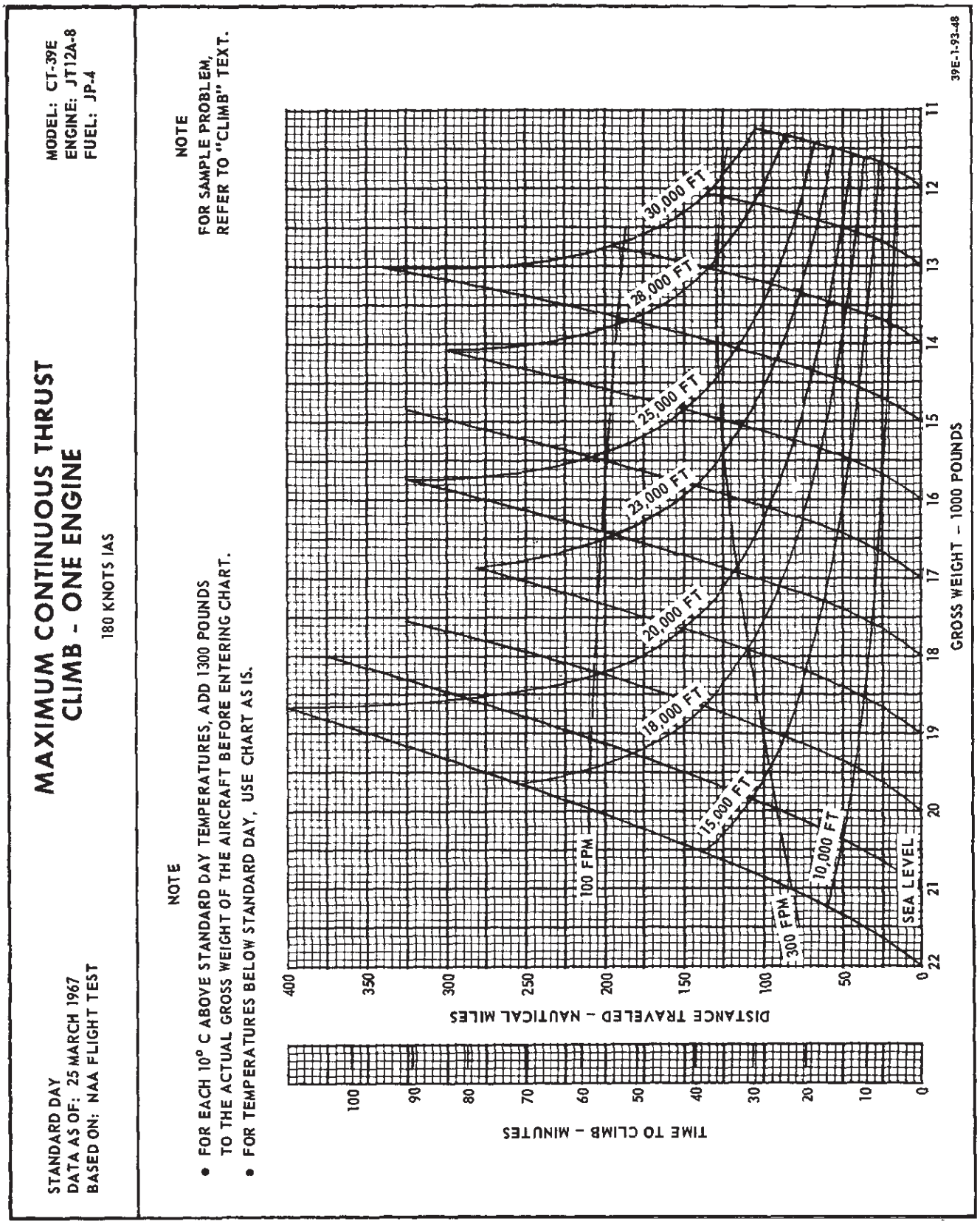

Figure 24-2. Maximum Continuous Thrust Climb (One Engine) — CT-39E

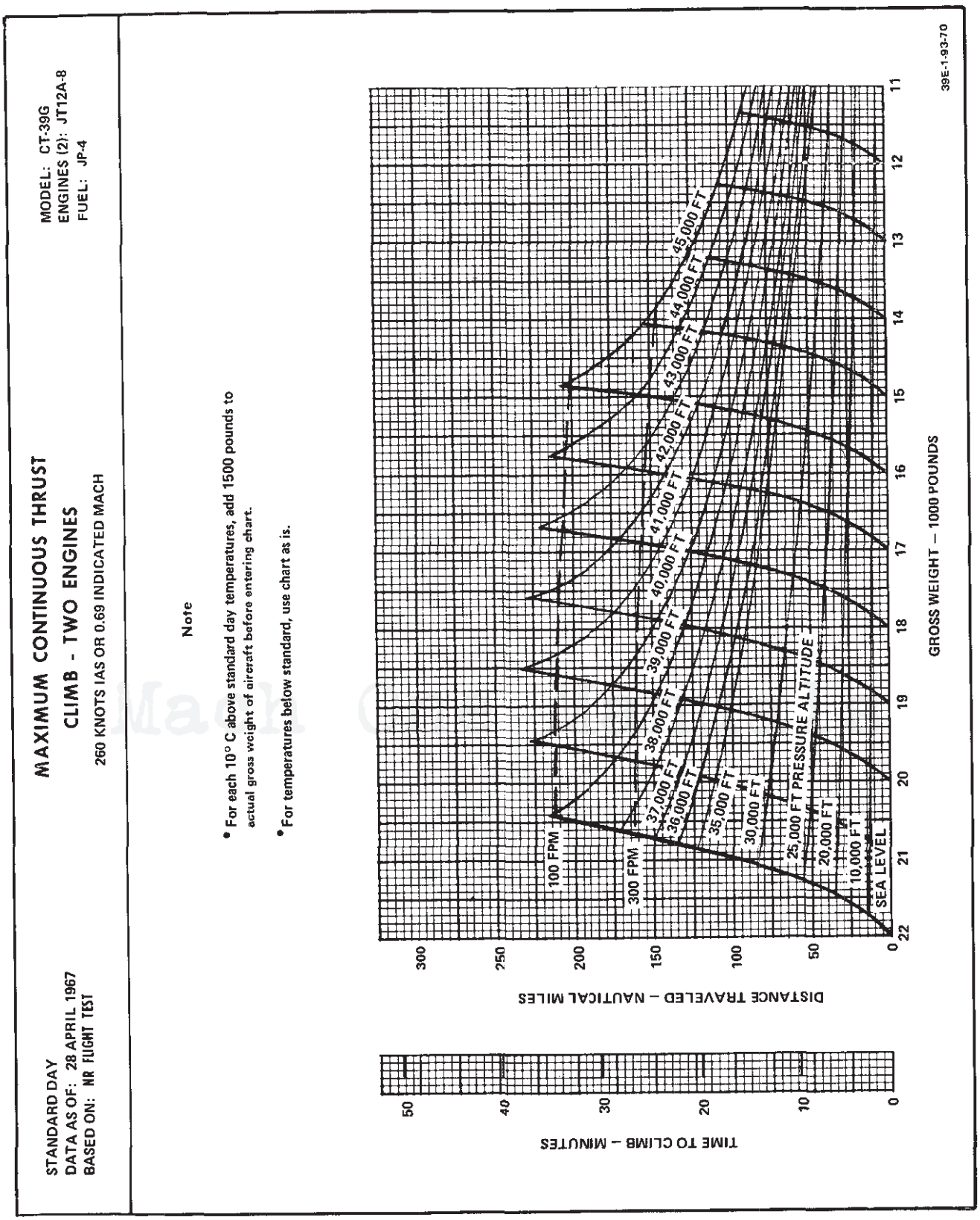

Figure 24-3. Maximum Continuous Thrust Climb (Two Engines) — CT-39G

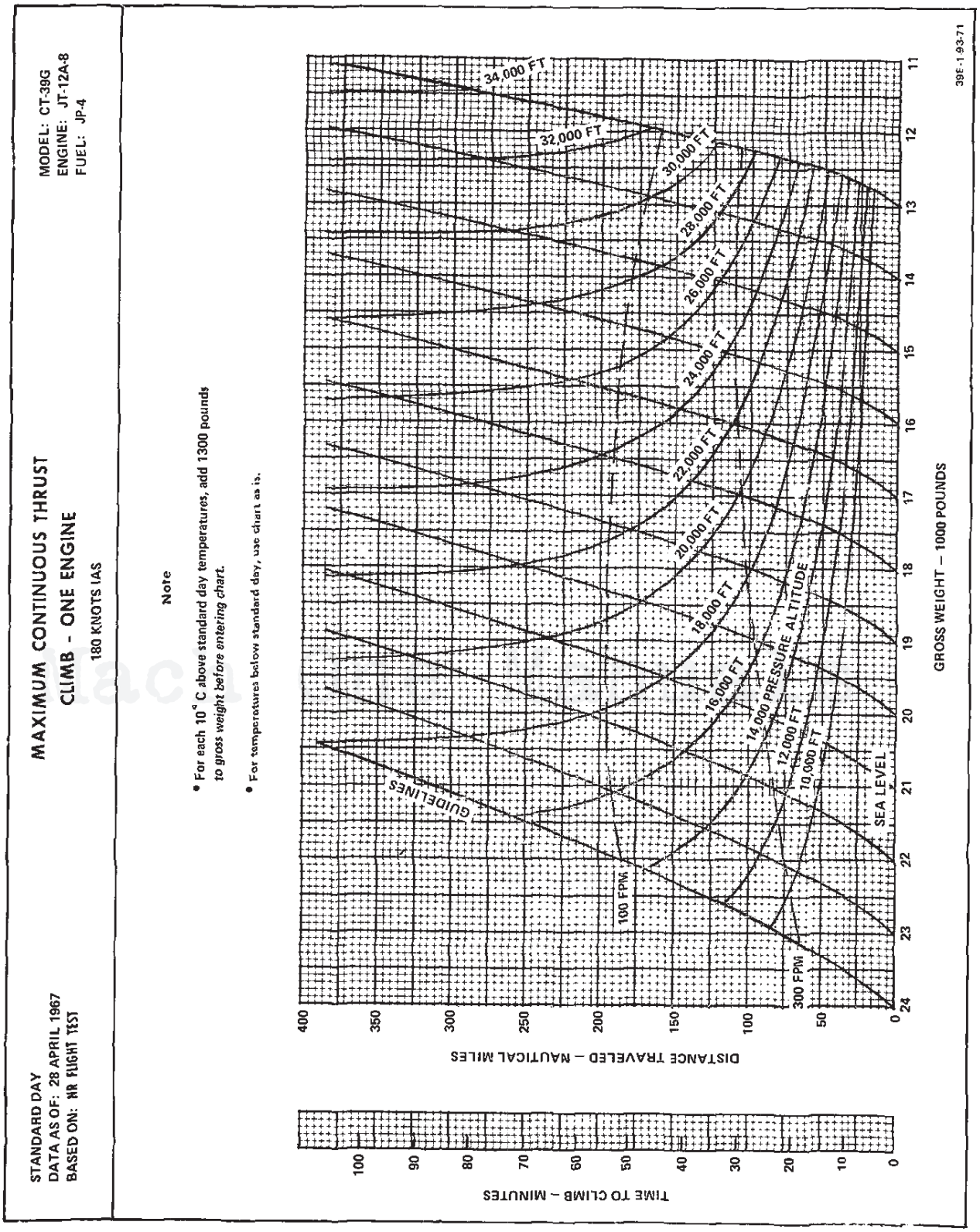

Figure 24-4. *Maximum Continuous Thrust Climb (One Engine) — CT-39G*

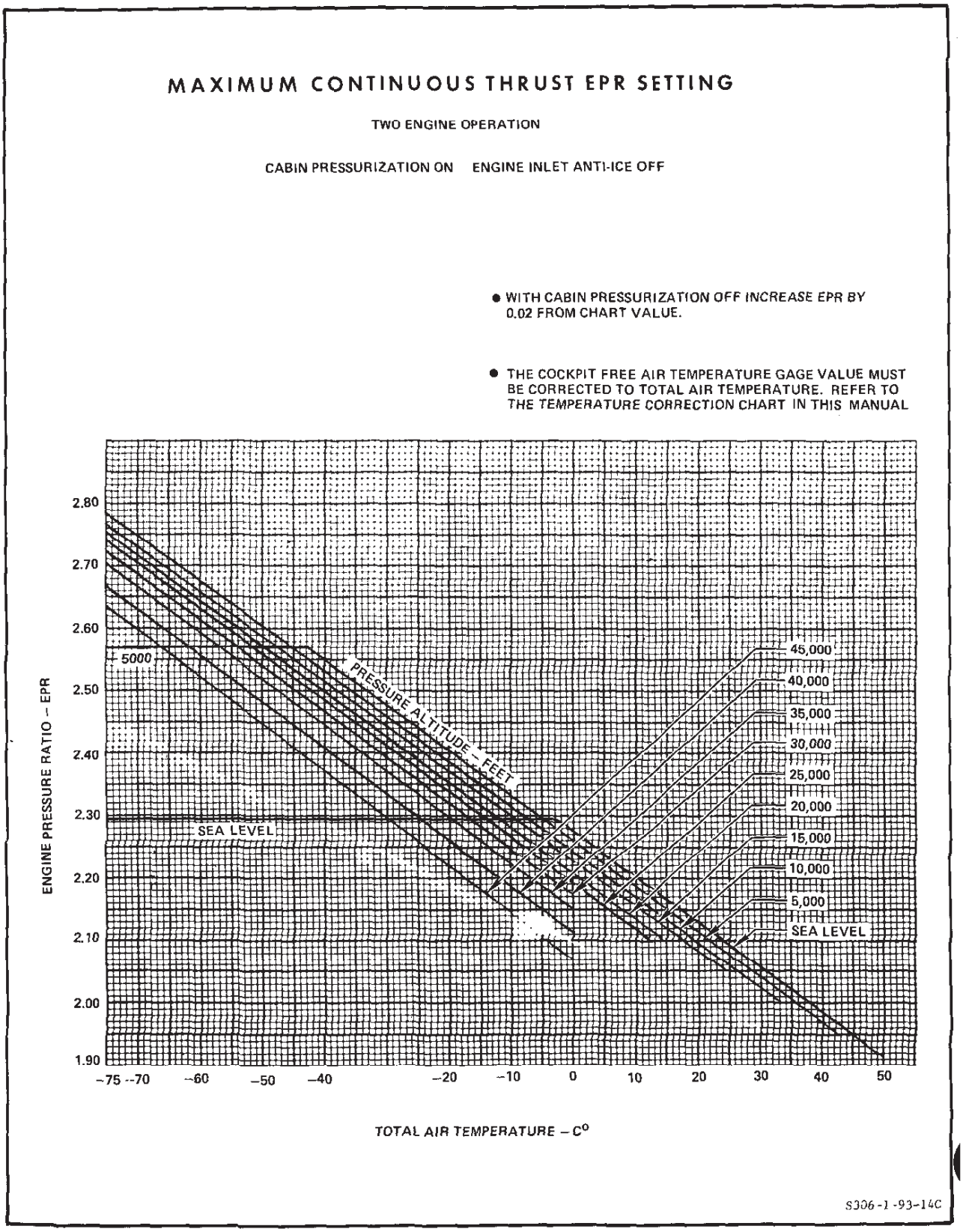

Figure 24-5. Maximum Continuous Thrust EPR Setting — Two Engine (Cabin Pressurization ON — Anti-Ice OFF)

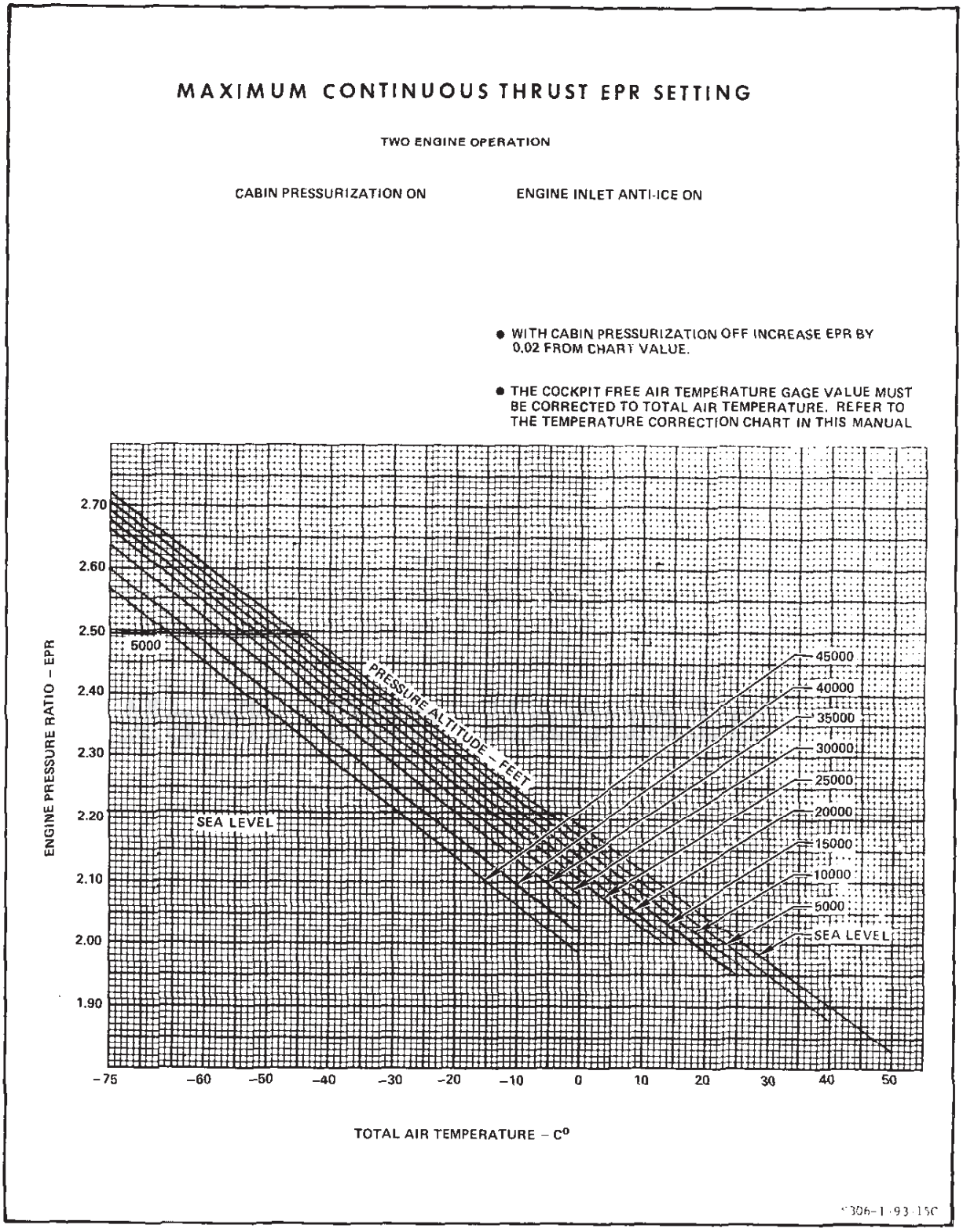

Figure 24-6. Maximum Continuous Thrust EPR Setting — Two Engine (Cabin Pressurization ON — Anti-Ice ON)

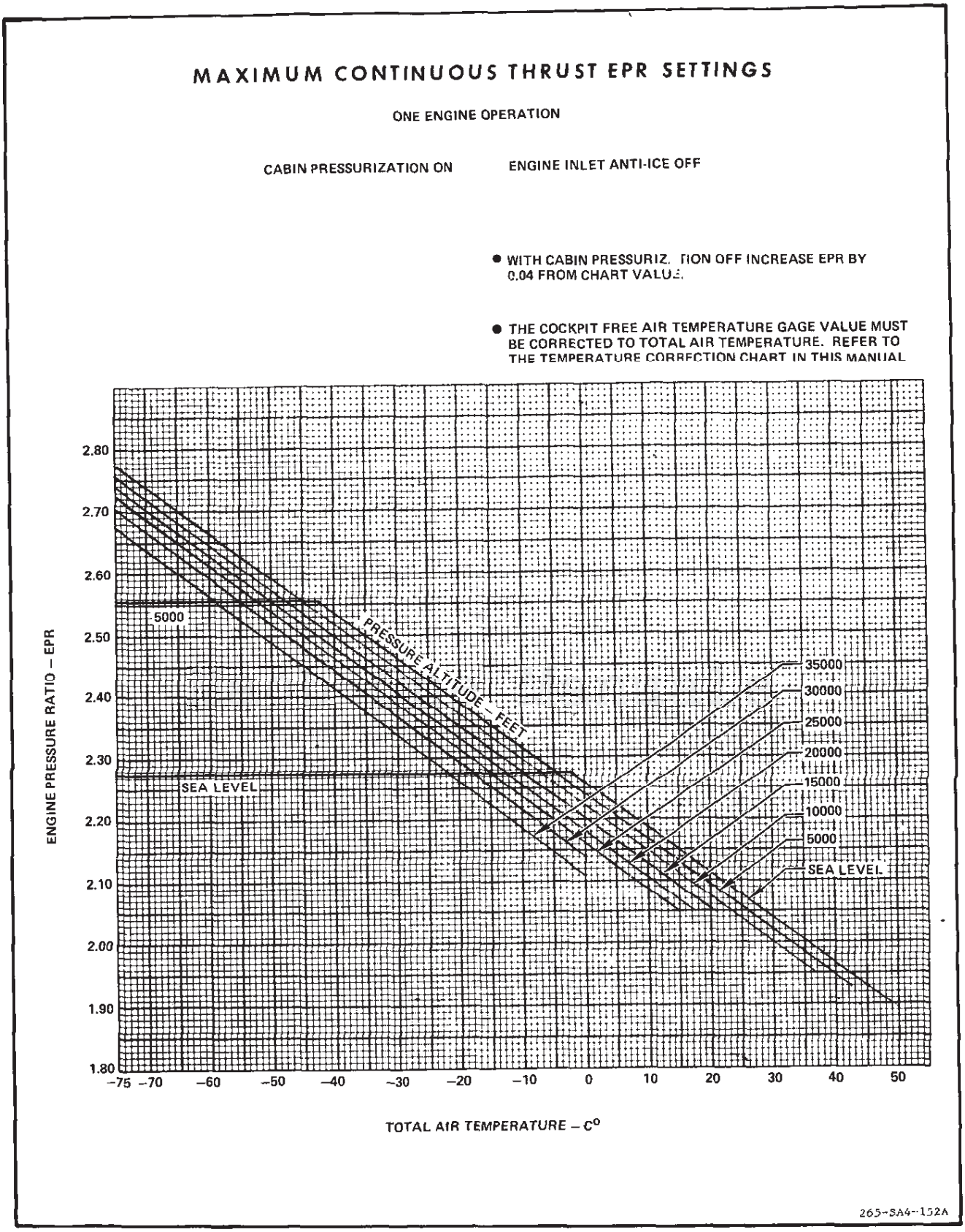

Figure 24-7. Maximum Continuous Thrust EPR Setting — One Engine (Cabin Pressurization ON — Anti-Ice OFF)

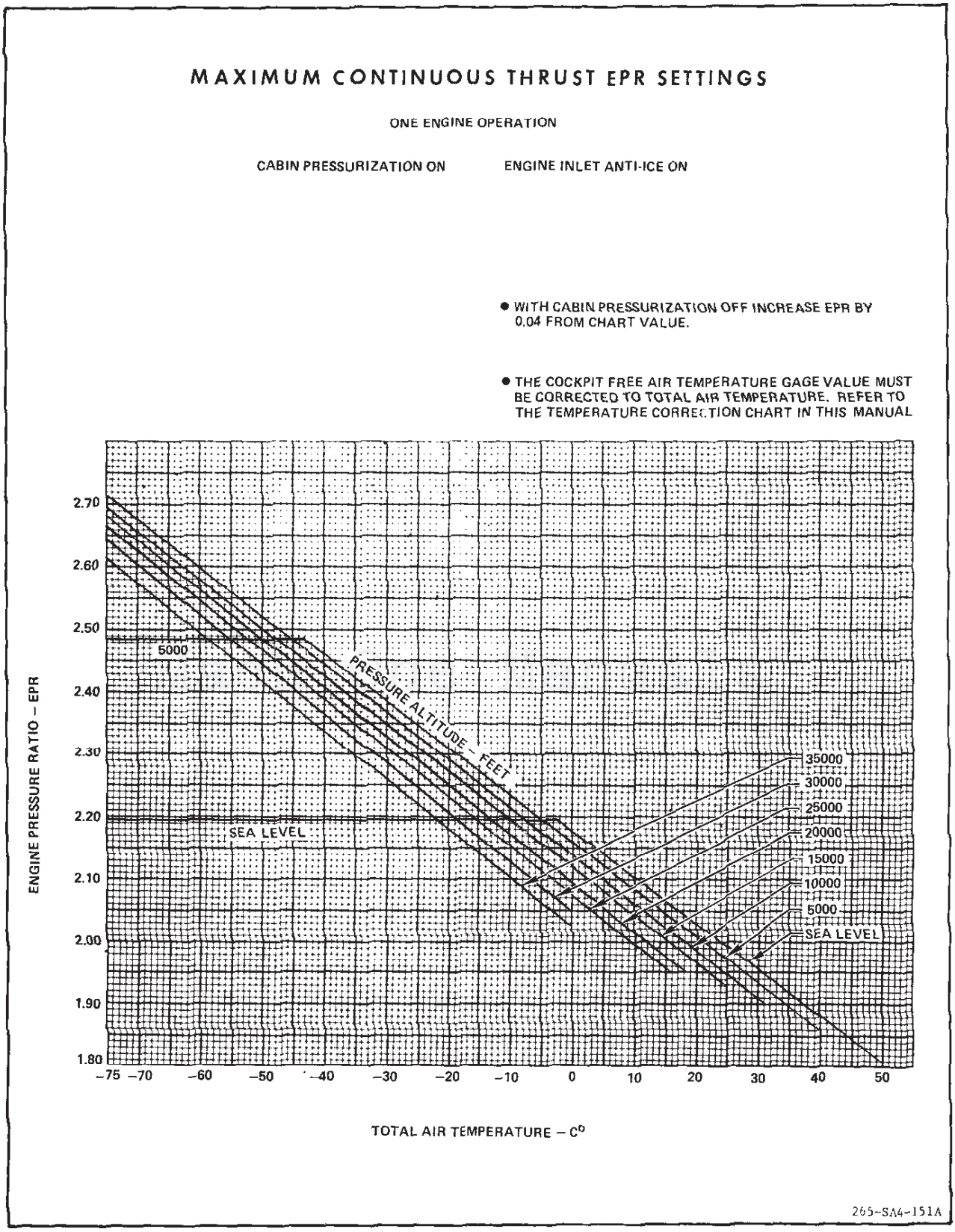

Figure 24-8. Maximum Continuous Thrust EPR Setting — One Engine (Cabin Pressurization ON — Anti-Ice ON)

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CHAPTER 25

Range

25.1 RANGE

As engine performance is affected by Mach number, revolutions per minute, and altitude, the drag curve indicates flexibility in both the optimum cruise Mach number and altitude. Within this flexibility, there is a definite range or span of Mach and altitude that will produce near maximum range. Variations of ambient temperature can have an important effect on the long-range cruise performance. The maximum thrust at which an aircraft can cruise is maximum continuous thrust. For some conditions, the thrust loss because of an increase in ambient temperature has only the small effect of requiring a higher cruise engine setting (EPR) and perhaps a slightly higher specific fuel consumption. However, increased ambient temperature also increases the true airspeed at a constant Mach number; consequently, specific range is not seriously affected. Remember that Mach number is the choice parameter for precise cruise control. If he cruise were to be performed at the optimum cruise altitude and the optimum altitude were thrust-limited on a standard day, then ambient temperatures above standard would reduce the range by reducing the altitude for cruise.

25.2 OPERATIONAL RANGE PROFILE

The operational range profile gives the time, fuel, distance, and altitude relationship to operational range for no-wind conditions. (See Figures 25-1 and 25-2). This relationship is based on a sequence of takeoff, maximum continuous thrust climb to any desired altitude up to 40,000 feet, and maximum-range at operational speeds. A cruise-climb condition from 40,000 to 45,000 feet is also given. A cruise table provides recommended Mach numbers and approximate operating conditions for both cruise at constant altitude and for cruise-climb procedure.

The fuel curves include a 380-pound allowance for start, taxi, takeoff, and acceleration to climb speed, plus the fuel used in climb to each altitude, as well as the fuel required for operational-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 and time consumed during the maximum continuous thrust climb from sea level to cruising altitude.

For cruise at a constant altitude, set up the recommended Mach number at the intersection of the climb path and cruise altitude. As the flight progresses, the thrust setting shall be decreased gradually as fuel is consumed to maintain the recommended Mach number. For cruise-climb, level off at 40,000 feet and set up the thrust setting and Mach number given for cruise-climb procedure. The aircraft will automatically climb as fuel is consumed to seek the cruise-climb altitude for its particular gross weight. Maintain the initial thrust setting and Mach number throughout the remainder of cruise-climb. The chart does not include fuel for descent and landing.

The chart 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 and fuel, compute groundspeed and air distance to be traveled (groundspeed equals TAS minus headwinds). Divide distance by groundspeed to obtain time to travel. Enter profile with time to travel and move to the cruise altitude to obtain the fuel required with wind.

25.2.1 Sample Problem (Figure 25-1). Find the necessary speed time and fuel required to cruise 550 nm at 30,000 feet with a headwind of 60 knots.

1. TAS as shown at 30,000 feet — (.66 Mach 390 Knots)
2. Groundspeed (390 minus 60) — 330 Knots
3. Time required to fly 550 nm with headwind (550 ÷ 330) — 1 Hour, 39 Minutes

4. Enter chart at time (with wind) and 30,000 feet to get fuel required — 3,200 Pounds.

25.3 MAXIMUM RANGE PROFILE

The maximum range profile (Figures 25-3 through 25-6) is similar in content and method of operation to the operational range profile with the exception of differing cruise speeds and thrust requirements. As the title implies, the maximum range profile primarily considers maximum range per pound of fuel. When using the maximum range profile chart, use the recommended thrust setting and Mach number, as indicated on the cruise chart, for the desired conditions. The speeds given are for an average weight configuration.

25.4 SINGLE-ENGINE RETURN PROFILE

The single-engine return profile (Figures 25-7 and 25-8) shows the relationship of fuel remaining to distance. The chart may be entered at the initial altitude with the fuel remaining to determine the range available or with the distance to be flown to determine the fuel required. No fuel reserve has been included. The legend on the chart shows average cruise airspeed and approximate fuel flow from sea level to 30,000 feet and for cruise climb.

25.5 NAUTICAL-MILES-PER-POUND-OF-FUEL CHARTS

Cruise data (zero wind) throughout the speed range from maximum endurance to takeoff thrust is shown in the nautical-miles-per-pound-of-fuel charts. The charts include specific range (nautical miles per pound), fuel flow, true Mach number, and true airspeed. Also included are curves of maximum range, operational range, and maximum continuous thrust. Because of the extended range of the aircraft, the range data is presented in several sets of charts. (See Figures 25-9 and 25-10.)

Aircraft cruising range is the product of specific range multiplied by pounds of fuel available for cruise:

$$\text{Range} = \frac{\text{NM}}{\text{Lb}} \times \text{Lb of Fuel}$$

When there is wind to be considered, multiply specific range by the range factor (groundspeed divided by true airspeed) to obtain the specific range for wind. For greatest accuracy in determining cruising range, consider small amounts of fuel at a time rather than the total fuel available. By this method, the total cruising range is the sum of the individual ranges obtained for each amount of fuel. The maximum range line is a baseline and, when range is being computed for other than maximum range, the guidelines are followed from this baseline to the desired point. In this manner, range may be determined for maximum endurance, operational range, maximum continuous thrust, or a desired true Mach number or true airspeed.

25.6 CRUISE CONTROL

The following two-engine tabular cruise data (CT-39G) (Figures 25-11 through 25-15) gives true Mach, true airspeed, indicated airspeed, fuel flow in pounds per hour for two engines, and specific range in nautical miles per pound of fuel for average gross weights ranging from 12,000 pounds through 20,000 pounds. It covers pressure altitudes from 15,000 feet through 45,000 feet with three ambient temperatures for each altitude shown. This arrangement of data is given for the following three cruise speeds:

1. Maximum range speed — 407 KTAS/Mach .71-Average Speed Above 39,000 Feet (variable Mach)

2. Recommended cruise speed — 430 KTAS/ Mach .75

3. Maximum speed cruise — 459 KTAS/Mach .80 or Maximum Cruise Power, Whichever Is Limiting

A sample use of cruise control data is shown below:

Given:

1. Cruise Mach number — .75.

2. Cruise altitude — 41,000 Feet.

3. Average cruise weight — 14,000 Pounds.

4. Ambient temperature — 46.5 °C (ISA + 10 °C).

Find:

True airspeed and fuel flow.

Solution:

1. Select table on Figure 25-12 (sheet 3 of 3).

2. Enter table at 41,000 feet.

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3. Proceed down to average gross weight (for this segment of flight) 14,000 pounds.

4. Read answers:

   a. True airspeed — 440 Knots.

   b. Indicated airspeed — 220 Knots.

   c. Fuel flow — 1,304 Pounds/Hour Total.

25.7 MAXIMUM RANGE SPEED VERSUS WIND

The maximum range speed versus wind chart (Figure 25-16) is used to establish a maximum range cruise when headwind or tailwind component is known. The chart enables obtaining a revised maximum range true airspeed for an altitude and gross weight that minimizes the range lost during cruise in headwind conditions or optimizes the range gained in tailwind conditions. This is possible because of the tendency of the nautical-miles-per-pound-of-fuel curves to flatten out at true airspeeds slightly above or below the maximum range speed. This resultant spread in true airspeeds has essentially the same specific range as maximum range true airspeed and only minor changes in fuel flow. It is, therefore, possible to reduce fuel flow and increase range with a tailwind or increase fuel flow and minimize range loss with a headwind.

The best maximum range cruise true airspeed is read directly on the scale below the intersection of the wind velocity grid and the altitude plot.

Entering the appropriate nautical-miles-per-pound-of-fuel chart with this revised true airspeed and the aircraft gross weight to obtain specific range enables prediction of total range for any segment.

25.8 RESERVE FUEL

Minimum fuel requirements for the CT-39E/G aircraft shall be computed using OPNAVINST 3710.7; however, in no case shall planned reserve fuel be less than 1,000 pounds.

25.9 MAXIMUM CRUISE THRUST ENGINE PRESSURE RATIO SETTING

Maximum cruise thrust EPR setting data is provided for the JT12A-8 engine for any ram-air temperature and pressure altitude, pressurization on or pressurization off. (See Figures 25-17 and 25-18.) This EPR setting data is the maximum approved for cruise flight and is intended to prolong engine life. Since maximum continuous thrust and maximum cruise thrust are the same on the JT12A-6A engines, maximum cruise thrust EPR charts are not provided for these engines. To find the maximum cruise thrust EPR setting, enter the chart with ram-air temperature. (The ram-air temperature correction for the indicated Mach number and indicated outside air temperature can be determined from Figure 22-7.) Proceed vertically upward until intersecting either the diagonal or the horizontal pressure altitude line, whichever is intersected first. Then move horizontally to the left and read maximum cruise thrust EPR setting.

25.10 ALTERNATE AIRPORT

The alternate airport data chart (CT-39G) (Figure 25-15) provides the fuel consumed in flying to an alternate airport at maximum range cruise speeds after aborting during the approach at the primary destination because of runway congestions or sudden degradation in weather. The data are presented for initial climb from an abort at sea level to representative cruise altitudes as well as for one condition of a climb from 20,000 feet to an altitude of 35,000 feet because of a change in plans or redirection from ground control.

The fuel consumption shown includes the total fuel used from the destination airport to landing at the alternate airport. No provisions for reserve fuel have been included.

To determine total fuel required to fly to an alternate airport, enter chart with alternate airport distance. Proceed vertically up to desired cruise altitude. Then proceed horizontally to left and read total fuel required from pullup at sea level at primary airport to landing at alternate airport. If necessary, a trade-off between cruising altitudes can be made to minimize fuel consumption.

EXAMPLE (Figure 25-15):

1. Alternate airport distance (200 nm desired)

2. Cruise altitude (10,000 feet)

3. Total fuel used from destination to landing (1,660 pounds).

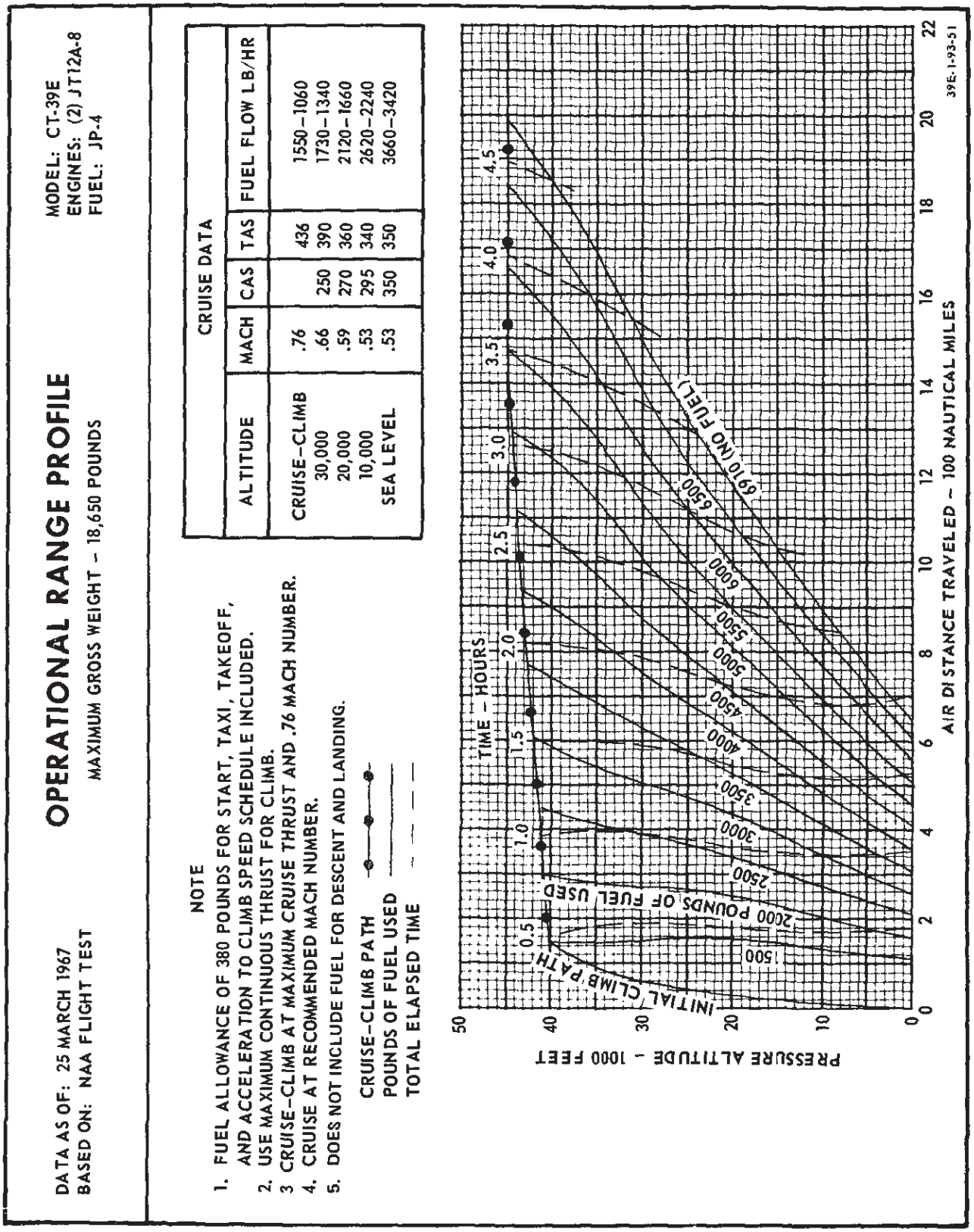

Figure 25-1. Operational Range Profile — CT-39E

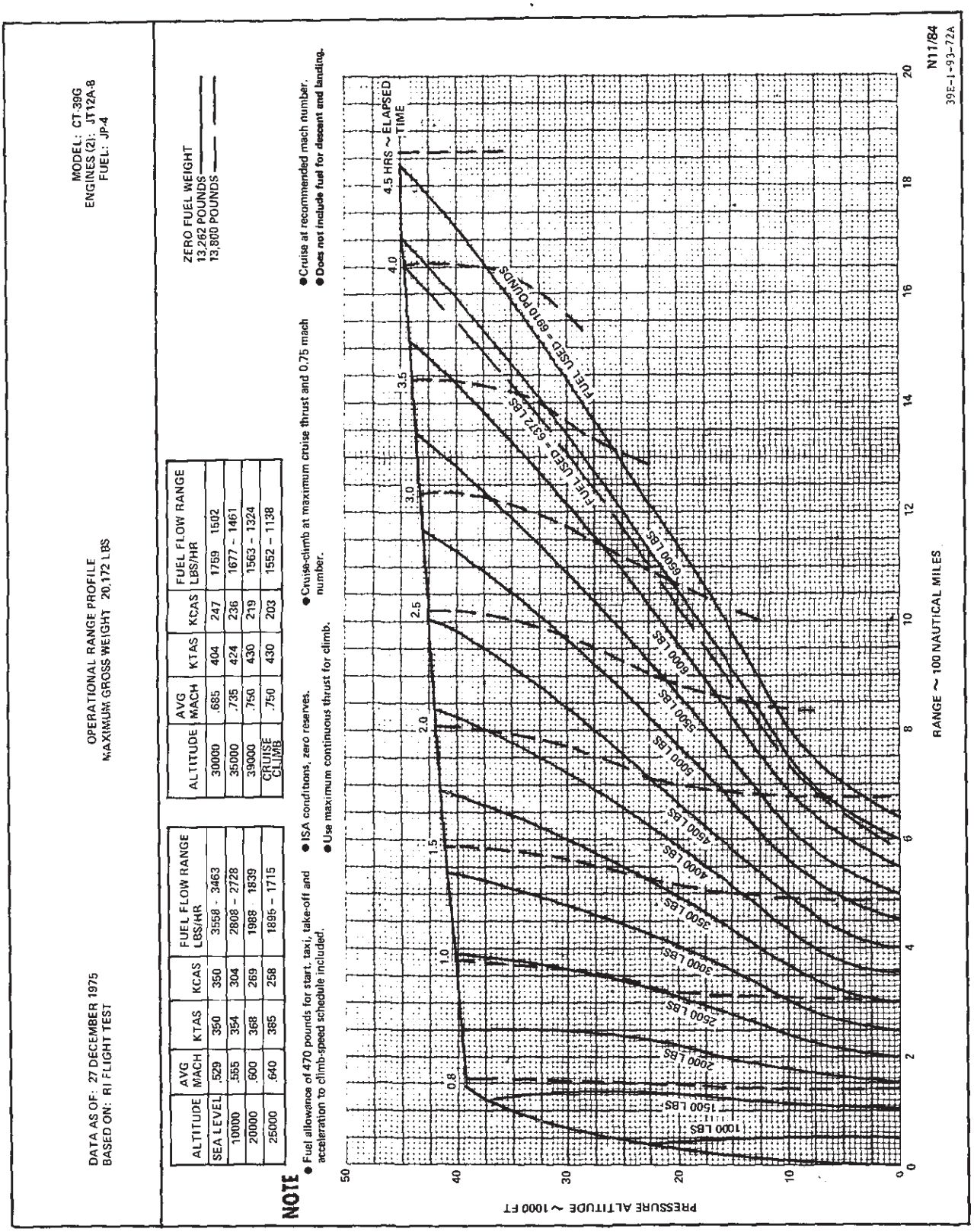

Figure 25-2. Recommended Cruise Range Profile — CT-39G

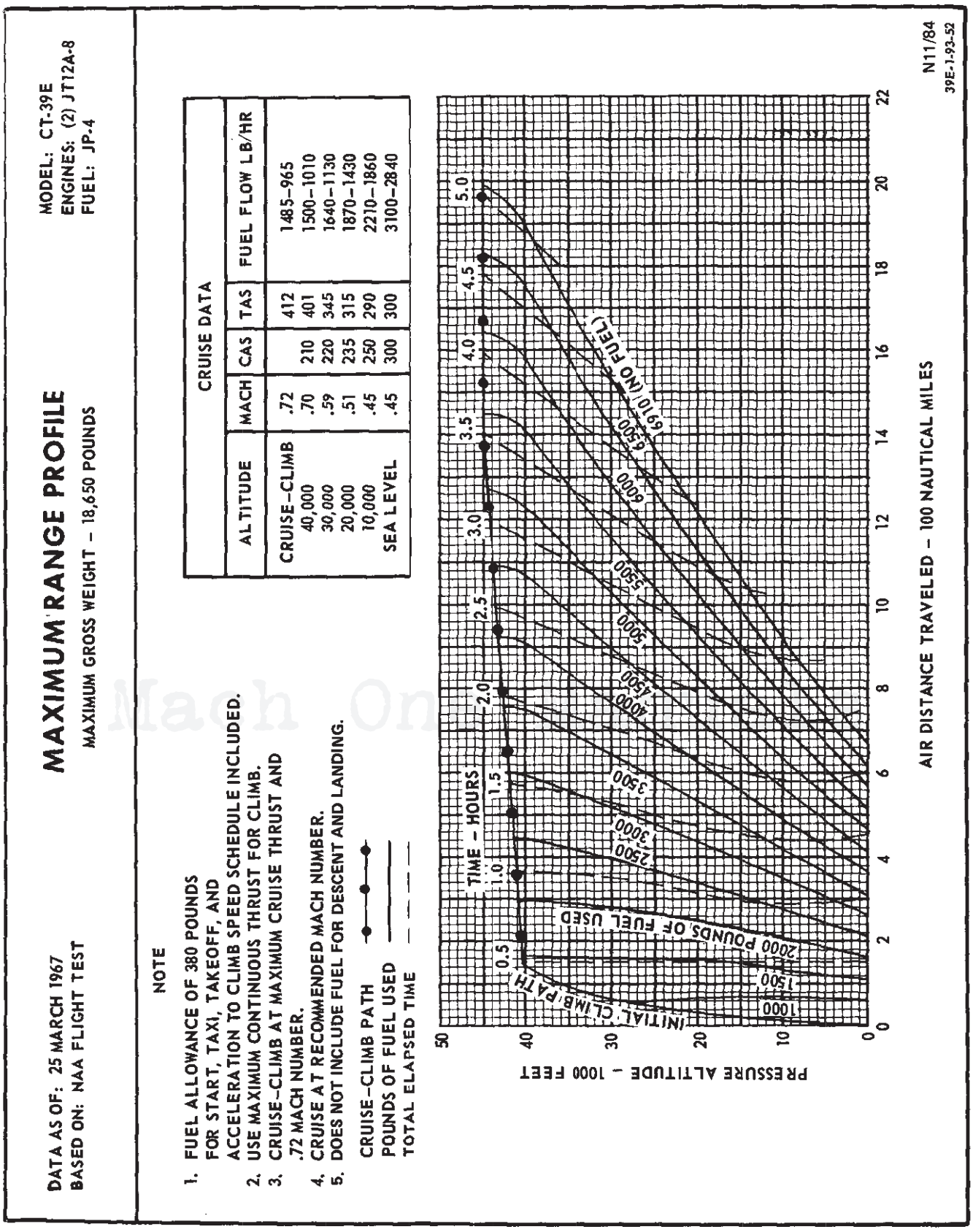

Figure 25-3. Maximum Range Profile — CT-39E

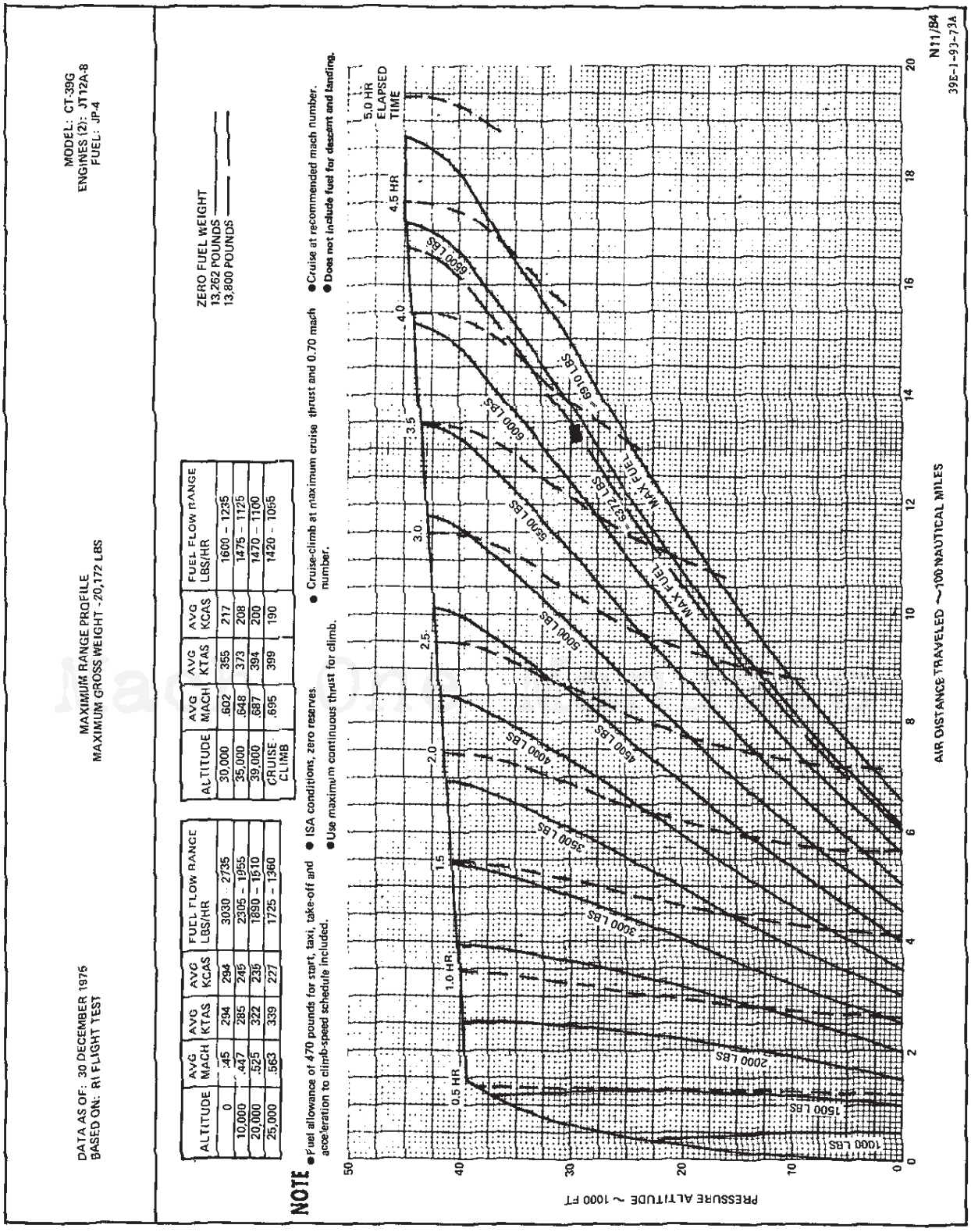

Figure 25-4. Maximum Range Profile — CT-39G

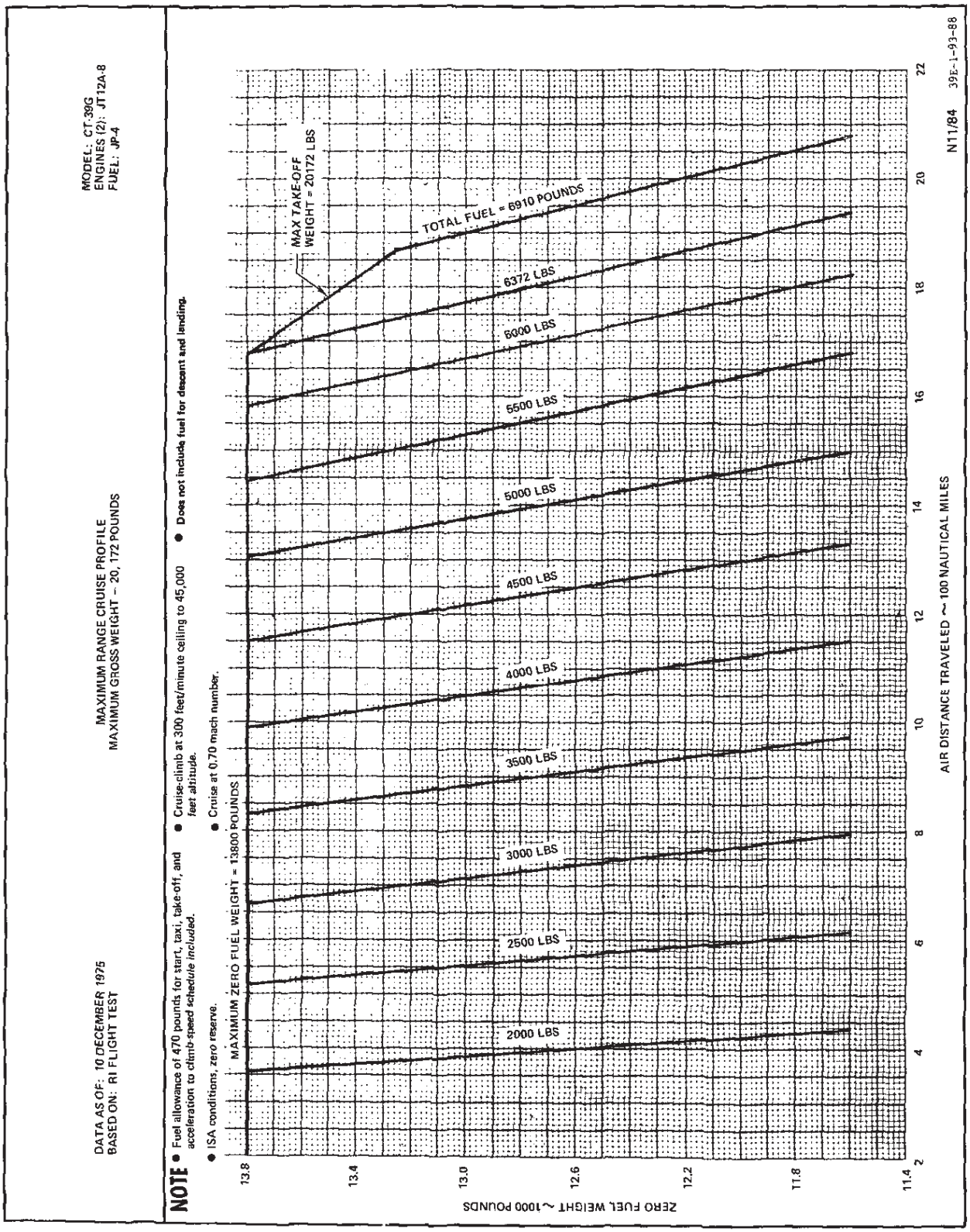

Figure 25-5. Maximum Range Cruise Profile — CT-39G

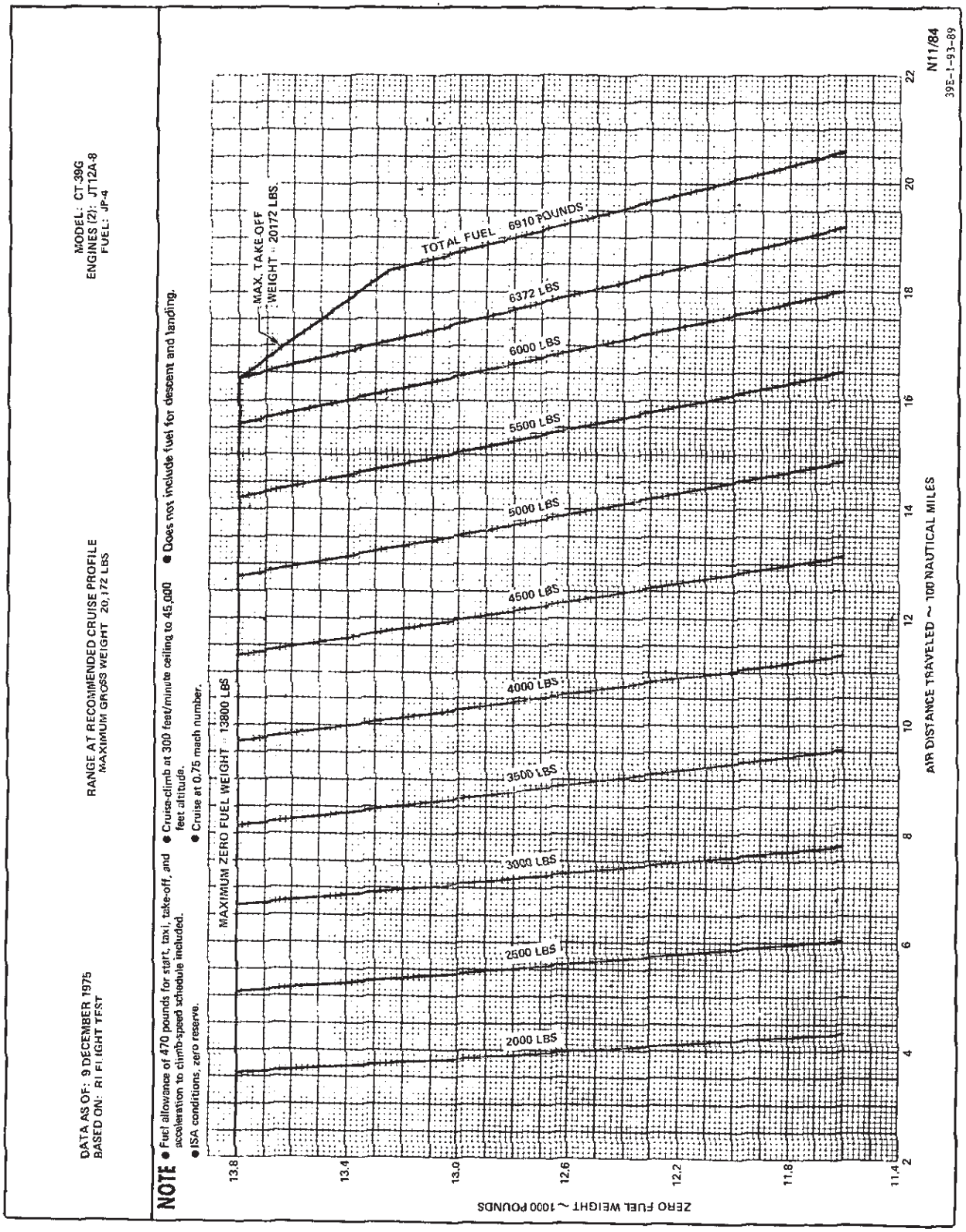

Figure 25-6. Range at Recommended Cruise Profile — CT-39G

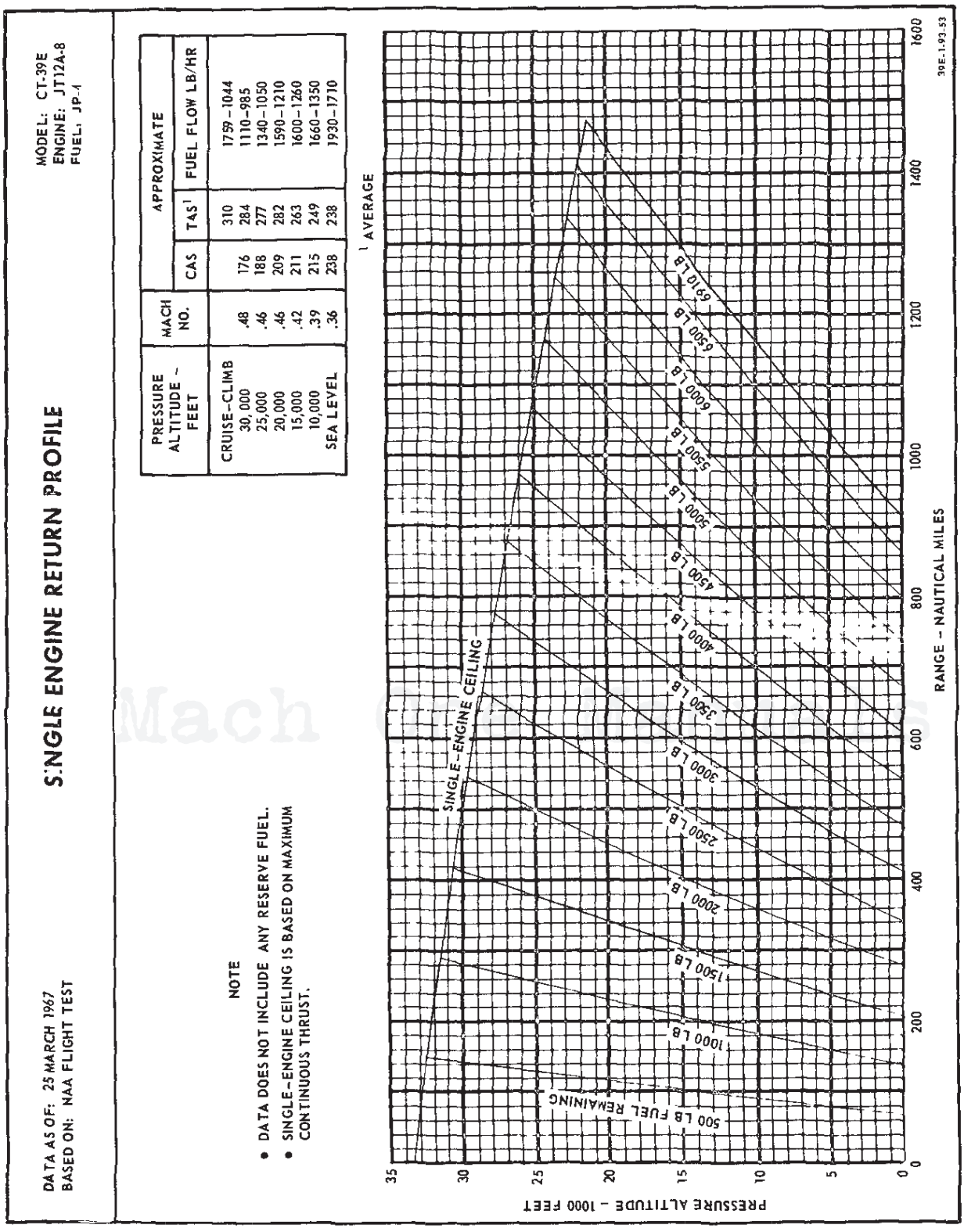

Figure 25-7. *Single-Engine Return Profile* — CT-39E

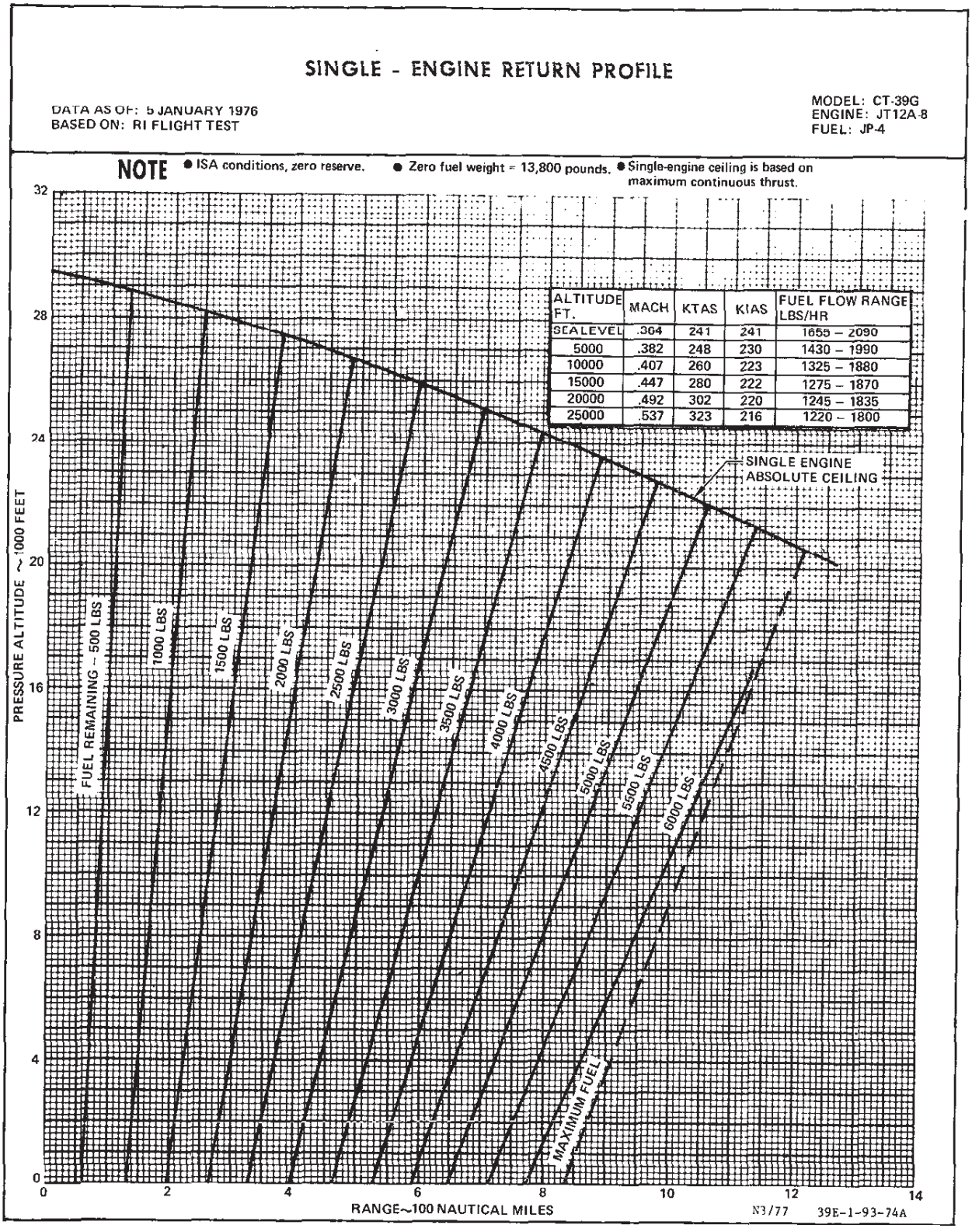

Figure 25-8 Single-Engine Return Profile — CT-39G

NAUTICAL MILES PER POUND OF FUEL

DATA AS OF: 25 MARCH 1967
BASED ON: NAA FLIGHT TEST

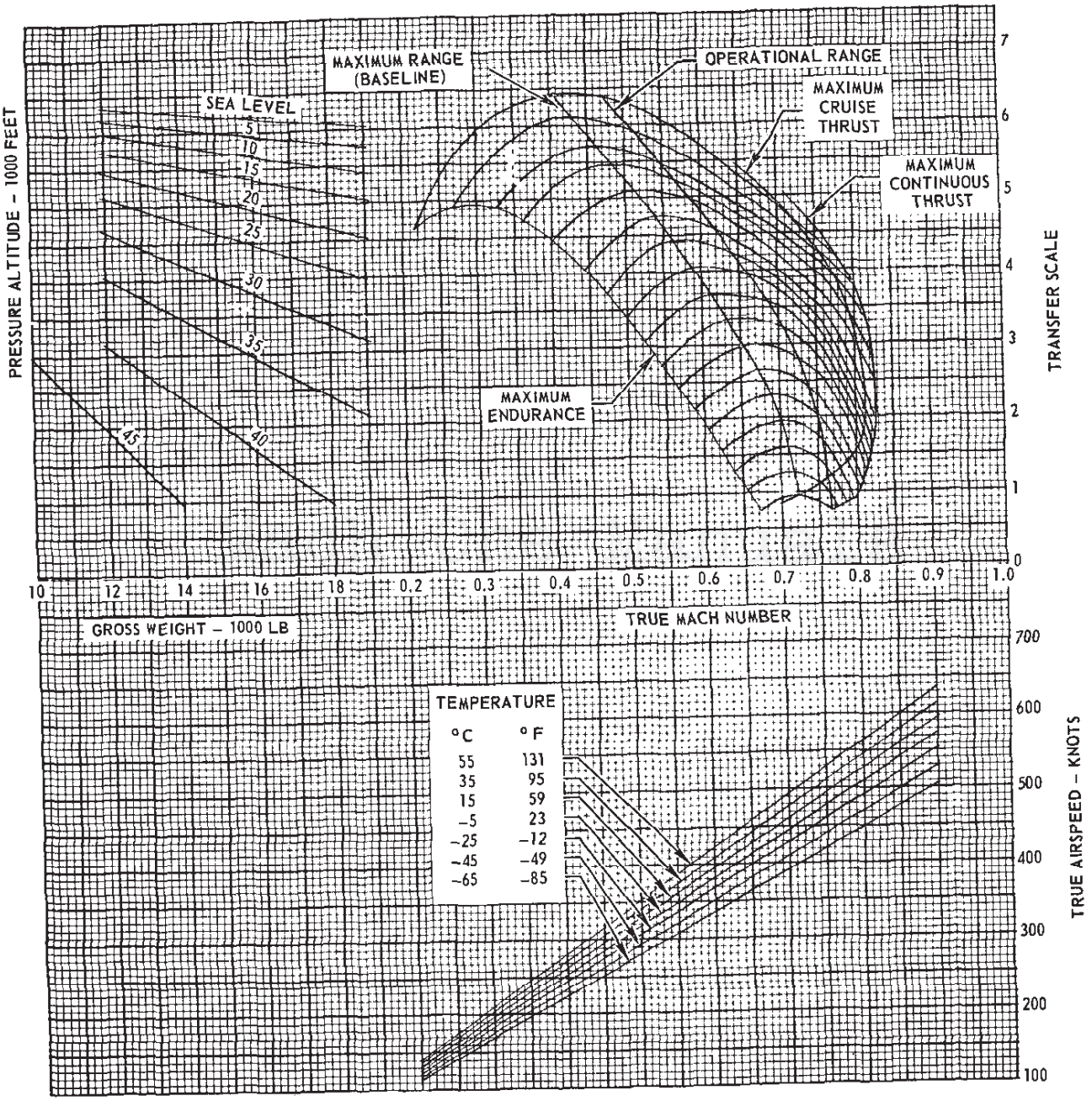

Figure 25-9. Nautical Miles Per Pound of Fuel (Sea Level to 30,000 Feet) — CT-39E (Sheet 1 of 2)

SEA LEVEL TO 30,000 FEET

MODEL: CT-39E
ENGINES: (2) JT12A-8
FUEL: JP-4

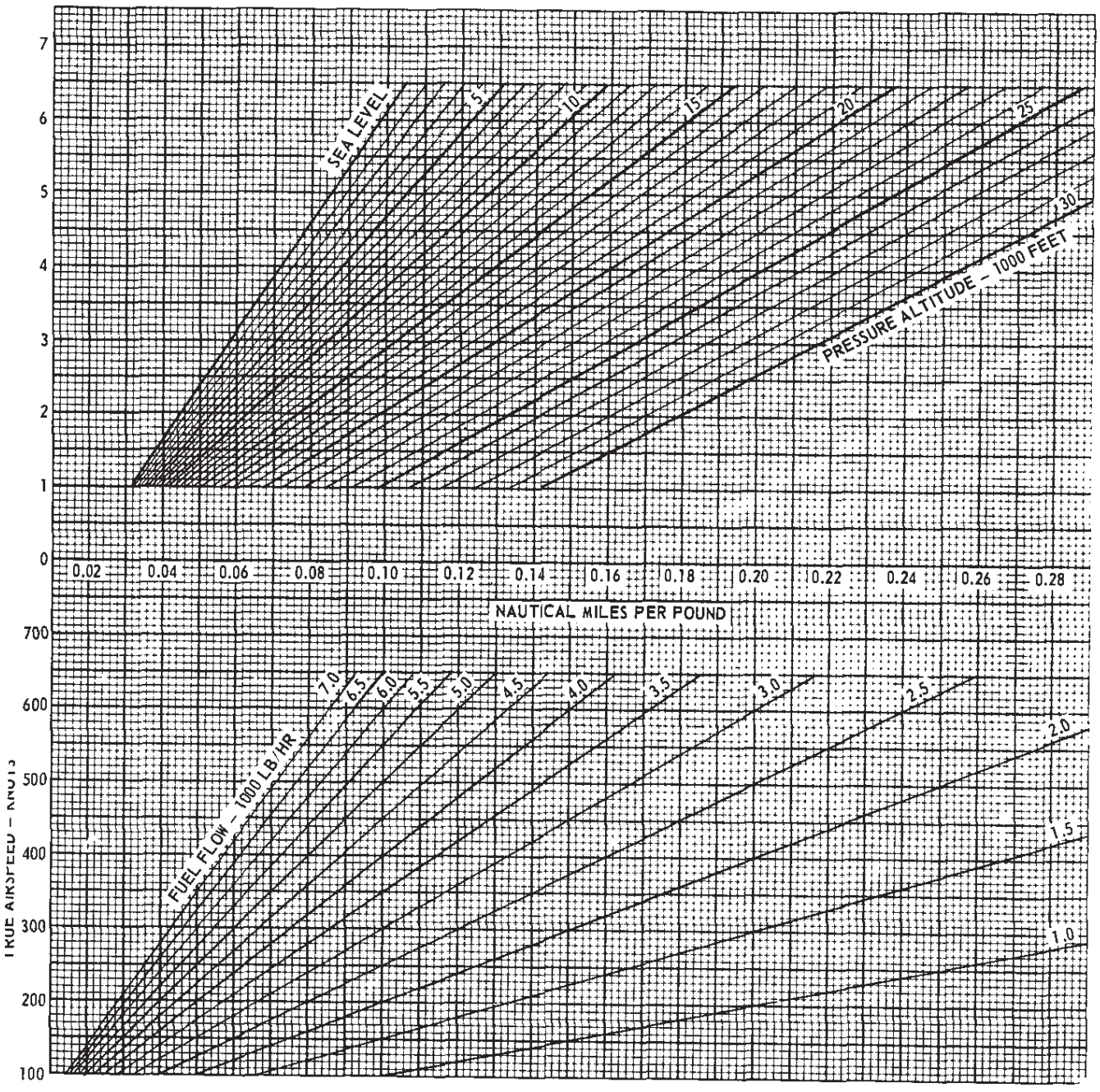

Figure 25-9. Nautical Miles Per Pound of Fuel (Sea Level to 30,000 Feet) — CT-39E (Sheet 2 of 2)

NAUTICAL MILES PER POUND OF FUEL

DATA AS OF: 25 MARCH 1967
BASED ON: NAA FLIGHT TEST

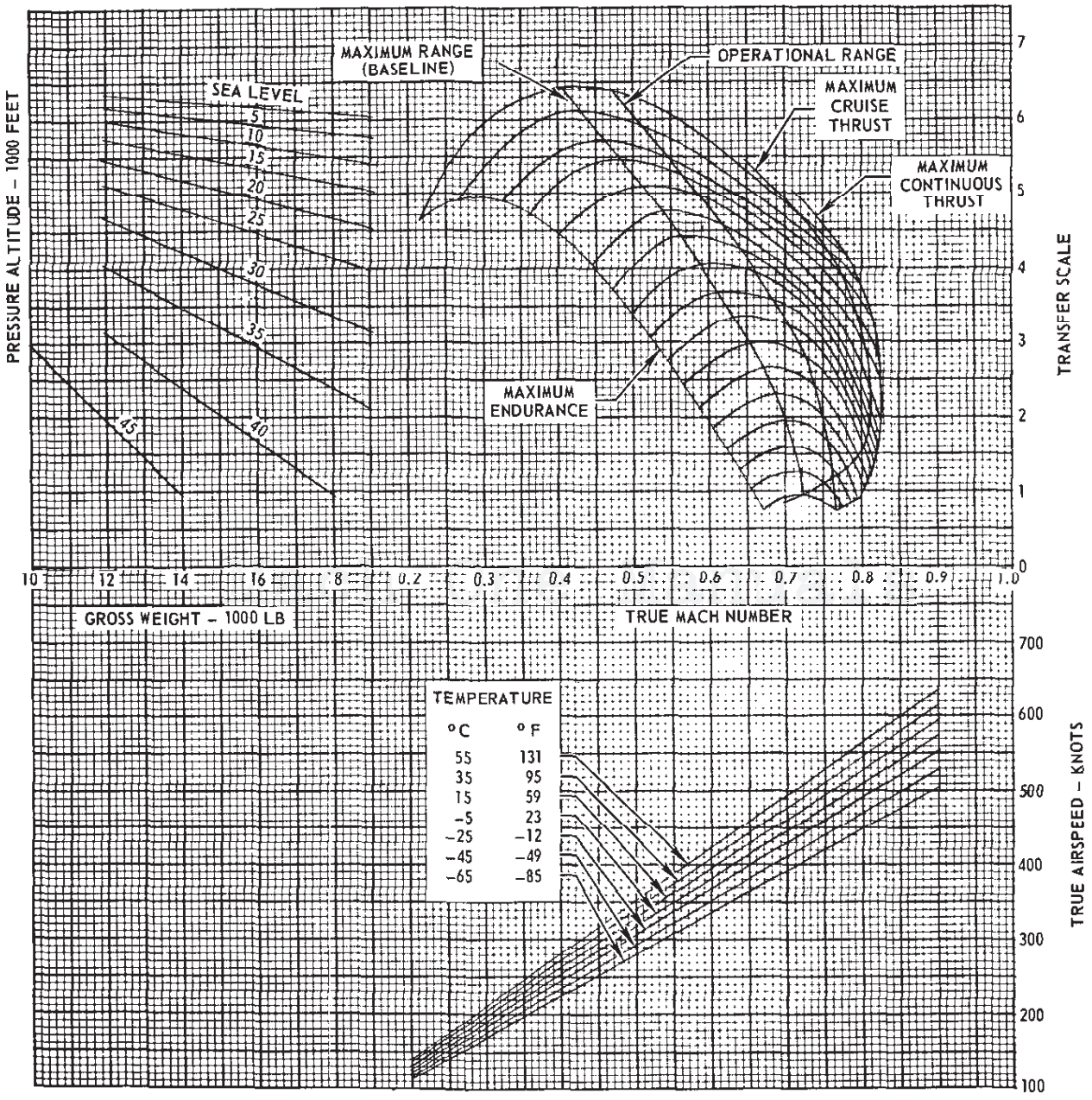

Figure 25-10. Nautical Miles Per Pound of Fuel (30,000 to 45,000 Feet) — CT-39E (Sheet 1 of 2)

30,000 FEET TO 45,000 FEET

MODEL: CT-39E
ENGINES: (2) JT12A-8
FUEL: JP-4

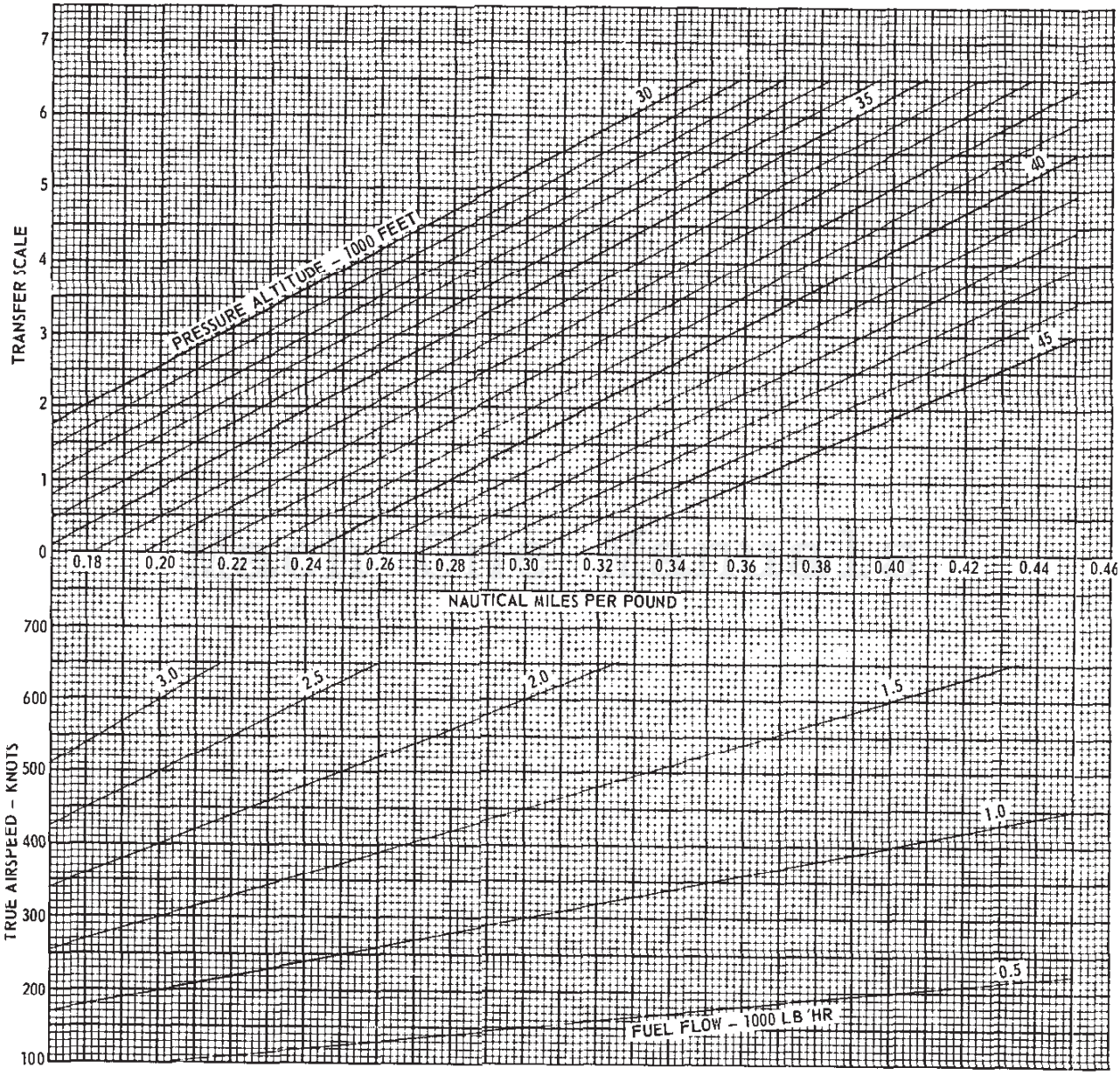

Figure 25-10. Nautical Miles Per Pound of Fuel (30,000 to 45,000 Feet) — CT-39E (Sheet 2 of 2)

MODEL: CT-39G
DATA AS OF: APRIL 1975

MAXIMUM RANGE SPEED
TWO ENGINE

GROSS WT. - LBS.	ALT.	15,000 Feet			20,000 Feet			25,000 Feet			29,000 Feet		
	STD. TEMP.	−14.7°C (5.5°F)			−24.6°C (−12.3°F)			−34.5°C (−30.1°F)			−42.4°C (−44.4°F)		
	ACT. TEMP.	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C
12,000	TRUE M	.443	.443	.443	.466	.466	.466	.501	.501	.501	.539	.539	.539
	TAS	272.	277.	283.	280.	286.	292.	295.	302.	308.	312.	319.	326.
	IAS	222.	222.	222.	212.	212.	212.	206.	206.	206.	204.	204.	204.
	FF-LBS	1518.	1548.	1578.	1383.	1412.	1440.	1211.	1238.	1263.	1135.	1161.	1185.
	NM/LB	.179	.179	.179	.203	.203	.203	.244	.244	.244	.275	.275	.275
13,000	TRUE M	.453	.453	.453	.479	.479	.479	.515	.515	.515	.553	.553	.553
	TAS	278.	284.	289.	288.	294.	300.	303.	310.	316.	320.	327.	334.
	IAS	228.	228.	228.	218.	218.	218.	212.	212.	212.	209.	209.	209.
	FF-LBS	1581.	1612.	1643	1433.	1463.	1492.	1275.	1303.	1330.	1198.	1225.	1251.
	NM/LB	.176	.176	.176	.201	.201	.201	.238	.238	.238	.267	.267	.267
14,000	TRUE M	.463	.463	.463	.492	.492	.492	.530	.530	.530	.566	.566	.566
	TAS	284.	290.	296.	296.	302.	308.	312.	319.	326.	328.	335.	342.
	IAS	233.	233.	233.	224.	224.	224.	218.	218.	218.	214.	214.	214.
	FF-LBS	1645.	1678.	1710.	1486.	1517.	1547.	1345.	1374.	1402.	1262.	1290.	1318.
	NM/LB	.173	.173	.173	.199	.199	.199	.232	.232	.232	.260	.260	.260
15,000	TRUE M	.473	.473	.473	.506	.506	.506	.544	.544	.544	.580	.580	.580
	TAS	290.	296.	302.	304.	311.	317.	320.	327.	334.	336.	343.	351.
	IAS	238.	238.	238.	231.	231.	231.	224.	224.	224.	220.	220.	220.
	FF-LBS	1712.	1746.	1779.	1550.	1582.	1614.	1414.	1445.	1475.	1330.	1360.	1389.
	NM/LB	.170	.170	.170	.196	.196	.196	.227	.227	.227	.252	.252	.252
16,000	TRUE M	.484	.484	.484	.518	.518	.518	.557	.557	.557	.593	.593	.593
	TAS	297.	303.	309.	312.	318.	324.	328.	335.	342.	343.	351.	358.
	IAS	243.	243.	243.	236.	236.	236.	230.	230.	230.	225.	225.	225.
	FF-LBS	1783.	1819.	1854.	1617.	1650.	1683.	1484.	1516.	1548.	1398.	1430.	1460.
	NM/LB	.167	.167	.167	.193	.193	.193	.221	.221	.221	.245	.245	.245
17,000	TRUE M	.494	.494	.494	.530	.530	.530	.570	.570	.570	.606	.606	.606
	TAS	303.	309.	315.	319.	326.	332.	336.	343.	350.	351.	359.	366.
	IAS	249.	249.	249.	242.	242.	242.	235.	235.	235.	230.	230.	230.
	FF-LBS	1853.	1890.	1926.	1687.	1722.	1756.	1557.	1590.	1623.	1470.	1502.	1535.
	NM/LB	.164	.164	.164	.189	.189	.189	.216	.216	.216	.239	.239	.239
18,000	TRUE M	.504	.504	.504	.541	.541	.541	.582	.582	.582	.618	.618	.618
	TAS	309.	316.	322.	326.	332.	339.	343.	350.	358.	358.	366.	374.
	IAS	254.	254.	254.	247.	247.	247.	241.	241.	241.	235.	235.	235.
	FF-LBS	1923.	1961.	1998.	1757.	1793.	1829.	1628.	1663.	1698.	1542.	1576.	1610.
	NM/LB	.161	.161	.161	.185	.185	.185	.211	.211	.211	.232	.232	.232
19,000	TRUE M	.514	.514	.514	.551	.551	.551	.594	.594	.594	.629	.629	.629
	TAS	316.	322.	328.	332.	338.	345.	350.	357.	365.	364.	372.	380.
	IAS	259.	259.	259.	252.	252.	252.	246.	246.	246.	240.	240.	240.
	FF-LBS	1993.	2033.	2072.	1826.	1864.	1901.	1701.	1738.	1774.	1613.	1649.	1685.
	NM/LB	.158	.158	.158	.182	.182	.182	.206	.206	.206	.226	.226	.226
20,000	TRUE M	.524	.524	.524	.560	.560	.560	.603	.603	.603	.638	.638	.638
	TAS	322.	328.	334.	337.	344.	351.	355.	363.	370.	369.	378.	386.
	IAS	264.	264.	264.	256.	256.	256.	250.	250.	250.	243.	243.	243.
	FF-LBS	2065.	2107.	2147.	1894.	1934.	1972.	1768.	1806.	1843.	1682.	1719.	1756.
	NM/LB	.156	.156	.156	.178	.178	.178	.201	.201	.201	.220	.220	.220

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Figure 25-11. Cruise Control — Maximum Range Speed CT-39G (Sheet 1 of 3)

MAXIMUM RANGE SPEED
TWO ENGINE

MODEL: CT-39G
DATA AS OF: APRIL 1975

GROSS WT LBS	ALT.	31,000 Feet			33,000 Feet			35,000 Feet			37,000 Feet		
	STD. TEMP.	−46.4°C (−51.5°F)			−50.4°C (−58.7°F)			−54.3°C (−65.8°F)			−56.5°C (−69.7°F)		
	ACT. TEMP.	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C
12,000	TRUE M	.558	.558	.558	.576	.576	.576	.597	.597	.597	.617	.617	.617
	TAS	320.	327.	334.	327.	335.	342.	336.	344.	352.	346.	354.	362.
	IAS	202.	202.	202.	200.	200.	200.	198.	198.	198.	196.	196.	196.
	FF-LBS	1099.	1124.	1148.	1062.	1086.	1110.	1035.	1059.	1083.	1010.	1034.	1058.
	NM/LB	.291	.291	.291	.308	.308	.308	.325	.325	.325	.342	.342	.342
13,000	TRUE M	.572	.572	.572	.590	.590	.590	.610	.610	.610	.630	.630	.630
	TAS	328.	336.	343.	335.	343.	351.	343.	352.	359.	353.	361.	370.
	IAS	207.	207.	207.	205.	205.	205.	203.	203.	203.	200.	200.	200.
	FF-LBS	1161.	1188.	1214.	1125.	1151.	1177.	1098.	1124.	1150.	1074.	1100.	1125.
	NM/LB	.283	.283	.283	.298	.298	.298	.3132	.313	.313	.329	.329	.329
14,000	TRUE M	.585	.585	.585	.603	.603	.603	.624	.624	.624	.643	.643	.643
	TAS	336.	343.	351.	343.	351.	358.	351.	360.	368.	360.	369.	377.
	IAS	212.	212.	212.	210.	210.	210.	208.	208.	208.	205.	205.	205.
	FF-LBS	1225.	1252.	1280.	1190.	1217.	1244.	1166.	1194.	1221.	1140.	1168.	1194.
	NM/LB	.274	.274	.274	.288	.288	.288	.301	.301	.301	.316	.316	.316
15,000	TRUE M	.598	.598	.598	.617	.617	.617	.637	.637	.637	.656	.656	.656
	TAS	343.	351.	358.	351.	359.	367.	359.	367.	375.	367.	376.	385.
	IAS	217.	217.	217.	215.	215.	215.	212.	212.	212.	209.	209.	209.
	FF-LBS	1291.	1320.	1349.	1260.	1289.	1318.	1234.	1264.	1292.	1209.	1238.	1266.
	NM/LB	.266	.266	.266	.278	.278	.278	.291	.291	.291	.304	.304	.304
16,000	TRUE M	.611	.611	.611	.630	.630	.630	.650	.650	.650	.669	.669	.669
	TAS	350.	358.	366.	358.	366.	374.	366.	375.	383.	375.	384.	392.
	IAS	222.	222.	222.	220.	220.	220.	217.	217.	217.	213.	213.	213.
	FF-LBS	1361.	1392.	1422.	1330.	1361.	1391.	1305.	1336.	1366.	1280.	1310.	1340.
	NM/LB	.258	.258	.258	.269	.269	.269	.280	.280	.280	.293	.293	.293
17,000	TRUE M	.624	.624	.624	.643	.643	.643	.662	.662	.662	.681	.681	.681
	TAS	358.	366.	374.	365.	374.	382.	373.	382.	390.	381.	391.	399.
	IAS	227.	227.	227.	224.	224.	224.	221.	221.	221.	218.	218.	218.
	FF-LBS	1433.	1466.	1498.	1403.	1435.	1467.	1375.	1407.	1439.	1351.	1383.	1415.
	NM/LB	.250	.250	.250	.261	.261	.261	.271	.271	.271	.282	.282	.282
18,000	TRUE M	.636	.636	.636	.655	.655	.655	.674	.6741	.674	.692	.692	.692
	TAS	365.	373.	381.	372.	381.	389.	379.	388.	397.	388.	397.	406.
	IAS	232.	232.	232.	229.	229.	229.	225.	225.	225.	221.	221.	221.
	FF-LBS	1505.	1540.	1573.	1475.	1509.	1542.	1447.	1481.	1515.	1424.	1458.	1491.
	NM/LB	.242	.242	.242	.252	.252	.252	.262	.262	.262	.272	.272	.272
19,000	TRUE M	.647	.647	.647	.666	.665	.666	.684	.684	.684	.703	.703	.703
	TAS	371.	380.	388.	378.	387.	396.	385.	394.	403.	394.	403.	412.
	IAS	236.	236.	236.	233.	233.	233.	229.	229.	229.	225.	225.	225.
	FF-LBS	1577.	1613.	1648.	1547.	1582.	1618.	1518.	1554.	1589.	1500.	1536.	1571.
	NM/LB	.235	.235	.235	.245	.245	.245	.254	.254	.254	.262	.262	.262
20,000	TRUE M	.656	.656	.656	.675	.675	.675	.693	.693	.693	.712	.712	.712
	TAS	376.	385.	393.	384.	392.	401.	390.	399.	408.	399.	408.	418.
	IAS	240.	240.	240.	236.	236.	236.	232.	232.	232.	228.	228.	228.
	FF-LBS	1646.	1683.	1720.	1616.	1653.	1690.	1590.	1628.	1664.	1577.	1614.	1651.
	NM/LB	.229	.229	.229	.237	.237	.237	.245	.245	.245	.253	.253	.253

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Figure 25-11. Cruise Control — Maximum Range Speed CT-39G (Sheet 2 of 3)

MODEL: CT-39G
DATA AS OF: APRIL 1975

MAXIMUM RANGE SPEED
TWO ENGINE

GROSS WT. - LBS.	ALT.	39,000 Feet			41,000 Feet			43,000 Feet			45,000 Feet		
	STD. TEMP.	−56.5°C (−69.7°F)			−56.5°C (−69.7°F)			−56.5°C (−69.7°F)			−56.5°C (−69.7°F)		
	ACT. TEMP.	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C
12,000	TRUE M	.638	.638	.638	.659	.659	.659	.680	.680	.680	.701	.701	.701
	TAS	357.	366.	374.	369.	378.	387.	381.	390.	399.	393.	402.	411.
	IAS	194.	194.	194.	191.	191.	191.	189.	189.	189.	186.	186	186.
	FF-LBS	990.	1014.	1037.	975.	999.	1021.	962.	985.	1008.	951.	974.	996.
	NM/LB	.361	.361	.361	.378	.378	.378	.396	.396	.396	.413	.413	.413
13,000	TRUE M	.651	.651	.651	.672	.672	.672	.693	.693	.693	.714	.714	.714
	TAS	365.	373.	382.	376.	385.	394.	388.	397.	406.	400.	409.	419.
	IAS	198.	198.	198.	195.	195.	195.	193.	193.	193.	190.	190.	190.
	FF-LBS	1055.	1081.	1105.	1041.	1066.	1091.	1030.	1055.	1079.	1021.	1046.	1070.
	NM/LB	.345	.345	.345	.361	.361	.361	.377	.377	.377	.391	.391	.391
14,000	TRUE M	.664	.664	.664	.685	.685	.685	.706	.706	.706	.726	.726	.717
	TAS	372.	381.	389.	384.	393.	402.	395.	405.	414.	407.	416.	421.
	IAS	202.	202.	202.	199.	199.	199.	197.	197.	197.	193.	193.	191.
	FF-LBS	1123.	1150.	1176.	1110.	1137.	1163.	1101.	1127.	1153.	1095.	1121.	1134.
	NM/LB	.331	.331	.331	.346	.346	.346	.359	.359	.359	.371	.371	.371
15,000	TRUE M	.676	.676	.676	.697	.697	.697	.718	.718	.718	–	–	–
	TAS	379.	388.	396.	390.	400.	409.	402.	412.	421.	–	–	–
	IAS	206.	206.	206.	203.	203.	203.	200.	200.	200.	–	–	–
	FF-LBS	1191.	1219.	1247.	1181.	1209.	1237.	1174.	1202.	1230.	–	–	–
	NM/LB	.318	.318	.318	.330	.330	.330	.342	.342	.342	–	–	–
16,000	TRUE M	.689	.689	.689	.709	.709	.709	.730	.730	–	–	–	–
	TAS	386.	395.	404.	397.	407.	416.	409.	419.	–	–	–	–
	IAS	210.	210.	210.	207.	207.	207.	204.	204.	–	–	–	–
	FF-LBS	1264.	1294.	1324.	1255.	1285.	1314.	1252.	1282.	–	–	–	–
	NM/LB	.305	.305	.305	.316	.316	.316	.326	.326	–	–	–	–
17,000	TRUE M	.701	.701	.701	.721	.721	.721	–	–	–	–	–	–
	TAS	393.	402.	411.	404.	413.	423.	–	–	–	–	–	–
	IAS	214.	214.	214.	211.	211.	211.	–	–	–	–	–	–
	FF-LBS	1339.	1371.	1402.	1333.	1364.	1396.	–	–	–	–	–	–
	NM/LB	.293.	.293	.293	.303	.303	.303	–	–	–	–	–	–
18,000	TRUE M	.712	.712	.712	.715	.715	–	–	–	–	–	–	–
	TAS	399.	408.	418.	401.	410.	–	–	–	–	–	–	–
	IAS	218.	218.	218.	209.	209.	–	–	–	–	–	–	–
	FF-LBS	1415.	1449.	1482.	1384.	1417.	–	–	–	–	–	–	–
	NM/LB	.282	.282	.282	.289	.289	–	–	–	–	–	–	–
19,000	TRUE M	.722	.722	.718	–	–	–	–	–	–	–	–	–
	TAS	404.	414.	421.	–	–	–	–	–	–	–	–	–
	IAS	221.	221.	220.	–	–	–	–	–	–	–	–	–
	FF-LBS	1494.	1529.	1556.	–	–	–	–	–	–	–	–	–
	NM/LB	.271	.271	.271	–	–	–	–	–	–			
20,000	TRUE M	.711	.711	–	–	–	–	–	–	–	–	–	–
	TAS	398.	408.	–	–	–	–	–	–	–	–	–	–
	IAS	218.	218.	–	–	–	–	–	–	–	–	–	–
	FF-LBS	1538.	1575.	–	–	–	–	–	–	–	–	–	–
	NM/LB	.259	.259.	–	–	–	–	–	–	–	–	–	–

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Figure 25-11. Cruise Control — Maximum Range Speed CT-39G (Sheet 3 of 3)

RECOMMENDED CRUISE SPEED
TWO ENGINE

MODEL: CT-39G
DATA AS OF: APRIL 1975

GROSS WT LBS	ALT	15,000 Feet			20,000 Feet			25,000 Feet			29,000 Feet		
	STD. TEMP	-14.7°C (5.5°F)			-24.6°C (-12.3°F)			-34.5°C (-30.1°F)			-42.4°C (-44.4°F)		
	ACT TEMP	-10°C	ISA	+10°C	-10°C	ISA	+10°C	-10°C	ISA	+10°C	-10°C	ISA	+10°C
12,000	TRUE M	.570	.570	.570	.600	.600	.600	.637	.637	.637	.674	.674	.674
	TAS	350.	357.	364.	361.	368.	376.	375.	383.	391.	390.	399.	407.
	IAS	288.	288.	288.	275.	275.	275.	265.	265.	265.	258.	258.	258.
	FF-LBS	2082.	2123.	2164.	1834.	1872.	1909.	1641.	1677.	1712.	1522.	1556.	1589.
	NM/LB	.168	.168	.168	.197	.197	.197	.229	.229	.229	.256	.256	.256
13,000	TRUE M	.570	.570	.570	.600	.600	.600	.637	.637	.637	.674	.674	.674
	TAS	350.	357.	364.	361.	368.	376.	375.	383.	391.	390.	399.	407.
	IAS	288.	288.	288.	275.	275.	275.	265.	265.	265.	258.	258.	258.
	FF-LBS	2100.	2142.	2183.	1854.	1893.	1930.	1664.	1700.	1735.	1547.	1582.	1616.
	NM/LB	.167	.167	.167	.195	.195	.195	.226	.226	.226	.252	.252	.252
14,000	TRUE M	.570	.570	.570	.600	.600	.600	.637	.637	.637	.674	.674	.674
	TAS	350.	357.	364.	361.	368.	376.	375.	383.	391.	390.	399.	407.
	IAS	288.	288.	288.	275.	275.	275.	265.	265.	265.	258.	258.	258.
	FF-LBS	2120.	2162.	2203.	1877.	1916.	1954.	1689.	1725.	1761.	1574.	1610.	1644.
	NM/LB	.165	.165	.165	.192	.192	.192	.222	.222	.222	.248	.248	.248
15,000	TRUE M	.570	.570	.570	.600	.600	.600	.637	.637	.637	.674	.674	.674
	TAS	350.	357.	364.	361.	368.	376.	375.	383.	391.	390.	399.	407.
	IAS	288.	288.	288.	275.	275.	275.	265.	265.	265.	258.	258.	258.
	FF-LBS	2141.	2183.	2225.	1901.	1940.	1979.	1715.	1752.	1789.	1604.	1640.	1675.
	NM/LB	.164	.164	.164	.190	.190	.190	.219	.219	.219	.243	243	.243
16,000	TRUE M	.570	.570	.570	.600	.600	.600	.637	.637	.637	.674	.674	.674
	TAS	350.	357.	364.	361.	368.	376.	375.	383.	391.	390.	399.	407.
	IAS	288.	288.	288.	275.	275.	275.	265.	265.	265.	258.	258.	258.
	FF-LBS	2163.	2206.	2249.	1927.	1967.	2006.	1744.	1781.	1818.	1635.	1672.	1708.
	NM/LB	.162	.162	.162	.187	.187	.187	.215	.215	.215	.239	.239	.239
17,000	TRUE M	.570	.570	.570	.600	.600	.600	.637	.637	.637	.674	.674	.674
	TAS	350.	357.	364.	361.	368.	376.	375.	383.	391.	390.	399.	407.
	IAS	288.	288.	288.	275.	275.	275.	265.	265.	265.	258.	258.	258.
	FF-LBS	2187.	2231.	2274.	1954.	1995.	2035.	1774.	1813.	1850.	1669.	1707.	1743.
	NM/LB	.160	.160	.160	.185	.185	.185	.211	.211	.211	.234	.234	.234
18,000	TRUE M	.570	.570	.570	.600	.600	.600	.637	.637	.637	.674	.674	.674
	TAS	350.	357.	364.	361.	368.	376.	375.	383.	391.	390.	399.	407.
	IAS	288.	288.	288.	275.	275.	275.	265.	265.	265.	258.	258.	258.
	FF-LBS	2213.	2257.	2300.	1984.	2025.	2065.	1807.	1846.	1884.	1705.	1744.	1781.
	NM/LB	.158	.158	.158	.182	.182	.182	.208	.208	.208	.229	.229	.229
19,000	TRUE M	.570	.570	.570	.600	.600	.600	.637	.637	.637	.674	.674	.674
	TAS	350.	357.	364.	361.	368.	376.	375.	383.	391.	390.	399.	407.
	IAS	288.	288.	288.	275.	275.	275.	265.	265.	265.	258.	258.	258.
	FF-LBS	2240.	2285.	2329.	2015.	2057.	2098.	1842.	1882.	1921.	1744.	1783.	1821.
	NM/LB	.156	.156	.156	.179	.179	.179	.204	.204	.204	.224	.224	.224
20,000	TRUE M	.570	.570	.570	.600	.600	.600	.637	.637	.637	.674	.674	.674
	TAS	350.	357.	364.	361.	368.	376.	375.	383.	391.	390.	399.	407.
	IAS	288.	288.	288.	275.	275.	275.	265.	265.	265.	258.	258.	258.
	FF-LBS	2269.	2314.	2359.	2048.	2091.	2132.	1879.	1919.	1959.	1785.	1825.	1864.
	NM/LB	.154	.154	.154	.176	.176	.176	.200	.200	.200	.219	.219	.219

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Figure 25-12. Cruise Control — Recommended Cruise Speed CT-39G (Sheet 1 of 3)

MODEL: CT-39G
DATA AS OF: APRIL 1975

RECOMMENDED CRUISE SPEED
TWO ENGINE

GROSS WT. - LBS.	ALT.	31,000 Feet			33,000 Feet			35,000 Feet			37,000 Feet		
	STD. TEMP.	−46.4°C (−51.5°F)			−50.4°C (−58.7°F)			−54.3°C (−65.8°F)			−56.5°C (−69.7°F)		
	ACT. TEMP.	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C
12,000	TRUE M	.695	.695	.695	.715	.715	.715	.735	.735	.735	.750	.750	.750
	TAS	399.	408.	417.	406.	416.	425.	414.	424.	433.	420.	430.	440.
	IAS	255.	255.	255.	252.	252.	252.	248.	248.	248.	242.	242.	242.
	FF-LBS	1473.	1507.	1540.	1424.	1457.	1489.	1380.	1413.	1445.	1330.	1362.	1393.
	NM/LB	.271	.271	.271	.285	.285	.285	.300	.300	.300	.316	.316	.316
13,000	TRUE M	.695	.695	.695	.715	.715	.715	.735	.735	.735	.750	.750	.750
	TAS	399.	408.	417.	406.	416.	425.	414.	424.	433.	420.	430.	440.
	IAS	255.	255.	255.	252.	252.	252.	248.	248.	248.	242.	242.	242.
	FF-LBS	1500.	1534.	1567.	1452.	1486.	1519.	1410.	1444.	1476.	1363.	1396.	1427.
	NM/LB	.266	.266	.266	.280	.280	.280	.293	.293	.293	.308	.308	.308
14,000	TRUE M	.695	.695	.695	.715	.715	.715	.735	.735	.735	.750	.750	.750
	TAS	399.	408.	417.	406.	416.	425.	414.	424.	433.	420.	430.	440.
	IAS	255.	255.	255.	252.	252.	252.	248.	248.	248.	242.	242.	242.
	FF-LBS	1528.	1563.	1597.	1482.	1517.	1550.	1443.	1477.	1510.	1399.	1433.	1465.
	NM/LB	.261	.261	.261	.274	.274	.274	.287	.287	.287	.300	.300	.300
15,000	TRUE M	.695	.695	.695	.715	.715	.715	.735	.735	.735	.750	.750	.750
	TAS	399.	408.	417.	406.	416.	425.	414.	424.	433.	420.	430.	440.
	IAS	255.	255.	255.	252.	252.	252.	248.	248.	248.	242.	242.	242.
	FF-LBS	1559.	1595.	1629.	1515.	1550.	1595.	1478.	1513.	1548.	1438.	1472.	1506.
	NM/LB	.256	.256	.256	.268	.268	.268	.280	.280	.280	.292	.292	.292
16,000	TRUE M	.695	.695	.695	.715	.715	.715	.735	.735	.735	.750	.750	.750
	TAS	399.	408.	417.	406.	416.	425.	414.	424.	433.	420.	430.	440.
	IAS	255.	255.	255.	252.	252.	252.	248.	248.	248.	242.	242.	242.
	FF-LBS	1592.	1629.	1664.	1551.	1587.	1622.	1517.	1552.	1588.	1480.	1515.	1550.
	NM/LB	.250	.250	.250	.262	.262	.262	.273	.273	.273	.284	.284	.284
17,000	TRUE M	.695	.695	.695	.715	.715	.715	.735	.735	.735	.750	.750	.750
	TAS	399.	408.	417.	406.	416.	425.	414.	424.	433.	420.	430.	440.
	IAS	255.	255.	255.	252.	252.	252.	248.	248.	248.	242.	242.	242.
	FF-LBS	1628.	1665.	1702.	1589.	1626.	1662.	1558.	1594.	1631.	1525.	1561.	1597.
	NM/LB	.245	.245	.245	.256	.256	.256	.266	.266	.266	.275	.275	.275
18,000	TRUE M	.695	.695	.695	.715	.715	.715	.735	.735	.735	.750	.750	.750
	TAS	399.	408.	417.	406.	416.	425.	414.	424.	433.	420.	430.	440.
	IAS	255.	255.	255.	252.	252.	252.	248.	248.	248.	242.	242.	242.
	FF-LBS	1666.	1704.	1741.	1629.	1667.	1704.	1601.	1639.	1676.	1573.	1611.	1647.
	NM/LB	.239	.239	.239	.249	.249	.249	.258	.258	.258	.267	.267	.267
19,000	TRUE M	.695	.695	.695	.715	.715	.715	.735	.735	.735	.750	.750	.750
	TAS	399.	408.	417.	406.	416.	425.	414.	424.	433.	420.	430.	440.
	IAS	255.	255.	255.	252.	252.	252.	248.	248.	248.	242.	242.	242.
	FF-LBS	1707.	1746.	1784.	1672.	1711.	1749.	1648.	1687.	1725.	1625.	1664.	1702.
	NM/LB	.234	.234	.234	.243	.243	.243	.251	.251	.251	.259	.259	.259
20,000	TRUE M	.695	.695	.695	.715	.715	.715	.735	.735	.735	.750	.750	.750
	TAS	399.	408.	417.	406.	416.	425.	414.	424.	433.	420.	430.	440.
	IAS	255.	255.	255.	252.	252.	252.	248.	248.	248.	242.	242.	242.
	FF-LBS	1750.	1790.	1829.	1718.	1758.	1797.	1698.	1738.	1778.	1679.	1720.	1758.
	NM/LB	.228	.228	.228	.236	.236	.236	.244	.244	.244	.250	.250	.250

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Figure 25-12. Cruise Control — Recommended Cruise Speed CT-39G (Sheet 2 of 3)

RECOMMENDED CRUISE SPEED
TWO ENGINE

MODEL: CT-39G
DATA AS OF: APRIL 1975

GROSS WT. - LBS.	ALT.	39,000 Feet			41,000 Feet			43,000 Feet			45,000 Feet		
	STD. TEMP.	−56.5°C (−69.7°F)			−56.5°C (−69.7°F)			−56.5°C (−69.7°F)			−56.5°C (−69.7°F)		
	ACT. TEMP.	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C
12,000	TRUE M	.750	.750	.750	.750	.750	.750	.750	.750	.750	.750	.750	.750
	TAS	420.	430.	440.	420.	430.	440.	420.	430.	440.	420.	430.	440.
	IAS	231.	231.	231.	220.	220.	220.	210.	210.	210.	201.	201.	201.
	FF-LBS	1239.	1269.	1298.	1161.	1189.	1216.	1094.	1120.	1145.	1037.	1062.	1086.
	NM/LB	.339	.339	.339	.362	.362	.362	.384	.384	.384	.405	.405	.405
13,000	TRUE M	.750	.750	.750	.750	.750	.750	.750	.750	.750	.750	.750	.750
	TAS	420.	430.	440.	420.	430.	440.	420.	430.	440.	420.	430.	440.
	IAS	231.	231.	231.	220.	220.	220.	210.	210.	210.	201.	201.	201.
	FF-LBS	1275.	1306.	1336.	1201.	1230.	1258.	1138.	1165.	1192.	1087.	1113.	1138.
	NM/LB	.329	.329.	.329	.350	.350	.350	.369	.369	.369	.386	.386	.386
14,000	TRUE M	.750	.750	.750	.750	.750	.750	.750	.750	.750	.750	.750	.717
	TAS	420.	430.	440.	420.	430.	440.	420.	430.	440.	420.	430.	421.
	IAS	231.	231.	231.	220.	220.	220.	210.	210.	210.	201.	201.	191.
	FF-LBS	1315.	1347.	1377.	1245.	1275.	1304.	1186.	1215.	1242.	1141.	1168.	1134.
	NM/LB	.319	.319	.319	.337	.337	.337	.354	.354	.354	.368	.368	.371
15,000	TRUE M	.750	.750	.750	.750	.750	.750	.750	.750	.741	–	–	–
	TAS	420.	430.	440.	420.	430.	440.	420.	430.	435.	–	–	–
	IAS	231.	231.	231.	220.	220.	220.	210.	210.	207.	–	–	–
	FF-LBS	1358.	1391.	1422.	1292.	1323.	1353.	1239.	1268.	1275.	–	–	–
	NM/LB	.309	.309	.309	.325	.325	.325	.339	.339	.341	–	–	–
16,000	TRUE M	.750	.750	.750	.750	.750	.750	.735	.735	–	–	–	–
	TAS	420.	430.	440.	420.	430.	440.	412.	422.	–	–	–	–
	IAS	231.	231.	231.	220.	220.	220.	206.	206.	–	–	–	–
	FF-LBS	1404.	1438.	1471.	1343.	1375.	1407.	1264.	1294.	–	–	–	–
	NM/LB	.299	.299	.299	.313.	.313	.313	.326	.326	–	–	–	–
17,000	TRUE M	.750	.750	.750	.750	.750	.724	–	–	–	–	–	–
	TAS	420.	430.	440.	420.	430.	425.	–	–	–	–	–	–
	IAS	231.	231.	231.	220.	220.	212.	–	–	–	–	–	–
	FF-LBS	1454.	1489.	1523.	1398.	1431.	1403.	–	–	–	–	–	–
	NM/LB	.289	.289	.289	.300	.301	.303	–	–	–	–	–	–
18,000	TRUE M	.750	.750	.750	.715	.715	–	–	–	–	–	–	–
	TAS	420.	430.	440.	401.	410.	–	–	–	–	–	–	–
	IAS	231.	231.	231.	209.	209.	–	–	–	–	–	–	–
	FF-LBS	1507.	1543.	1579.	1385.	1418.	–	–	–	–	–	–	–
	NM/LB	.279	.279	.279	.289	.289	–	–	–	–	–	–	–
19,000	TRUE M	.750	.750	.718	–	–	–	–	–	–	–	–	–
	TAS	420.	430.	421.	–	–	–	–	–	–	–	–	–
	IAS	231.	231.	220.	–	–	–	–	–	–	–	–	–
	FF-LBS	1564.	1602.	1557.	–	–	–	–	–	–	–	–	–
	NM/LB	.269	.269	.271	–	–	–	–	–	–	–	–	–
20,000	TRUE M	.711	.711	–	–	–	–	–	–	–	–	–	–
	TAS	398.	408.	–	–	–	–	–	–	–	–	–	–
	IAS	218.	218.	–	–	–	–	–	–	–	–	–	–
	FF-LBS	1539.	1575.	–	–	–	–	–	–	–	–	–	–
	NM/LB	.259	.259	–	–	–	–	–	–	–	–	–	–

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Figure 25-12. Cruise Control — Recommended Cruise Speed CT-39G (Sheet 3 of 3)

MODEL: CT-39G
DATA AS OF: APRIL 1975

MAXIMUM SPEED CRUISE
TWO ENGINES

GROSS WT. - LBS.	ALT.	15,000 Feet			20,000 Feet			25,000 Feet			29,000 Feet		
	STD. TEMP.	−14.7°C (5.5°F)			−24.6°C (−12.3°F)			−34.5°C (−30.1°F)			−42.4°C (−44.4°F)		
	ACT. TEMP.	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C
12,000	TRUE M	.705	.705	.705	.781	.781	.781	.800	.800	.800	.800	.800	.800
	TAS	433.	442.	450.	470.	480.	489.	471.	481.	491.	463.	473.	484.
	IAS	359.	359.	359.	364.	364.	364.	338.	338.	338.	311.	311.	311.
	FF-LBS	2981.	3041.	3099.	3030.	3093.	3155.	2633.	2690.	2745.	2215.	2264.	2313.
	NM/LB	.145	.145	.145	.155	.155	.155	.179	.179	.179	.209	.209	.209
13,000	TRUE M	.705	.705	.705	.781	.781	.781	.800	.800	.800	.800	.800	.800
	TAS	433.	442.	450.	470.	480.	489.	471.	481.	491.	463.	473.	484.
	IAS	359.	359.	359.	364.	364.	364.	338.	338.	338.	311.	311.	311.
	FF-LBS	2995.	3054.	3113.	3047.	3111.	3171.	2654.	2711.	2768.	2239.	2290.	2339.
	NM/LB	.145	.145	.145	1.54	.154	.154	.178	.178	.178	.207	.207	.207
14,000	TRUE M	.705	.705	.705	.781	.781	.781	.800	.800	.800	.800	.800	.800
	TAS	433.	442.	450.	470.	480.	489.	471.	481.	491.	463.	473.	484.
	IAS	359.	359.	359.	364.	364.	364.	338.	338.	338.	311.	311.	311.
	FF-LBS	3009.	3069.	3128.	3066.	3129.	3191.	2677.	2735.	2791.	2268.	2319.	2369.
	NM/LB	.144	.144	.144	.153	.153	.153	.176	.176	.176	.204	.204	.204
15,000	TRUE M	.705	.705	.705	.781	.781	.781	.800	.800	.800	.800	.800	.800
	TAS	433.	442.	450.	470.	480.	489.	471.	481.	491.	463.	473.	484.
	IAS	359.	359.	359.	364.	364.	364.	338.	338.	338.	311.	311.	311.
	FF-LBS	3024.	3084.	3143.	3085.	3149.	3211.	2701.	2759.	2816.	2298.	2350.	2400.
	NM/LB	.143	.143	.143	.152.	.152	.152	.174	.174	.174	.201	.201	.201
16,000	TRUE M	.705	.705	.705	.781	.781	.781	.800	.800	.800	.800	.800	.800
	TAS	433.	442.	450.	470.	480.	489.	471.	481.	491.	463.	473.	484.
	IAS	359.	359.	359.	364.	364.	364.	338.	338.	338.	311.	311.	311.
	FF-LBS	3040.	3101.	3160.	3105.	3169.	3232.	2727.	2786.	2844.	2331.	2383.	2434.
	NM/LB	.142	.142	.142	.151	.151	.151	.173	.173	.173	.199	.199	.199
17,000	TRUE M	.705	.705	.705	.781	.781	.781	.800	.800	.800	.800	.800	.800
	TAS	433.	442.	450.	470.	480.	489.	471.	481.	491.	463.	473.	484.
	IAS	359.	359.	359.	364.	364.	364.	338.	338.	338.	311.	311.	311.
	FF-LBS	3058.	3119.	3178.	3126.	3191.	3255.	2756.	2816.	2875.	2364.	2417.	2469.
	NM/LB	.142	.142	.142	.150	.150	.150	.171	.171	.171	.196	.196	.196
18,000	TRUE M	.705	.705	.705	.781	.781	.781	.800	.800	.798	.800	.800	.799
	TAS	433.	442.	450.	470.	480.	489.	471.	481.	490.	463.	473.	483.
	IAS	359.	359.	359.	364.	364.	364.	338.	338.	337.	311.	311.	310.
	FF-LBS	3076.	3137.	3198.	3149.	3214.	3278.	2787.	2848.	2879.	2402.	2456.	2495.
	NM/LB	.141	.141	.141	.149	.149	.149	.169	.169	.170	.193	.193	.194
19,000	TRUE M	.705	.705	.705	.781	.781	.781	.800	.800	.795	.800	.800	.796
	TAS	433.	442.	450.	470.	480.	489.	471.	481.	488.	463.	473.	480.
	IAS	359.	359.	359.	364.	364.	364.	338.	338.	335.	311.	311.	308.
	FF-LBS	3096.	3157.	3218.	3173.	3238.	3303.	2819.	2880.	2873.	2442.	2497.	2487.
	NM/LB	.140	.140	.140	.148	.148	.148	.167	.167	.170	.190	.190	.193
20,000	TRUE M	.705	.705	.705	.781	.781	.781	.800	.800	.792	.800	.800	.791
	TAS	433.	442.	450.	470.	480.	489.	471.	481.	486.	463.	473.	478.
	IAS	359.	359.	359.	364.	364.	364.	338.	338.	334.	311.	311.	307.
	FF-LBS	3116.	3178.	3239.	3198.	3264.	3329.	2853.	2914.	2872.	2484.	2540.	2484.
	NM/LB	.139	.139	.139	.147	.147	.147	.165	.165	.169	.186	.186	.192

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Figure 25-13. Cruise Control — Maximum Speed CT-39G (Sheet 1 of 3)

MAXIMUM SPEED CRUISE
TWO ENGINE

MODEL: CT-39G
DATA AS OF: APRIL 1975

GROSS WT LBS	ALT.	31,000 Feet			33,000 Feet			35,000 Feet			37,000 Feet		
	STD. TEMP.	−46.4°C (−51.5°F)			−50.4°C (−58.7°F)			−54.3°C (−65.8°F)			−56.5°C (−69.7°F)		
	ACT. TEMP.	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C
12,000	TRUE M	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800
	TAS	459.	469.	480.	455.	465.	476.	450.	461.	471.	448.	459.	469.
	IAS	297.	297.	297.	285.	285.	285.	272.	272.	272.	260.	260.	260.
	FF-LBS	2033.	2080.	2125.	1869.	1913.	1955.	1721.	1761.	1801.	1589.	1627.	1664.
	NM/LB	.226	.226	.226	.243	.243	.243	.262	.262	.262	.282	.282	.282
13,000	TRUE M	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800
	TAS	459.	469.	480.	455.	465.	476.	450.	461.	471.	448.	459.	469.
	IAS	297.	297.	297.	285.	285.	285.	272.	272.	272.	260.	260.	260.
	FF-LBS	2062.	2109.	2155.	1900.	1944.	1988.	1754.	1796.	1836.	1628.	1667.	1705.
	NM/LB	.222	.222	.222	.239	.239	.239	.257	.257	.257	.275	.275	.275
14,000	TRUE M	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800
	TAS	459.	469.	480.	455.	465.	476.	450.	461.	471.	448.	459.	469.
	IAS	297.	297.	297.	285.	285.	285.	272.	272.	272.	260.	260.	260.
	FF-LBS	2093.	2141.	2187.	1933.	1978.	2022.	1793.	1835.	1877.	1670.	1710.	1749.
	NM/LB	.219	.219	.219	.235	.235	.235	.251	.251	.251	.268	.268	.268
15,000	TRUE M	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800
	TAS	459.	469.	480.	455.	465.	476.	450.	461.	471.	448.	459.	469.
	IAS	297.	297.	297.	285.	285.	285.	272.	272.	272.	260.	260.	260.
	FF-LBS	2126.	2174.	2222.	1971.	2016.	2061.	1834.	1877.	1920.	1716.	1757.	1797.
	NM/LB	.216	.216	.216	.231	.231	.231	.246	.246	.246	.261	.261	.261
16,000	TRUE M	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800	.800	.796
	TAS	459.	469.	480.	455.	465.	476.	450.	461.	471.	448.	459.	466.
	IAS	297.	297.	297.	285.	285.	285.	272.	272.	272.	260.	260.	258.
	FF-LBS	2161.	2210.	2259.	2011.	2057.	2103.	1877.	1921.	1965.	1768.	1810.	1799.
	NM/LB	.212	.212	.212	.226	.226	.226	.240	.240	.240	.253	.253	.259
17,000	TRUE M	.800	.800	.800	.800	.800	.800	.800	.800	.798	.798	.798	.790
	TAS	459.	469.	480.	455.	465.	476.	450.	461.	470.	447.	458.	463.
	IAS	297.	297.	297.	285.	285.	285.	272.	272.	271.	259.	259.	256.
	FF-LBS	2200.	2250.	2299.	2053.	2101.	2147.	1927.	1973.	1980.	1798.	1841.	1804.
	NM/LB	.209	.209	.209	.221	.221	.221	.234	.234	.237	.249	.249	.257
18,000	TRUE M	.800	.800	.798	.800	.800	.796	.800	.800	.792	.791	.791	.780
	TAS	459.	469.	478.	455.	465.	473.	450.	460.	466.	442.	453.	458.
	IAS	297.	297.	296.	285.	285.	283.	272.	272.	269.	256.	256.	253.
	FF-LBS	2241.	2292.	2313.	2099.	2148.	2138.	1966.	2013.	1974.	1775.	1817.	1795.
	NM/LB	.205	.205	.207	.217	.217	.221	.229	.229	.236	.249	.249	.255
19,000	TRUE M	.800	.800	.794	.799	.799	.790	.794	.794	.785	.783	.783	.767
	TAS	459.	469.	476.	454.	464.	470.	447.	458.	462.	439.	449.	451.
	IAS	297.	297.	295.	284.	284.	281.	270.	270.	266.	254.	254.	248.
	FF-LBS	2284.	2336.	2307.	2128.	2177.	2134.	1966.	2013.	1971.	1782.	1825.	1780.
	NM/LB	.201	.201	.206	.213	.213	.220	.227	.227	.235	.246	.246	.253
20,000	TRUE M	.798	.798	.789	.794	.794	.784	.788	.788	.775	.771	.771	.750
	TAS	457.	468.	473.	451.	461.	466.	444.	455.	457.	432.	442.	440.
	IAS	296.	296.	293.	282.	282.	278.	268.	268.	263.	249.	249.	242.
	FF-LBS	2298.	2351.	2307.	2119.	2168.	2132.	1966.	2013.	1960.	1765.	1807.	1759.
	NM/LB	.199	.199	.205	.213	.213	.219	.226	.226	.233	.245	.245	.250

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Figure 25-13. Cruise Control — Maximum Speed CT-39G (Sheet 2 of 3)

MODEL: CT-39G
DATA AS OF: APRIL 1975

MAXIMUM SPEED CRUISE
TWO ENGINE

GROSS WT. - LBS.	ALT.	39,000 Feet			41,000 Feet			43,000 Feet			45,000 Feet		
	STD. TEMP.	−56.5°C (−69.7°F)			−56.5°C (−69.7°F)			−56.5°C (−69.7°F)			−56.5°C (−69.7°F)		
	ACT. TEMP.	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C	−10°C	ISA	+10°C
12,000	TRUE M	.800	.800	.800	.800	.800	.800	.800	.800	.793	.793	.793	.781
	TAS	448.	459.	469.	448.	459.	469.	448.	459.	465.	444.	455.	459.
	IAS	248.	248.	248.	237.	237.	237.	226.	226.	224.	213.	213.	210.
	FF-LBS	1477.	1512.	1546.	1379.	1412.	1444.	1300.	1332.	1307.	1186.	1214.	1185.
	NM/LB	.303	.303	.303	.325	.325	.325	.345	.345	.356	.375	.375	.387
13,000	TRUE M	.800	.800	.800	.800	.800	.795	.794	.794	.783	.781	.781	.760
	TAS	448.	459.	469.	448.	459.	466.	445.	455.	459.	437.	448.	446.
	IAS	248.	248.	248.	237.	237.	235.	224.	224.	221.	210.	210.	204.
	FF-LBS	1518.	1555.	1590.	1429.	1463.	1446.	1306.	1337.	1305.	1176.	1204.	1166.
	NM/LB	.295	.295	.295	.314	.314	.322	.341	.341	.352	.372	.372	.383
14,000	TRUE M	.800	.800	.798	.796	.796	.786	.784	.784	.766	.756	.756	.717
	TAS	448.	459.	468.	446.	457.	461.	439.	450.	450.	424.	434.	421.
	IAS	248.	248.	247.	235.	235.	232.	221.	221.	216.	202.	202.	191.
	FF-LBS	1567.	1604.	1619.	1447.	1481.	1445.	1301.	1332.	1295.	1157.	1185.	1134.
	NM/LB	.286	.286	.289	.308	.308	.319	.338	.338	.348	.366	.366	.371
15,000	TRUE M	.800	.800	.792	.788	.788	.773	.766	.766	.741	–	–	–
	TAS	447.	458.	464.	442.	452.	454.	429.	439.	435.	–	–	–
	IAS	248.	248.	245.	233.	233.	228.	215.	215.	208.	–	–	–
	FF-LBS	1610.	1648.	1614.	1445.	1479.	1437.	1285.	1316.	1277.	–	–	–
	NM/LB	.278	.278	.287	.306	.306	.316	.334	.334	.341	–	–	–
16,000	TRUE M	.793	.793	.783	.775	.775	.755	.735	.735	–	–	–	–
	TAS	444.	455.	459.	435.	445.	443.	412.	422.	–	–	–	–
	IAS	246.	246.	242.	229.	229.	222.	206.	206.	–	–	–	–
	FF-LBS	1610.	1648.	1612.	1432.	1466.	1424.	1263.	1293.	–	–	–	–
	NM/LB	.276	.276	.285	.303	.303	.311	.326	.326	–	–	–	–
17,000	TRUE M	.785	.785	.770	.756	.756	.724	–	–	–	–	–	–
	TAS	440.	450.	452.	423.	434.	425.	–	–	–	–	–	–
	IAS	243.	243.	238.	222.	222.	212.	–	–	–	–	–	–
	FF-LBS	1604.	1643.	1602.	1416.	1449.	1403.	–	–	–	–	–	–
	NM/LB	.274	.274	.282	.299	.299	.303	–	–	–	–	–	–
18,000	TRUE M	.772	.772	.752	.715	.715	–	–	–	–	–	–	–
	TAS	433.	443.	441.	401.	410.	–	–	–	–	–	–	–
	IAS	239.	239.	231.	209.	209.	–	–	–	–	–	–	–
	FF-LBS	1595.	1633.	1585.	1385.	1418.	–	–	–	–	–	–	–
	NM/LB	.272	.272	.278	.289	.289	–	–	–	–	–	–	–
19,000	TRUE M	.753	.753	.718	–	–	–	–	–	–	–	–	–
	TAS	422.	432.	421.	–	–	–	–	–	–	–	–	–
	IAS	232.	232.	220.	–	–	–	–	–	–	–	–	–
	FF-LBS	1576.	1613.	1557.	–	–	–	–	–	–	–	–	–
	NM/LB	.268	.268	.271	–	–	–	–	–	–	–	–	–
20,000	TRUE M	.711	.711	–	–	–	–	–	–	–	–	–	–
	TAS	398.	408.	–	–	–	–	–	–	–	–	–	–
	IAS	218.	218.	–	–	–	–	–	–	–	–	–	–
	FF-LBS	1539.	1575.	–	–	–	–	–	–	–	–	–	–
	NM/LB	.259	.259	–	–	–	–	–	–	–	–	–	–

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Figure 25-13. Cruise Control — Maximum Speed CT-39G (Sheet 3 of 3)

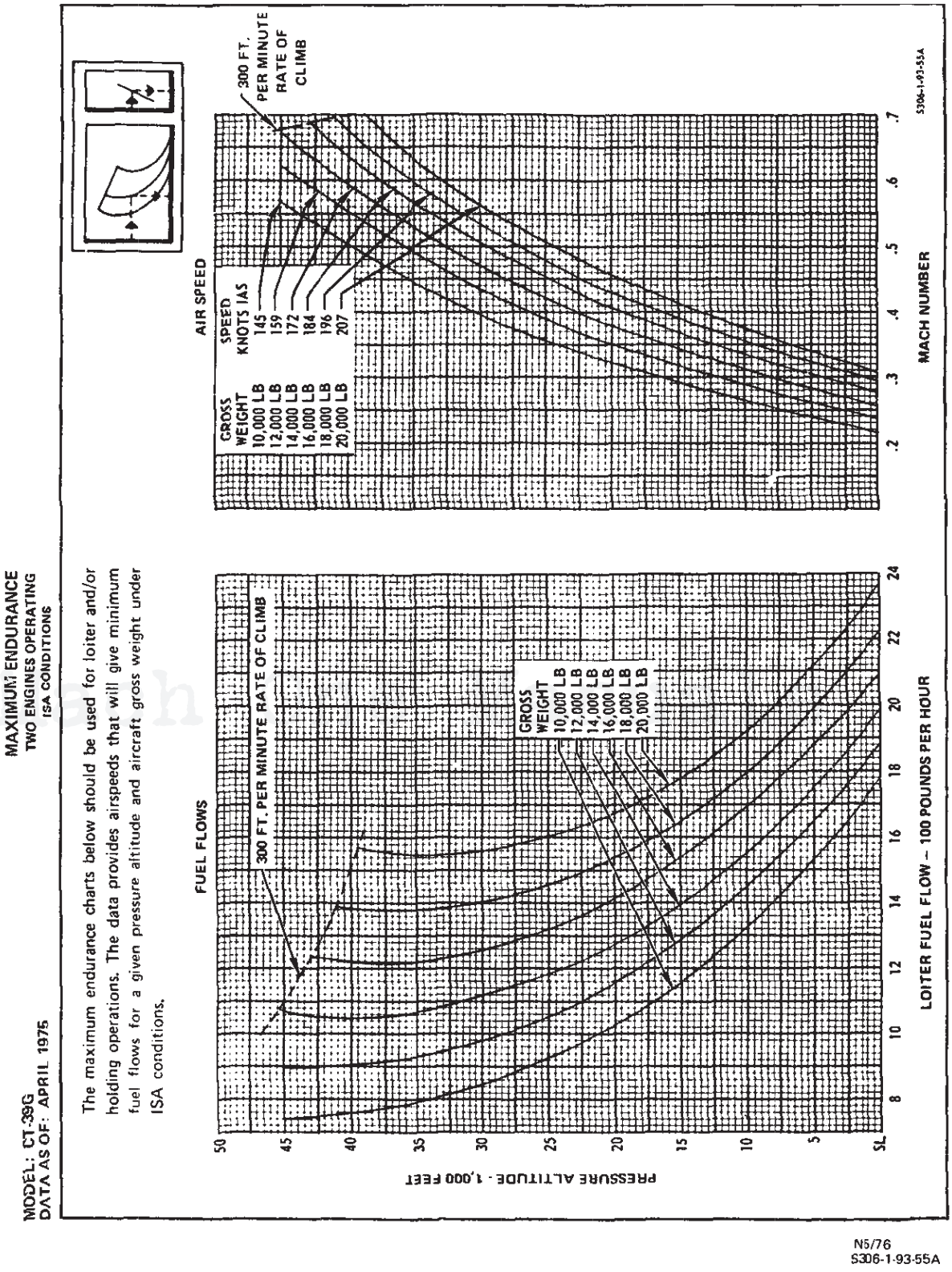


Figure 25-14. Maximum Endurance — CT-39G

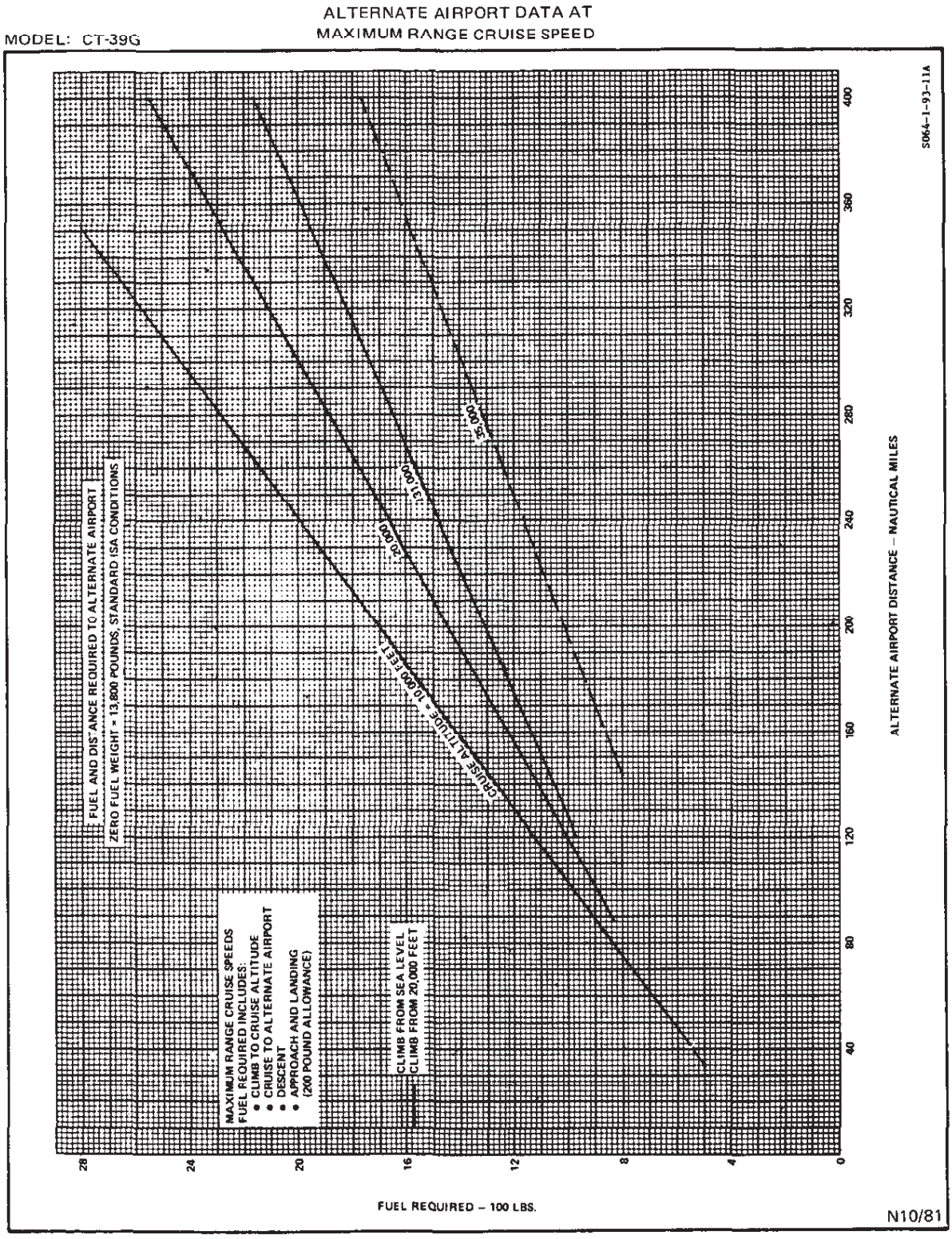


Figure 25-15. Alternate Airport Data — Maximum Range Cruise Speeds (CT-39G)

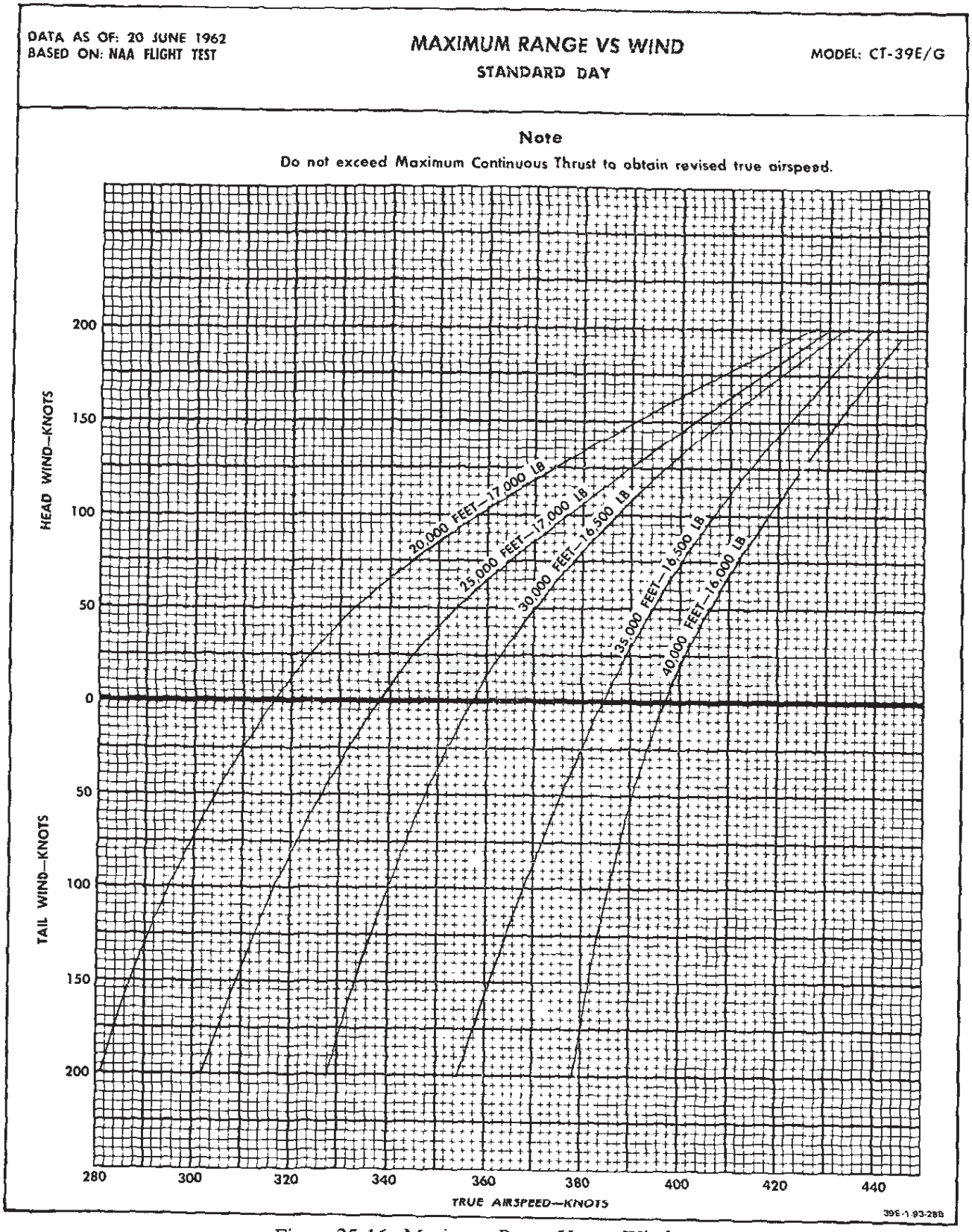


Figure 25-16. Maximum Range Versus Wind

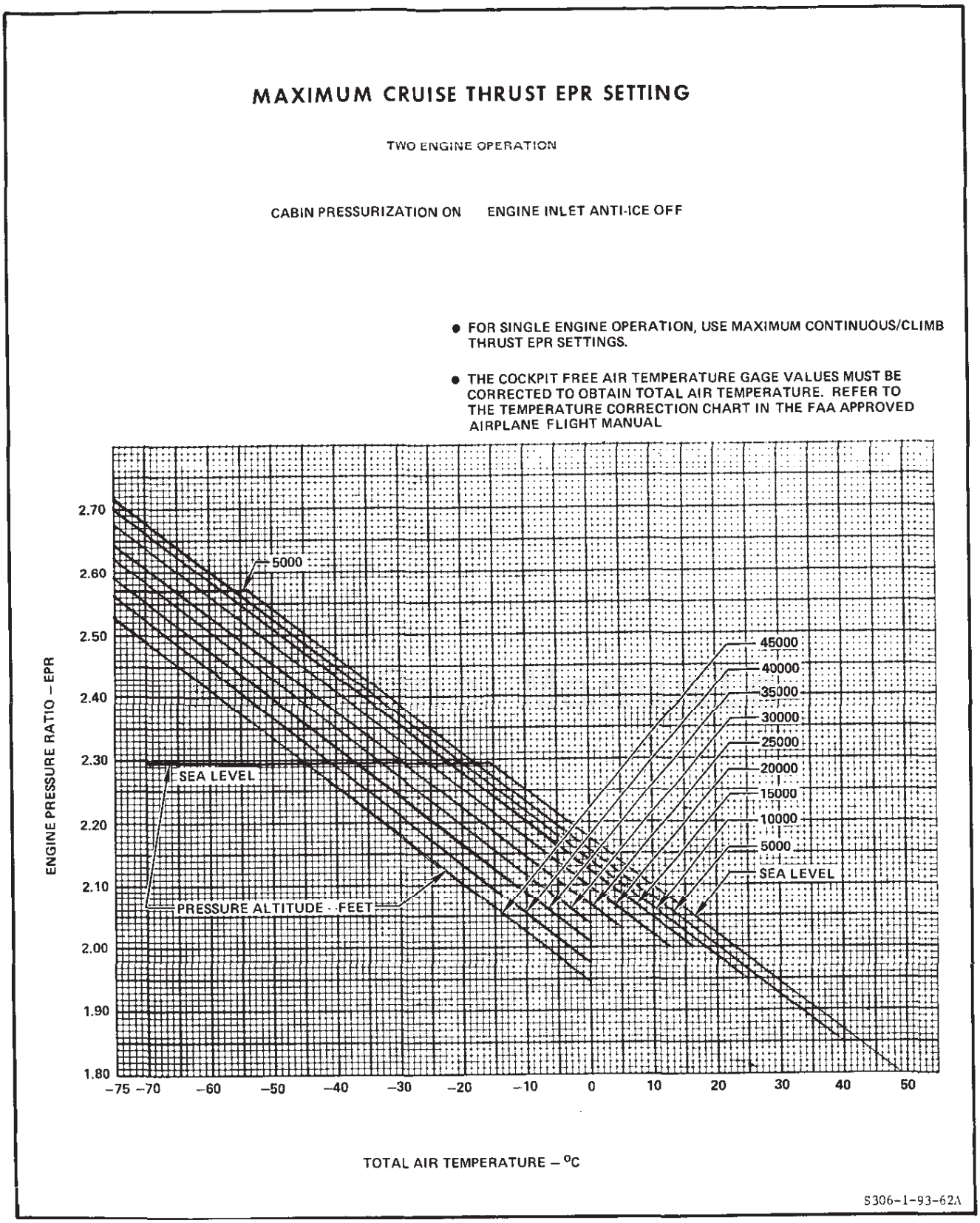


Figure 25-17. Maximum Cruise Thrust EPR Setting (Cabin Pressurization ON — Anti-Ice OFF)

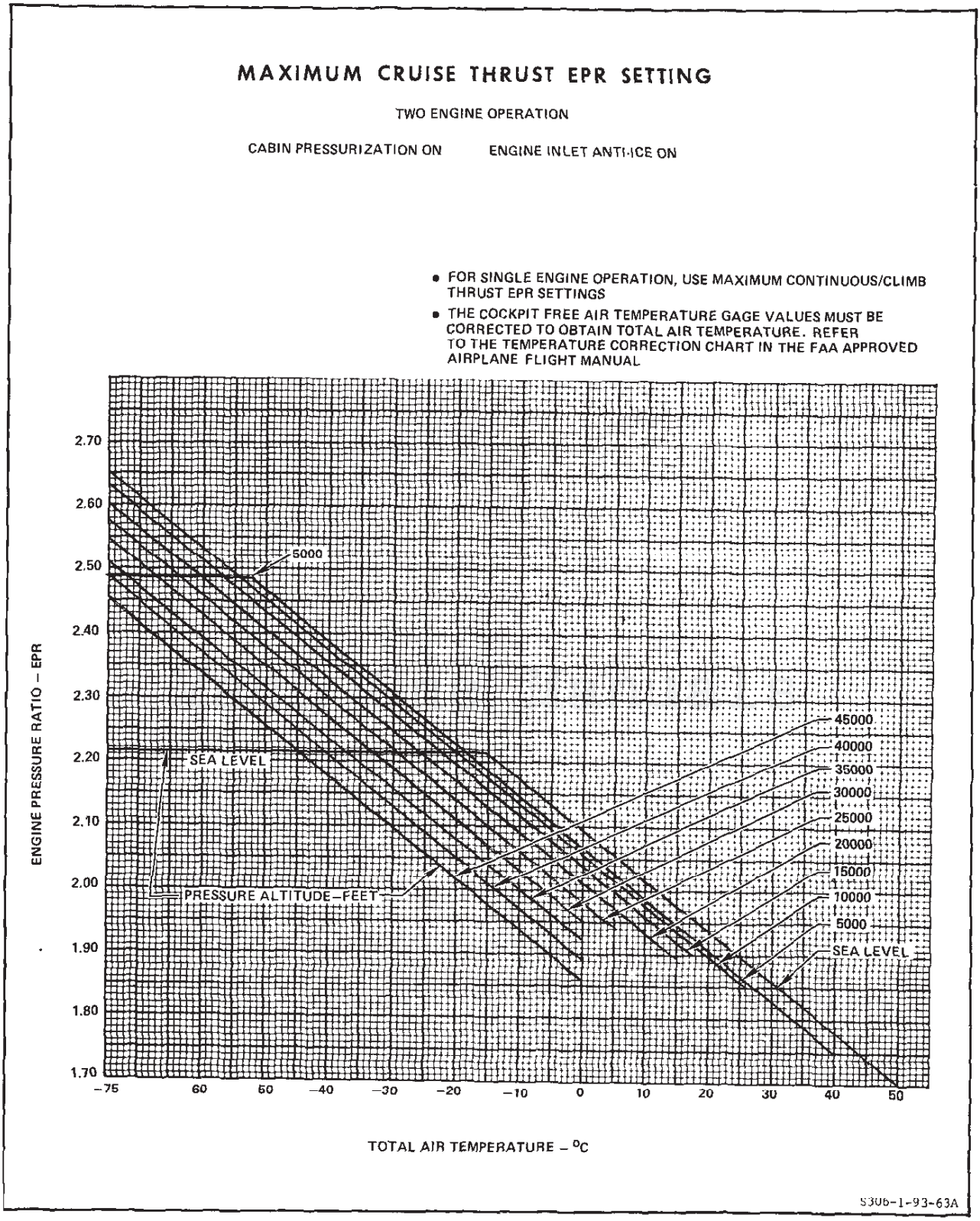


Figure 25-18. Maximum Cruise Thrust EPR Setting (Cabin Pressurization ON — Anti-Ice ON)

Mach One Manuals

CHAPTER 26

Descent

26.1 DESCENTS

Distance, time, fuel consumed, and rate of descent are given for emergency descent (idle thrust, speedbrake open) and normal descent (idle thrust, speedbrake closed). Two additional charts are provided to enable planning of missions involving penetration descents and emergency or power-off descents.

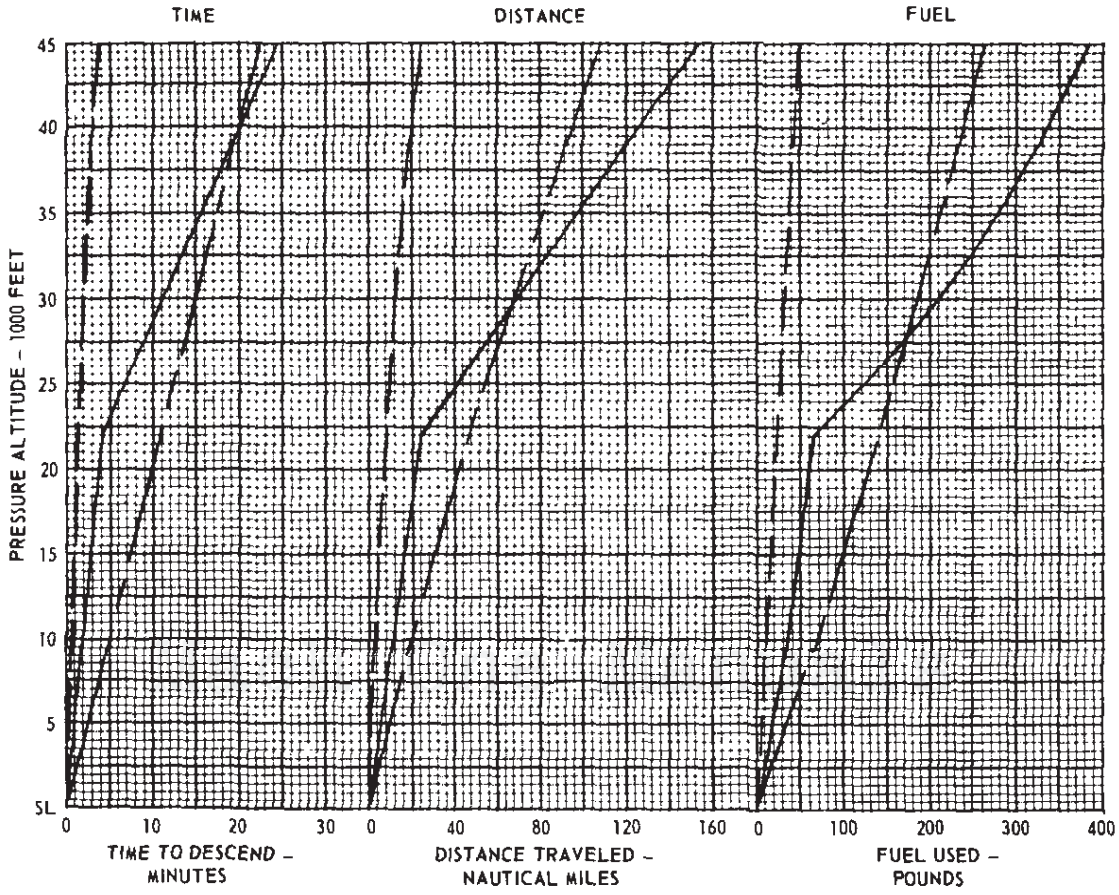
26.1.1 Normal, Emergency, and Idle Thrust Descents. The descent data chart (Figure 26-1) compares fuel, time, and distance to descend between normal, emergency, and idle thrust descents. To make the time, fuel, and distances accurate, maintain the prescribed parameters for the particular descent.

26.1.2 Normal or Emergency Descents. The normal or emergency descent chart shows aircraft configuration for and performance during the normal and the emergency descents. (See Figure 26-2.) To make good the time and distance in either of these descents, maintain the descent speed constant after establishing aircraft configuration.

26.1.3 Penetration Descents. The penetration descent chart (Figure 26-3) shows distance, time, fuel consumed, and rate of descent for penetrations with speedbrake open. To obtain performance as presented on the chart, establish revolutions per minute, position speedbrake for the descent, and maintain the indicated airspeed.

26.1.4 Zero Thrust Descents. The descent chart (Figure 26-4) shows time, distance, and rate of descent for an emergency descent.

USE ON CT-39E/G AIRCRAFT
EQUIPPED WITH
JT12A-8 ENGINES



NOTES:

——— NORMAL DESCENT AT 0.65M OR 300 KNOTS IAS. SPEED BRAKE CLOSED
ABOVE 22,000 FT, ESTABLISH AIRPLANE RATE OF DESCENT AT 1100 FT/MIN.
BELOW 22,000 FT, ESTABLISH AIRPLANE RATE OF DESCENT AT 5500 FT/MIN WITH IDLE THRUST.
MAXIMUM CABIN RATE OF DESCENT IS 500 FT/MIN

— — — EMERGENCY DESCENT AT 0.765M OR 325 KNOTS IAS SPEED BRAKE OPEN.

— — IDLE THRUST DESCENT AT 200 KNOTS IAS.
ESTABLISH AIRPLANE RATE OF DESCENT AT 2000 FT/MIN.

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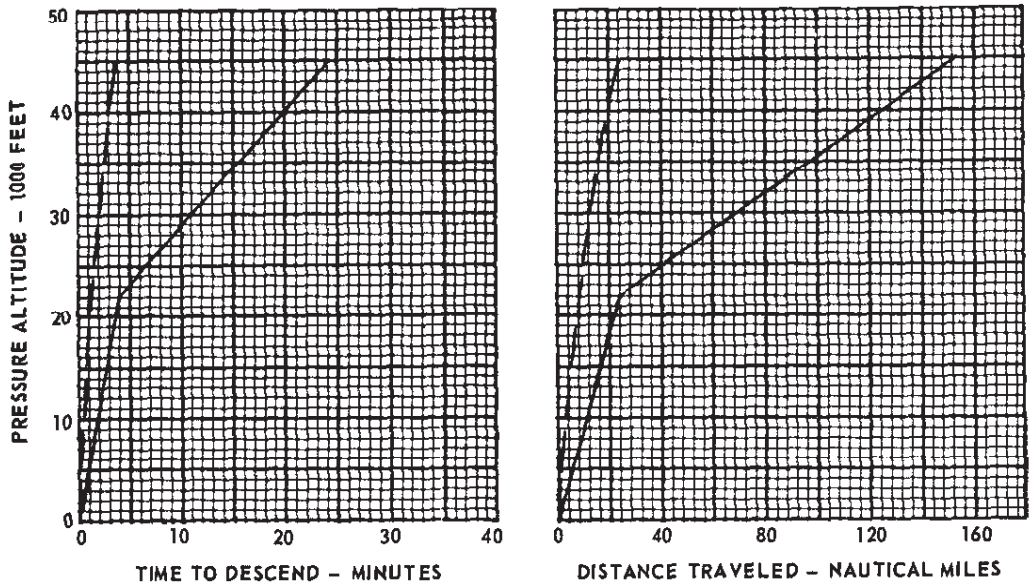
Figure 26-1. Descent Data-Standard ISA Conditions

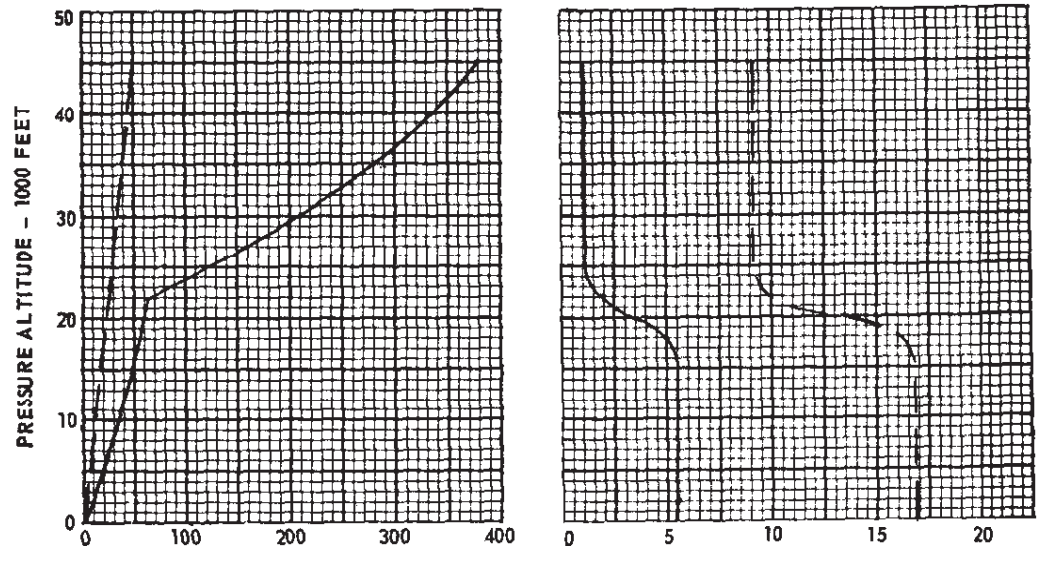
DATA AS OF: 25 MARCH 1967
BASED ON: NAA FLIGHT TEST AND ESTIMATED FUEL FLOWS

DESCENTS

NORMAL OR EMERGENCY

MODEL: CT-39E/G
ENGINES: (2) JT12A-8





LEGEND

——— NORMAL DESCENT
– – – EMERGENCY DESCENT

NOTE

NORMAL DESCENT – .65 Mach or 300 knots IAS, speed brake closed. Above 22,000 feet establish aircraft rate of descent at 1100 ft/min. Below 22,000 feet establish aircraft rate of descent at 5500 ft/min with idle thrust.

EMERGENCY DESCENT – Use idle thrust, speed brake open. Descent at .77 indicated Mach number or 325 knots IAS, whichever is less.

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Figure 26-2. Descent — Normal or Emergency

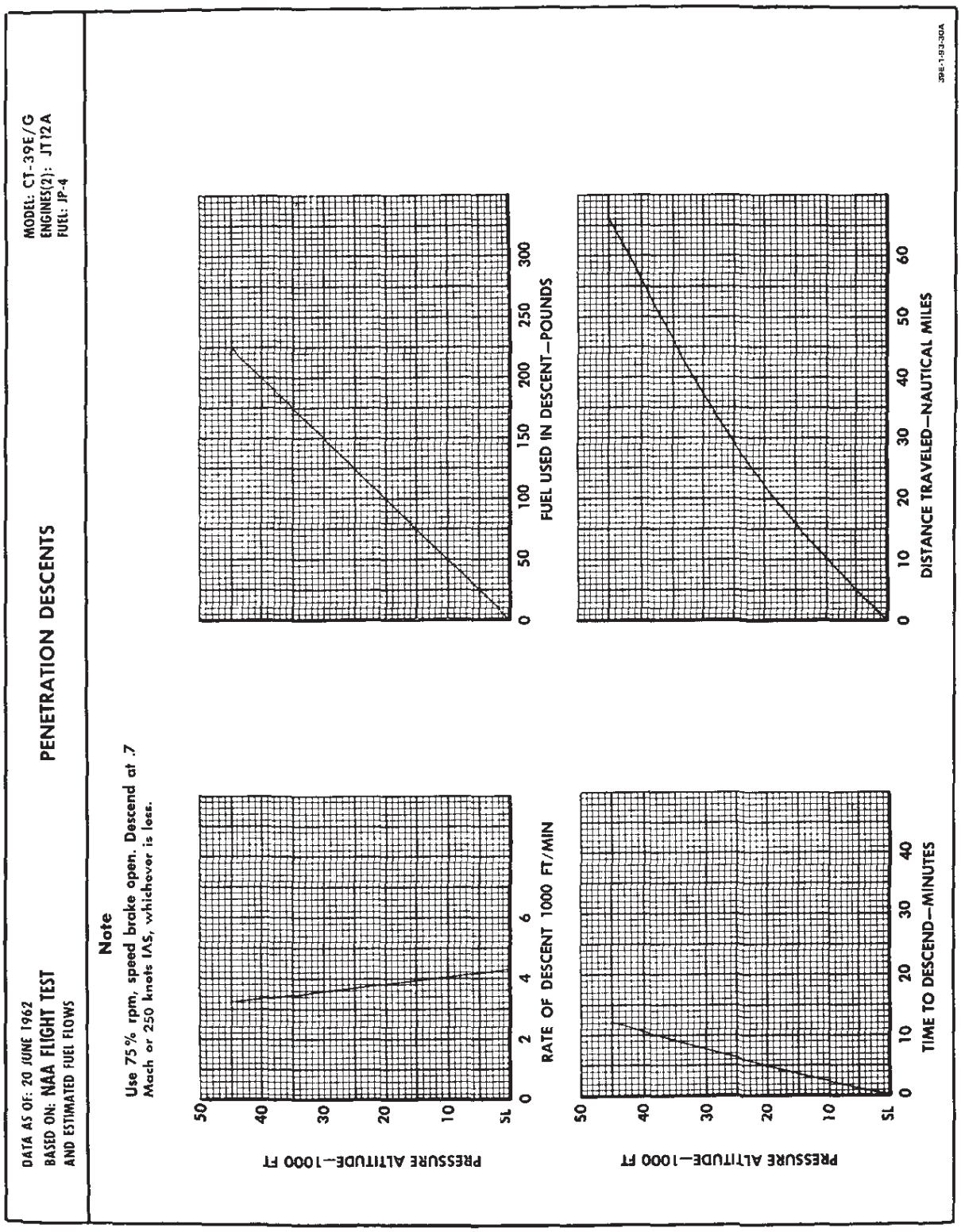


Figure 26-3. Penetration Descents

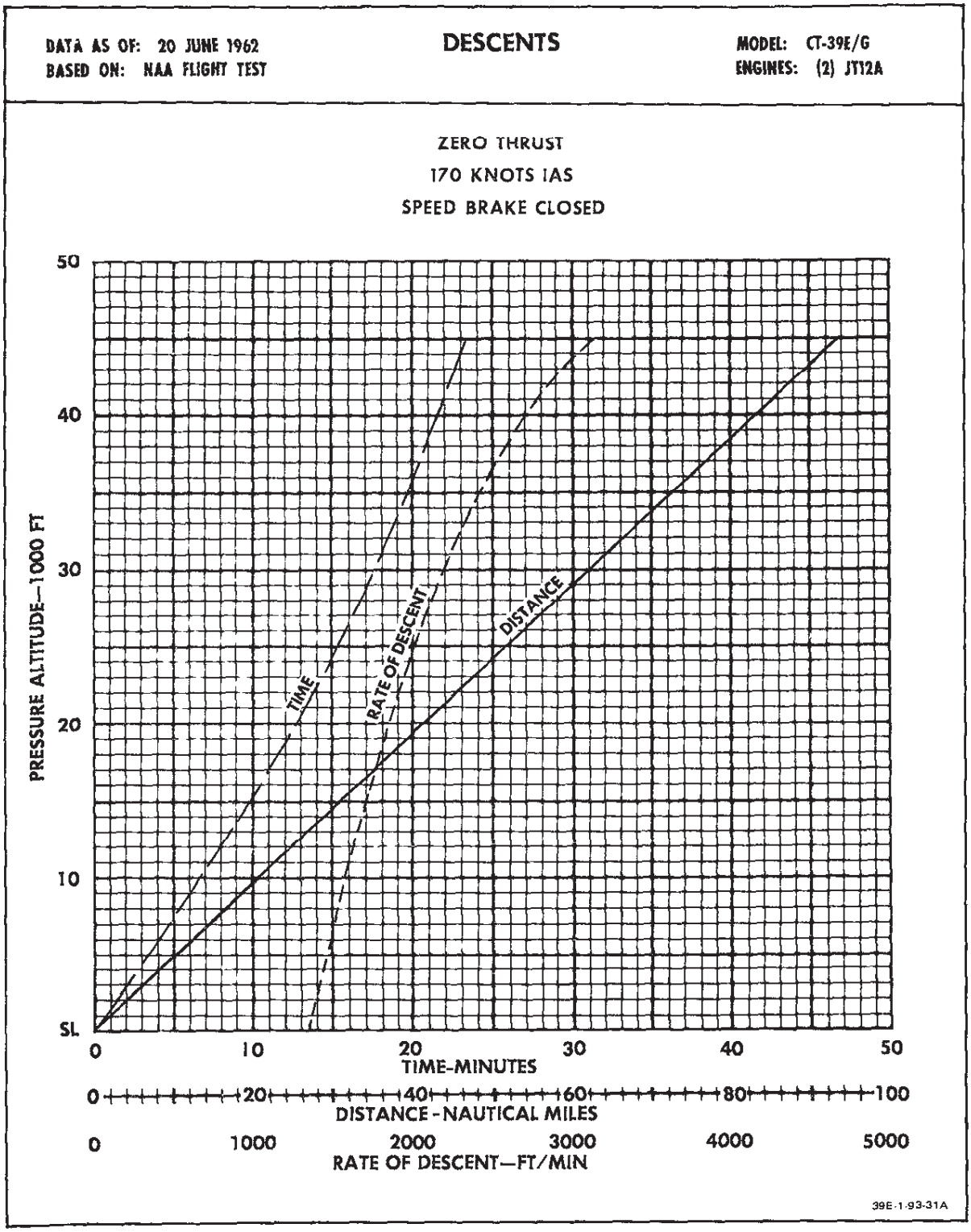


Figure 26-4. Descents — Zero Thrust

Mach One Manuals

CHAPTER 27

Landing

27.1 FLIGHT TAKEOFF THRUST ENGINE PRESSURE RATIO SETTING

The flight takeoff thrust EPR setting chart shows the EPR required for takeoff thrust operation under conditions such as a waveoff. (See Figures 27-1 and 27-2.) Flight takeoff thrust settings are given for any indicated free air temperature and pressure altitude up to 10,000 feet. Flight takeoff thrust is not used above 10,000 feet.

27.2 LANDING DISTANCES — OVER OBSTACLE

Note

See Figure 23-11 for crosswind landing data.

The landing distance chart (Figure 27-3) shows the distance required to stop for a recommended landing. The distance is shown in ground-roll distance plus air distance required when landing over a 50-foot obstacle. This chart is based on full flaps during the approach and touchdown. A note on the chart shows the percentage of decrease in distance for landings with the speedbrake extended. Temperature, altitude, runway condition, and gross weight compensations are included. The final approach speed for various weights is shown with a note to show approximate touchdown speeds. A sample problem demonstrates the use of the chart.

27.2.1 Example

A is air temperature (75 °F).

B is pressure altitude (2,000 feet).

C is gross weight (13,000 pounds).

D is distance from 50-foot obstacle to stop (2,450 feet; zero wind and zero slope).

E is slope baseline.

F is uphill slope (1° up).

G is wind baseline.

H is headwind (10 knots).

J is distance from 50-foot obstacle to stop (2,250 feet; for wind and runway slope).

K is runway condition (RCR 15).

L is total distance from 50-foot obstacle to stop (3,000 feet; for wind, slope, and runway condition)

27.3 LANDING DISTANCES — GROUND ROLL

Landing ground-roll distances are shown in Figure 27-4 for the recommended landing configuration of flaps down and speedbrake closed. Temperature, altitude, and runway condition compensations are included.

27.3.1 Example

A is runway air temperature (75 °F).

B is airport pressure altitude (2,000 feet).

C is landing weight (13,000 pounds).

D is slope baseline.

E is runway slope (1° up).

F is wind baseline.

G is headwind (10 knots).

H is wet runway condition.

J is landing distance ground roll (2,700 feet)

27.4 REVERSE THRUST EPR SETTING

When landing at any airport with a pressure altitude above 2,500 feet, the thrust reverser levers should only be actuated to the precomputed EPR setting as given

in Figures 27-5 through 27-8. When landing at airports with a pressure altitude below 2,500 feet, the adjustable thrust reverser lever stop may be set and only that reverse thrust will be available after landing. (Refer to paragraph 2.10.2, Thrust Reverser Lever Stop Assemblies.)

The reverser thrust EPR charts are used to obtain the maximum allowable reverse thrust during landing. To precompute a reverse thrust EPR setting, use the reported field ambient temperature and pressure altitude. Charts are presented for one- and two-engine operation and engine inlet anti-icing on and off. All four charts are based on cabin pressurization on.

Enter the chart with the reported pressure altitude and extend vertically to reported ambient temperature (°F) line; extend to the left side of the chart for the maximum allowable reverse thrust EPR setting.

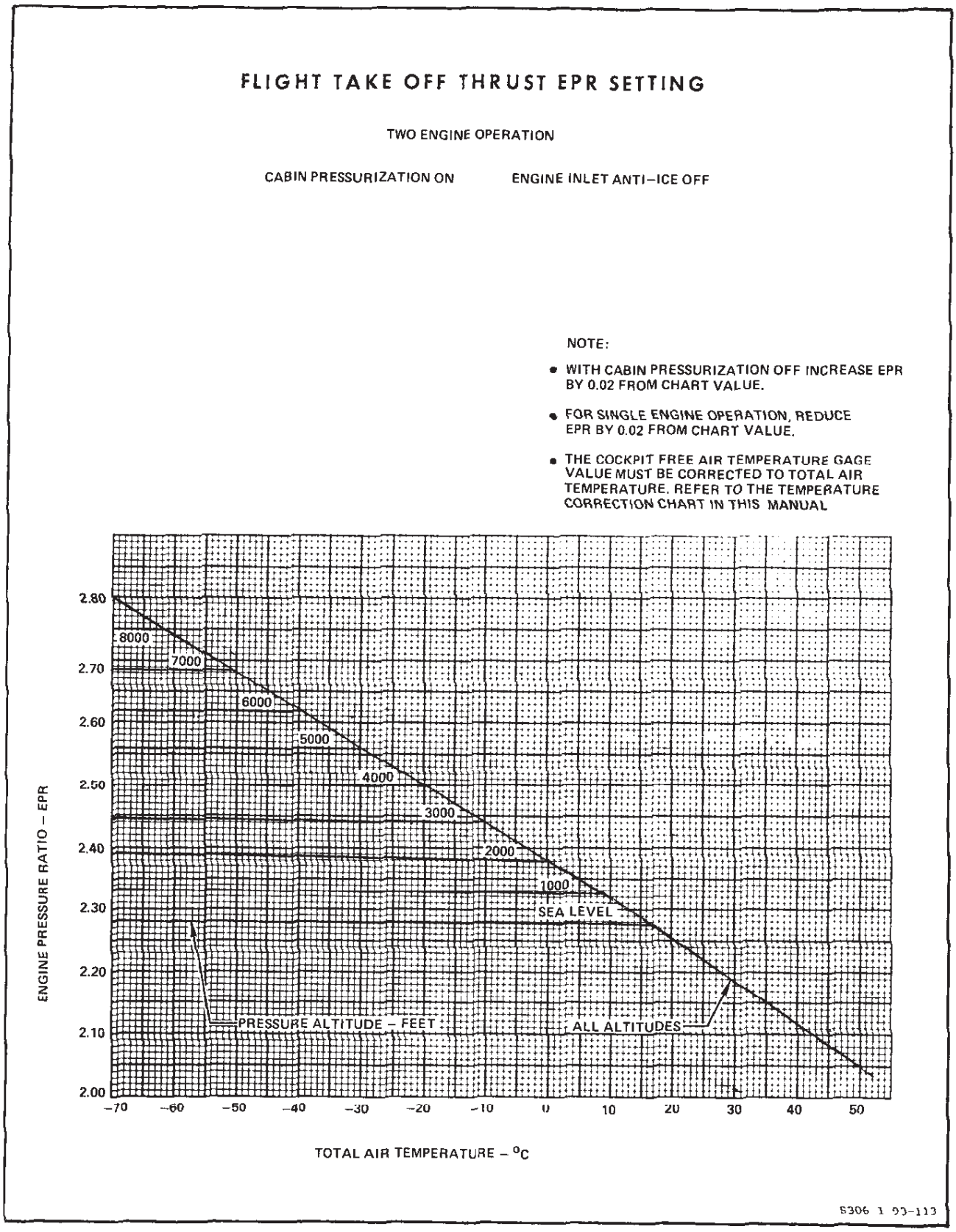


Figure 27-1. Flight Takeoff Thrust EPR Setting (Cabin Pressurization ON — Anti-Ice OFF)

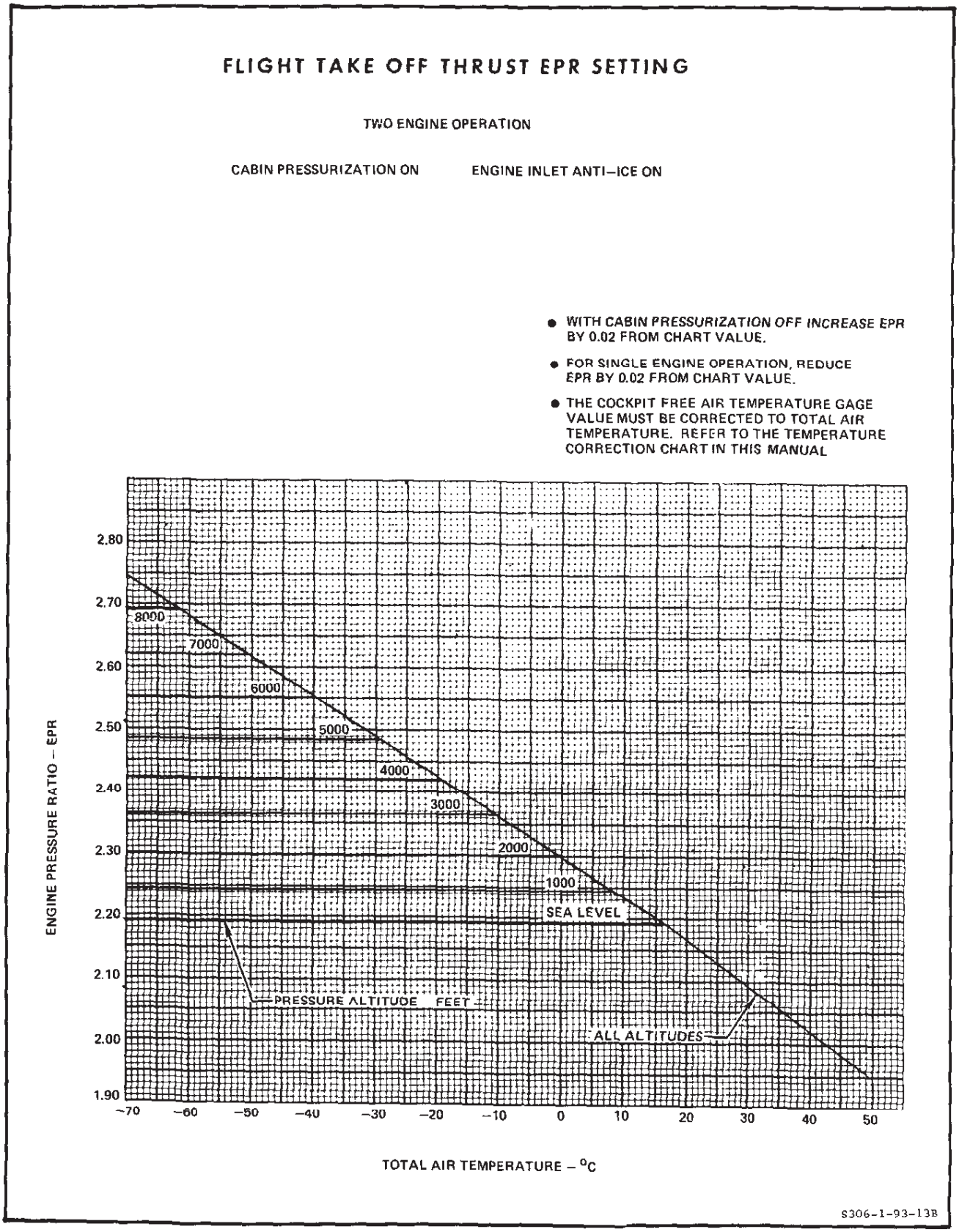


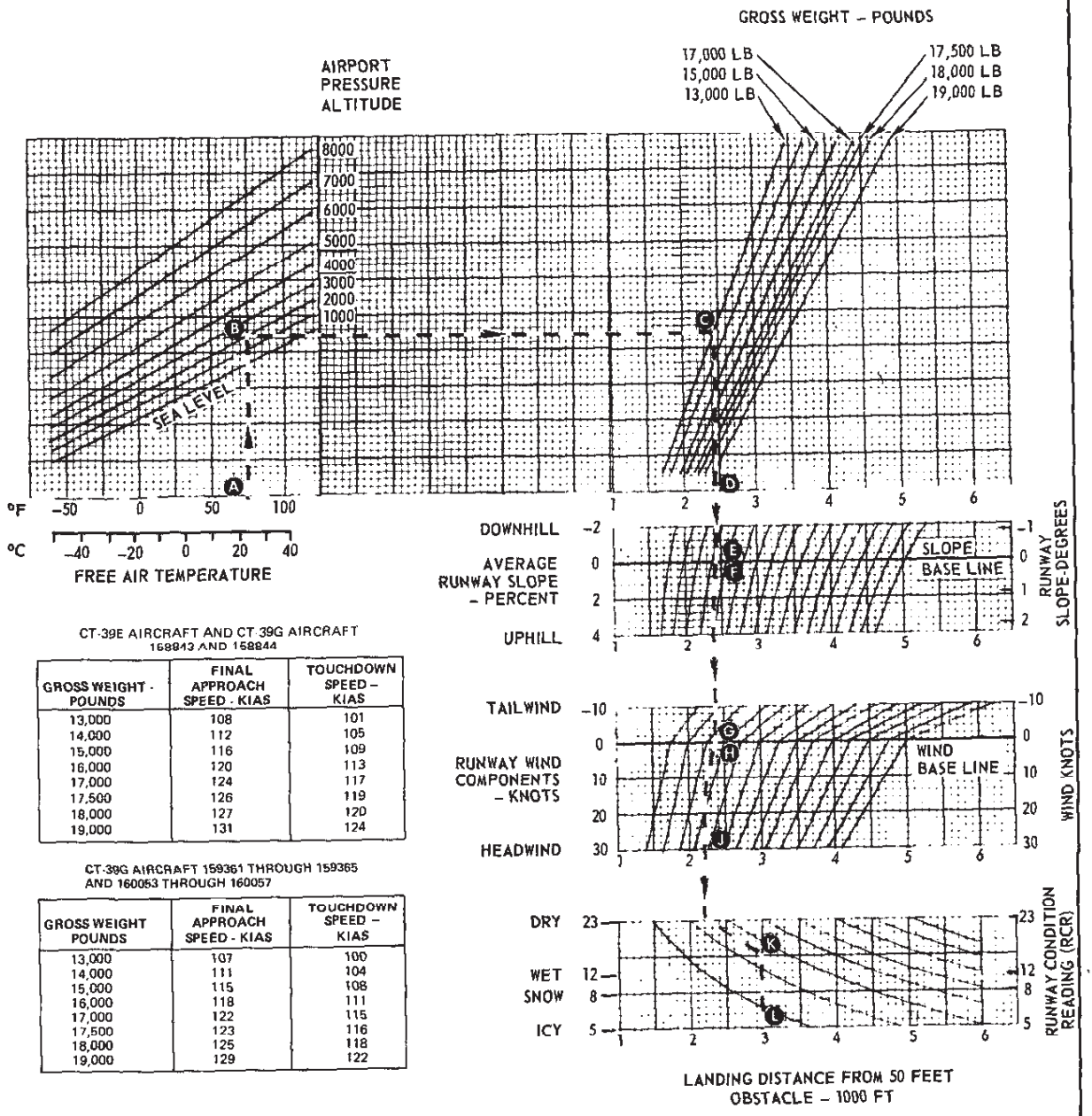
Figure 27-2. Flight Takeoff Thrust EPR Setting (Cabin Pressurization ON — Anti-Ice ON)

DATA AS OF: 3 MAY 1968
BASED ON: NAA FLIGHT TEST

LANDING DISTANCE - OVER OBSTACLE

FLAPS FULL DOWN - OPTIMUM BRAKING - SPEED BRAKE CLOSED ANTI-SKID ON

MODEL: CT-39E/G
ENGINES(2): JT12A
FUEL: JP-4



CT-39E AIRCRAFT AND CT-39G AIRCRAFT 158843 AND 158844

GROSS WEIGHT - POUNDS	FINAL APPROACH SPEED - KIAS	TOUCHDOWN SPEED – KIAS
13,000	108	101
14,000	112	105
15,000	116	109
16,000	120	113
17,000	124	117
17,500	126	119
18,000	127	120
19,000	131	124

CT-39G AIRCRAFT 159361 THROUGH 159365 AND 160053 THROUGH 160057

GROSS WEIGHT POUNDS	FINAL APPROACH SPEED - KIAS	TOUCHDOWN SPEED – KIAS
13,000	107	100
14,000	111	104
15,000	115	108
16,000	118	111
17,000	122	115
17,500	123	116
18,000	125	118
19,000	129	122

NOTE

- SPEED BRAKE OPEN WILL DECREASE DISTANCE BY 6 PERCENT
- FOR ANTI-SKID INOPERATIVE INCREASE CHART DISTANCES 30%.
- NO ALLOWANCE INCLUDED FOR THRUST REVERSER OPERATION
- WITH MAIN LANDING GEAR TIRE PRESSURE OF 214 (+5-0) PSI, ADD 200 FEET TO THE COMPUTED DISTANCE

N3/77
39E-1-91-32B

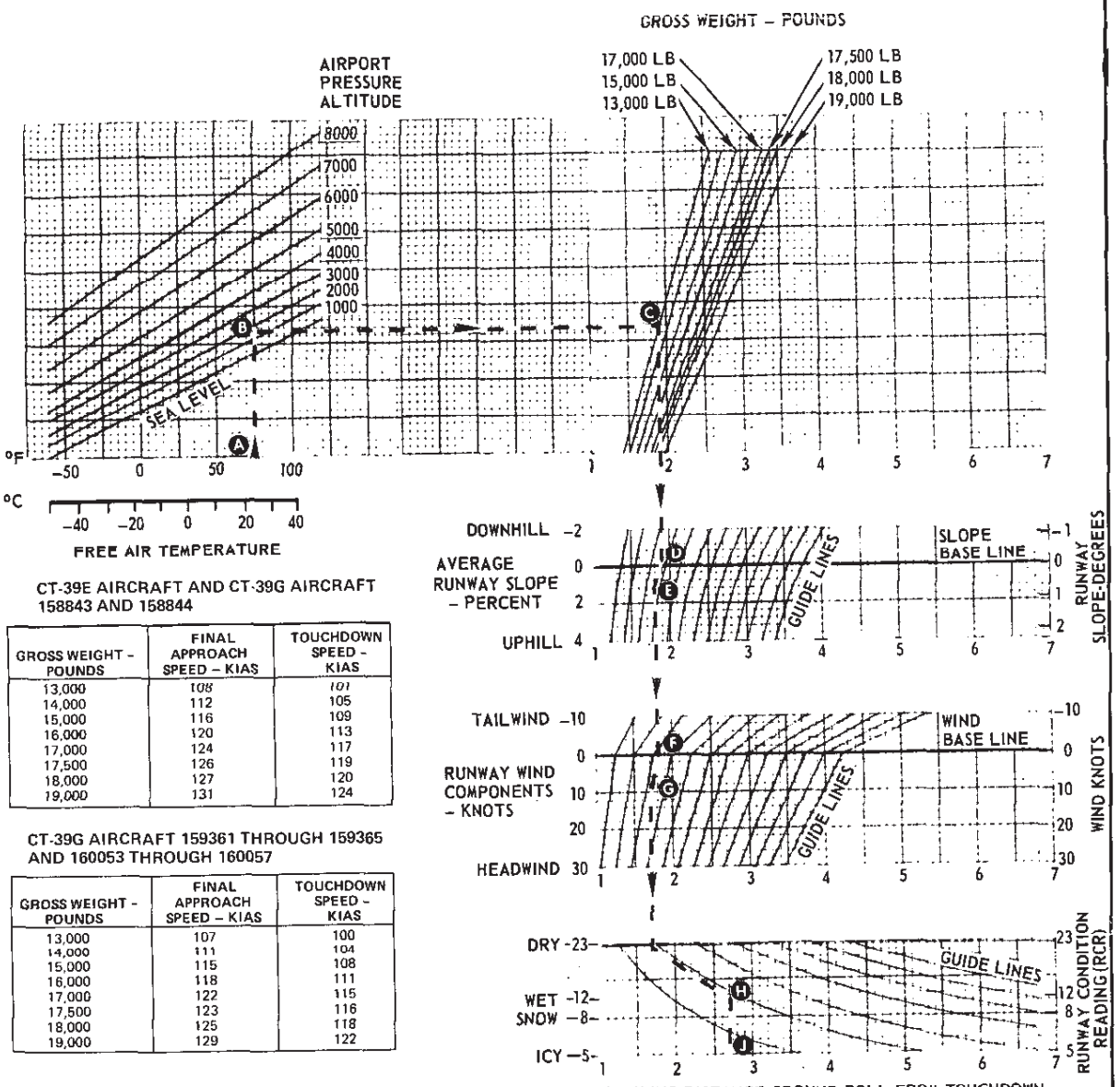
Figure 27-3. Landing Distance — Over Obstacle

DATA AS OF: 3 MAY 1968
BASED ON: NAA FLIGHT TEST

LANDING DISTANCE - GROUND ROLL

FLAPS DOWN-OPTIMUM BRAKING -
SPEED BRAKE CLOSED
ANTI-SKID ON

MODEL: CT-39E/G
ENGINES: (2) JT12A
FUEL: JP-4



CT-39E AIRCRAFT AND CT-39G AIRCRAFT 158843 AND 158844

GROSS WEIGHT - POUNDS	FINAL APPROACH SPEED – KIAS	TOUCHDOWN SPEED - KIAS
13,000	108	101
14,000	112	105
15,000	116	109
16,000	120	113
17,000	124	117
17,500	126	119
18,000	127	120
19,000	131	124

CT-39G AIRCRAFT 159361 THROUGH 159365 AND 160053 THROUGH 160057

GROSS WEIGHT - POUNDS	FINAL APPROACH SPEED – KIAS	TOUCHDOWN SPEED - KIAS
13,000	107	100
14,000	111	104
15,000	115	108
16,000	118	111
17,000	122	115
17,500	123	116
18,000	125	118
19,000	129	122

NOTE

- FOR EACH 5 KNOTS ABOVE RECOMMENDED TOUCHDOWN SPEED, INCREASE GROUND ROLL BY 10 PERCENT.
- SPEED BRAKE OPEN WILL DECREASE DISTANCE BY 6 PERCENT.
- FOR ANTI-SKID INOPERATIVE INCREASE CHART DISTANCES 26%.
- NO ALLOWANCE INCLUDED FOR THRUST REVERSER OPERATION.
- WITH MAIN LANDING GEAR TIRE PRESSURE OF 214 (+5-0) PSI, ADD 200 FEET TO THE COMPUTED DISTANCE.

N3/77
39E-1-93-33B

Figure 27-4. Landing Distance — Ground Roll

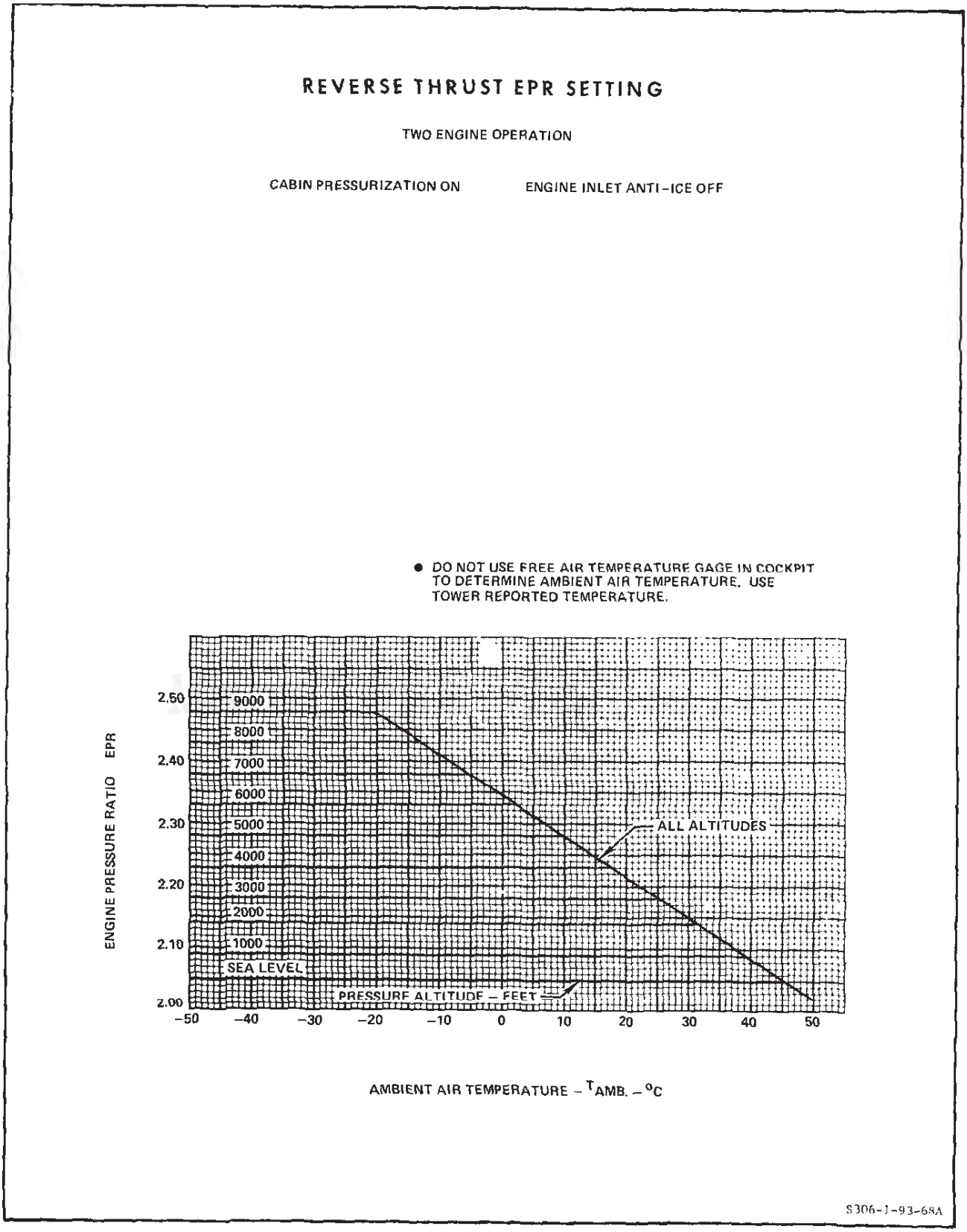


Figure 27-5. Reverse Thrust EPR Setting — Two-Engine Operation (Cabin Pressurization ON — Anti-Ice OFF)

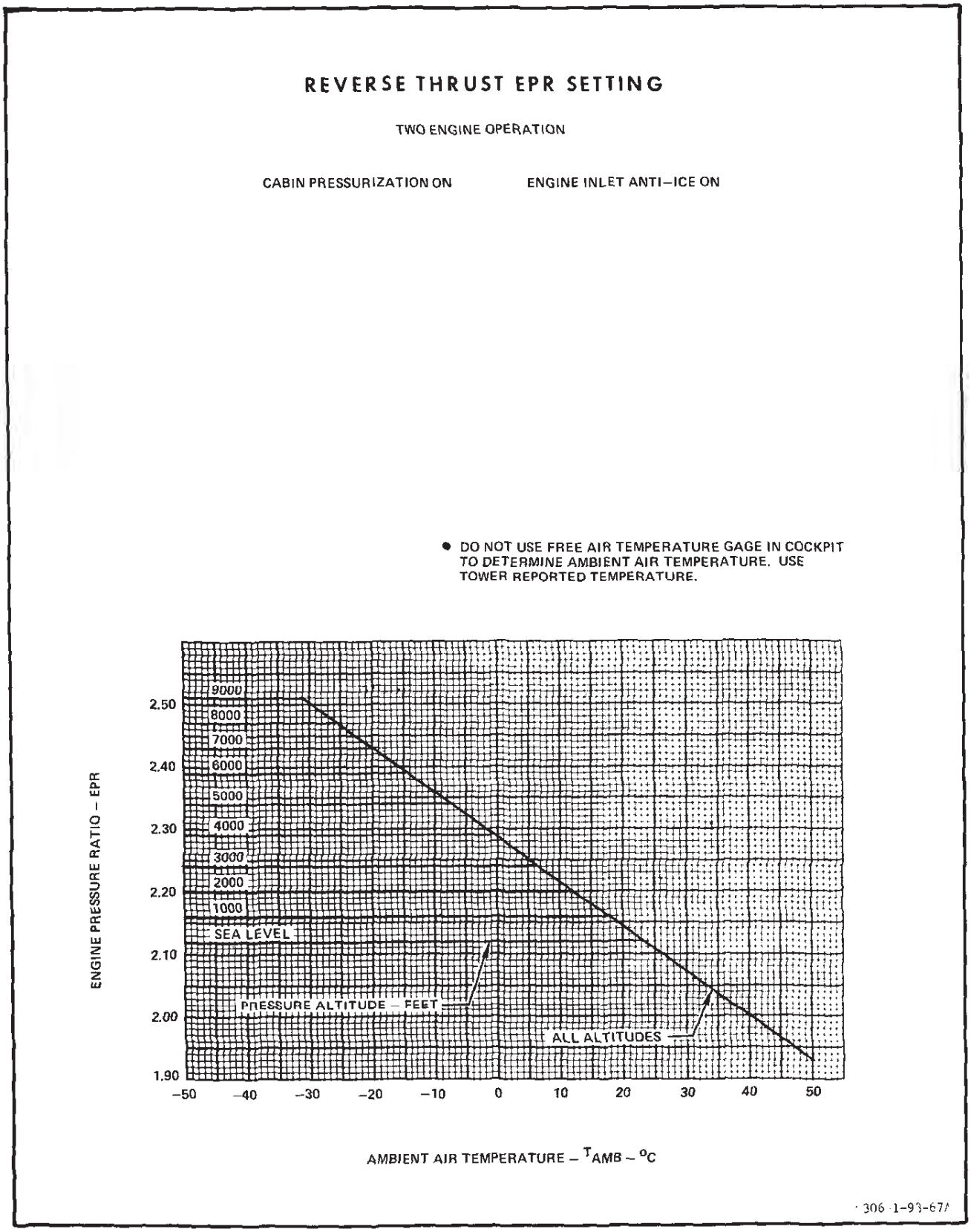


Figure 27-6. Reverse Thrust EPR Setting — Two-Engine Operation (Cabin Pressurization ON — Anti-Ice ON)

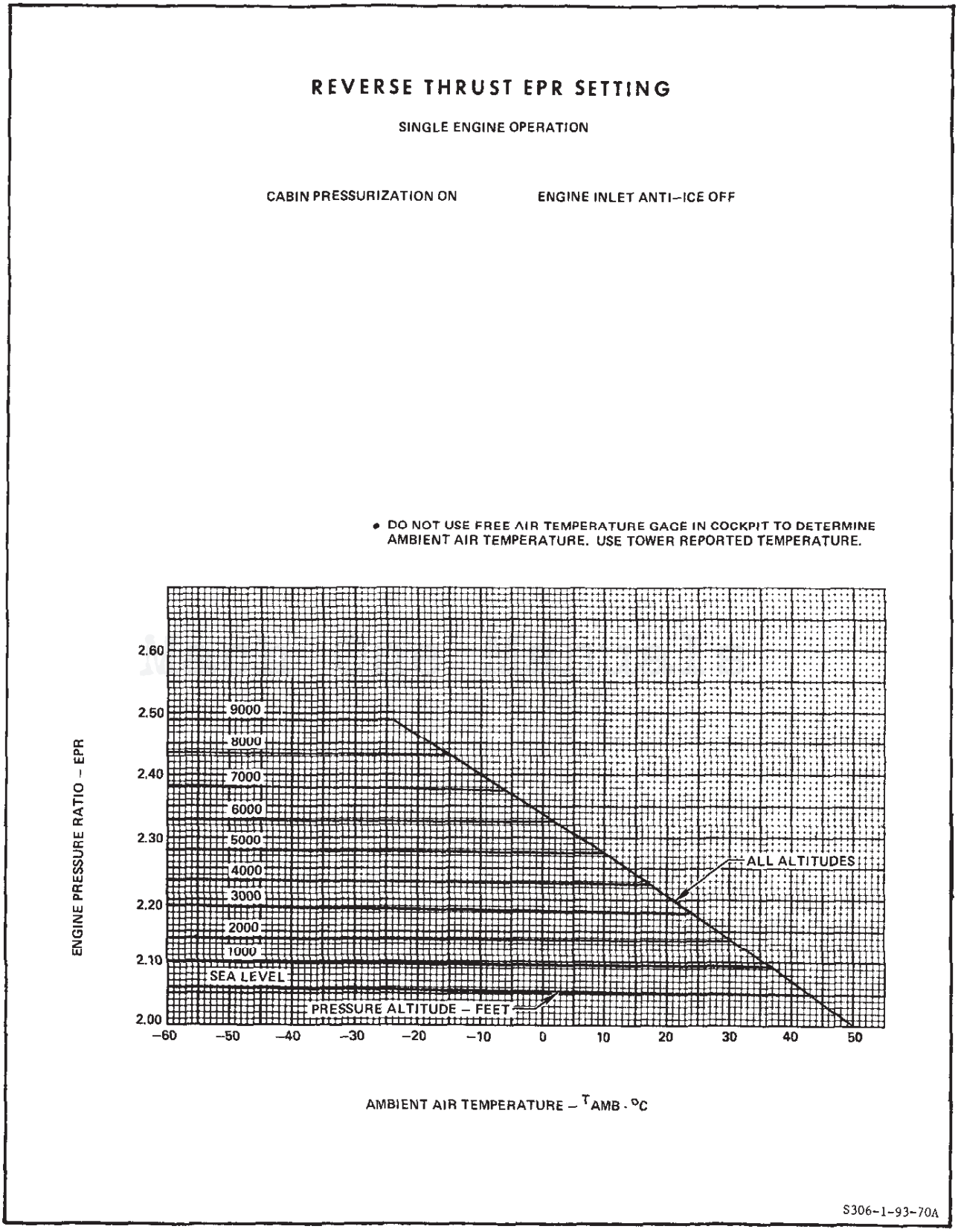


Figure 27-7. Reverse Thrust EPR Setting — One-Engine Operation (Cabin Pressurization ON — Anti-Ice OFF)

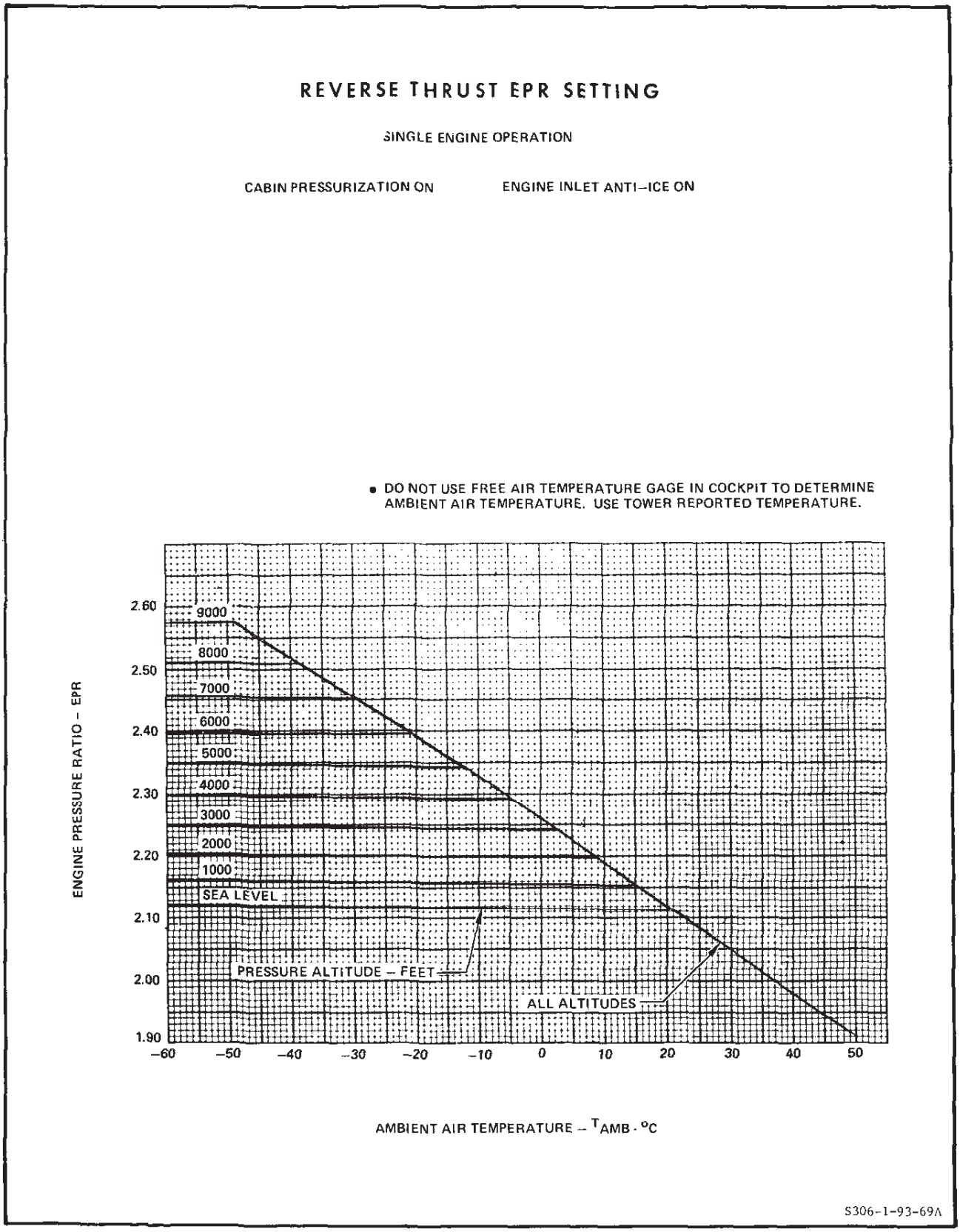


Figure 27-8. Reverse Thrust EPR Setting — One-Engine Operation (Cabin Pressurization ON — Anti-Ice ON)

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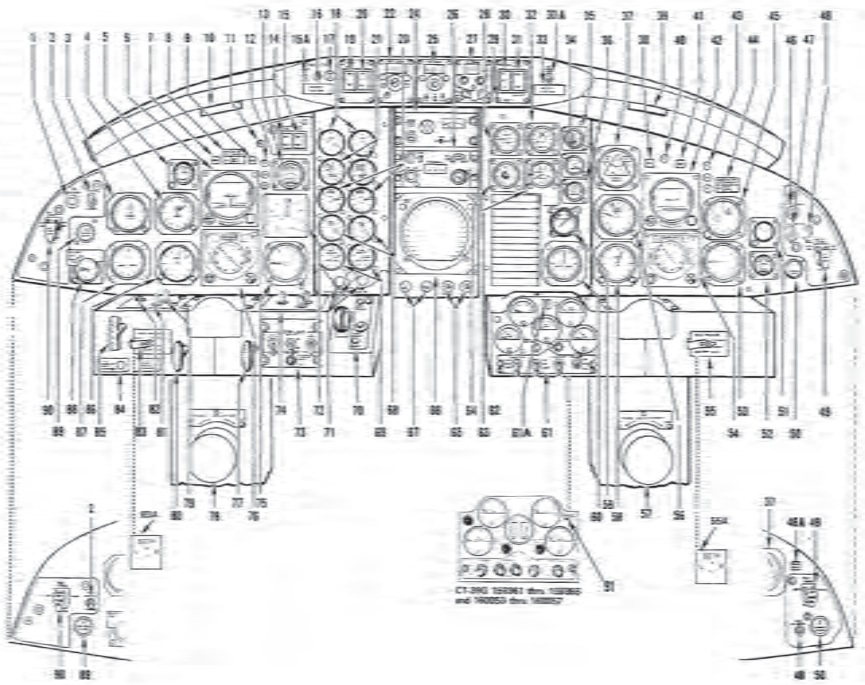
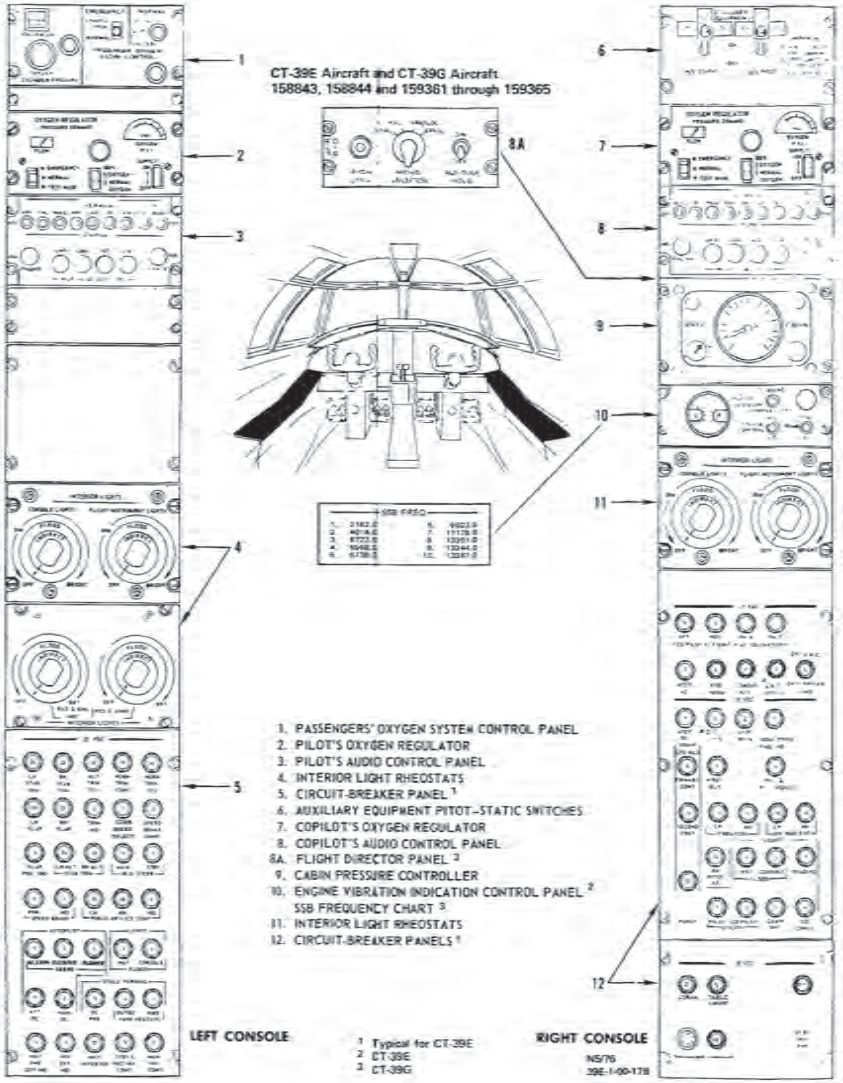
(TYPICAL)
CT-39E Aircraft and CT-39G Aircraft 158843, 160844 and 160861 through 160868



[1] CT-39E aircraft
[2] CT-39G aircraft
[3] CT-39E aircraft 157952 through 157994 (MACH/AIRSPEED INDICATOR on CT-39E aircraft 158090 through 158093 and CT-39G aircraft
[4] CT-39E aircraft 157952 through 157994 (combined with Items 4 and 56 on CT-39E aircraft 158090 through 158093 and CT-39G aircraft
[5] CT-39E aircraft and CT-39G aircraft 158843and 158844
[6] CT-39G aircraft 160861 through 160868

Cockpit — Forward View (Typical)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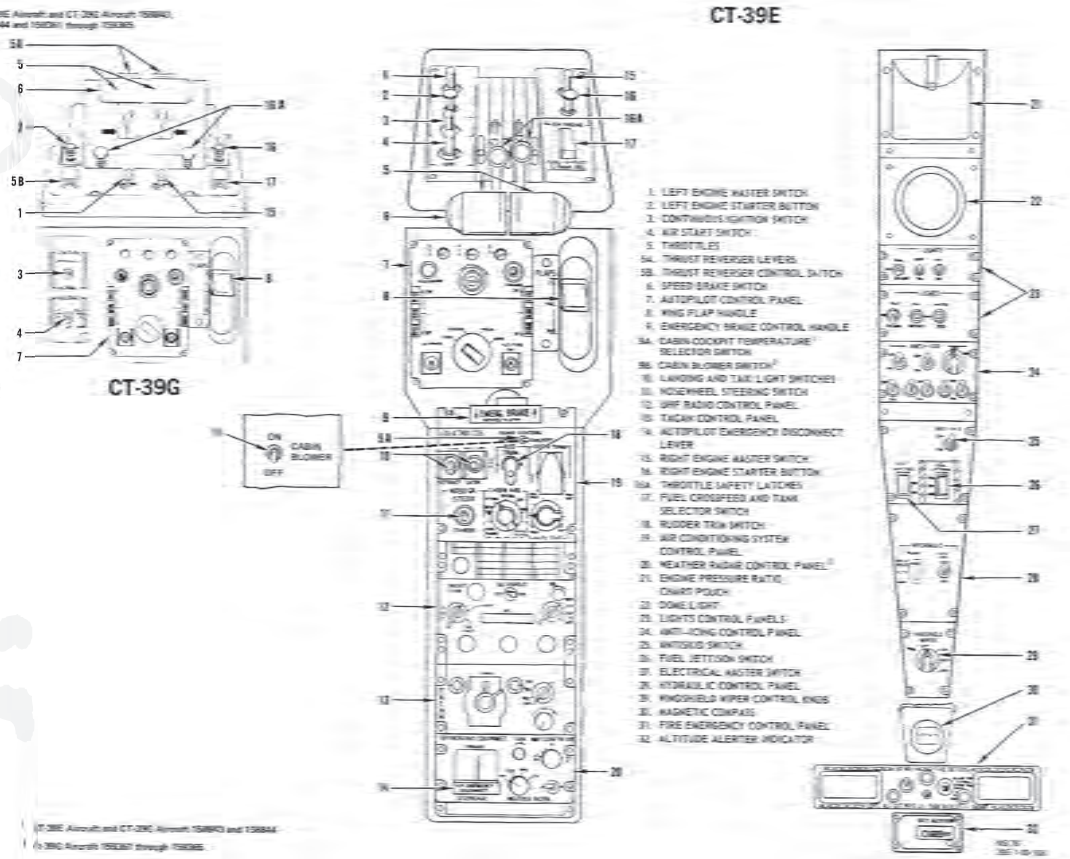
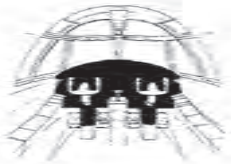
Consoles



Center Pedestal and Overhead Control Panels

CT-39G AIRCRAFT 160657
THROUGH 160667



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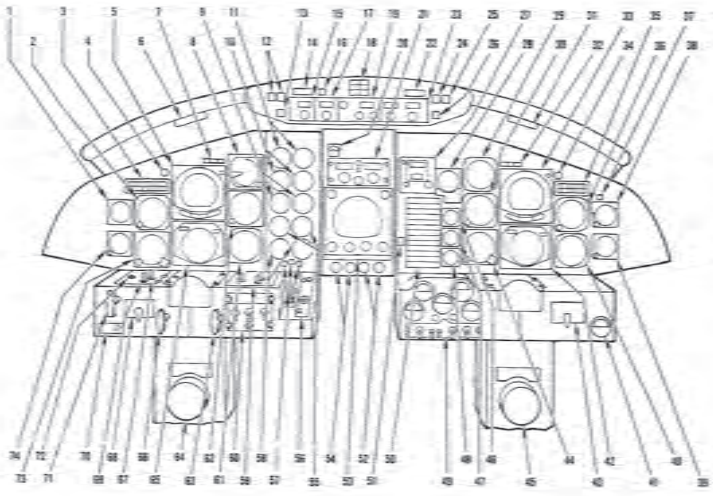
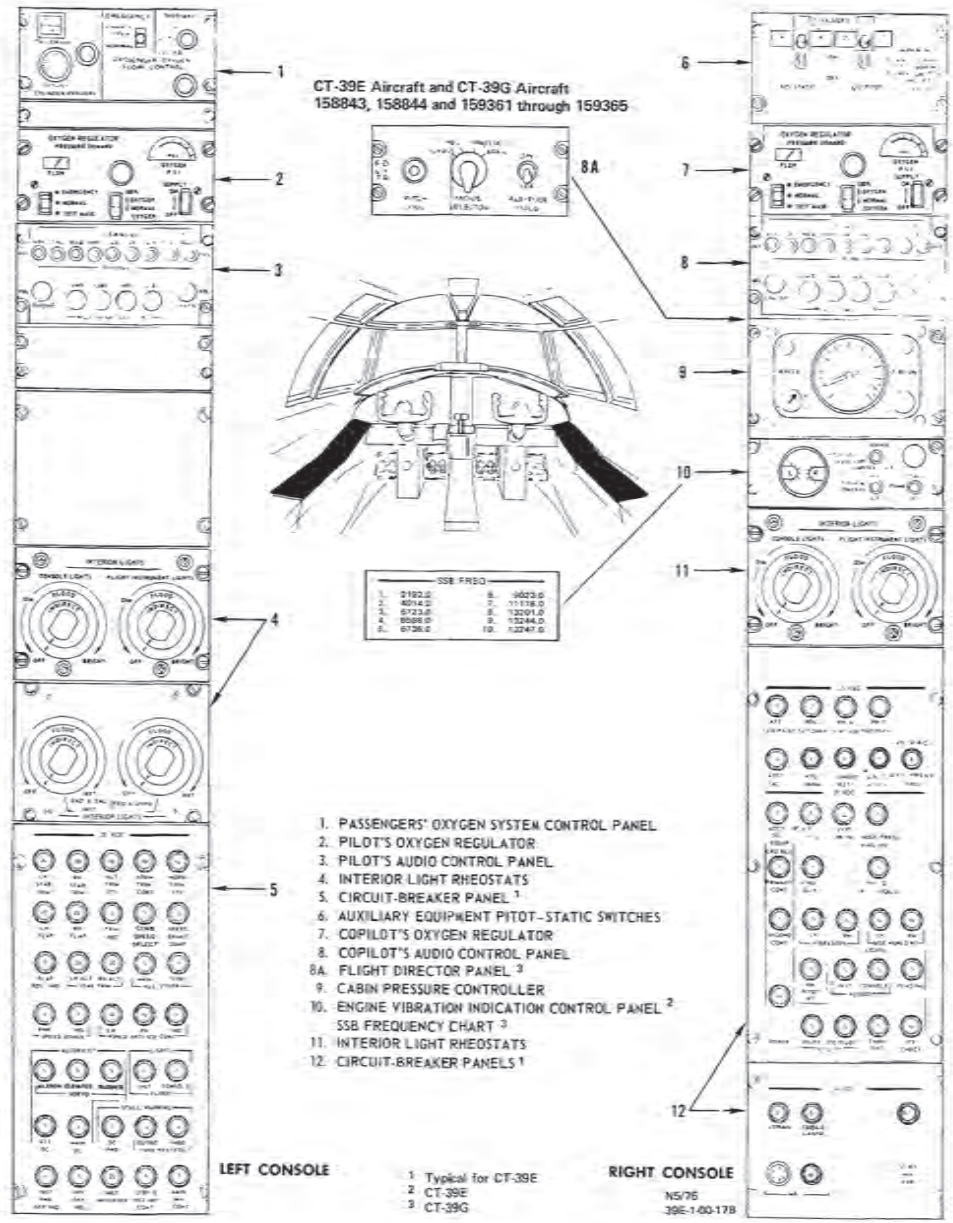


Cockpit — Forward View (CT-39G)



Side Consoles — CT-39G

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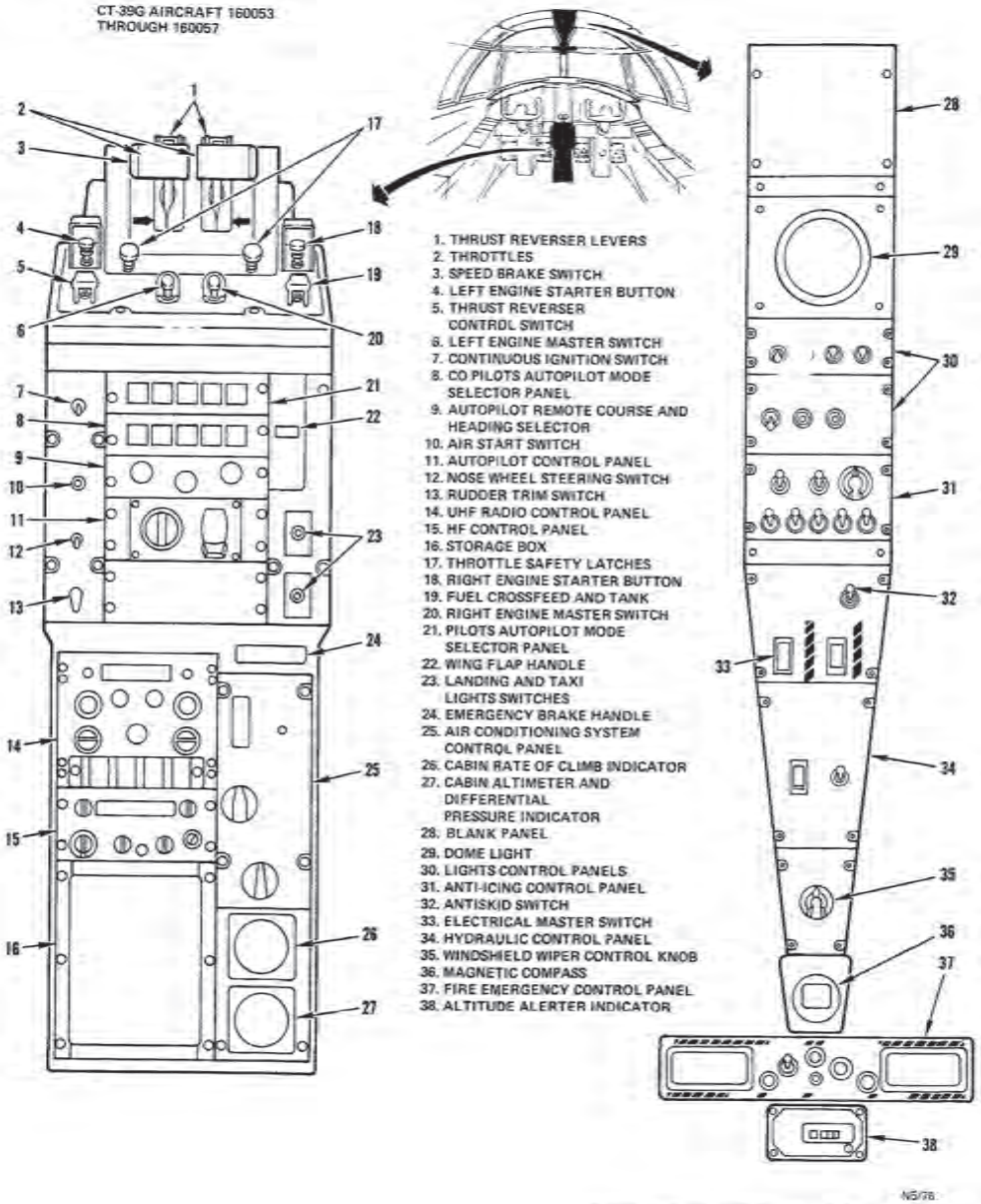
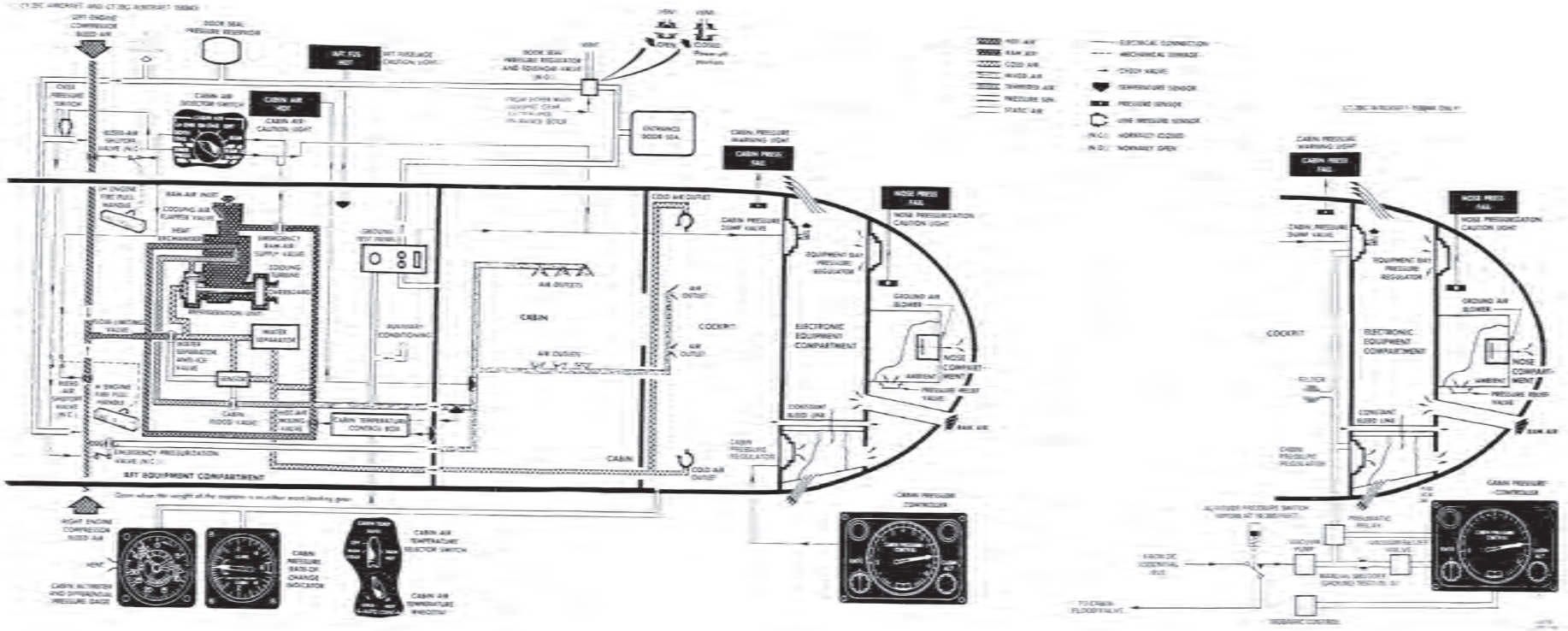
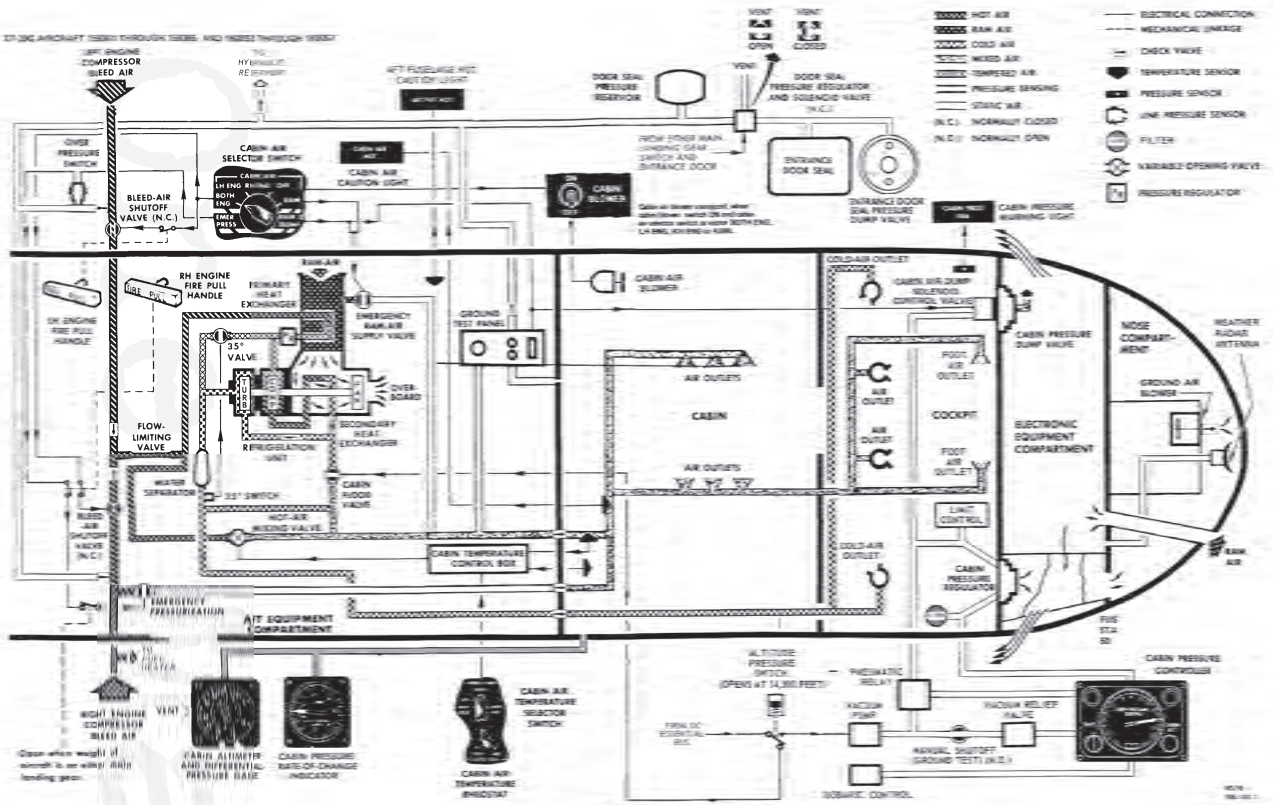


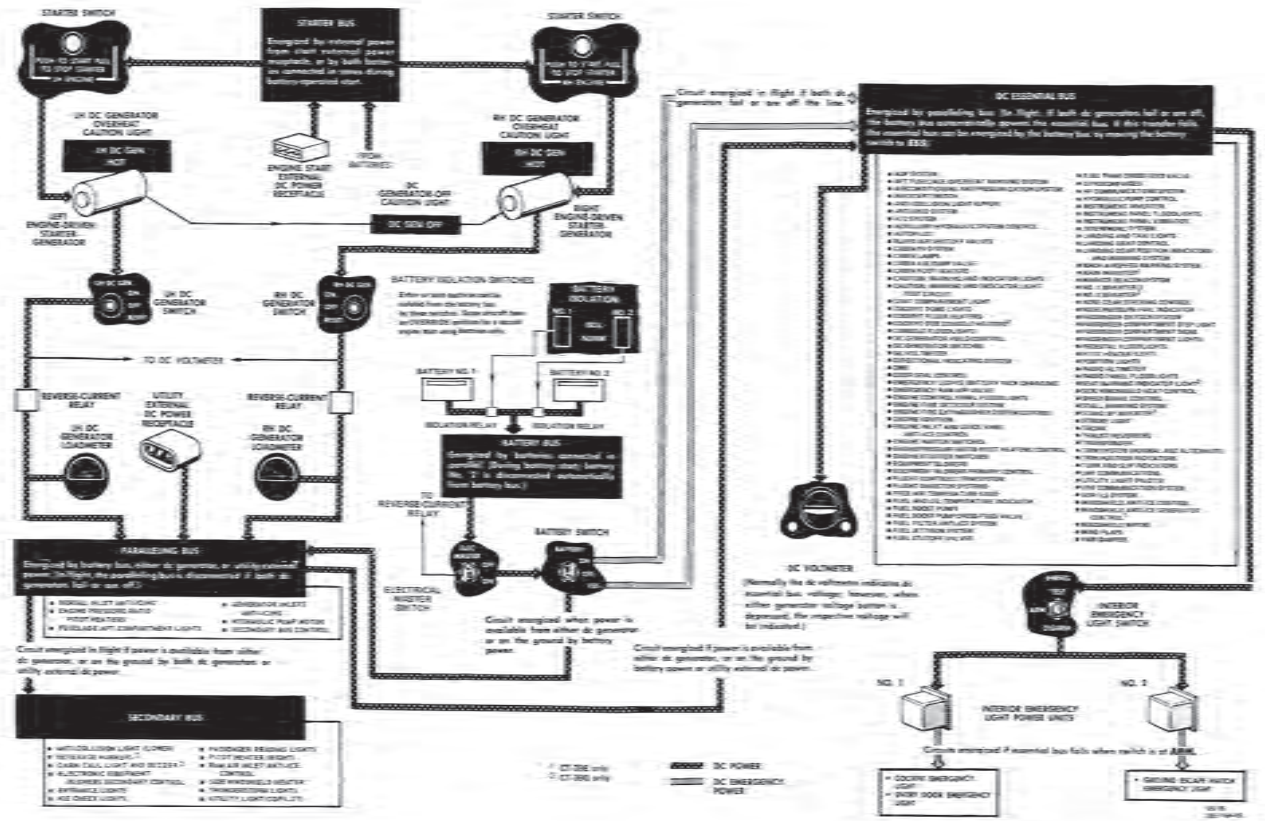
Center Pedestal and Overhead Control Panel — CT-39G



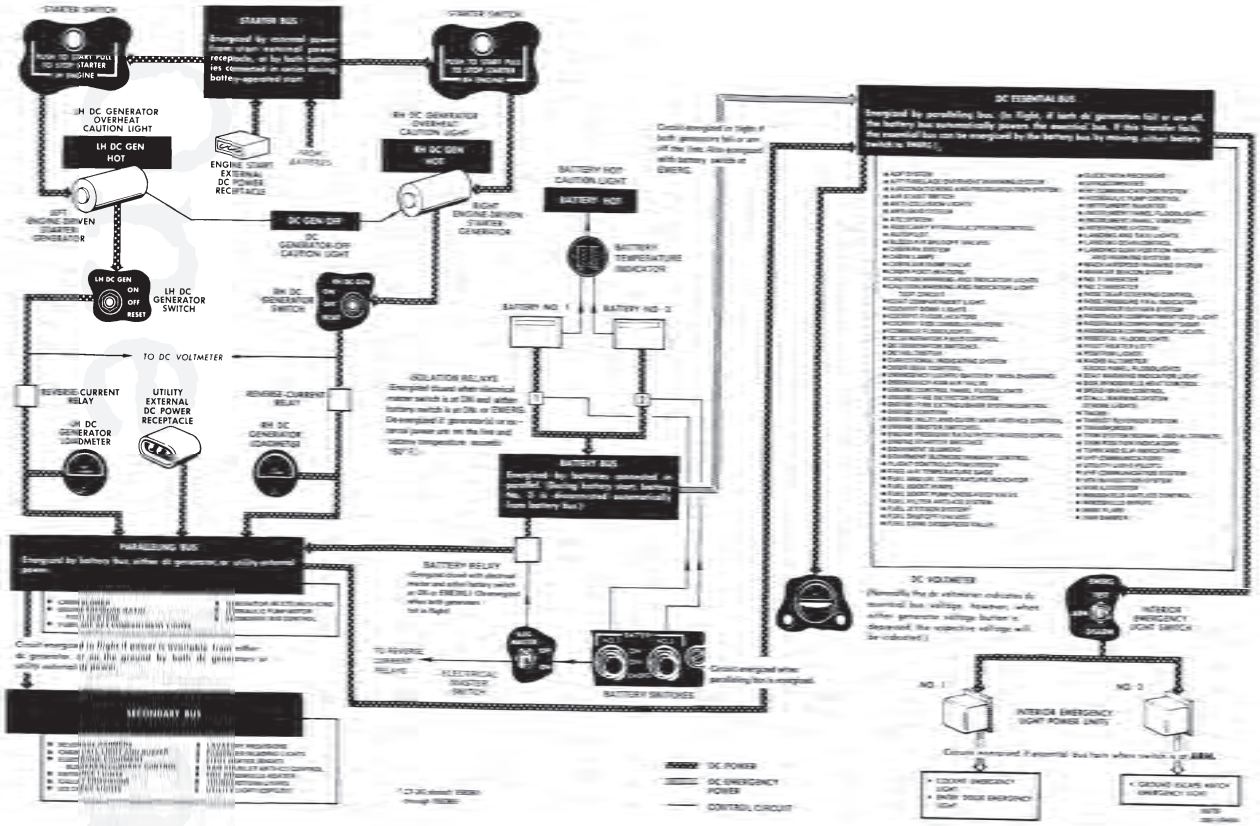
Air-Conditioning and Pressurization System
(Sheet 1 of 2)



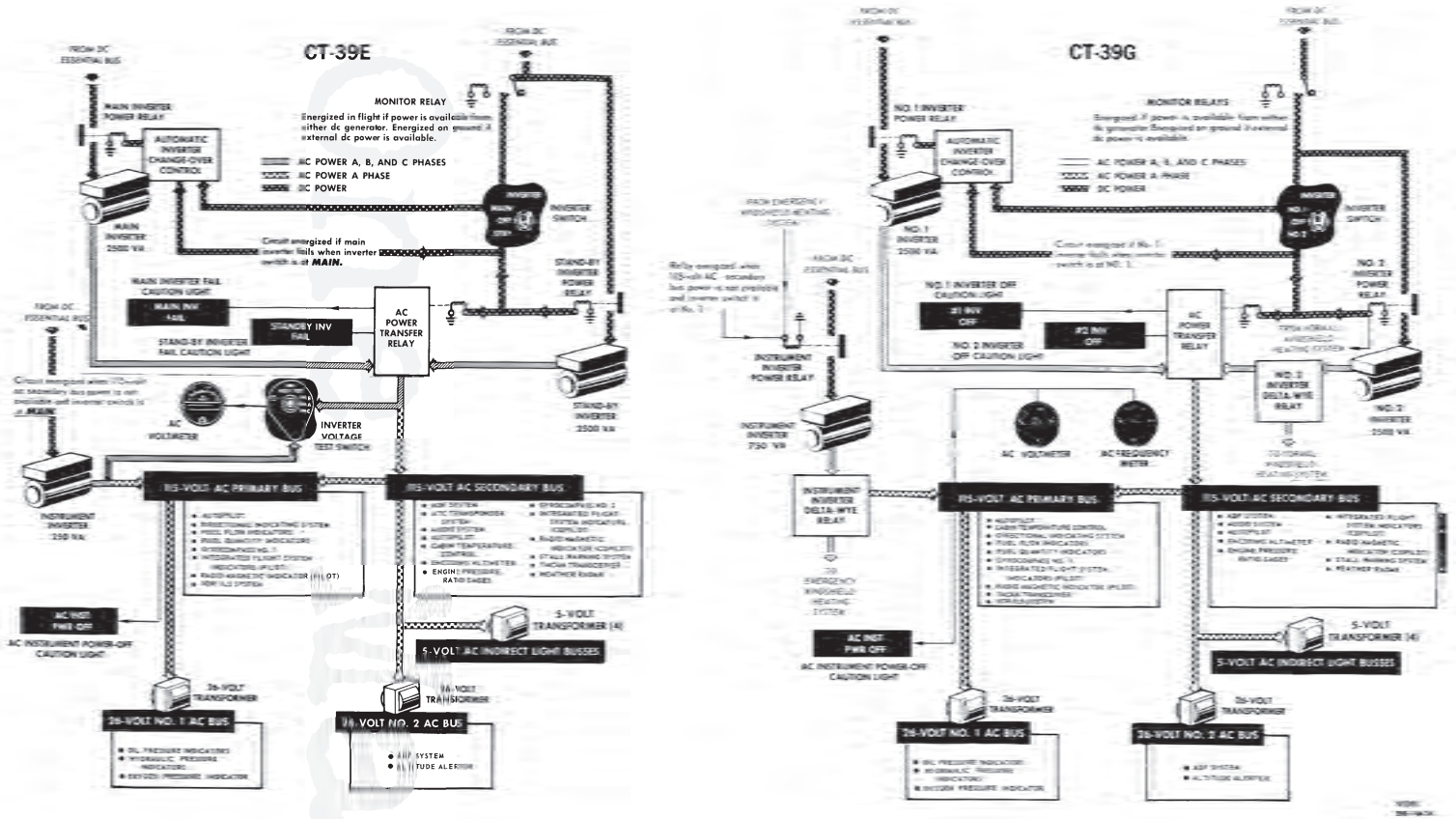
Air-Conditioning and Pressurization System (Sheet 2 of 2)



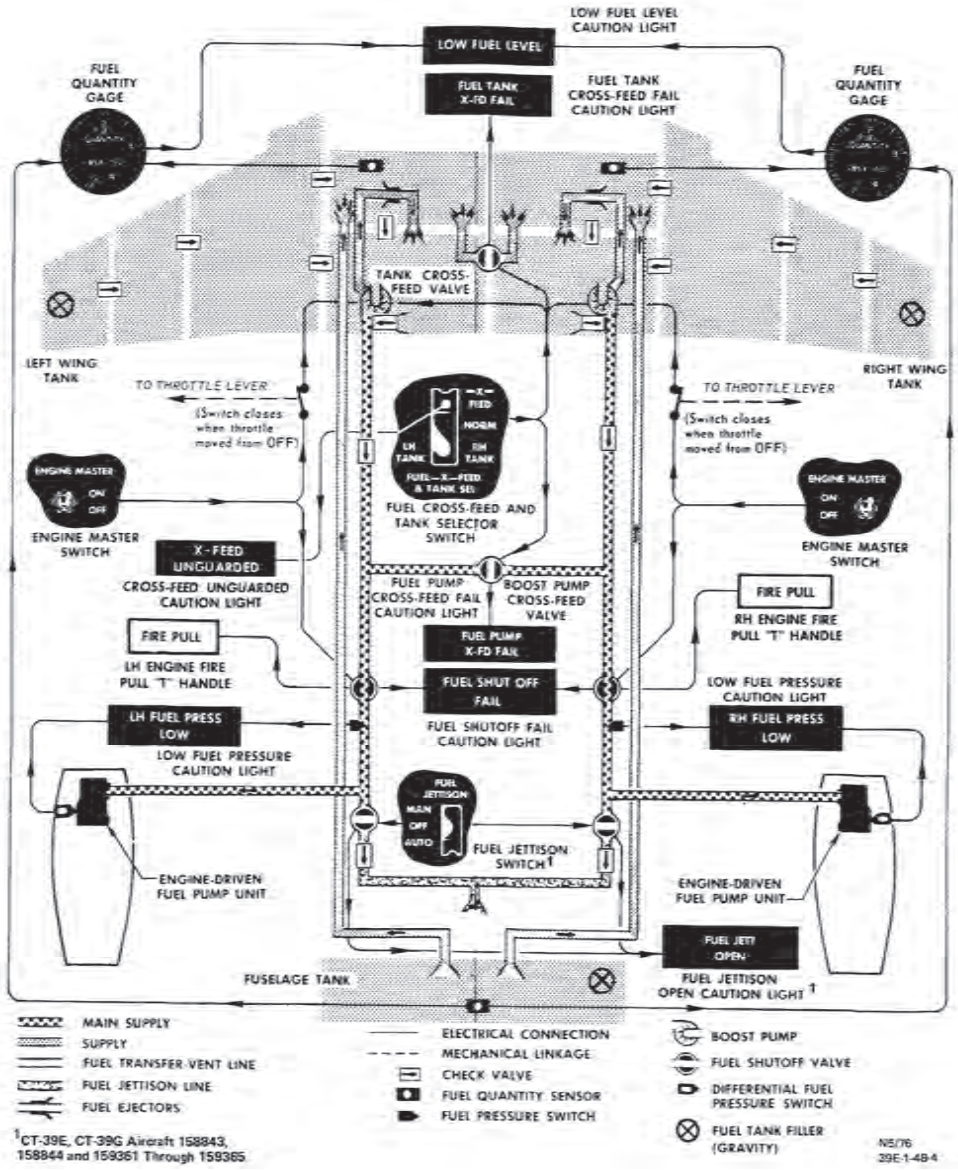
Dc Electrical Power Distribution (CT-39E Aircraft and CT-39G Aircraft 158843 and 158844)



DC Electrical Power Distribution (CT-39G Aircraft 159361 Through 159365 and 160053 and 160057)

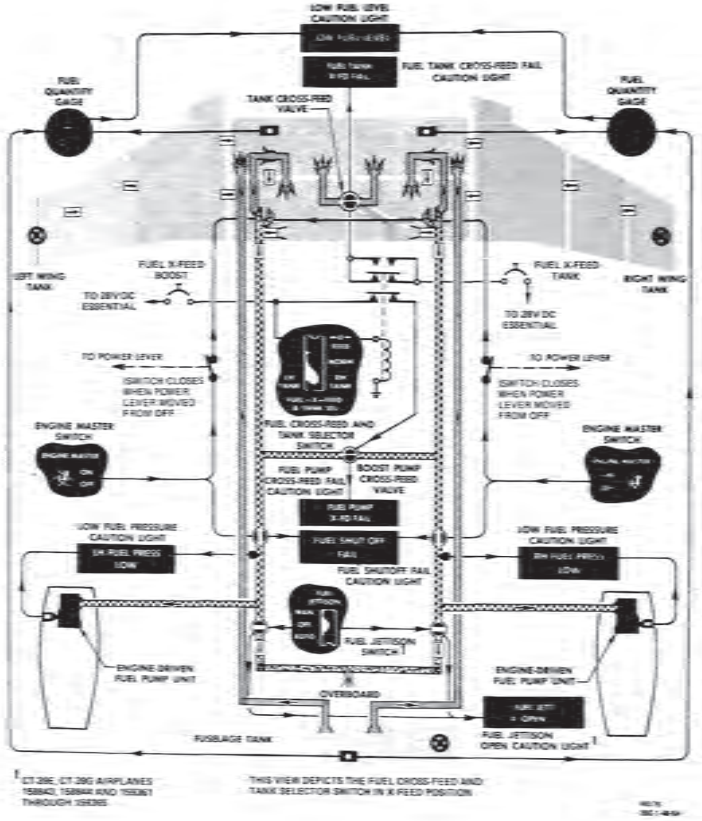


Ac Electrical Power Distribution

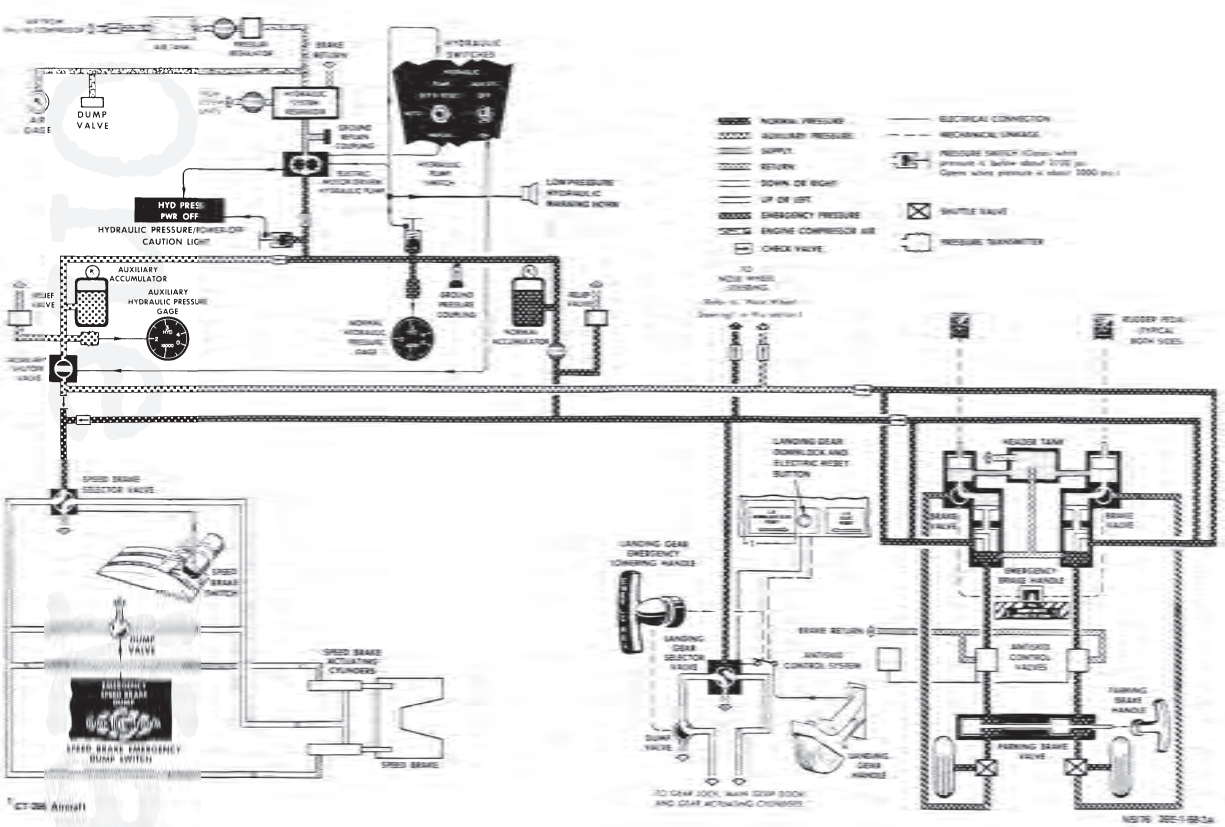


Aircraft Fuel System (Sheet 1 of 2)

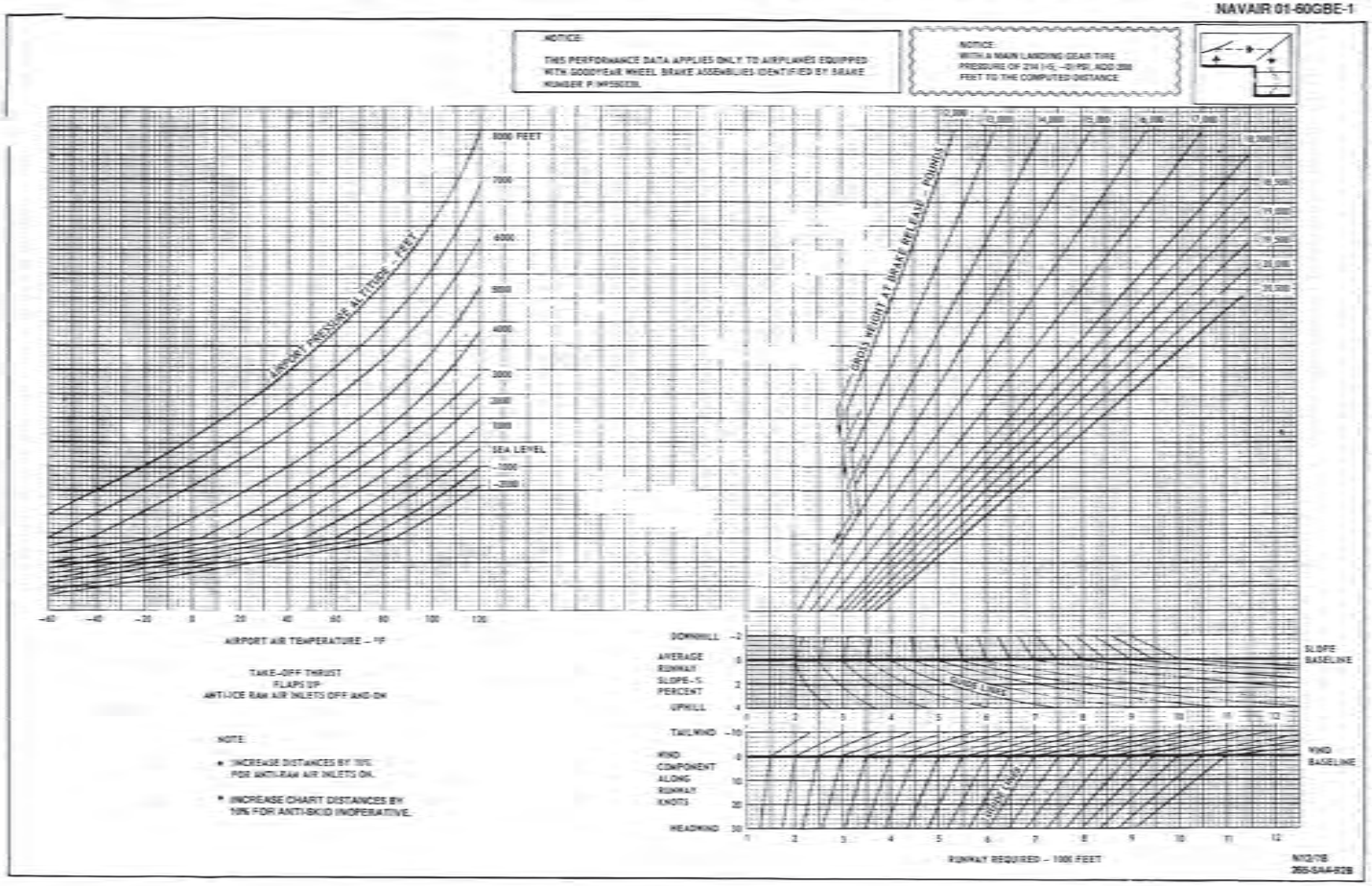
Mach One Manuals



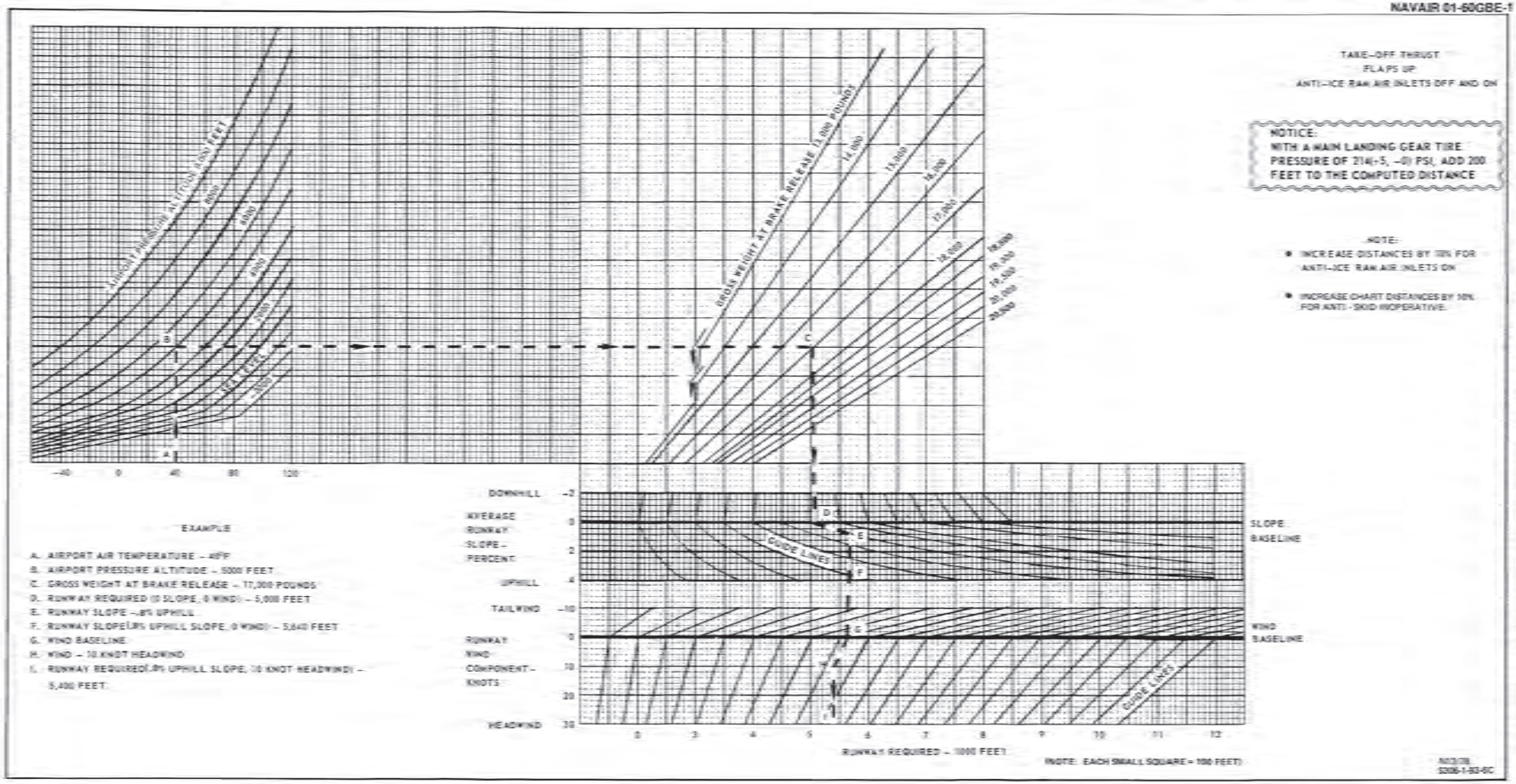
Aircraft Fuel System (Sheet 2 of 2)



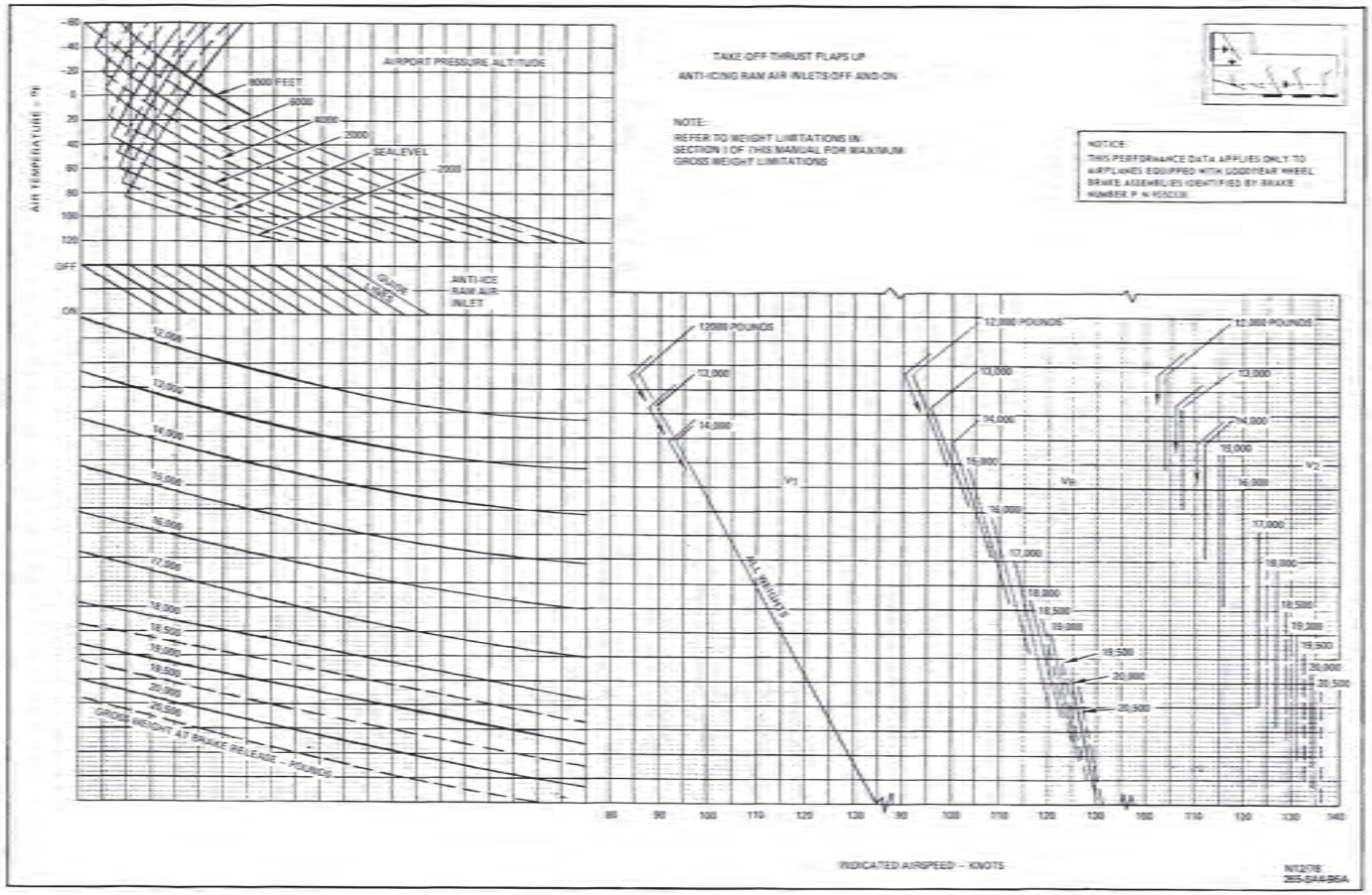
T-38 Aircraft



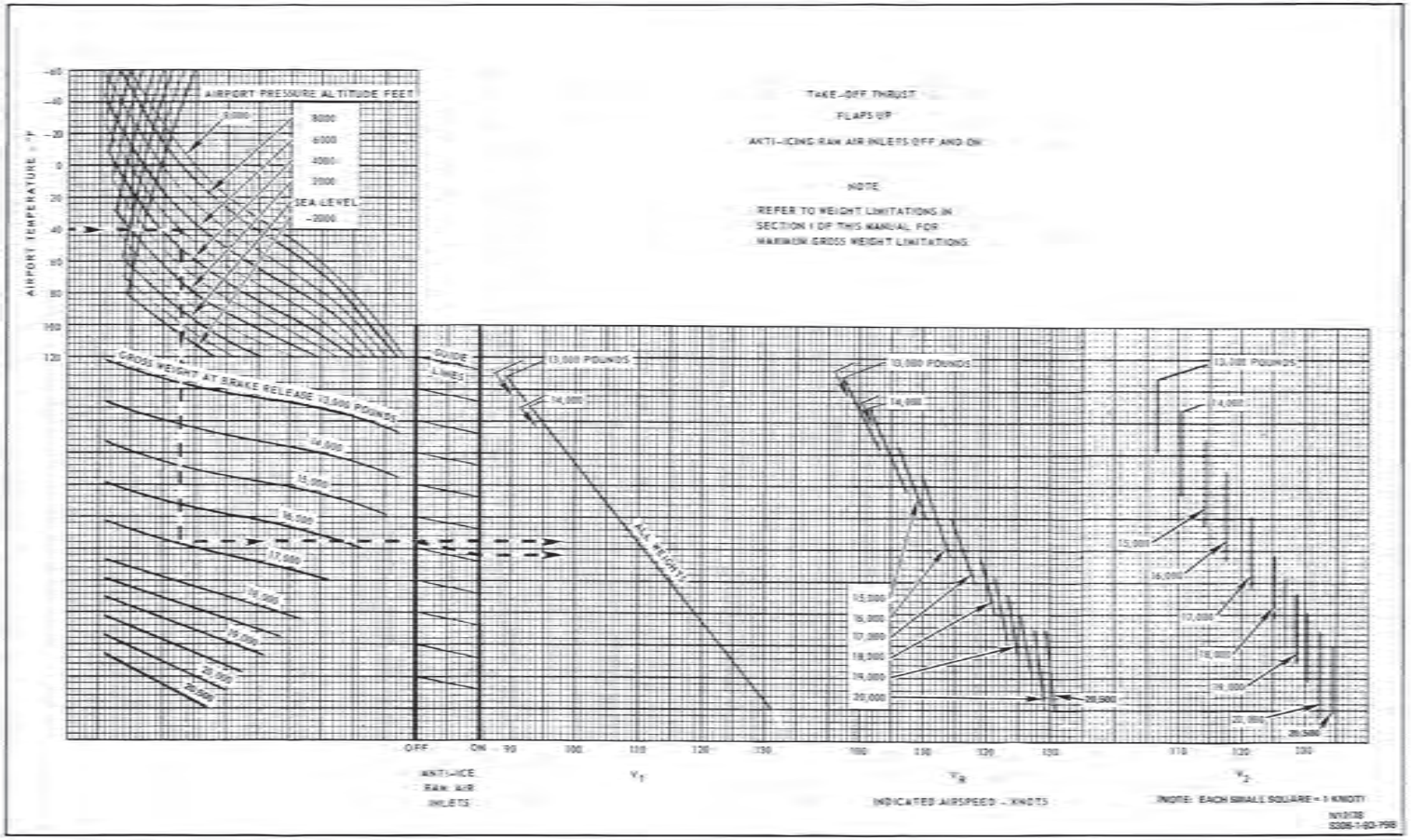
Takeoff Runway Requirements (One Engine — Cabin Engine — Cabin Pressurization ON — Antiskid ON — 9550038 Brake) — CT-39E



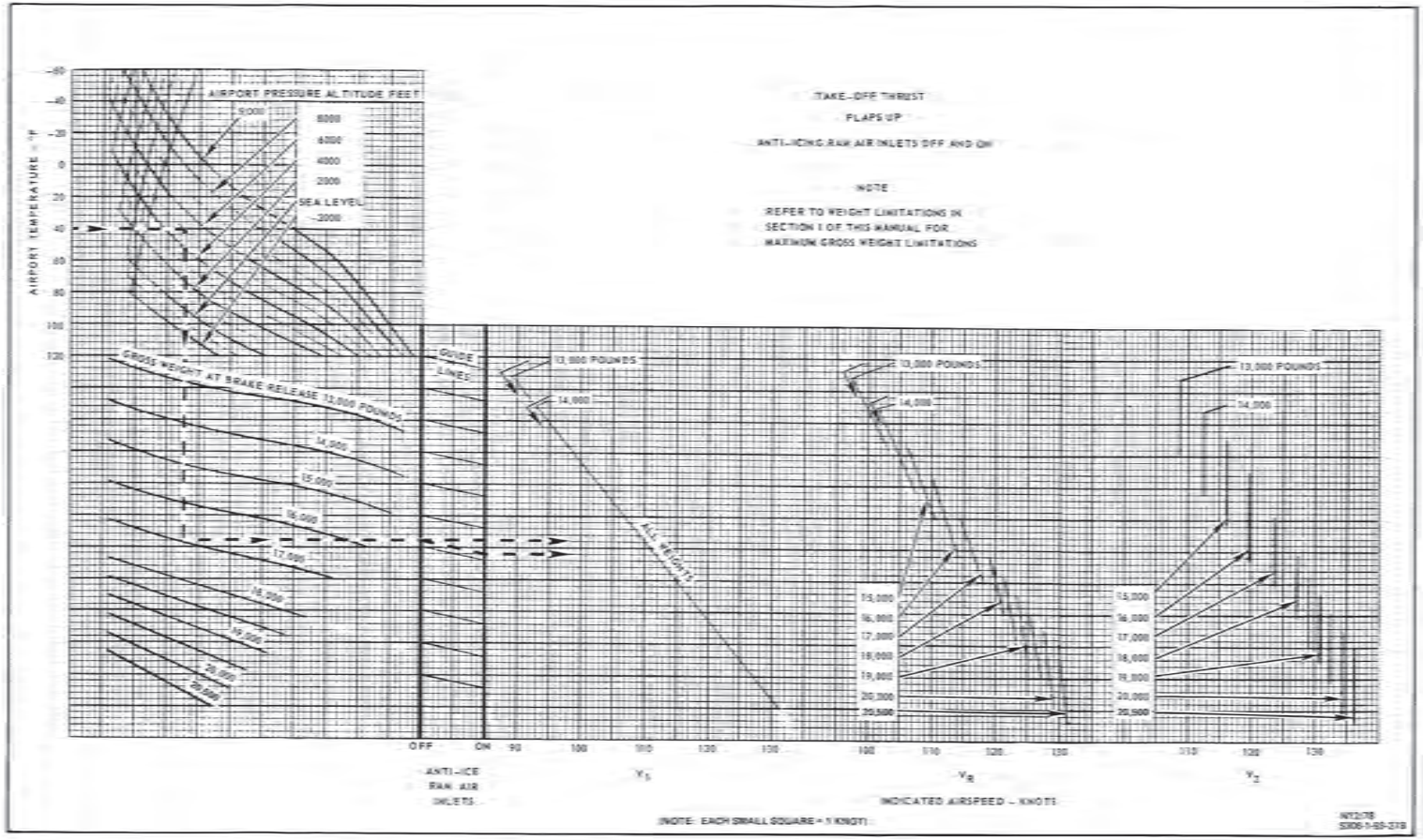
Takeoff Runway Requirements (One Engine — Cabin Pressurization ON — Antiskid ON) — CT-39G



V_1, V_R, and V_2 Speeds (Cabin Pressurization ON —Antiskid ON — 9550338 Brake) — CT-39E



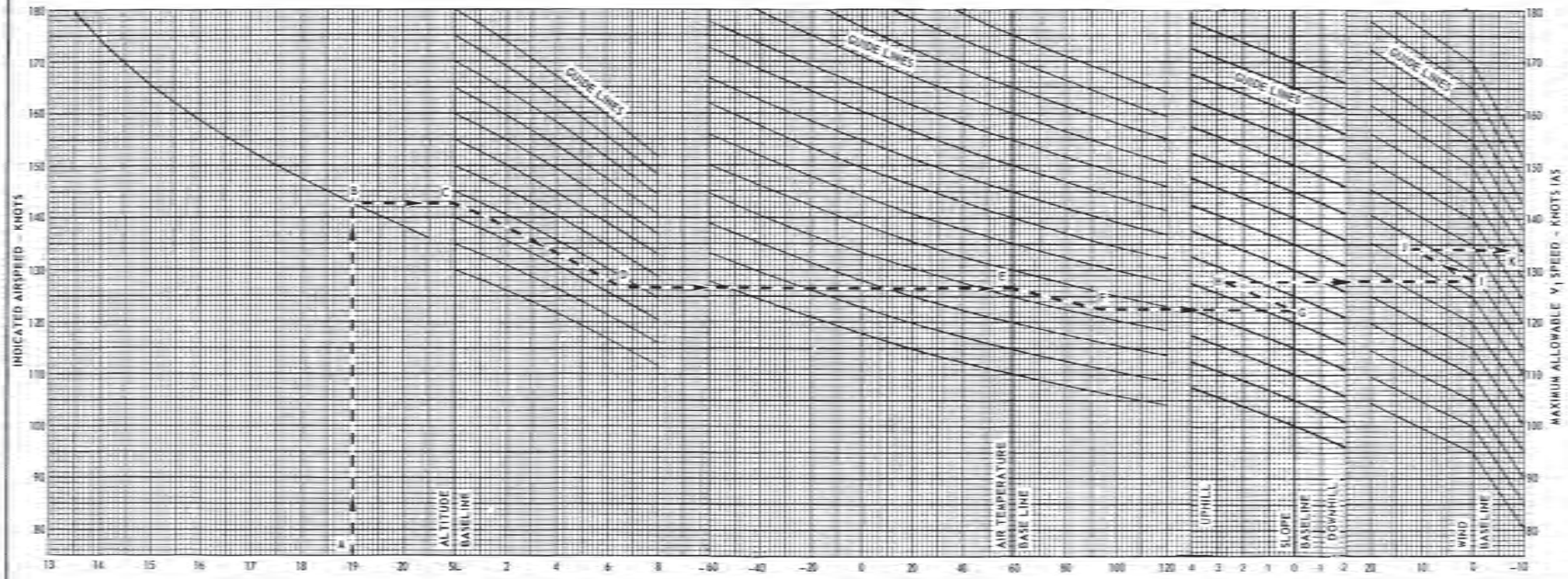
V_1, V_R, and V_2 Speeds (Cabin Pressurization ON — Antiskid ON) — CT-39G Except 158843



V_1, V_R, and V_2 Speeds (Cabin Pressurization ON — Antiicid ON) — CT-39G 158843

NOTICE:
THIS PERFORMANCE DATA APPLIES ONLY TO CT-39E AIRCRAFT EQUIPPED WITH GOODYEAR WHEEL BRAKE ASSEMBLIES IDENTIFIED BY BRAKE NUMBER 9550338.

FLAPS UP



EXAMPLE

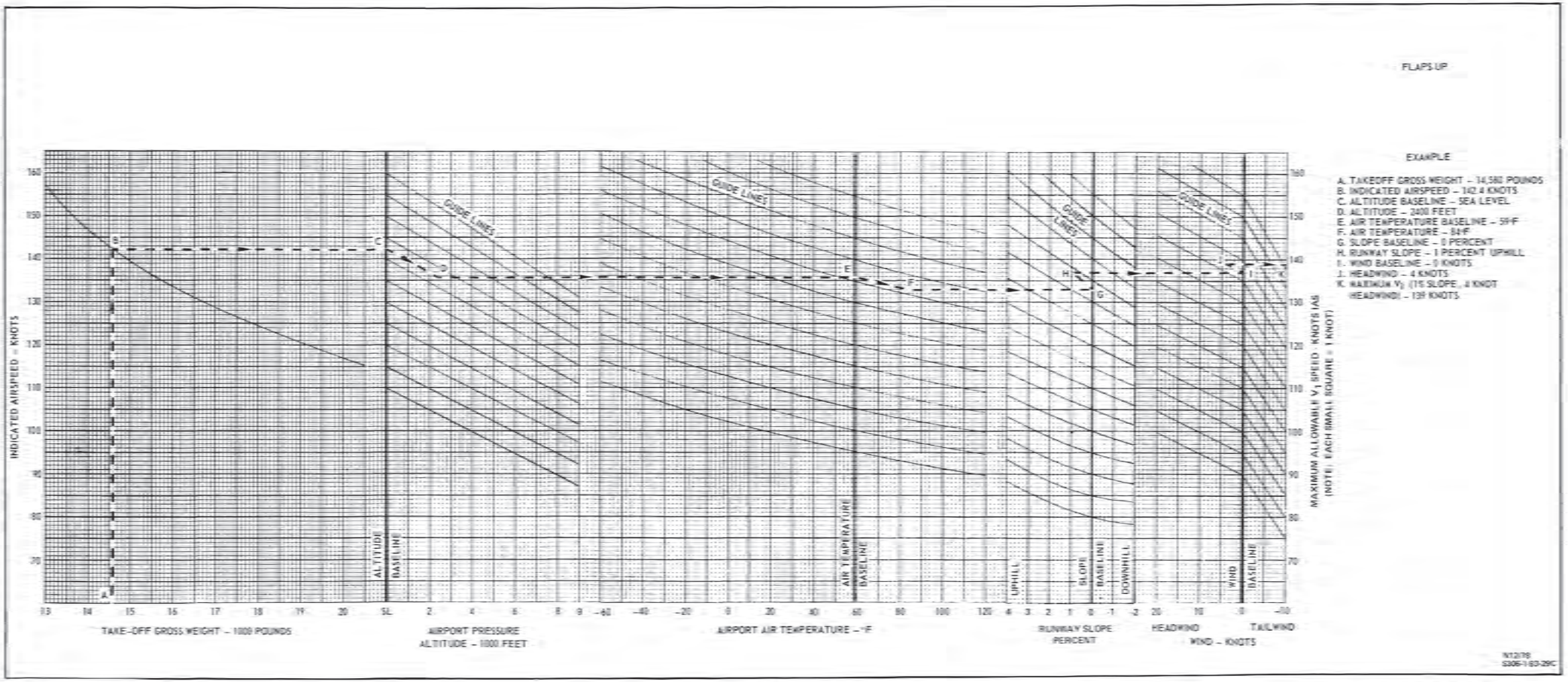
A. TAKE-OFF GROSS WEIGHT – 19,200 POUNDS
B. INDICATED AIRSPEED – 142.7 KNOTS
C. ALTITUDE BASELINE – SEA LEVEL
D. ALTITUDE – 6200 FEET
E. AIR TEMPERATURE BASELINE – 59°F
F. AIR TEMPERATURE – 94°F
G. SLOPE BASELINE – 0 PERCENT
H. RUNWAY SLOPE – 2.6 PERCENT UPHILL
I. WIND BASELINE – 0 KNOTS
J. WIND – 12 KNOTS HEADWIND
K. MAXIMUM V_1 (2.6 PERCENT SLOPE, 12 KNOT HEADWIND) – 134 KNOTS

NOTE

REFER TO WEIGHT LIMITATIONS IN SECTION I OF THIS MANUAL FOR MAXIMUM GROSS WEIGHT LIMITATIONS

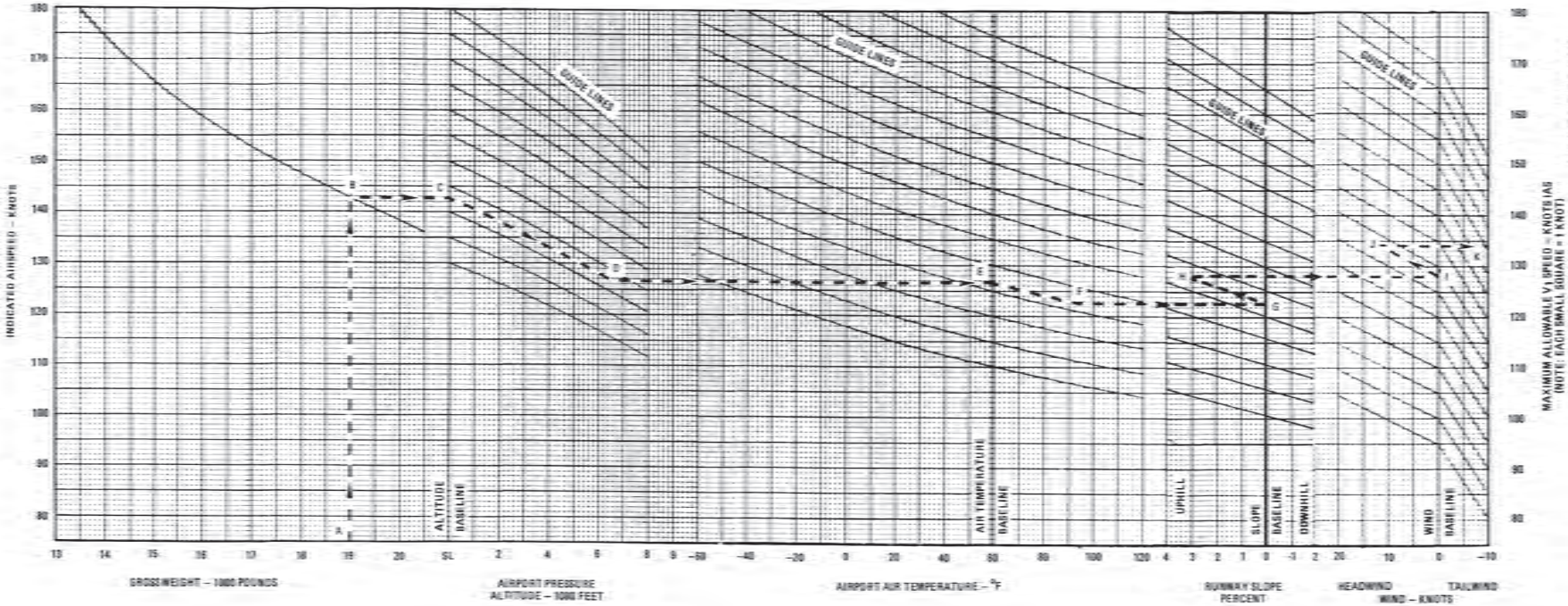
N12078
265-844-118C

V_1 Speed as Limited by Brake Kinetic Energy (Antiskid ON — 9550338 Brake) — CT-39E



V_1 Speed as Limited by Brake Kinetic Energy (Antiskid OFF) — CT-39G

FLAPS UP

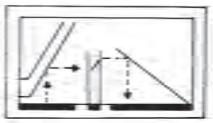


EXAMPLE

A. TAKEOFF GROSS WEIGHT – 19,000 POUNDS
B. INDICATED AIRSPEED – 142.7 KNOTS
C. ALTITUDE BASELINE – SEA LEVEL
D. ALTITUDE – 8500 FEET
E. AIR TEMPERATURE BASELINE – 98°F
F. AIR TEMPERATURE – 94°F
G. SLOPE BASELINE – 0 PERCENT
H. RUNWAY SLOPE – 2.6 PERCENT UPHILL
I. WIND BASELINE – 0 KNOTS
J. WIND – 12 KNOTS HEADWIND
K. MAXIMUM V_1 (2.6 PERCENT SLOPE, 12-KNOT HEADWIND) – 134 KNOTS

N11/94
5306-1-93-308

V_1 Speed as Limited by Brake Kinetic Energy (Antiskid ON) — CT-39G



TAKE-OFF THRUST

ONE ENGINE

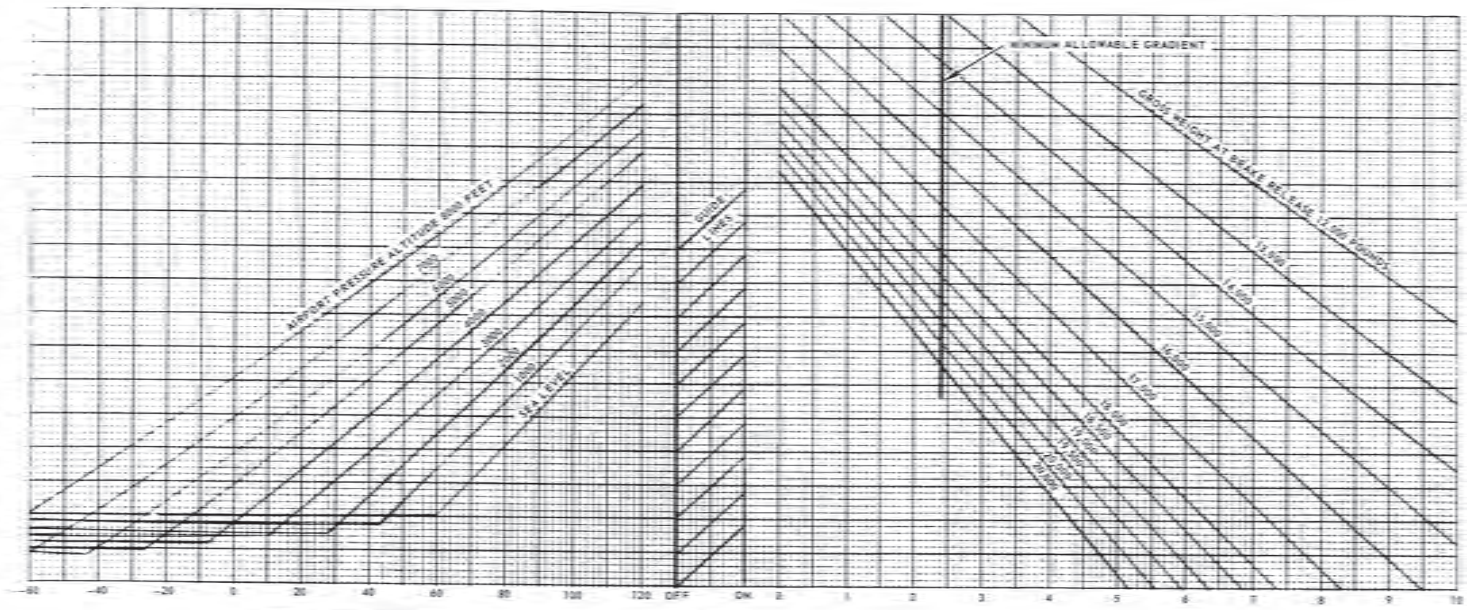
FLAPS UP

ANTI-ICE RAM AIR INLETS OFF and ON

V_2 CLIMB SPEED

NOTE

REFER TO WEIGHT LIMITATIONS IN SECTION I OF THIS MANUAL FOR MAXIMUM GROSS WEIGHT LIMITATIONS.



N12178
395-BAA-639

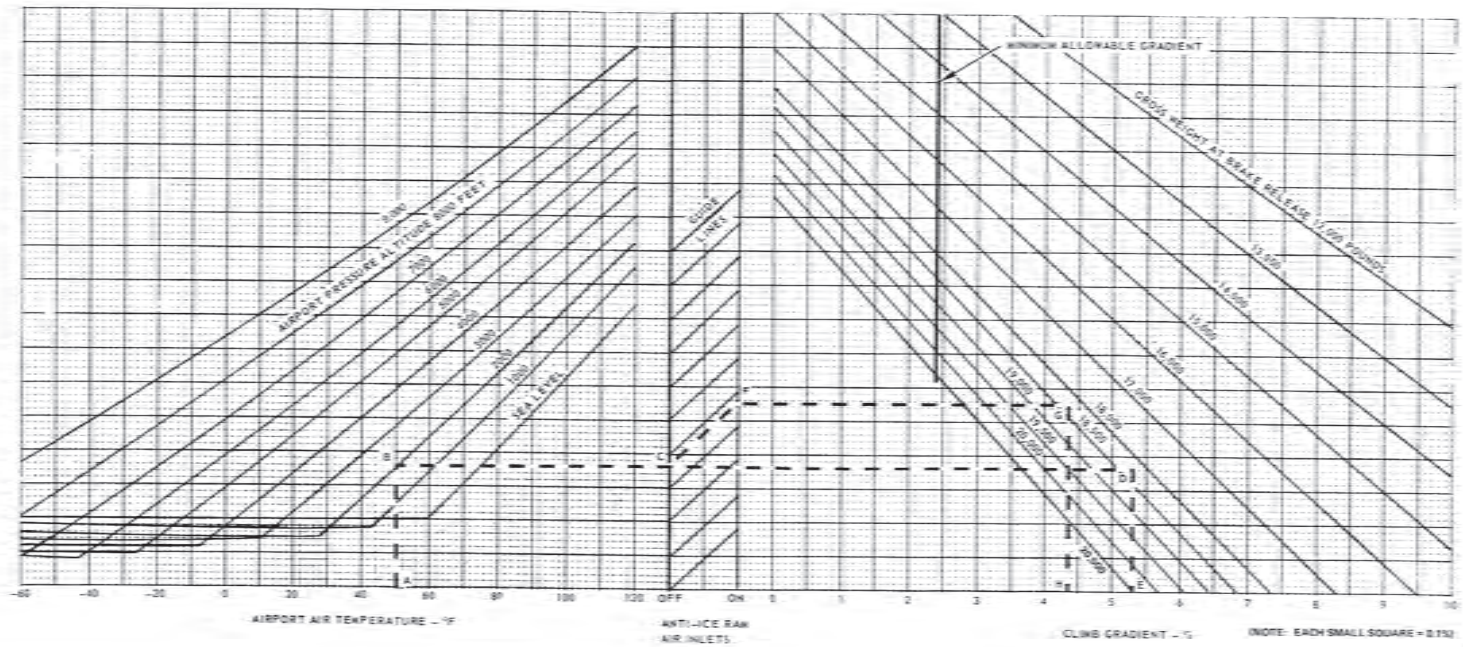
Little-Lakeott (Gear UP — Cabin Pressurization ON) — CT-39E

EXAMPLE

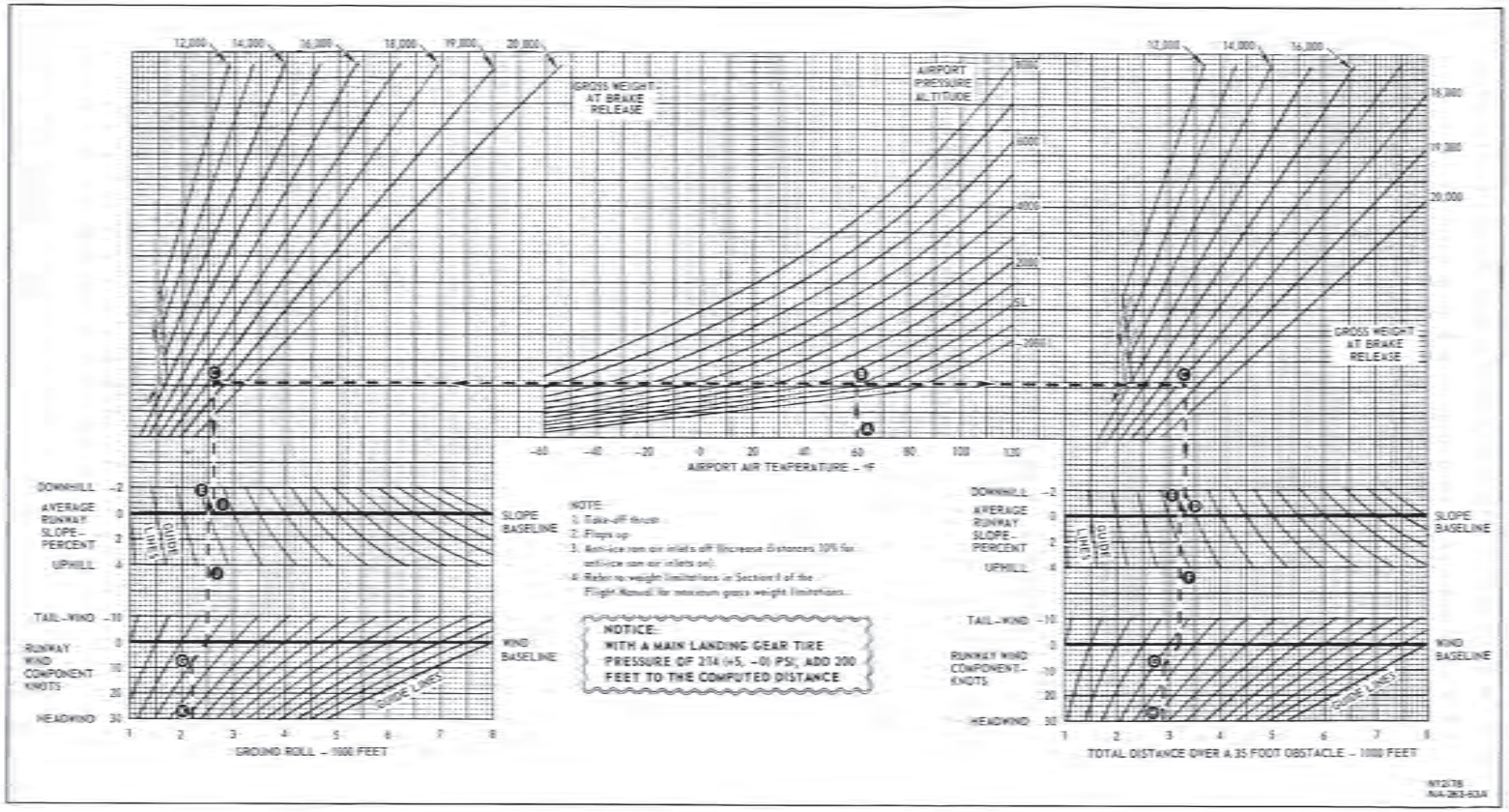
A. AIRPORT AIR TEMPERATURE – 50°F
B. AIRPORT PRESSURE ALTITUDE – 2000 FEET
C. ANTI-ICE RAM AIR INLETS – OFF
D. GROSS WEIGHT AT BRAKE RELEASE – 18,300 POUNDS
E. CLIMB GRADIENT (ANTI-ICE RAM AIR INLETS OFF) – 5.35%
F. ANTI-ICE RAM AIR INLETS – ON
G. GROSS WEIGHT AT BRAKE RELEASE – 18,300 POUNDS
H. CLIMB GRADIENT (ANTI-ICE RAM AIR INLETS ON) – 4.35%

TAKE-OFF THRUST

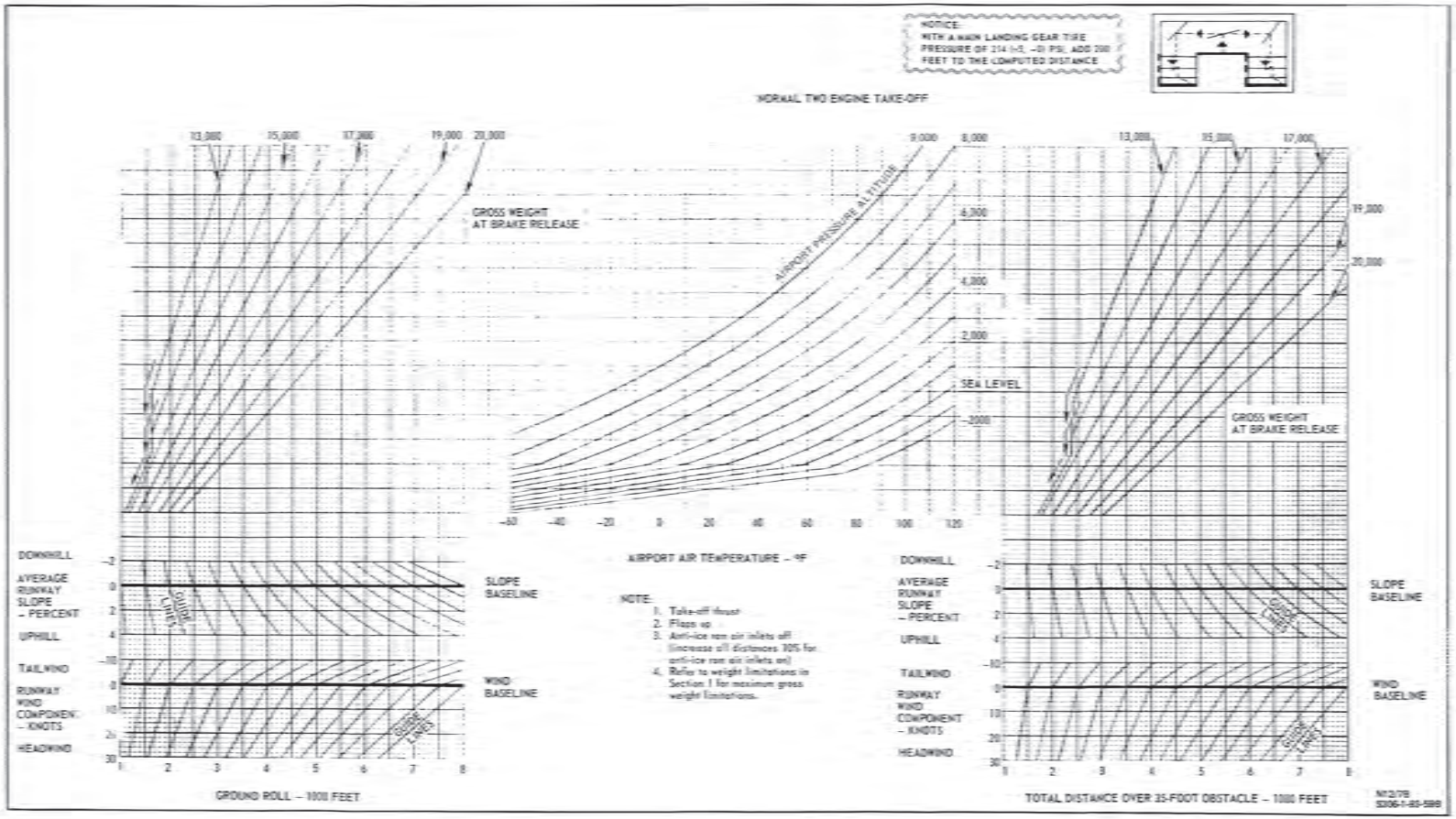
ONE ENGINE
FLAPS UP
ANTI-ICE RAM AIR INLETS OFF AND ON
V_2 CLIMB SPEED



Climb-Takeoff (Gear UP — Cabin Pressurization ON) — CT-39G



Takeoff Runway Requirements (Two-Engine Operation — Cabin Pressurization ON) — CT-39E



Takeoff Runway Requirements (Two-Engine Operation — Cabin Pressurization ON) — CT-39G

LIST OF EFFECTIVE PAGES

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Original	1 (Reverse Blank)
Original	3 (Reverse Blank)
Original	5 thru 27 (Reverse Blank)
Original	29 (Reverse Blank)
Original	31 (Reverse Blank)
Original	33 (Reverse Blank)
Original	35 thru 37 (Reverse Blank)
Original	39 thru 43 (Reverse Blank)
Original	45 (Reverse Blank)
Original	1-1 thru 1-6
Original	2-1 thru 2-155 (Reverse Blank)
Original	3-1 thru 3-14
Original	4-1 thru 4-7 (Reverse Blank)
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Original	5-1, 5-2
Original	49 (Reverse Blank)
Original	6-1 (Reverse Blank)
Original	7-1 (Reverse Blank)
Original	8-1 thru 8-26
Original	9-1 thru 9-11 (Reverse Blank)
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Original	10-1 thru 10-7 (Reverse Blank)
Original	11-1 (Reverse Blank)
Original	53 thru 56
Original	12-1 (Reverse Blank)
Original	13-1 thru 13-4
Original	14-1 thru 14-4
Original	15-1 thru 15-15 (Reverse Blank)
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Original	25-1 thru 25-29 (Reverse Blank)
Original	26-1 thru 26-5 (Reverse Blank)
Original	27-1 thru 27-10
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Original	FO-25 (Reverse Blank)
Original	FO-26 (Reverse Blank)
Original	LEP-1 (Reverse Blank)

Mach One Manuals