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# ROYAL AUSTRALIAN AIR FORCE

# DEFENCE INSTRUCTION (AIR FORCE)

# AAP 7212.007-1

# FLIGHT MANUAL
# PC9/A

**Date of Issue:** Jun 89

(R. FUNNELL)
Air Marshal
Chief of the Air Staff

**Sponsor:** HQTC - DOAT

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# FRONTISPIECE




PC9/A

# FOREWORD

## AUTHORITY

Users are to regard this Flight Manual as an authoritative publication. It is compiled from data available from operating, technical, manufacturing and safety sources, and represents the best level of information available. These instructions provide you with a general knowledge of the aircraft, its characteristics, and specific normal and emergency operating procedures. Instructions in this manual are for a pilot inexperienced in the operation of the aircraft.

## OPERATING INSTRUCTIONS

This manual provides the best possible operating instructions, however, on occasions these instructions may prove to be a poor substitute for sound judgement. Multiple emergencies, adverse weather, terrain and other considerations may require modification of the procedure.

## PERMISSIBLE OPERATIONS

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

## EQUIPMENT LOCATION

Throughout this manual, any cockpit equipment which is not specifically mentioned as being in the 'front cockpit only', or in the 'rear cockpit only', will be found in both cockpits.

## AMENDMENT ACTION

To assist in maintaining this publication in an up-to-date condition users are to bring to the notice of higher authority without delay any errors, omissions or suggestions for improvement. This should be done through your Unit Flight Manuals Officer. Attention is drawn to DI(AF) OPS 5-2.

## WARNINGS, CAUTIONS AND NOTES

The following definitions apply to 'Warnings', Cautions' and 'Notes' found throughout the Manual:

**WARNING**

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

**CAUTION**

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

**Note**

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

## CONTROL AND IDENTIFICATION MARKINGS

The use of block capitals in the text, when identifying switches, controls, etc, indicates the actual markings on that item.

## AIRSPEEDS

All airspeeds quoted in this manual are "indicated" unless otherwise stated.

## PROCEDURAL STEPS

All procedures and checklist items are numbered sequentially with arabic numerals.

## USE OF THE MANUAL

To use the Flight Manual correctly, it is essential to understand the division of the Manual into its sections and the subsequent division of the sections. Each section has a table of contents, and best use will be obtained from the Manual by becoming familiar with the table of contents for each section. The index enables easy reference to a particular topic or item by page number.

## CHECKLISTS

The Flight Manual contains amplified checklists. Abbreviated Checklists are issued as a separate publication (DI(AF) AAP 7212.007-1CL). The items in the Checklist have the same number as the amplified checks in the Flight Manual.

# AMENDMENT CERTIFICATE

It is certified that the amendments promulgated in the undermentioned Amendment Lists have been incorporated in this copy of the Publication:

| Amendment List | | Topic/Sect Affected | *Amendment Effect | Amended By | Date |
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| No | Date of Issue | | | | |
| 1 | JUL 89 | SECT 2&3 | VARIOUS PAGES | | 14SEP89 |
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# RAAF SUPPLEMENT STATUS SHEET

**Instructions**

This sheet accompanies each issue/cancellation/amendment of RAAF Supplements to the Flight Manual.

Insert each attached RAAF Supplement in accordance with its instructions. The information in each RAAF Supplement supersedes that in the published text to which reference is made.

This sheet is to be inserted immediately after the Amendment Certificate, replacing the previous RAAF Supplement Status Sheet.

Distributees are responsible for bringing the content of the attached RAAF Supplements to the attention of affected personnel.

**General**

RAAF Supplements to Flight Manuals are issued as a means of incorporating information of a temporary nature, as an interim method of directing attention to changes to the aircraft operating procedures or limitations pending issue of a formal amendment and as a means of incorporating information where it is not practicable to amend the published text. Periodically, information contained in RAAF Supplements will be consolidated and issued as an amendment to the publication.

The following pages lists the status of all current RAAF Supplements applicable to this manual. Also listed is the Issue date of each associated publication, current at the date of issue of this sheet.

## COMPLIANCE CERTIFICATE

Certified that AL 17 has been incorporated and that the publication has all applicable RAAF Supplements inserted in the locations specified and have been page checked in accordance with the details contained thereon.

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# RAAF SUPPLEMENT STATUS SHEET

## RAAF Supplement Status Sheet

This page provides a listing of the current status of the Flight Manuals, Checklists and RAAF Supplements applicable to the PC9/A aircraft. Supplements are listed in numerical sequence, if you are missing one listed, a copy may be obtained from your Unit TPO/Library.

| Current Flight Manuals | Date |
|---|---|
| DI(AF) AAP 7212.007-1 | JUN 89 |

| Current Aircrew Checklists | Date |
|---|---|
| DI(AF) AAP 7212.007-1CL | JUN 96 |

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SECTION 1

# DESCRIPTION AND OPERATION

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SECTION 1

# DESCRIPTION AND OPERATION

## THE AIRCRAFT

### Introduction

The Pilatus PC9/A is a single engine, low-wing tandem two-seat training aircraft, powered by a Pratt and Whitney PT6A-62 turbo prop engine flat rated to 950 shaft horsepower. The aircraft was built in Australia by Hawker de Havilland under licence from Pilatus Aircraft Ltd. Standard front cockpit instrument panel and side consoles are shown at Figure FO-1 and rear cockpit instrument panel and side consoles are shown at Figure FO-2. The general arrangement of the aircraft is shown at Figure 1-1.

### Aircraft Dimensions

The main dimensions, shown at Figure 1-2 are:

a. Wing span - 10.24 m (33 ft 7 in)

b. Fuselage length - 10.175 m (33 ft 4 in)

c. Height - 3.26 m (10 ft 8 in) (top of fin)

### Aircraft Gross Weight

The approximate gross weight of the aircraft in its normal configuration with two crew and 860 lb of internal fuel, or one crew and full fuel, is 2250 kg.

## THE ENGINE

### Introduction

The Pratt and Whitney PT6A-62 turbo prop engine is a free turbine engine which is flat rated (from 1150 SHP) to a maximum power of 950 SHP. Maximum Cruise Power (MCP) is 900 SHP. Fuel type is Aviation Turbine Kerosene (F34 with FSII (S748)). The engine consists of two independent contra-rotating assemblies: a compressor turbine driving a three stage axial compressor combined with a single stage centrifugal compressor (assembled as a single unit) and a two stage power turbine driving the propeller shaft through a two stage planetary gearbox located on the front of the engine. Figure 1-3 shows left and right side views of the powerplant.

To partly counter the effect of high engine power settings on the control of the aircraft, the engine is mounted with its fore and aft axis depressed: 2° below and 2° to the right of the aircraft axis. All engine driven accessories, with the exception of the propeller overspeed governor and constant speed unit, are mounted on the accessory gearbox at the rear of the engine. The components are driven by the compressor by means of a coupling shaft which extends the drive through a tube at the centre of the oil tank.

### Air Flow Through the Engine

Figure 1-4 shows the airflow through the engine together with details of cooling airflow through the compressor turbine vanes. Inlet air enters the engine through an annular plenum chamber, formed by the compressor inlet case. From the inlet, air is directed forward to the compressor. The compressor consists of three axial stages combined with a centrifugal stage, assembled as an integral unit. The compressed air passes through a diffuser which turns the air through 90° in direction and converts velocity into static pressure. The diffused air then passes through straightening vanes to the annulus surrounding the combustion chamber liner assembly. The liner assembly has perforations of various sizes that allow entry of compressor delivery air. The flow of air changes direction 180° as it enters the combustion chamber and mixes with fuel. The fuel/air mixture is ignited and the resultant expanding gases leave the combustion chamber, reverse direction in the exit duct zone and pass through the compressor turbine inlet guide vanes to the single stage compressor turbine. The still expanding gases are then directed forward to drive the power turbine section, before leaving the engine via the exhaust ducts.

The two stage power turbine, consisting of the first stage inlet guide vane and turbine and the second stage inlet guide vane and turbine, drives the propeller shaft via a reduction gearbox. The exhaust gas from the power turbine is collected and ducted in the bifurcated exhaust duct assembly and directed to atmosphere via twin opposed exhaust stubs. Interturbine temperature is monitored by an integral busbar, probe and harness assembly installed between the compressor and power turbines with the probes projecting into the gas path. A terminal block mounted on the gas generator case provides a connection point to cockpit instrumentation.

### ENGINE FUEL CONTROL SYSTEM

The Engine Fuel Control System consists of an oil-to-fuel heater, high pressure fuel pump, Fuel Control Unit (FCU), flow divider and dual fuel manifolds. The Engine Fuel Control System is shown at Figures FO1-3 and FO1-4.

### Fuel Control Unit

The FCU is comprised of four sections: governing. pneumatic. metering and manual override. The FCU provides automatic scheduling of the fuel required to achieve and maintain the engine power selected by the Power Control Lever (PCL).

The PCL is mechanically connected to the governor section of the FCU via a metal ribbon cable. Movement of the PCL controls a governor valve which regulates the supply of governing air (Py) and reference air (Px) to the pneumatic section. Compressor bleed air is the source for Py and Px. The pressure differential between these air supplies acts on bellows in the pneumatic section to drive the fuel metering valve. A special feature of FCU operation is the protection provided by the Electronic Limiting System (ELS). which acts on Py air in the pneumatic section.

### ELECTRONIC LIMITING SYSTEM

The ELS consists of the Electronic Limiting Unit (ELU). an electrically-driven interface valve operating on the FCU governing air pressure (Py). an isolating solenoid valve. signal sensors and an ELS ISOLATE/EMERGency FUEL control panel. The primary function of the ELS is to limit engine operation to maximum values of torque and Inter-Turbine Temperature (ITT). Two sets of torque and ITT limits are available: Maximum Continuous Power (MCP) and Maximum Power (MAX). In addition, the ELS provides secondary protection against power turbine overspeed in the event of malfunction of the constant speed unit (CSU) and overspeed governor.

The ELU continuously monitors torque. ITT. propeller speed (Np) and gas generator speed (Ng). Outputs of the ELU control the interface valve. isolating solenoid valve, and drive the torque indicators in the cockpit. Power for the ELU and associated components is supplied from the front generator busbar via the ELU/TORQUE circuit breaker. A schematic diagram of the ELS is shown at Figure FO1-11.

### Normal Operation

During normal non-limiting engine operations. the FCU schedules fuel flow to the engine without ELS intervention. Under these conditions. the isolating solenoid valve is open and the interface valve is closed. Whenever the PCL is advanced to a position which would otherwise result in one or both of the MCP

## *PC9/A GENERAL ARRANGEMENT*

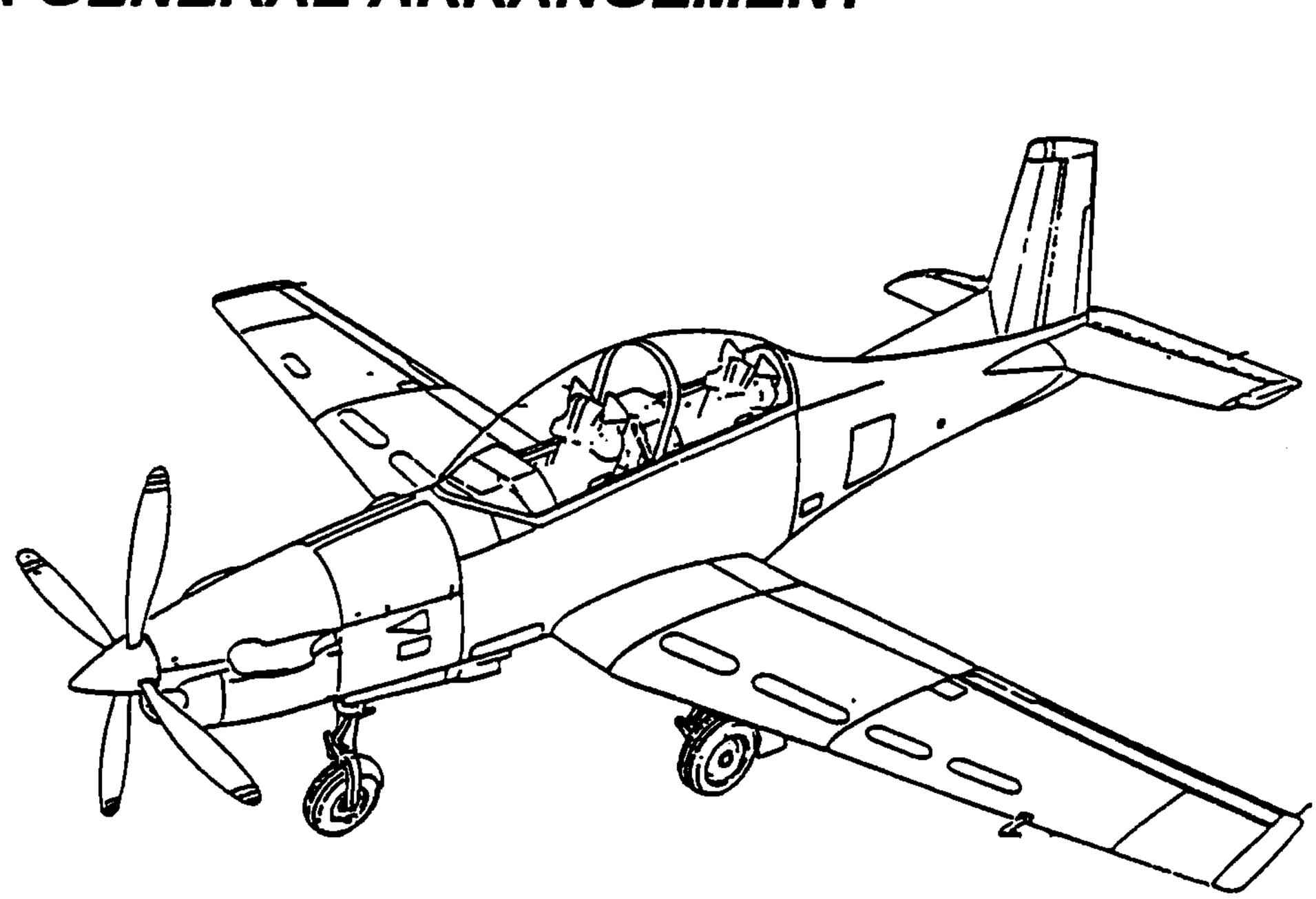
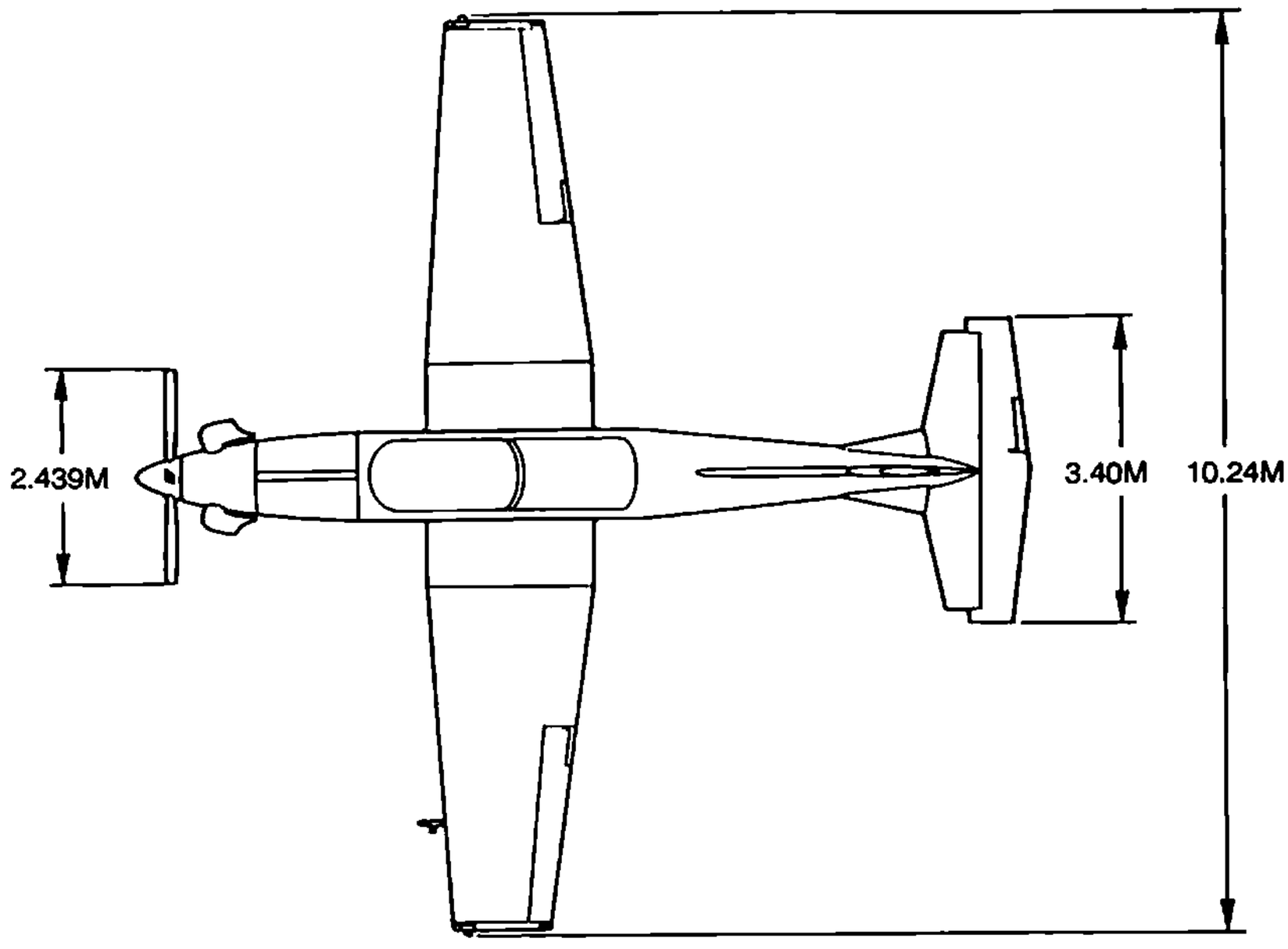
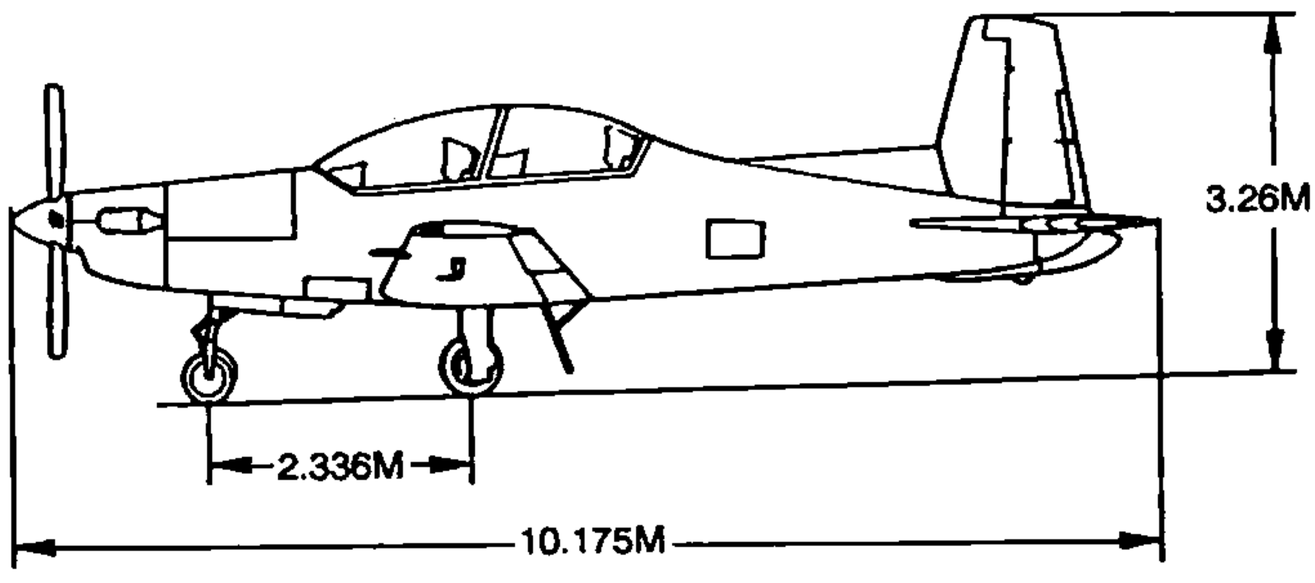


**Figure 1-1**

# AIRCRAFT DIMENSIONS



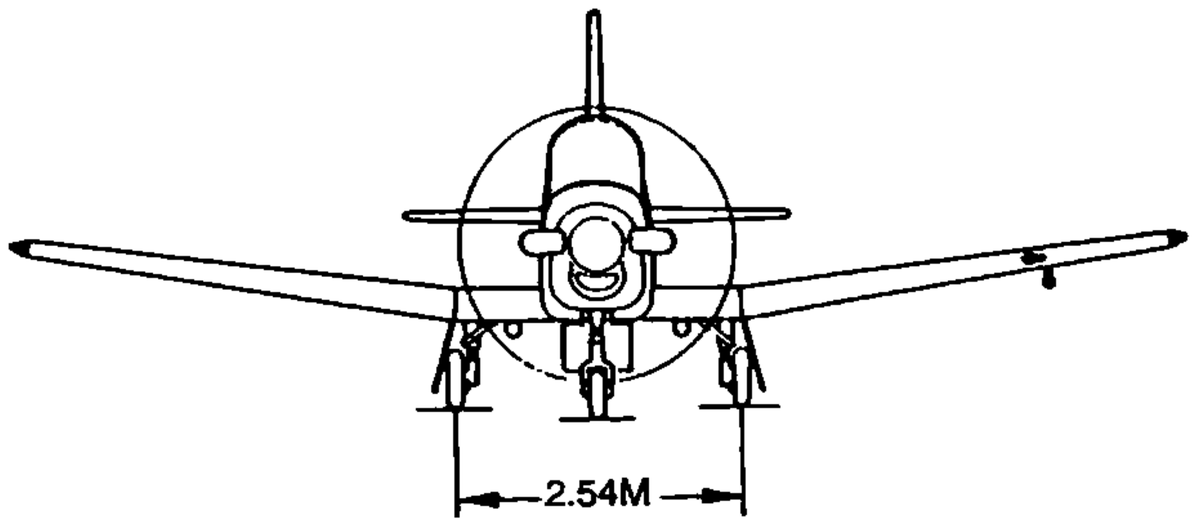




*Figure 1-2*

# POWER PLANT INSTALLATION

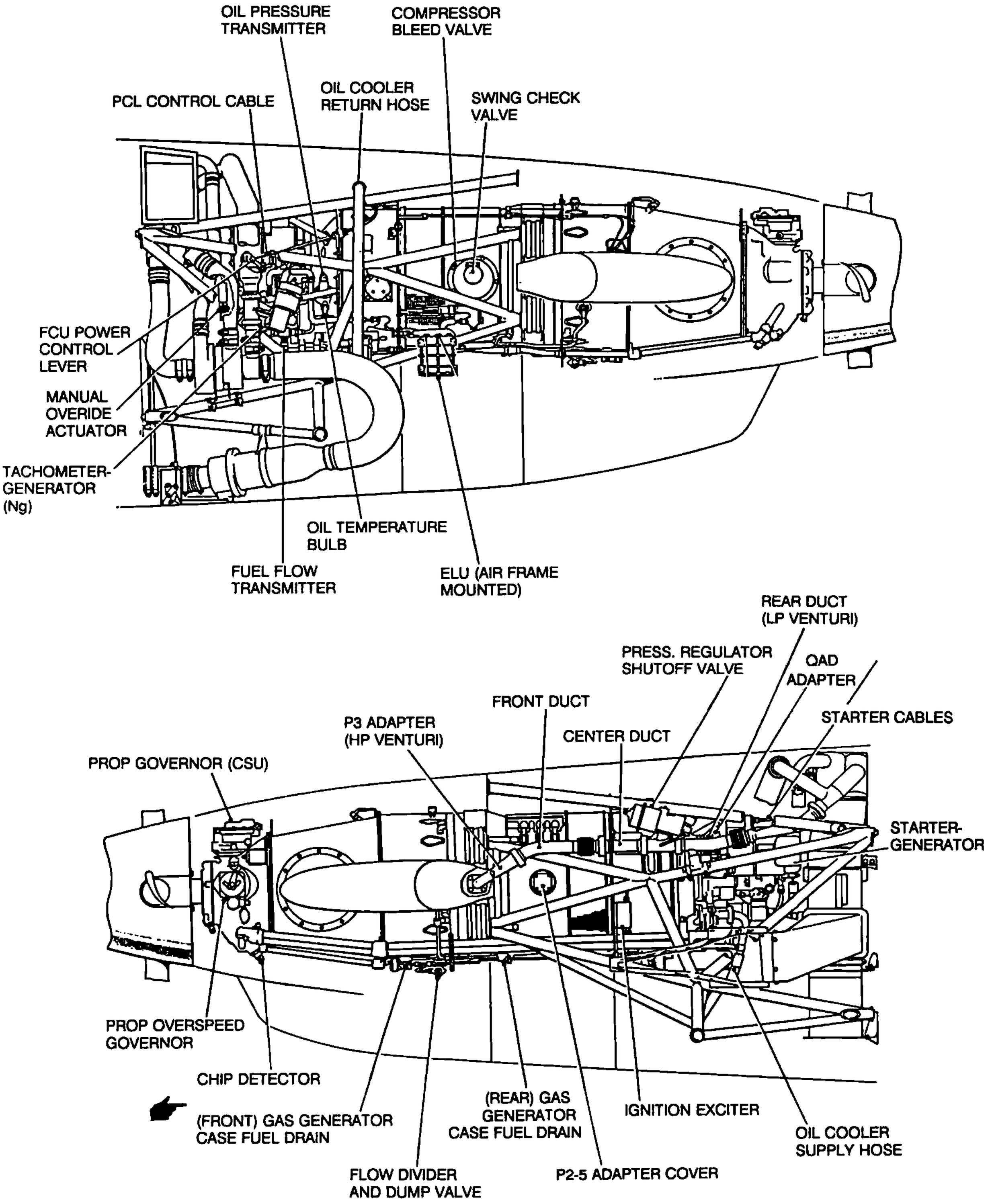
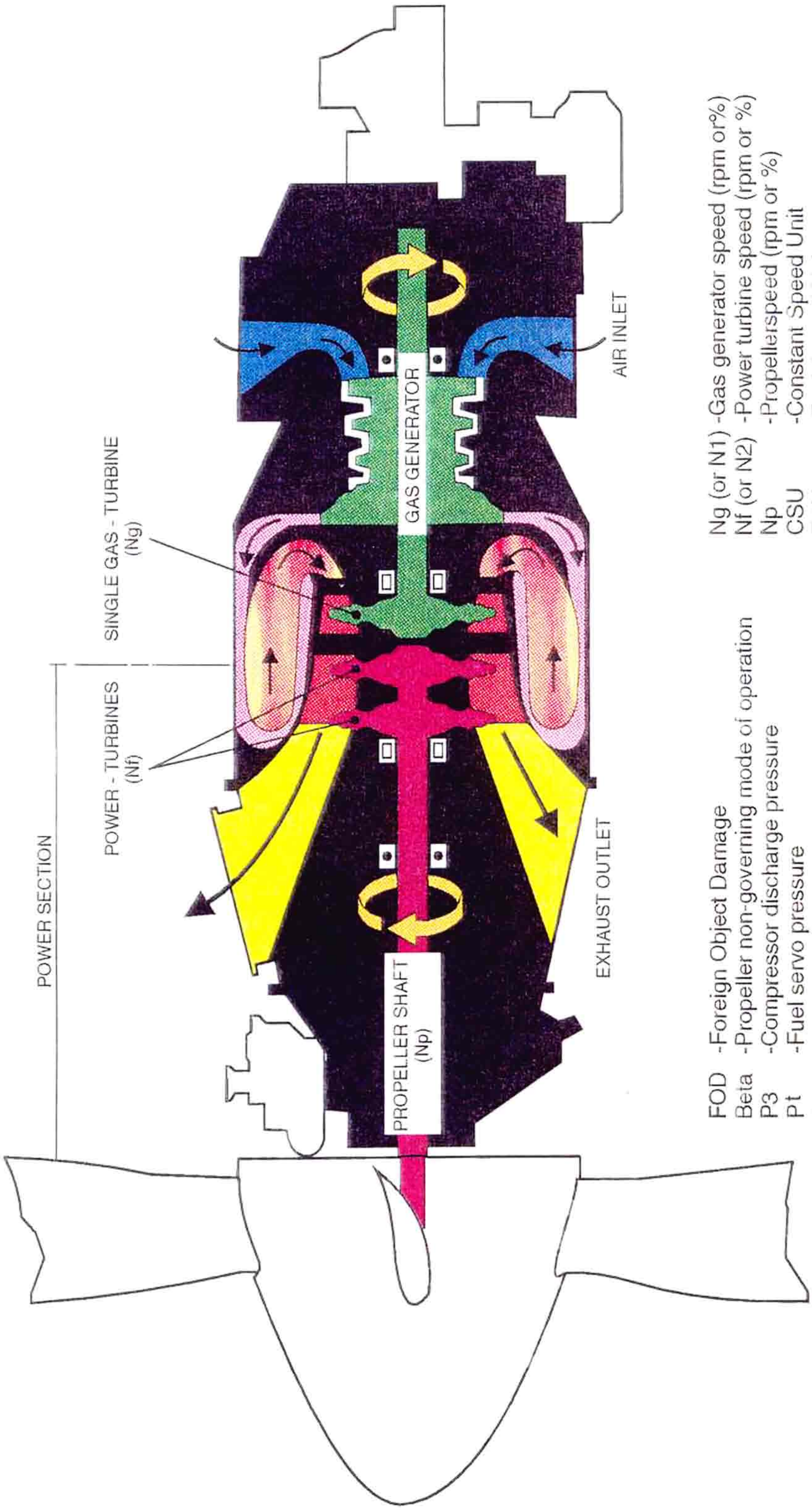



***Figure 1-3***




**PT6A-62 TURBOPROP -ENGINE SYMBOLS AND THEIR MEANINGS**

FIGURE 1-4

# EMERGENCY FUEL CONTROL PANEL

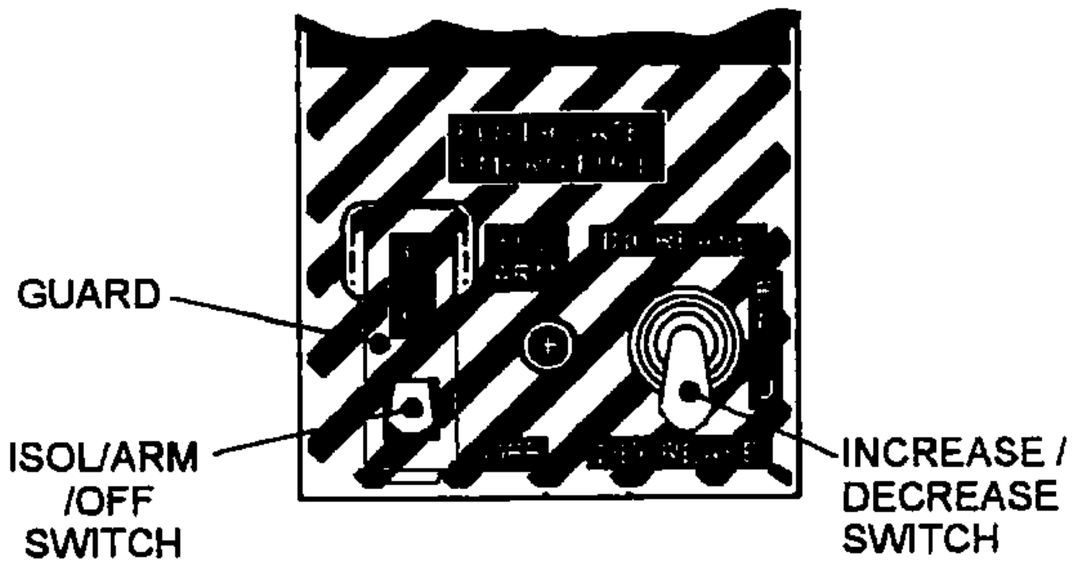



**Figure 1-5**

limits being exceeded, the ELU opens the interface valve to bleed Py air pressure thereby limiting FCU fuel scheduling. When engine parameters decrease below MCP the interface valve closes and the FCU resumes total control of fuel scheduling. Engine limiting to MAX limits is selected by means of a microswitch activated by movement of the PCL to the MAX position.

During aerobatic manoeuvres torque indication may drop due to momentary reduction in engine oil pressure. If this occurs when limiting is enabled,

the ELU will compare torque with other engine parameters and freeze the interface valve to maintain selected power. Normal ELS operation will resume if torque indication recovers within 5 seconds. If the ELS is not limiting during manoeuvres, power is again unaffected although torque indication may vary due to oil pressure fluctuations.

A number of engine parameters are recorded by the ELU. Information recorded includes cumulative time that engine limits have been exceeded, total engine operating time, number of engine starts and duration of engine operation at MAX power. The recording function is enabled whenever the ELU is powered and the engine is operating.

## ELS AND FUEL SCHEDULING MALFUNCTION

### ELS Malfunction

In the event of an ELU malfunction, the ELU logic will act to freeze the interface valve in its last valid position and illuminate the Central Warning System (CWS) ELU and WARNING press to reset captions. If the malfunction occurs when the engine is operating below MCP or MAX limits, the interface valve will freeze closed, allowing unrestricted power availability up to the mechanical FCU stops. Power limits must therefore be monitored manually. If the malfunction occurs when the limiting function is active, the interface valve will freeze partially open. Power is then available only up to the setting corresponding to the position of the interface valve. Following illumination of the ELU caption, the ELU should be disengaged by retarding the PCL to IDLE. Selecting PCL to IDLE activates a microswitch which in conjunction with the illuminated ELU caption causes the isolating solenoid valve to close, thereby preventing further ELS operation. Following this action, power limits must be monitored manually.

There is a possibility that the ELU will not recognise the malfunction of other ELS components. In this case the ELU will continue to function but may cause power fluctuations or restriction of PCL authority (possibly restricting power as low as the FCU metering valve stop at approximately 44% Ng). Under these conditions the ELS can be isolated by selecting the ELS ISOLATE/EMERG FUEL switch to ISOL/ARM. This will cause the isolating solenoid valve to close, the ELS limiting function to be disabled, and illumination of the CWS ELU and WARNING press to reset captions. With the PCL at a high power setting, TQ and/or ITT limits may be exceeded unless the PCL is quickly retarded.

Loss of power through the ELU/TORQUE circuit breaker will cause the isolating solenoid valve to close, the ELU caption to illuminate, and cockpit torque indication to be lost. Power limits must be monitored manually.

### Emergency Fuel Control

In the event of a malfunction of the automatic fuel scheduling system resulting in a loss of engine power, a manual override system, when operated, provides direct control of engine fuel scheduling. A toggle switch in each cockpit controls a linear actuator which applies force through a bellcrank to manually compress the bellows in the FCU pneumatic section, thereby opening the metering valve and scheduling fuel to the engine. The toggle switch is on the ELS ISOLATE/EMER FUEL control panel and is marked INCREASE and DECREASE. The toggle switch is armed by a guarded ELS ISOLATE/EMERG FUEL switch which must be lifted and selected to ISOL/ARM before the respective toggle switch will operate. The ELS ISOLATE/EMER FUEL control panel is shown at Figure 1-5.

During EMER FUEL operations, ELS power limiting is isolated and the CWS ELU and WARNING press to reset captions will be illuminated. Power limits must be monitored manually. If malfunction of the automatic fuel scheduling system results in power fluctuations, the EMERG FUEL system will override normal FCU operation only up to the power set by the toggle switch.

If both ELS ISOLATE/EMERG FUEL switches are turned OFF, the emergency actuator will retract

# RAAF SUPPLEMENT NO 8

# PCL ASSEMBLY GUARD

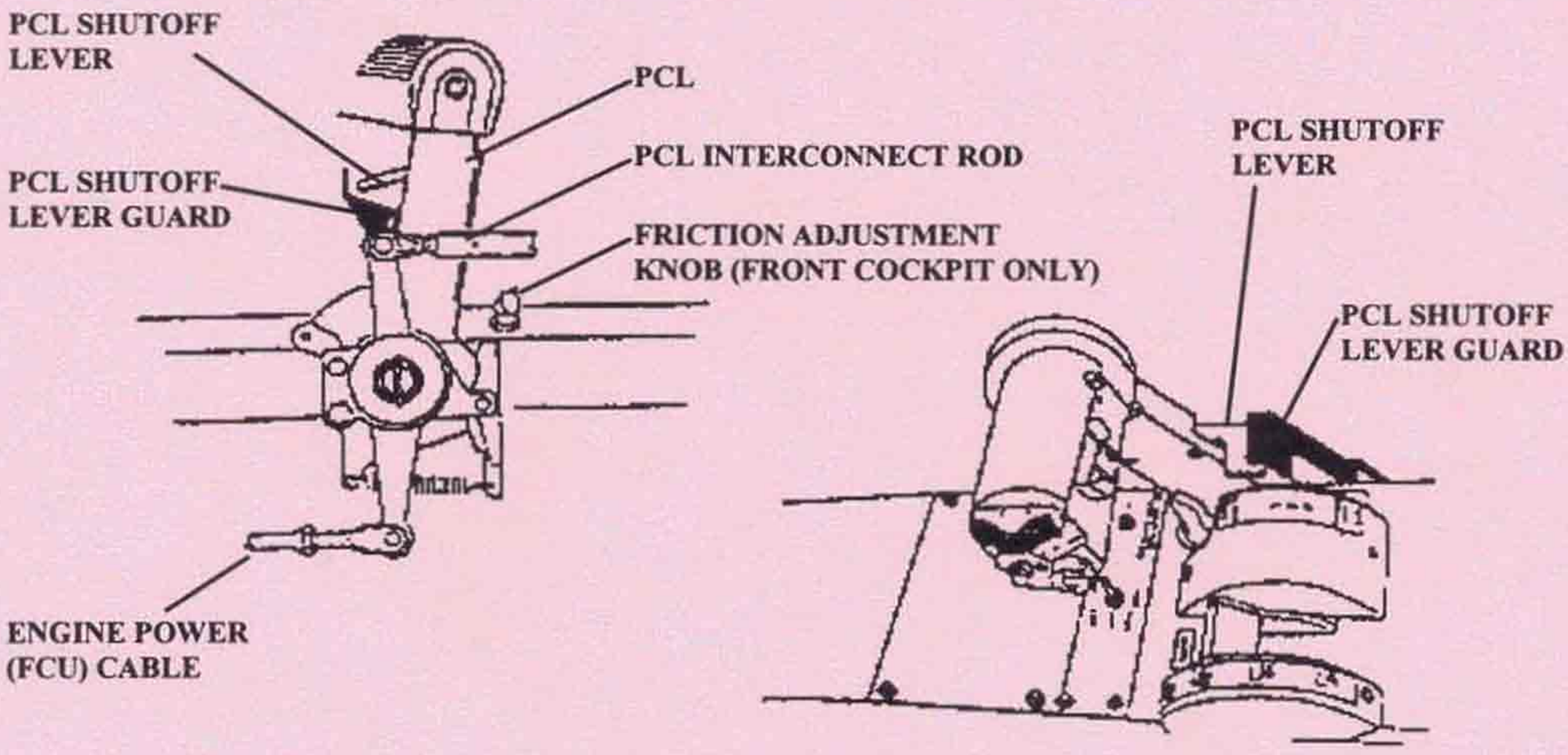
## Introduction

**1.** PC9/A modification 7212.007-163, which incorporates a guard on the PCL assembly, is being progressively introduced into the PC9/A fleet from Aug 96. This RAAF Supplement is raised to provide details of the modification and the associated operating procedures. Insert this RAAF Supplement facing page 1-7 of the Flight Manual PC9/A. This RAAF Supplement will be cancelled and the Flight Manual PC9/A amended when the entire PC9/A fleet has modification 7212.007-163 incorporated.

## Variation

**2.** This following text and diagrams are applicable to PC9/A aircraft with modification 7212.007-163 incorporated and are to be referred to in conjunction with the Power Control Lever section of the Flight Manual PC9/A.

A mechanical detent is engaged as the PCL is advanced from OFF to IDLE. This detent protects against inadvertent engine shutdown when retarding the PCL to IDLE. Engine shutdown is achieved by forcing the spring loaded PCL Shutoff Lever Guard forward, lifting the PCL Shutoff Lever and moving the PCL to OFF. As the PCL is moved from IDLE to OFF the engine fuel supply is shut off and the propeller feathering system is activated. The front and rear PCLs are mechanically interconnected. The PCL is shown below.




**Reference:** DOAT 10/502/1/Air Pt 5

**Date of Issue:** 11 September 1996

**Contents Listing**


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resulting in engine power reducing to idle (in the event of FCU failure), or a setting determined by PCL position (if normal FCU operation is available). If power to the actuator is lost with the system armed, the actuator will freeze.

Following practice use of the EMERG FUEL system, ELS operation can only be restored by turning the ELS ISOLATE/EMERG FUEL switches OFF and interrupting power to the ELU.

## POWER CONTROL LEVER

Engine power output is controlled by a manually operated lever mounted in a control box assembly located on the LH console of each cockpit. The lever is designated Power Control Lever (PCL) and enables selection of engine power in the range from IDLE to:

a. a 'soft' detent position marked MAX CRUISE which provides maximum continuous power, and

b. the maximum travel position marked MAX which is reached by pushing the PCL through the soft detent. This position provides take-off/maximum power. Operation of the engine at the MAX setting is restricted to not more than 10% of total operating time.

In the OFF position the PCL cuts off the engine fuel supply to shut down the engine, and also activates the propeller feathering system. To prevent inadvertent shutdown a mechanical stop and lever detent system is incorporated. The front and rear PCLs are mechanically interconnected. The PCL is shown at Figure 1–6.

**CAUTION**

Care should be exercised when operating switch/lever functions in the vicinity of the Detent Release Lever, as lifting of the lever while the PCL is at or near the idle stop can result in the PCL being inadvertently retarded through the stop into the OFF position.

## IGNITION SYSTEM

### Description

A dual high energy ignition system provides lightup of the engine during ground starting and in-flight relight in the event of engine flameout. The system has been developed to provide the engine with a capability for relight over a wide range of temperatures. The ignition system consists of an airframe mounted ignition exciter box, two individual high tension cable assemblies and two spark igniters. The system receives electrical power from the battery busbar (28 VDC supply through the IGNITION C/B).

***POWER CONTROL LEVER***

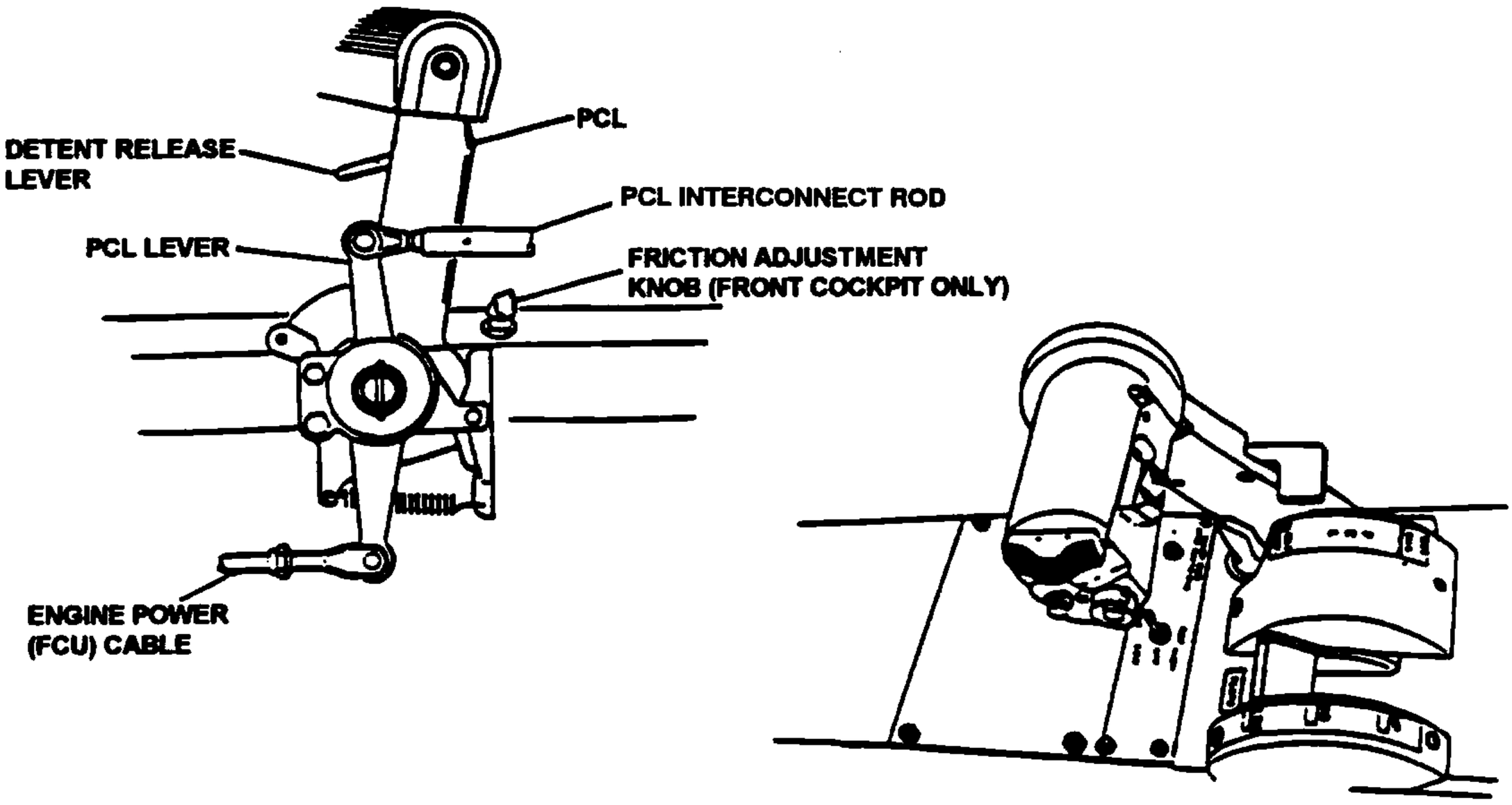



Figure 1-6

### Ignition Switch

A guarded IGNITION switch on the right side console provides direct control of the ignition system during ground starting and in-flight relighting. The system may be operated with the BAT switch selected OFF (with external power connected).

## STARTER SYSTEM

### Description

Engine starting is accomplished by inducing a mass airflow through the engine, delivering atomised fuel to the combustion chamber, and initiating ignition of the fuel/air mixture. A starter-generator, driving through the engine accessory gearbox, induces engine mass air-flow by rotating the gas generator (compressor and compressor turbine). Fuel is delivered to the combustion chamber when the PCL is selected from OFF to IDLE, and initial ignition of the fuel/air mixture is achieved by energising the spark igniters. The aircraft battery or an external power source provides the power required to start the engine.

### Starter Switch

A starter switch is provided on the right console in both cockpits. Selecting the front or rear STARTER switch to ON connects DC power from the battery busbar to the starter relay via the STARTER C/B. Prior to and during engine starting the normal operating parameters of several systems may be temporarily exceeded. To avoid unnecessary indications the associated monitoring and warning systems are inhibited until engine starting has been completed. The inhibiting circuits are connected in parallel with the starter relay energising circuit and are activated by selection of either starter switch.

## ENGINE INSTRUMENTATION

### Description

Engine parameters are displayed electrically on the Engine and Secondary Instruments System (ESIS), located in both cockpits. The ESIS comprises:

a. a dual channel Sensor Interface and Control Unit (SICU) located behind the front cockpit instrument panel,

b. two Engine and Secondary Instruments Display Panels (ESDPs), located on the right side of each cockpit instrument panel ,

c. system failure indicators, and

d. associated test and setting switches.

### Engine and Secondary Instruments Display Panel

The ESDP indicators, shown at Fig 1-7, provide a means of visually maintaining the following operating parameters:

a. Engine TORQUE,

b. interturbine temperature (ITT),

c. gas generator speed (NG),

d. outside air temperature (OAT),

e. propeller speed (NP),

f. oil temperature and oil pressure,

g. fuel used, fuel remaining and fuel flow,

h. DC busbar voltage and current, and

i. hydraulic system pressure (emergency hydraulic only).

### Sensor Interface and Control Unit

The SICU comprises two independent processing channels, designated the primary and secondary SICUs. Both SICU channels interface with the ESIS sensors and process the information received. Each SICU channel transmits the resultant information along independent databuses to both ESDPs. The transmitted signals drive Liquid Crystal Displays (LCD) which display the information in digital and/or analog form. The primary SICU channel is connected to DC power at the battery busbar. The secondary SICU channel is connected to DC power at the generator busbar. The system provides its own AC power requirements by using integral inverter circuits. This power connection arrangement, together with dual data buses ensures system performance continues unaffected by single busbar or system failures.

### System Failure Indicators

Two system failure indicators, one red (WARNING) and one yellow (CAUTION), are located on each ESDP. The indicators are illuminated under the fol-lowing conditions:

a. Steady red or yellow, where the detected failure affects the operation of the relevant system such as a defective sensor or an operating limit being exceeded. The affected system is annunciated by the appropriate system display flashing (red - 80 times/min, yellow - 40 times/min). This type of failure indication can not be reset and remains until the fault is cleared.

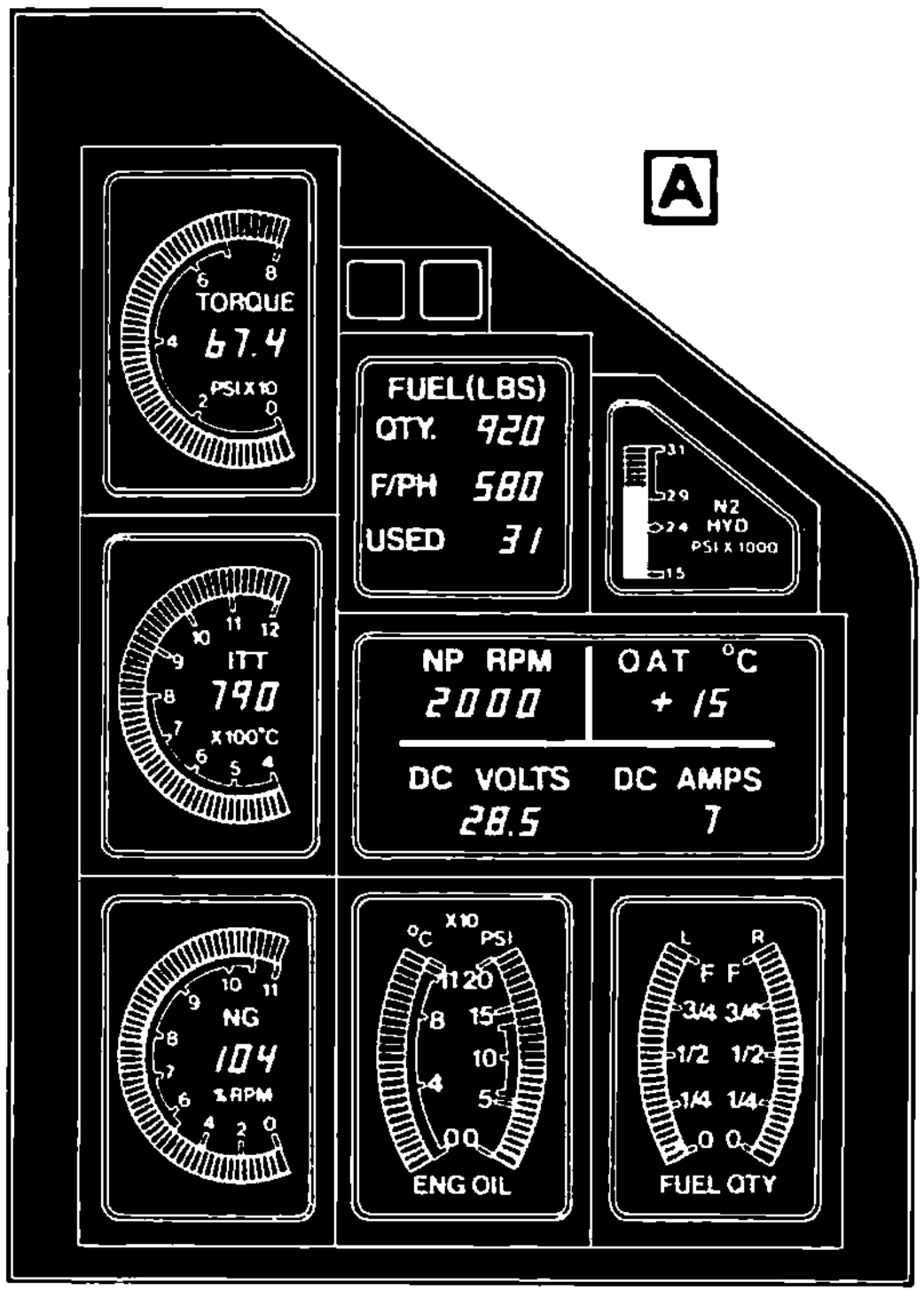
b. Flashing yellow, where the detected failure does not immediately affect the operation of the relevant system, such as a single SICU channel failure. The flashing yellow indicator can be cancelled by operation of the Central Warning System (CWS) master CAUTION PRESS TO RESET indicator in either the front or rear cockpits (the CAUTION indicator will not be illuminated, but it will produce the reset command needed to cancel the ESDP indicator).

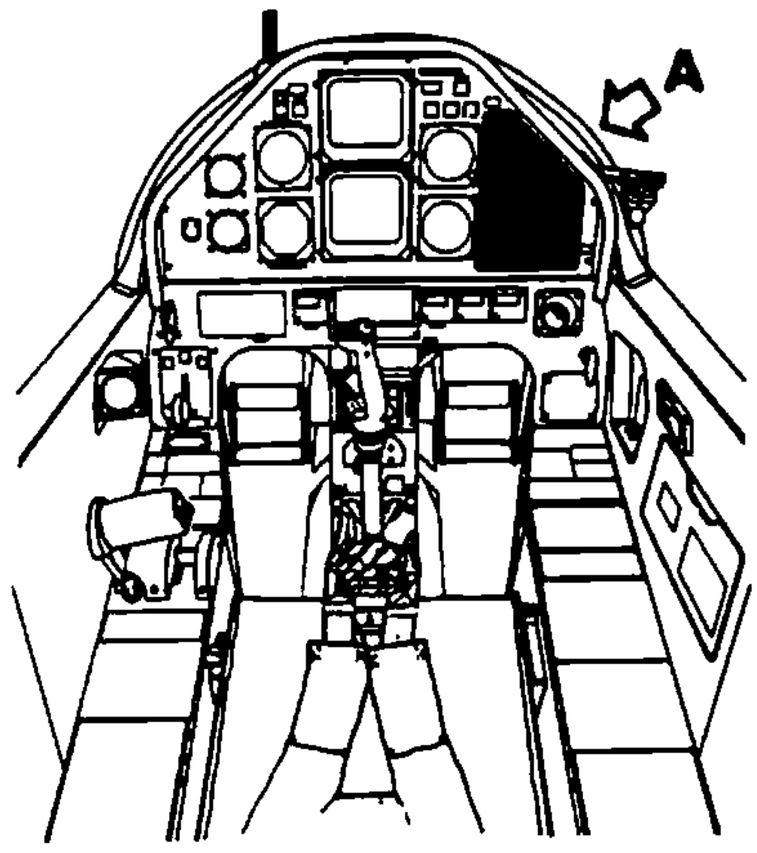
Confirmation of appropriate fault is annunciated by pressing the LAMP AND ENG INSTR SYS test switch located in the front cockpit. A fault code will appear in the FUEL QTY digital segment.

### Associated Test and Setting Switches

The LAMP AND ENG INSTR SYS push-switch (behind the PCL in the front cockpit) initiates a self-test which causes the recorded failure status codes (if any)

# ENGINE AND SECONDARY INSTRUMENTS DISPLAY PANEL




FRONT COCKPIT SHOWN
REAR COCKPIT SIMILAR

***Figure 1-7***

to be displayed after gradually turning on each indicator segment and flashing of the yellow CAUTION indicator. The status code is displayed for approximately 5 seconds on the FUEL QTY segment. ESDP lighting and control (dimming) is provided by the aircraft instrument lighting system. Fuel indication switches are used to zero the fuel USED totaliser and to set the aircraft fuel QTY (remaining) on a count down totaliser. The switches are located on the right side wall in the front cockpit. Zero fuel USED indication is achieved using a two position F USED RESET switch with marked positions RESET and OFF. A three position F QTY SET switch is used to increase (+) or decrease (-) the fuel QTY remaining. When using external tanks this switch will need to be operated to increase the total fuel quantity to indicate internal and external fuel load. Figure 1-8 shows the ESIS system.

**NOTE**

After having entered external fuel load do not activate F USED RESET switch.

## PROPELLER SYSTEM

### Description

The propeller is a Hartzell HC-D4N-2A/9512A four-bladed, variable pitch, feathering type, driven by the two stage turbine through a reduction gearbox. The propeller and the associated propeller control system comprise:

a. A variable pitch, feathering propeller driven by a two stage power turbine through a 16.62:1 reduction gear. Propeller blades are actuated by an integral oil pressure operated pitch-change mechanism.

b. A propeller governor/constant speed unit (CSU), driven from the propeller reduction gear.

c. A propeller overspeed governor, also driven from the reduction gear.

### Propeller Operation (Fig FO1–5)

At normal engine power settings, the propeller is driven at a constant speed of 2,000 RPM. With the engine running at IDLE the power turbine is not producing enough power to drive the propeller to 2,000 RPM and the blades are kept in the fully fine position. Propeller speed is maintained by the CSU by adjusting the propeller blade angle to the pitch required to absorb the engine power. Full pitch angles of the blades measured at the 30 inch station (marked with a yellow stripe) are 14° (fine) and 86° (feather). As flight conditions and power settings change, the CSU continues to vary blade angle as required to maintain propeller speed. An electronic pick-up in the CSU provides a propeller speed signal to the ELU and a pick-up in the overspeed governor provides a signal to the propeller speed (NP) indicator. In the event of a fail-

# ENGINE AND SECONDARY INSTRUMENT WARNING SYSTEM

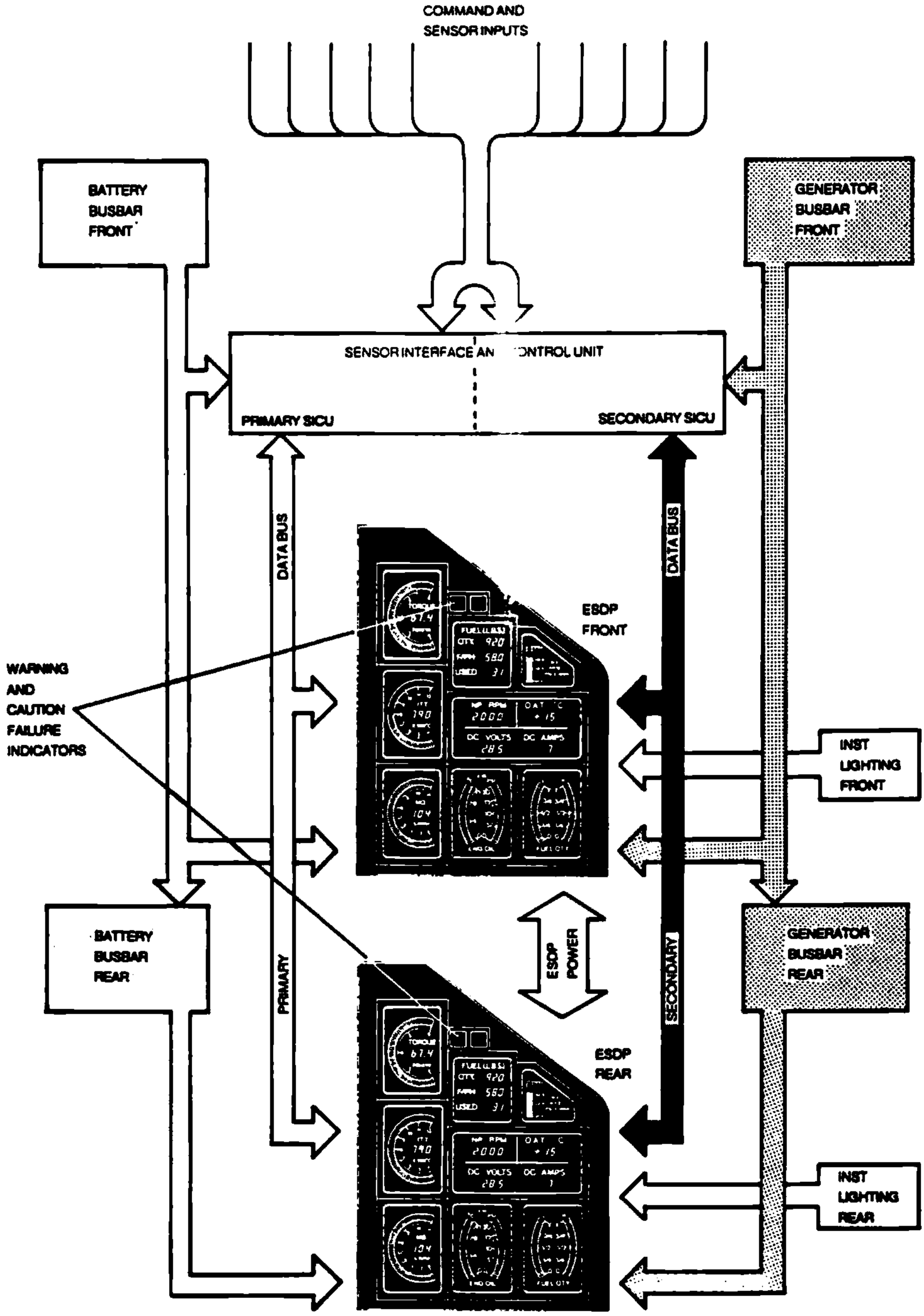



*Figure 1-8*

# RAAF SUPPLEMENT NO 7

# ENGINE OIL SYSTEM

## Introduction

**1.** Several incidents of high engine oil temperature and low engine oil pressure have occurred during high altitude operations of the PC9/A. Pilatus have advised that extended flight at an outside air temperature (OAT) less than -30°C may result in engine oil temperature rising higher than 90°C due to oil congealing in the oil cooling system. A copy of this supplement is to be inserted in two places; facing page 1-11 and page 5-8.

## Variation

**2.** High altitude operations are to be limited to a minimum indicated OAT of -20°C, unless operationally necessary.

**Reference:** DOAT 10/502/1/Air Pt5

**Date of Issue:** 6 June 1996

### Contents Listing


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ure of the normal governing system, the overspeed governor will operate to limit propeller speed to a figure slightly higher than 2,000 RPM (106%). The ELU also operates to limit power turbine overspeed.

### Propeller Feathering

The overspeed governor incorporates a feathering solenoid which is energised by placing the PCL to the OFF position. Selection of PCL to OFF causes the propeller blades to move to the feathered position. With a failed engine a feathered propeller produces significantly less drag than a windmilling propeller.

### Propeller Speed Indication

Propeller speed (NP) is indicated on a digital LCD located on the Engine and Secondary Display Panel (ESDP) in each cockpit. The speed sensor signal from the magnetic pick-up on the propeller overspeed governor is routed to the two Sensor Interface and Control Units (SICU). Each SICU converts the AC analog signal to its digital equivalent. The ESDPs select one of the SICU databuses for use, the other acting as standby in case of a fault on the selected SICU. The signal is then passed to the LCD presentation.

## ENGINE OIL SYSTEM

### Description

The components of the engine oil system consist of the:

a. oil cooler,

b. oil temperature/pressure indicating system, and,

c. the chip detector system.

Refer to Figure FO-6 for a schematic presentation of the engine oil system. The oil grade and specification used in the aircraft are shown in the servicing information. The oil system provides a constant supply of filtered oil to lubricate and cool engine bearings, reduction gears and all accessory drives throughout all normal and most aerobatic flight conditions, including inverted flight. Some restrictions apply to zero – G manoeuvers see Section 5. The oil cooler, shown at Figure 19, is located on the bottom of the engine compartment and is an air cooled plate and fin type heat exchanger. The oil cooler incorporates a thermostat/pressure relief valve which allows oil to bypass the cooler during cold operating conditions.

The engine is equipped with a chip detector connected to the CWS annunciator panel (CHIP light), to give warning of ferrous particle contamination of the oil system. The oil pressure and oil temperature indicating system comprises:

a. a temperature sensitive bulb in the accessory gearbox, and

b. an oil pressure transducer in the accessory gearbox.

The temperature and pressure signals are processed by the SICU and are displayed on the ESDPs.

## AIRCRAFT FUEL SYSTEM

### Introduction

The aircraft fuel system stores fuel in integral wing tanks and delivers filtered fuel to the engine fuel system at a rate and pressure in excess of the maximum engine requirement. The system maintains fuel supply during engine starting and all ground and flight conditions including aerobatics. An aerobatic tank (21 lb capacity) is located in the forward left fuselage. It provides uninterrupted fuel flow to the engine during negative G conditions and for the maximum permitted period (60 seconds) of inverted flight. The total useable capacity of the system is 535 litres (925 lb). The storage capacity of the aircraft fuel supply system can be increased by installing underwing tanks on the centre hardpoints under each wing.

### Fuel Storage

An integral tank is formed in each wing structure by sealing off part of the wing cavity forward of the main spar. The rear inboard part of the tank space is enclosed by a vertical wall to form a collector tank. Fuel feed from wing tank to collector tank is by transfer jet pump in the wing tank. Each wing tank is refueled through an overwing filler point using gravity feed. The collector tank is filled from the wing tanks, through the flap valves, whilst refueling. The filler is positioned far enough inboard to ensure that sufficient air space remains within the tank, after refueling, to allow for fuel thermal expansion and tank venting. The bottom of the collector tank forms the wing tank sump and is connected to a drain valve for fuel sampling, water contamination drainage and tank draining. Fig 1-10 shows the fuel system layout and position of indicators and controls.

### Fuel Supply

During normal engine running, fuel is delivered from the wing tanks to the engine high pressure fuel system by:

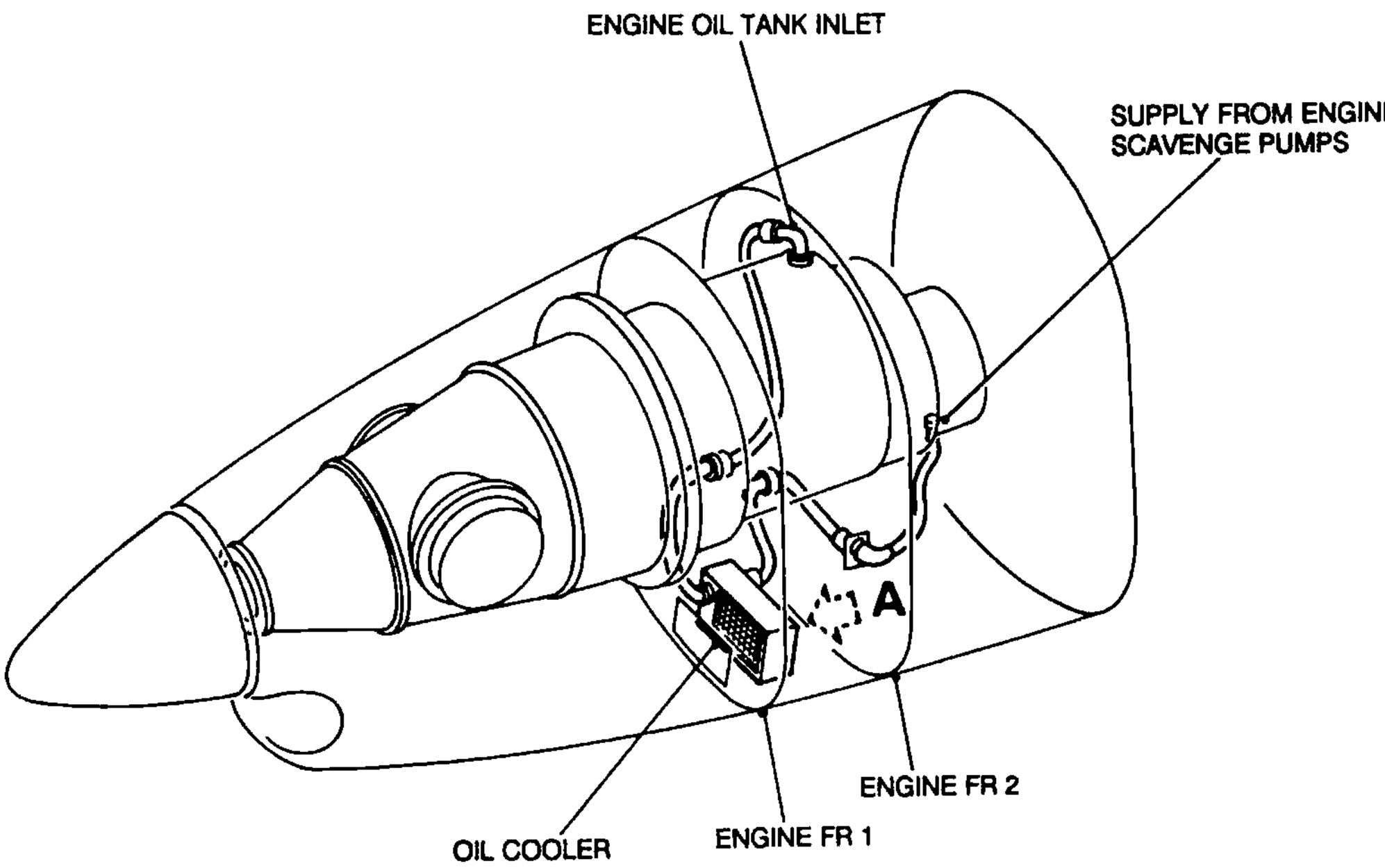
a. a transfer jet pump in each wing tank which supples fuel to a collector tank formed at the inboard end of each wing tank,

b. a delivery jet pump in each collector tank which supplies fuel to the engine-driven fuel pump (EDP), and

c. the EDP, mounted on the engine accessory gearbox, which receives the combined flows from the delivery jet pumps and supplies the fuel to the engine. The EDP also supplies the motive fuel flow to operate the two delivery jet pumps and the two transfer jet pumps.

For engine starting, when the EDP is not being driven, and as a back-up system should the normal delivery system fail, an alternative delivery system is provided. This system is activated by the low pressure switch and utilises electrically operated booster pumps; one

# ENGINE OIL COOLER



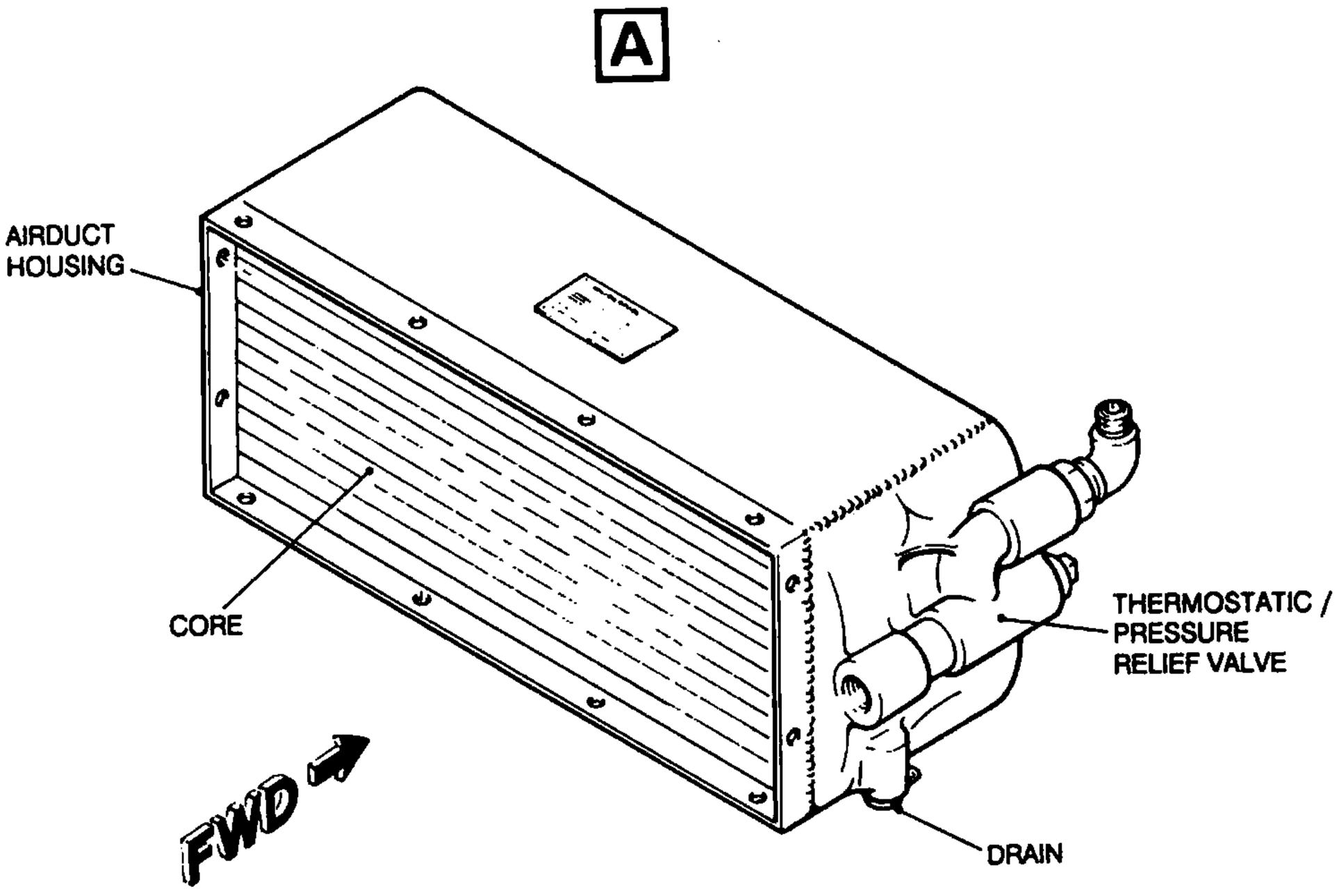




*Figure 1-9*

# Fuel System Layout Indicators And Controls

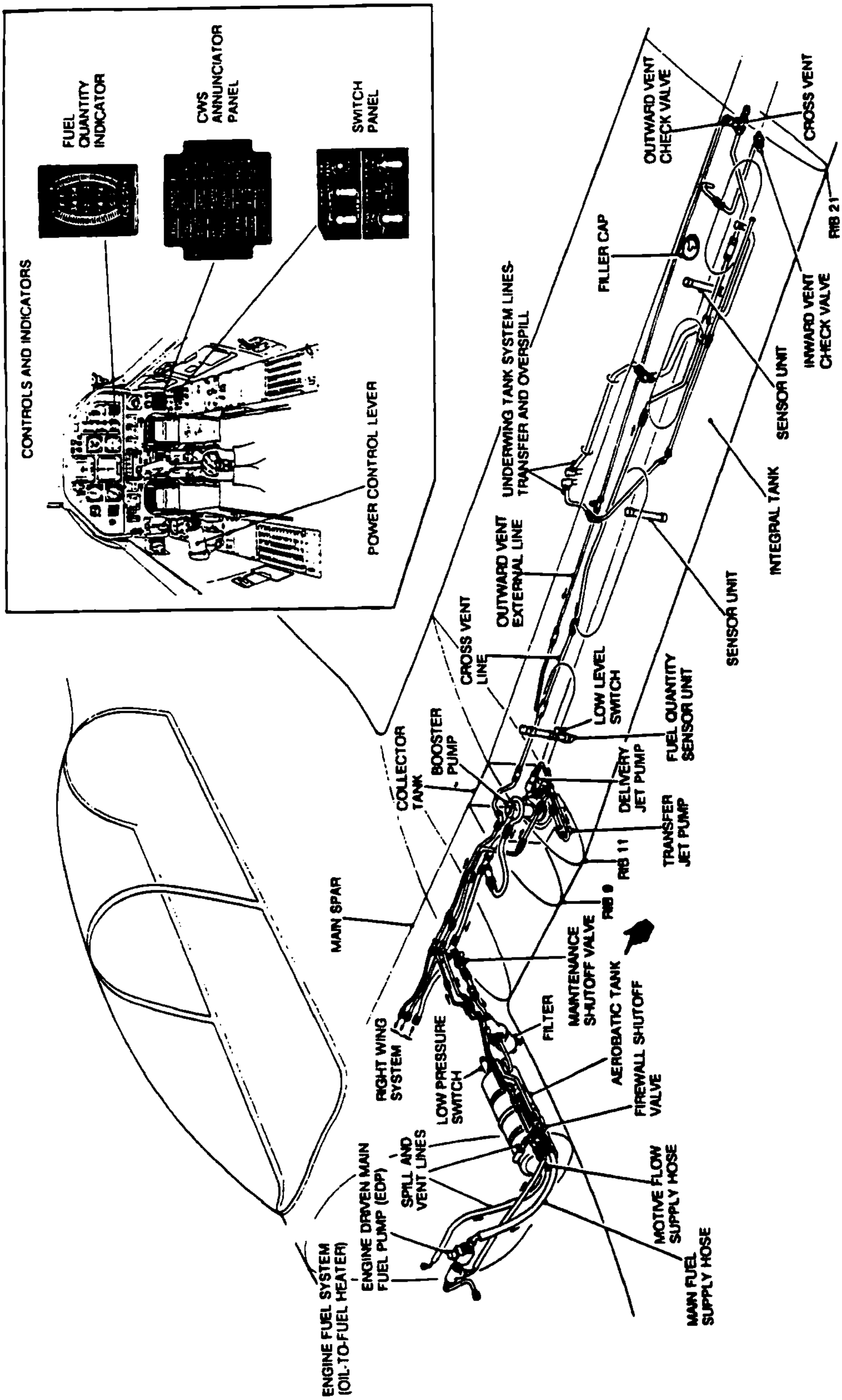
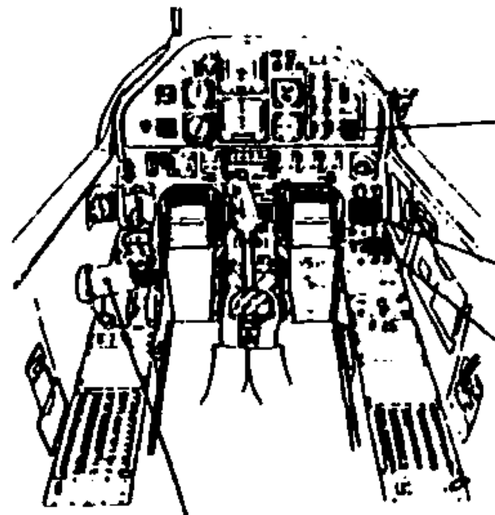



*Figure 1-10*

CONTROLS AND INDICATORS



FUEL QUANTITY INDICATOR

CWS ANNUNCIATOR PANEL

SWITCH PANEL

POWER CONTROL LEVER

ENGINE FUEL SYSTEM (OIL-TO-FUEL HEATER)

ENGINE DRIVEN MAIN FUEL PUMP (EDP)

SPILL AND VENT LINES

LOW PRESSURE SWITCH

FILTER

MAINTENANCE SHUTOFF VALVE

AEROBATIC TANK

FIREWALL SHUTOFF VALVE

MOTIVE FLOW SUPPLY HOSE

MAIN FUEL SUPPLY HOSE

MAIN SPAR

RIGHT WING SYSTEM

RIB 9

RIB 11

COLLECTOR TANK

BOOSTER PUMP

DELIVERY JET PUMP

TRANSFER JET PUMP

CROSS VENT LINE

LOW LEVEL SWITCH

FUEL QUANTITY SENSOR UNIT

OUTWARD VENT EXTERNAL LINE

UNDERWING TANK SYSTEM LINES-TRANSFER AND OVERSPILL

FILLER CAP

OUTWARD VENT CHECK VALVE

SENSOR UNIT

INTEGRAL TANK

SENSOR UNIT

INWARD VENT CHECK VALVE

CROSS VENT

RIB 21

in each collector tank. The booster pumps supply fuel to the engine fuel system via a bypass in the EDP, and also provides motive flow to the transfer jet pumps. Fig FO1-7 shows a schematic layout of the aircraft fuel system.

## Fuel System Operation

With the PCL in the OFF position, the booster pumps are activated by selecting the BOOST PUMP switches to ON. Booster pump activation is indicated by illumination of the L and R FUEL P advisory captions on the Central Warning System (CWS) annunciator panel. The fuel flow from each booster pump passes through a check valve and the combined flow is delivered into the aerobatic tank via the maintenance valve and the fuel filter. As the aerobatic tank fills, any trapped air is vented through the spill and vent line, back to the wing tanks via the cross vent line. A restrictor in the spill and vent line allows rapid venting of air but restricts fuel flow sufficiently to maintain system pressure. A low pressure switch is mounted on the outlet from the aerobatic tank and is set to operate if system fuel pressure drops. When the aerobatic tank is full, fuel under booster pump pressure is fed from the aerobatic tank, through the firewall shutoff valve and through the EDP bypass to the engine fuel system. Fuel under booster pump pressure is also supplied back to the transfer jet pumps, ensuring that the correct fuel levels in the collector tanks are maintained. The delivery jet pumps are inoperative; check valves in the pumps being closed by booster pump pressure. As the engine fuel system is now primed and being supplied by the booster pumps, the engine can be started. After engine start the EDP is operating, and the BOOST PUMP switches can be selected to ARM. The L and R FUEL P advisory captions on the CWS annunciator panel will extinguish. Fuel is now being supplied by the EDP to the engine fuel system. A pressure relief valve incorporated in the pump allows any excess pressure in the EDP outlet to be relieved into the EDP inlet. The EDP also delivers a sufficient flow back to the delivery and transfer jet pumps, to provide the jet pumps with motive flow. The motive flow induces the delivery and transfer jet pumps into operation. The transfer jet pumps supply fuel into the collector tanks, while the delivery jet pumps supply fuel to the EDP, through a check valve, by the same route as booster pump flow. Should the fuel system pressure drop during normal operation, the fuel pressure switch contacts will close. This energises a relay, activates the CAUTION PRESS TO RESET light and illuminates the FUEL PX warning caption on the CWS annunciator panel. Contacts in this relay will close to activate both booster pumps to restore system pressure. As soon as the system pressure rises, the pressure switch contacts will open and the FUEL PX warning caption will extinguish. Activation and cancellation of the FUEL PX caption occurs almost instantaneously. A hold-on circuit (through the PMP RESET buttons) ensures that the relay remains energised and the booster pumps operating after the pressure switch contacts open. The electrical supply to the booster pumps causes illumination of the L and R FUEL P advisory captions located on the CWS annunciator panel. These advisory captions will remain illuminated until the booster pumps are deactivated. De-selection of the booster pumps after automatic activation is accomplished by pressing the PMP RESET button.

## Operation During Inverted Flight

During inverted flight or other negative G conditions, fuel will cease to be drawn from the wing tanks as fuel surges away from the inlet to the delivery jet pumps. However, the EDP will continue to supply fuel to the engine by drawing fuel from the aerobatic tank via an alternative inlet which opens during inverted flight or other negative G conditions. Sufficient fuel is available in the tank to supply the engine during the period of inverted flight. During inverted flight, fuel pressure falls and the pressure switch activates the booster pumps. On reversion to normal flight conditions, fuel supply from the wing tanks is restored. Any air which entered the aerobatic tank will be vented through the tank spill and vent line, allowing the tank to refill within a few seconds. The booster pumps can then be deactivated by pressing the PMP RESET button.

## Fuel Indicating System

Three capacitance-type sensor units and a low-level switch in each wing tank supply signals to operate:

a. a dual scale LCD fuel quantity indicator on the Engine and Secondary Display Panel (ESDP) located in each cockpit, which shows the fuel contents of each wing tank as a fraction of total tank capacity. This indication is temperature compensated and accurate to ±2%, and

b. a L and R FUEL L illuminating warning caption on each cockpit annunciator panel which indicates when fuel in the affected tank falls below approximately 60 lbs.

**NOTE**

If both fuel low level lights illuminate simultaneously, the useable fuel remaining will be approximately 205 lbs: comprising 60 lbs per fuel tank, and 85 lbs in the collector tanks, aerobatic tank, plumbing and filters.

A low pressure switch in the fuel delivery line operates under low fuel pressure conditions to illuminate a FUEL PX warning caption on each cockpit annunciator panel. Actuation of booster pumps is indicated by illumination of a L or R FUEL P advisory caption on each cockpit annunciator panel.

A flowmeter in the fuel supply line to the engine provides signals to operate a digital fuel flow and fuel used indicator on the ESDP in each cockpit. The flowmeter is accurate to ±2%.

**NOTE**

Under normal operations, the digital fuel remaining indication should be used as a primary fuel remaining reference supported by periodic checks with the fuel quantity analogue system.

**CAUTION**

If a significant dicrepancy exists between the digital and analogue fuel indicating systems and doubt exists as to the actual amount of useable fuel remaining, the aircraft is to be landed at the nearest suitable airfield.

The digital FUEL QTY and FUEL USED indications can be reset to indicate fuel quantity in the wing tanks and zero respectively by operating the F USED RESET switch located adjacent to the GENERATOR BUS circuit breaker panel. The FUEL QTY indications can be increased or decreased by operation of the F QTY SET switch found next to the F USED RESET switch as would be required when underwing tanks are fitted.

**NOTE**

- The PC9/A fuel quantity reset system is accurate only when the fuel tanks are full. Any attempt to reset the fuel quantity system with less than full fuel tanks will almost certainly result in an inaccurate fuel remaining indication on the digital fuel quantity indicator.

- The fuel quantity reset system is to be operated only when the aircraft tanks are full either:

  **a.** on the ground, or

  **b.** on completion of external fuel transfer.

## Underwing Tanks

To increase the operational range of the aircraft, two jettisonable underwing tanks can be fitted. The tanks are suspended on racks, which can be installed on the underwing centre hard points. The usable capacity of each tank is 240 lt (415 lb). Fuel transfer from underwing tanks to wing tanks is accomplished by a booster pump installed in each underwing tank. Fuel delivery rate from the underwing tanks to the wing tanks is in excess of engine requirements. When a wing tank is full, excess fuel is returned to the associated underwing tank, via the overspill return line. The tanks are jettisonable if they are installed on the centre hardpoint and the battery busbar voltage is greater than 18 VDC or if they are installed on the inner hard-point and the generator busbar is energised. Fig 1-11 shows the underwing fuel tank system.

Pull-out, self-sealing couplings are installed in the fuel supply and overspill return lines, ensuring no loss of fuel from the wing tanks should the underwing tanks be jettisoned. Electrical power is supplied to the tanks via two self-sealing couplings, which separate should the tanks be jettisoned. Simultaneous jettisoning of the underwing tanks is achieved by operating a guarded EMER ALL JTSN push-switch located in the front cockpit.

# *Underwing Fuel Tank System*

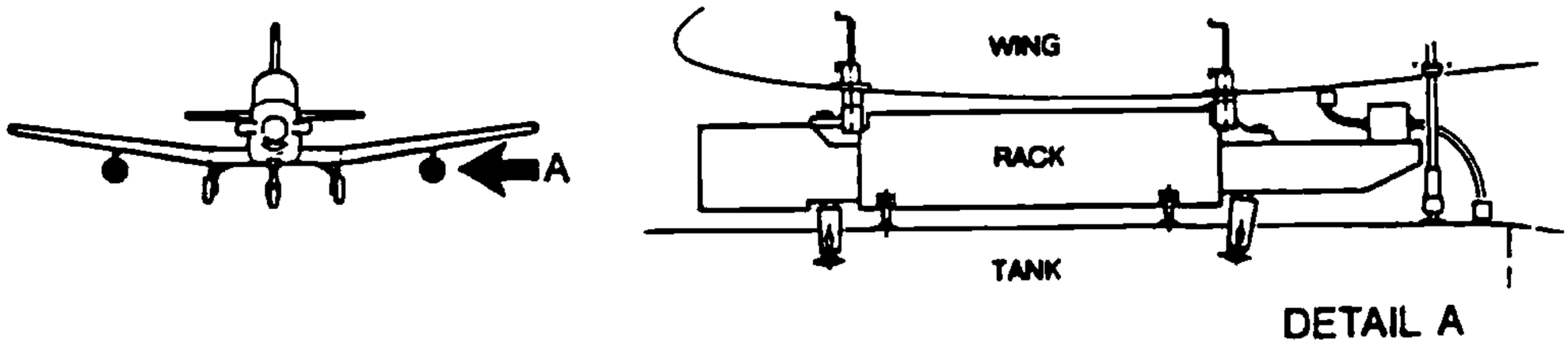



*Figure 1-11*

## ELECTRICAL SUPPLY SYSTEM

### Description

The electrical supply system for the aircraft consists of a primary 28 VDC system, a secondary 24 VDC system, and a dual output (26 V and 115 V, 400 Hz) AC system. DC electrical power is supplied by the dual-role starter-generator with a rated output of 28 V. The secondary DC power source is a 24 VDC 40 Ah nickel cadmium battery, which can be used for engine starting and which will provide temporary DC supply in the event of engine failure or malfunction of the generator in-flight. Under normal operating conditions, the starter-generator provides all necessary DC power and will also recharge the aircraft battery as necessary. DC power from an External Power Unit (EPU) can be supplied to the DC busbars via the external power receptacle. When external power is supplied to the aircraft DC system, the battery is isolated from the EPU system to prevent the possibility of the battery discharging through the EPU. LCD indicators, annunciator displays and power sockets installed in each cockpit provide the means to observe and test system operation. The AC/DC electrical system schematic is shown at Fig FO-8.

### Starter Generator

The starter-generator is mounted on the engine accessory drive housing. Generator output is regulated at 28 VDC, up to a load of 200 A. In normal operation the generator provides 28 VDC power, via the busbars to the aircraft systems, including a trickle-charge to the aircraft battery. When an EPU is connected the generator and battery are isolated from the busbars. If the generator output voltage exceeds 32 V the voltage regulator automatically disconnects the generator from the generator busbar and the battery then takes over to supply the busbars.

### Battery

The 24 V 40 Ah nickel-cadmium battery is connected to the DC power system and is brought on-line when the BAT switch is selected ON. A battery over-temperature warning system is provided.

### DC Power Distribution

A dual busbar arrangement is used for the main DC distribution system. Power provided by the engine-driven generator, the aircraft battery or an external supply is distributed to busbars (identified as 'battery' and 'generator' busbars), which are normally connected in parallel through a busbar cross-tie circuit breaker. A battery busbar and a generator busbar are located in each cockpit on the left and right sides respectively. In the event of a ground fault in either the battery or generator supply system the busbar cross-tie operates to break the busbar cross-tie connection, isolating the faulty system. Reset is achieved by opening and closing the BUS TIE C/B, located on the front BATTERY BUS

C/B panel. A 'battery direct' busbar is permanently connected to the aircraft battery and is used to provide power for selected lighting and emergency systems. A DC VOLTS and a DC AMPS LCD indicator, located on the Engine and Secondary Instrument System Display Panel (ESDP) in each cockpit, allows visual observation of the busbar voltage and aircraft load current. Captions on the annunciator panels of the Central Warning System (CWS) illuminate to indicate battery over-temperature (BAT HOT) and generator failure (GEN), generator busbar failure (GEN BUS), battery busbar failure (BAT BUS), and inverter failure (INV). Illumination of the GEN BUS or BAT BUS illuminates the WARNING PRESS TO RESET light. The BAT HOT, GEN and INV captions illuminate the CAUTION PRESS TO RESET light.

### External Power

External power is normally used for starting and can be connected to the aircraft through the EPU receptacle. located on the left hand rear fuselage. An EPU with a regulated 28 VDC output (nominal) should be used whenever possible.

**CAUTION**

To avoid malfunction of systems due to voltage peaking, the EPU should be set to 28 VDC and switched on before it is connected to the aircraft.

### Radio Busbars

A 'battery radio' busbar and a 'generator radio' busbar are located in each cockpit. Each radio busbar is connected to its main DC distribution busbar in the front cockpit only, via a remote circuit breaker and a filter. Two toggle switches (2 position), placarded AVIONICS BAT and GEN, are located in the front cockpit of the LH console. Each switch is in series with a RADIO BUS C/B. When an avionics switch is selected ON the appropriate remote circuit breaker operates and connects DC power to its respective radio busbar. In the event of an avionics switch malfunction in the ON position, the remote circuit breaker can be de-energised to disconnect DC power by opening the RADIO BUS C/B which is in series with the defective switch. The RADIO BUS C/Bs are located in the front cockpit, one on the BATTERY BUS C/B panel and the other on the GENERATOR BUS C/B panel.

### Battery Direct Busbar

The battery is permanently connected to the battery direct busbar. This arrangement provides an uninterrupted supply for the aircraft clocks, emergency hydraulic pressure gauge, emergency flaps and service compartment lighting. To avoid power drain on the battery, the Environmental Control System cooling fan can be operated only when the generator is on-line or when external power is connected to the aircraft.

### Command Control Transfer

A magnetic latching system allows command selection, from either cockpit, of the battery and generator switches. It is not possible for the battery or generator switches to be ON in both cockpits at the same time and a selection in one cockpit can be cancelled and countermanded from the other cockpit.

### Protection Systems

Undervoltage protection is provided to switch the generator off whenever the voltage falls below 23.6 VDC for approximately 1.5 seconds. Overload protection switches the generator OFF if voltage exceeds 150% of generator rating for approximately 10 seconds.

Manual reset of the system is achieved by depressing the GEN RESET switch, located on the right side console in both cockpits, for one to two seconds. Although multiple attempts to reset the generator will not prevent a fire risk, a maximum of five reset attempts is recommended.

The undervoltage and overload protection systems can be tested by exercising the OVERLOAD/OVERVOLT test switch, located next to the RESET button. Switch selection to OVERLOAD must be maintained for approximately 10 seconds to cause the generator to trip. The OVERVOLT selection will trip the generator within a few seconds. The test is normally done on the first flight of the day. The EPU receptacle has polarity reversal protection which prevents negative DC power being connected to the external power system. The aircraft battery is fitted with a temperature sensing device which detects overheat conditions. As a precaution against switch failure, two thermal switches wired in parallel are used in the sensing device. Closure of either switch completes a circuit in the CWS, causing the BAT HOT caption on the annunciator panels to illuminate.

### AC Electrical Power System

The AC electrical power system consists of two identical transistorised static inverters and an inverter selector switch. The system is protected by circuit breakers located in the front cockpit and is monitored by the CWS.

### Static Inverters

The static inverters, located in the avionics compartment, convert 28 VDC electrical power to 115 VAC and 26 VAC, 400 Hz. Each inverter is capable of supplying the total load requirements of the aireraft AC systems. Only one inverter is selected on-line at a time with the remaining inverter used as a standby source of power. The inverters, identified as INVERTER BAT and INVERTER GEN receive their 28 VDC

input separately from different busbars. This prevents complete loss of aircraft AC power in the event of a single DC busbar failure. A four pole, three position switch is used for inverter selection. The switch is located in the front cockpit, LH console, and is placarded INVERTER with marked positions BAT-OFF-GEN. The CWS monitors the 26 VAC output of the on-line inverter and the caption INV, on the front and rear cockpit CWS annunciator panels, illuminates when the voltage falls below 13 VAC. The 7.5 A circuit breaker is located in each of the front cockpit C/B panels. The circuit breakers are placarded INV BAT (battery busbar C/B panel) and INV GEN (generator busbar C/B panel). Both inverters are supplied with a positive DC input when power is connected to the DC distribution system. When the INVERTER switch is selected to the mid (OFF) position the inverters are prevented from converting the DC input to AC by internal inhibiting circuits. The inhibiting circuits are controlled by the inverter's remote on/off facility and are de-activated by connecting the remote on/off circuit to ground. Selecting the INVERTER switch to BAT or GEN enable the selected inverter and connects the nominal outputs of 115 AC and 26 VAC, 400 Hz to the aircraft AC distribution system.

### DC Electrical System Power Requirements

The following aircraft systems require DC power:

a. hydraulic system auto-selection, indication and warning,
b. landing gear selection, position indication and warning,
c. trim system selection, position indication and warning,
d. flap normal/emergency selection and position indication,
e. engine starting,
f. ignition,
g. engine and secondary instruments display panel,
h. propeller speed selection and indication,
i. flight instruments,
j. fuel booster pumps selection, actuation and indication,
k. fuel quantity indication system,
l. environmental control system selection and control,
m. heating elements of: pitot tube, static ports, AOA,
n. seat adjustment,
o. lighting,
p. master caution and annunciator lights,
q. aural warning, and
r. inverter operation.

### AC Electrical System Power Requirements

The following aircraft systems require AC power:

a. attitude indicator,
b. compass, and
c. avionics equipment.

## LIGHTING SYSTEMS

### Description

Interior and exterior lighting, shown at Fig 1-12, consists of the following:

a. Instrument, panel and map lights.
b. Utility cockpit lights.
c. Compartment lights.
d. Navigation lights.
e. Landing lights.

### INTERIOR LIGHTING

#### Instrument Lights

The instruments and panels are fitted with internal 5 VDC or 28 VDC white lighting, with map lights located on the left and right sidewalls of each cockpit. The map lights are equipped with rotatable lens cover to provide illumination area adjustment. During normal daylight conditions several indicator lighting circuits are connected to the cockpit generator busbar via the OFF contact of the INSTR LIGHTS switch. When the INSTR LIGHTS switch is selected ON the supply to these indicators is routed via an alternative (dimming) circuit, or is disconnected and the indicator illuminated by panel lighting only. The intensity of the instrument lighting is controlled by dimmer potentiometers on the right side consoles, via a master ON/ OFF switch. A separate dimmer is provided for the main panel and the side panels. A schematic of the cockpit lighting is shown at Fig 1-13. The following indicator lights are dimmed when the instrument lights are switched on with the master switch:

a. Nosewheel steering.
b. Air brake.
c. Landing gear position indicators.
d. CWS annunciators, MASTER CAUTION and MASTER WARNING lights.
e. Internal communications equipment lights.
f. Navigation advisory lights (MKR, BCN).
g. AOA indexer.

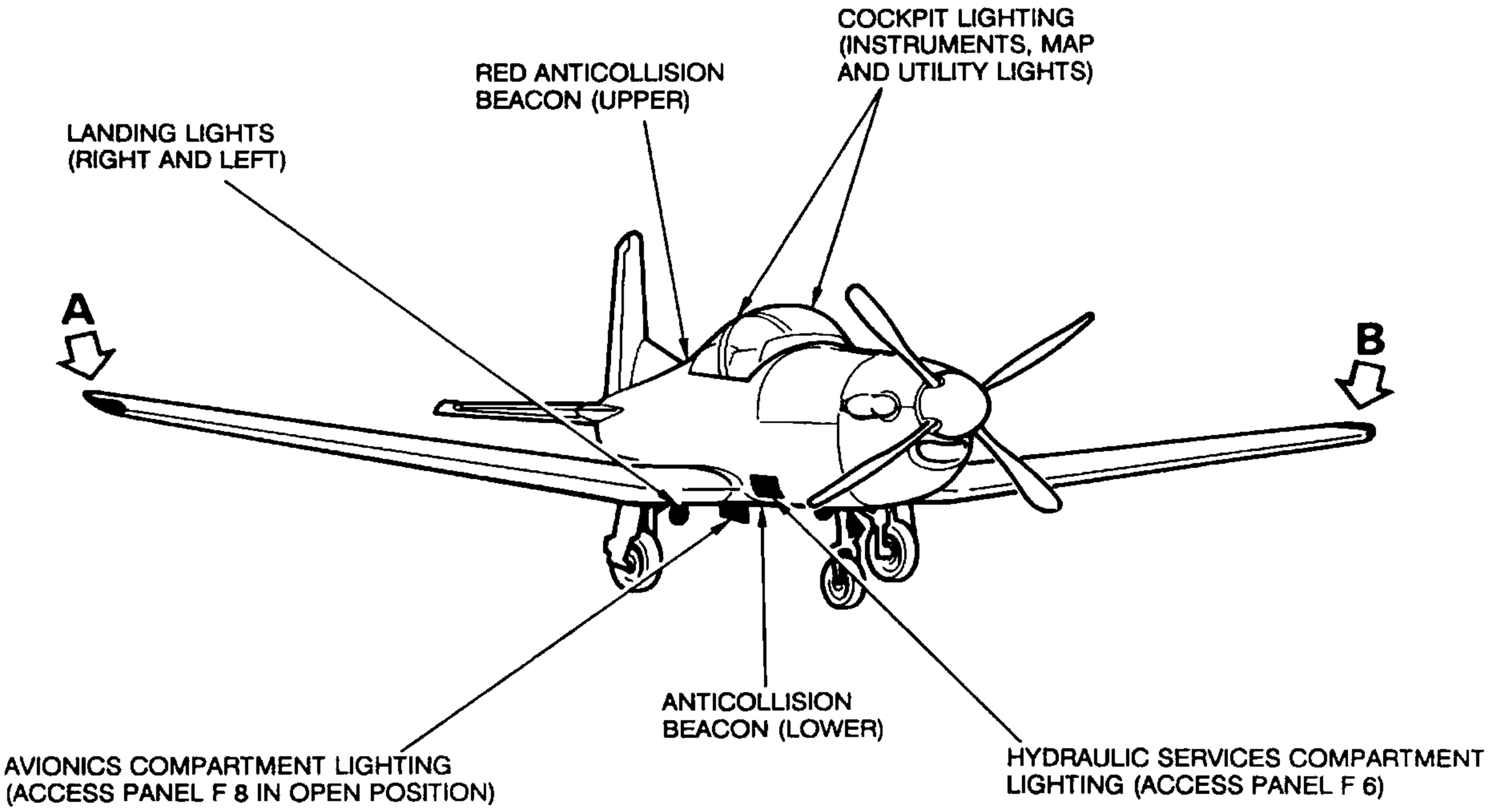
#### Utility Cockpit Lights

A dimmable flood light, held in a snap mounting, is located on the right side cockpit wall. Each light is fitted with a flexible lead and can be removed from its mounting and moved around the cockpit. An additional mount is located on the right side canopy sill of each cockpit, forward of the canopy breaker knife. The circuit breakers are located on the battery busbar and marked UTILITY LIGHT.

#### Compartment Lights

The hydraulic and electrical/avionic equipment

# AIRCRAFT LIGHTING



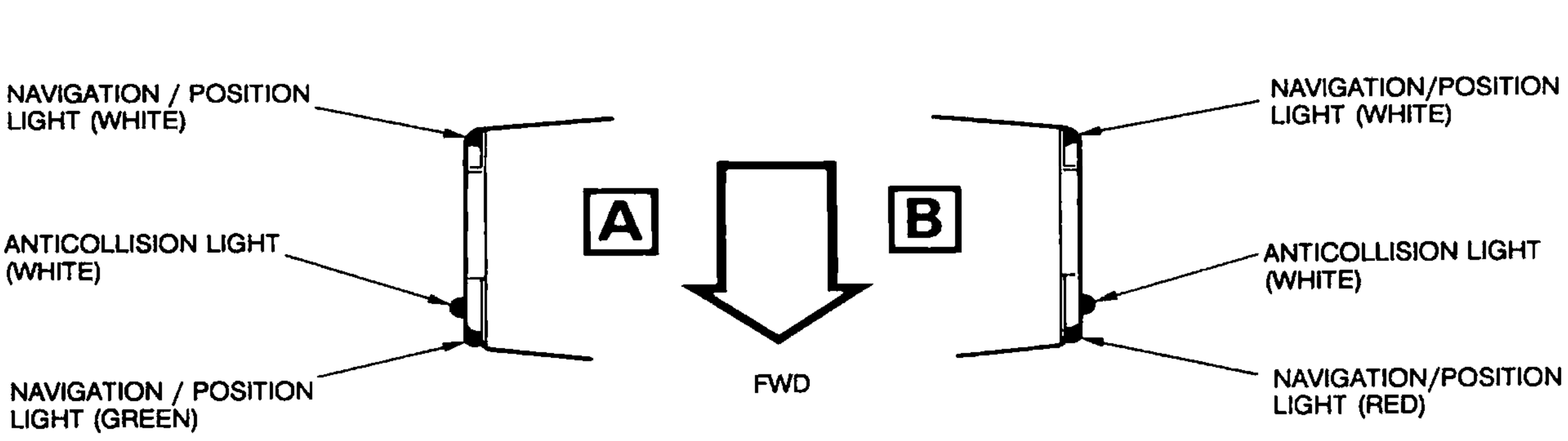




***Figure 1-12***

compartments are lit by lamps controlled by an access door-operated switch. The light in the hydraulic compartment is positioned to enable the hydraulic reservoir quantity indicator to be checked.

## EXTERIOR LIGHTING

### Navigation Lights

The following comprises the navigation lighting:

a. A red navigation light on the forward part of the left wing tip, and a green navigation light on the forward part of the right wing tip.

b. Single white navigation lights fitted on the rear section of both wing tips.

c. Single anti-collision beacons fitted on the upper and lower fuselage.

d. Single white strobe anti-collision lights fitted on the front section of both wing tips.

A three position toggle switch is located on the front cockpit right side console. The upper and lower beacons are selected on with both the centre and the forward selections (BCN or STROBE AND BCN). The strobes and beacons are selected on with the forward selection only (STROBE AND BCN). A two position toggle switch, marked NAV, controls power to the red, green and white navigation lights on each wing tip. The power for the navigation lights is via the NAV LIGHT C/B on the front generator busbar. The power for beacons is via the STROBE BEACON C/B on the front generator busbar. The strobe lights are powered via the front battery busbar via the STROBE LIGHT C/B.

# COCKPIT LIGHTING

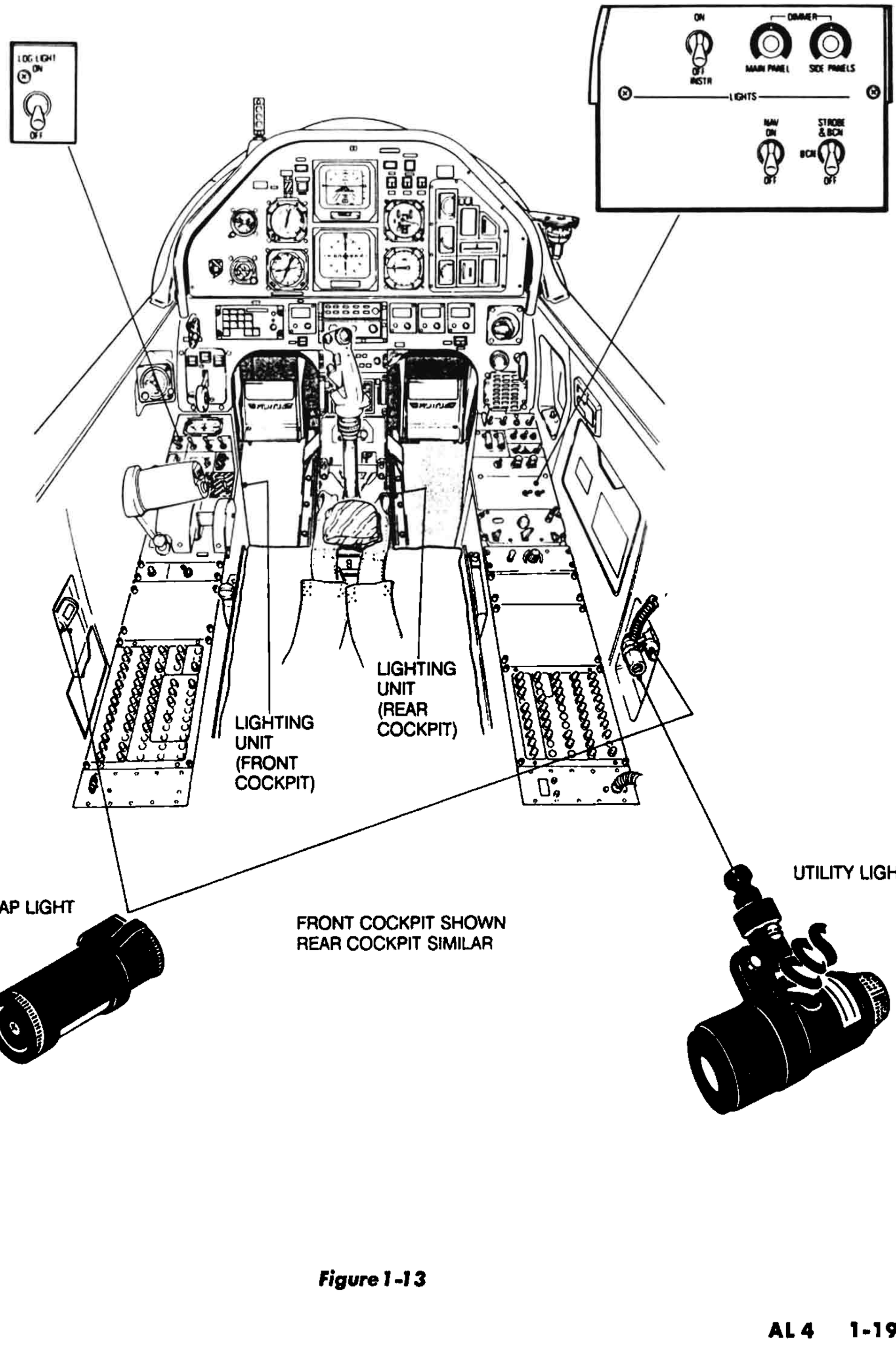



*Figure 1-13*

### Landing Lights

Single 250 W landing lights are located at the rear of each main landing gear leg compartment. The light retracts and extends with the landing gear. The landing light relays are routed through the rear cockpit landing gear control unit and the lights will not operate until all landing gear is fully down and locked. The right-hand light is pointed downwards a few degrees to provide optimum illumination for taxiing. The lights are controlled by a two position toggle switch located on the forward left side console of the front cockpit. The left light is powered by the front battery busbar and the right light is powered by the front generator busbar with corresponding circuit breaker locations.

## HYDRAULIC SYSTEM

### System Description

The hydraulic system provides hydraulic power to operate the landing gear, main gear doors, flaps, air brake and nose wheel steering. The system comprises an engine-driven pump, a power package, a selector manifold, a sampling valve, a firewall shutoff valve and an emergency package. A variable stroke swash-plate type pump is installed on and driven from the engine accessory gearbox. The pump is regulated to 3,000 psi. The N2 hydraulic indicator on the ESDP will read emergency accumulator pressure as supplied by the power package when the engine is running. Main hydraulic system pressure can only be monitored by illumination of the HYDR PX light. The power package is of modular construction and comprises an integral reservoir, pressure and return line filters, pressure and return line relief valves, a low pressure switch, a system selector valve, three check valves, pressure and return line servicing rig connections and a system air bleed valve. The assembly is installed in the hydraulic services compartment beneath the front cockpit floor. Access to the compartment is via a hinged access panel, forward of the right wing leading edge. A fluid quantity indicator rod with four, coloured, indicator bands is installed in the reservoir and protrudes forward from the power package. Fig FO-9 shows a schematic diagram of the aircraft hydraulic system.

### Selector Manifold

The selector manifold consists of a manifold housing and five solenoid valves which control the user systems. The manifold is installed in the hydraulic services compartment, aft of the power package.

### Firewall Shutoff Valve

The firewall shutoff valve is installed in the system return line, between the reservoir and the hydraulic pump. If a fire occurs in the engine compartment the valve can be closed to shutoff hydraulic fluid supply to the fire. The valve is interconnected and operates in conjunction with the fuel shutoff valve and the environmental control system (ECS) shutoff valve. Activation of the firewall shutoff valve is via the firewall shutoff handle on the left side console of the front cockpit. The firewall shutoff handle may be reset in flight.

### Emergency Package

The emergency package is of modular construction and comprises an accumulator, a nitrogen charging valve, two restrictors (protected by in-line gauze screens), a LG emergency extension selector valve, a pressure operated control valve, a pressure transducer, two pressure relief valves, a manual pressure release valve, a check valve and the flap emergency extension selector valve. The assembly is installed on the rear face of the rear wing spar in the air brake compartment.

### System Operation

With the engine running, the hydraulic pump supplies pressurised fluid to the power package where the fluid passes through a filter. A low pressure switch in the power package operates at 2,300 psi to complete the electrical circuit to energise the system selector valve. Pressurised fluid is routed from the power package to the selector manifold, nose wheel steering solenoid valve and emergency package accumulator. The selection of a user system will actuate the respective solenoid to direct pressure fluid to the selected system actuator. Return fluid from the actuator is routed to the reservoir in the power package. The return fluid passes through a low pressure filter in the power package. An indicating rod on the return line filter protrudes when the filter become 85% clogged, showing bypass valve operation. Pressurised fluid is supplied from the system pressure line to the nitrogen-charged accumulator in the emergency package to increase the volume of fluid in the accumulator. A pressure release valve allows the hydraulic pressure in the accumulator to be discharged manually to the system return line. This is achieved by pulling out (and holding) on a pressure release handle located in the hydraulic services compartment. This permits checking of the emergency package pre-charge N2 pressure. If system pressure falls below 1,850 psi the low pressure switch opens and, after a 2 second delay, de-energises the system selector valve solenoid, and the main pressure gallery in the power package is closed. This makes the main hydraulic system inoperative. The 2 second delay permits normal system service fluctuations. The CAUTION PRESS TO RESET light and the HYD PX caption will illuminate. Pressure relief is provided at 3,500 psi.

### Emergency Operation

In the event of a failure in the main hydraulic system, the landing gear and flaps can be extended using hydraulic pressure from the emergency package accumulator. NWS is unavailable. Manual operation of the landing gear emergency extension selector valve, via the EMER LDG GR handle in the front cockpit, directs pressure fluid to operate the shuttle valves in the landing gear and door actuators. The pressure fluid is then directed to the appropriate side of the actuators to open the doors and extend the landing gear. Electrical operation of the flaps emergency extension selector valve directs pressure fluid to the flaps actua-

# EXTERIOR LIGHT SCHEMATIC

*Figure 1-14*

(Deleted)

tor to extend the flaps to the TO (take-off) or LAND position. The landing gear and flaps can not be retracted if extended using the emergency method. The emergency system is totally independent of the main system, and utilises separate pipelines.

## FLIGHT CONTROL SYSTEM

### Introduction

Aircraft primary flight control surfaces consist of the ailerons, rudder and elevator. The control surfaces are manually operated from a conventional dual control column and rudder pedal arrangement, with connections to the control surfaces through a system of control rods, bellcranks, cables and levers. The location of flying controls and indicators is shown at Fig 1-15.

### Elevators

The elevator is mechanically operated by longitudinal movement of either control column. The elevator is statically mass balanced by the use of weights in the elevator horns. The linkage also incorporates two coil springs to enhance stability in level flight.

### Ailerons

The ailerons are mechanically operated by lateral movement of either of the mechanically interconnected control columns. Each aileron is statically mass balanced by the use of balance weights bolted to the aileron leading edge beak. Aerodynamic balancing is achieved by the use of sealed hinges.

### Rudder

The rudder is operated by movement of either set of rudder pedals, the front pedals being connected to the rear pedals and the rear pedals connected to the rudder operating lever by cables. The rudder is statically mass balanced by weights installed in the rudder horn.

### Trimming system

Trimming control is provided on all three control axes for use during flight. All trims act in the conventional sense with the rear cockpit having priority if trims are simultaneously selected. DC electric motors are used to move the relevant tabs. A combined elevator, aileron and rudder trim position indicator is provided on the lower left instrument panel. A trim emergency switch is located on the left side console in each cockpit. In the event of a runaway trim actuator, selection of either emergency trim switch to INT RPT will de-energise all three trim actuators simultaneously.

### Elevator Trim

An electrically operated trim tab is installed on the right hand elevator. The tab is connected to an electrical actuator installed in the elevator, the actuator being controlled by the fore and aft movement of a combined elevator/aileron trim switch installed on the top of each control column.

### Aileron Trim

Fixed, ground adjustable static balance tabs are provided on each aileron. Pilot adjustable trim is provided by an electrical actuator on the centre bellcrank spring box assembly. The electrical actuator is controlled by lateral movement of a combined elevator/aileron trim switch installed on the top of each control column.

### Rudder Trim

A combined trim and anti-balance tab is installed on the rudder, mechanically linked to the rudder so that the tab moves in the same direction as the rudder, but at a greater deflection angle. In-flight this movement increases the aerodynamic forces acting on the deflected rudder, thus increasing the required pedal input loads. An electrical actuator, installed in the vertical stabiliser, is connected to the tab operating lever to move the rudder trim tab. The electrical actuator is controlled by the rudder trim switch installed on the PCL in each cockpit.

### Gust Lock

A centralised gust lock assembly is installed in the aircraft to allow the primary control surfaces to be locked in position when the aircraft is parked. The ailerons and rudder are held in the neutral position and the elevator in a nose down position when the lock is engaged. The gust lock is engaged by centralising the rudder and aileron controls, raising the yoke which is hinge mounted to the front cockpit centre console and moving the control column forward until the peg on the yoke engages in the adapter on the forward face of the control column. To disengage the gust lock the yoke is raised until the peg is clear of the adapter, the control column moved rearwards and the yoke lowered to the stowed position.

## WING FLAP SYSTEM

### Description

The aircraft is fitted with hydraulically operated, electrically controlled, split flaps, with a normal system for extending and retracting and an emergency system for extending the flaps only. The system comprises two flap selectors, two selector valves for normal operation, an emergency extension selector valve, a flap actuator, two flap position indicators and associated microswitches. A flap selector is installed in the left console in each cockpit. The selectors are interconnected by a bowden cable so that operation of one selector is duplicated by the other selector. Selectors are placarded FLAPS and have three positions UP, TO (take-off) and LDG corresponding to 0, 23 and 50 degrees of flap deflection respectively.

### Normal Operation

Normal flap selection is controlled by two switches adjacent to and operating from the front cockpit flap selector. For redundancy, the switch is provided with two 28 VDC supplies, one from the front battery busbar (FLAP CONT1 C/B) the other from the front

# FLYING CONTROLS AND INDICATORS

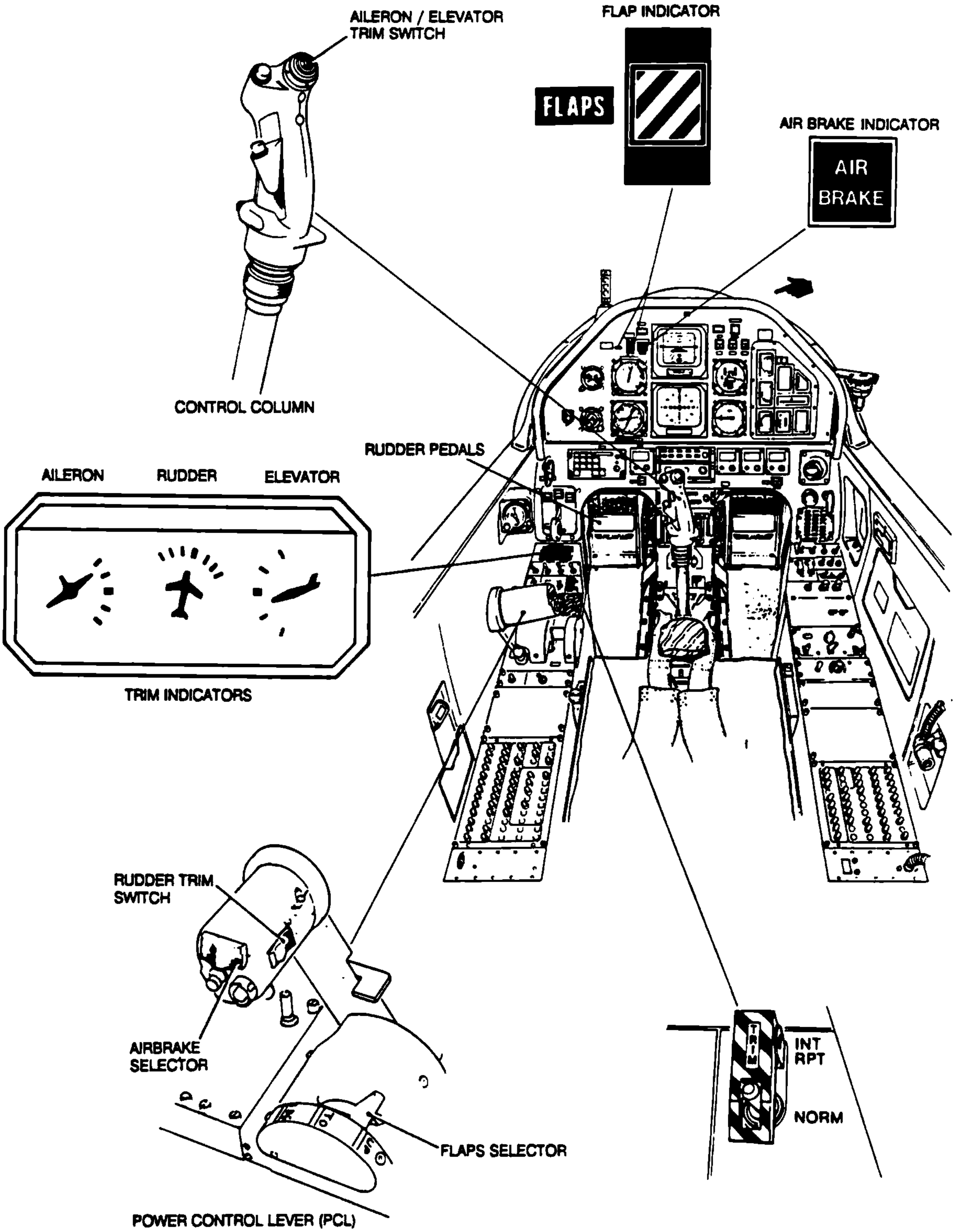



*Figure 1-15*

generator busbar (FLAPS CONT2 C/B). Two normal operation selector valves are solenoid operated and are located in the hydraulic power system selector manifold. For redundancy reasons, each selector valve solenoid has dual independent coils, one coil receiving its supply from the battery busbar, the other coil receiving its supply from the generator busbar. The three position hydraulic actuator is controlled by the selector valves.

### Emergency Operation

Emergency extension selection is also controlled by the front cockpit selector. The switch has no power supply during normal operation but is connected to the battery direct busbar (FLAPS EMERG C/B) via a microswitch when the EMER LDG GR handle is operated. The emergency selector valve is solenoid operated and is located in the hydraulic power system emergency pack.

### Flap Indicators

A magnetic-type FLAPS position indicator, located on the instrument panel in each cockpit, displays UP, TO or LAND captions. Captions are activated by the closure of one of three position indicating microswitches. Each relevant position indication microswitch is operated, at the appropriate flap position, by a separate cam. The three cams are mounted on the flap operating torque tube and the microswitches are mounted on the adjacent aircraft structure. The indicators display a cross-hatched pattern when the flaps are in transition between two positions or when electrical power is removed from the indicating system. The indicators are powered from the battery busbar (front) via the FLAPS CONT1 C/B. During emergency extension, the TO position microswitch provides control of the TO selection. The LAND position microswitch provides a flaps at LAND signal for a gear up/flaps land aural warning.

## AIR BRAKE SYSTEM

### Description

The aircraft is fitted with a flat plate air brake (located under the fuselage, behind the rear seat pilot), which can be extended into the airstream to increase aircraft total drag and hence increase the rate of airspeed deceleration. The air brake system is hydraulically operated and electrically controlled using hydraulic fluid at 3,000 psi pressure supplied by the aircraft hydraulic power system. The air brake system comprises two selectors and their associated selector switches located on the PCL, a selector valve, a hydraulic actuator, a relay, two air brake indicators, associated microswitches and a single plate-type air brake. The air brake has a single deflection angle of 70 degrees and cannot be selected to an intermediate position. The two position selector valve is solenoid operated and is installed in the hydraulic power system selector manifold. The two position hydraulic actuator is controlled by the selector valve. Power to the selector valve is supplied via the AIR BRAKE C/B on the front generator busbar. The indicator lights, powered from the 28 VDC instrument lighting bus, are fitted to the left of the EADI in each cockpit and illuminate whenever the air brake is extended.

### Air Brake Selectors

An air brake selector is located on the PCL in each cockpit. The selector has a two step rearward switch and a one step forward switch, operated by the thumb. The switch has a soft detent at the first rearward step position, and further thumb pressure on the switch passes the soft detent to operate the switch to the second step position. When released, the slide switch always spring-returns to the original neutral position. First pressure rearwards on the selector switch will select the air brake OUT. The air brake will retract when the switch is released. Second pressure rearwards on the selector switch will select the air brake out but the air brake will stay out when the switch is released. The air brake is selected IN by pressing the selector switch forward. The air brake automatically retracts when MAX power is selected at the PCL.

## LANDING GEAR SYSTEM

### System Description

The landing gear is a hydraulically actuated, retractable, tricycle-type system with a single wheel fitted to each gear leg. The system, shown at Fig 1-16, consists of the following components:

a. A tricycle undercarriage consisting of two inward-retracting, single wheel, oleo-pneumatic main legs and a rearwards-retracting, single wheel, oleo-pneumatic nose leg.

b. A hydraulically actuated extension and retraction system, including an emergency extension system.

c. A mechanically actuated, hydraulically operated nose wheel steering system.

d. A mechanically actuated, hydraulically operated, multi-piston brake system operating on each main wheel.

e. An electrical control and indicating system.

### System Operation

Normal landing gear extension and retraction is hydraulically actuated and controlled by an electrically sequencing system. Hydraulic fluid at 3,000 psi is supplied to the system by the hydraulic power system. In the normal operation, microswitches installed in the electrical circuits to the selector valves control the door opening/closing sequence and landing gear extension/retraction. The extension and retraction system comprises two landing gear control units, two selector valves, three landing gear leg actuators, a landing gear doors actuator, two main leg up-locks, a nose leg down-lock spring strut, main door locks and the operating/sequencing circuits. The extension and retraction sequence is shown at Fig 1-17.

# LANDING GEAR SYSTEM

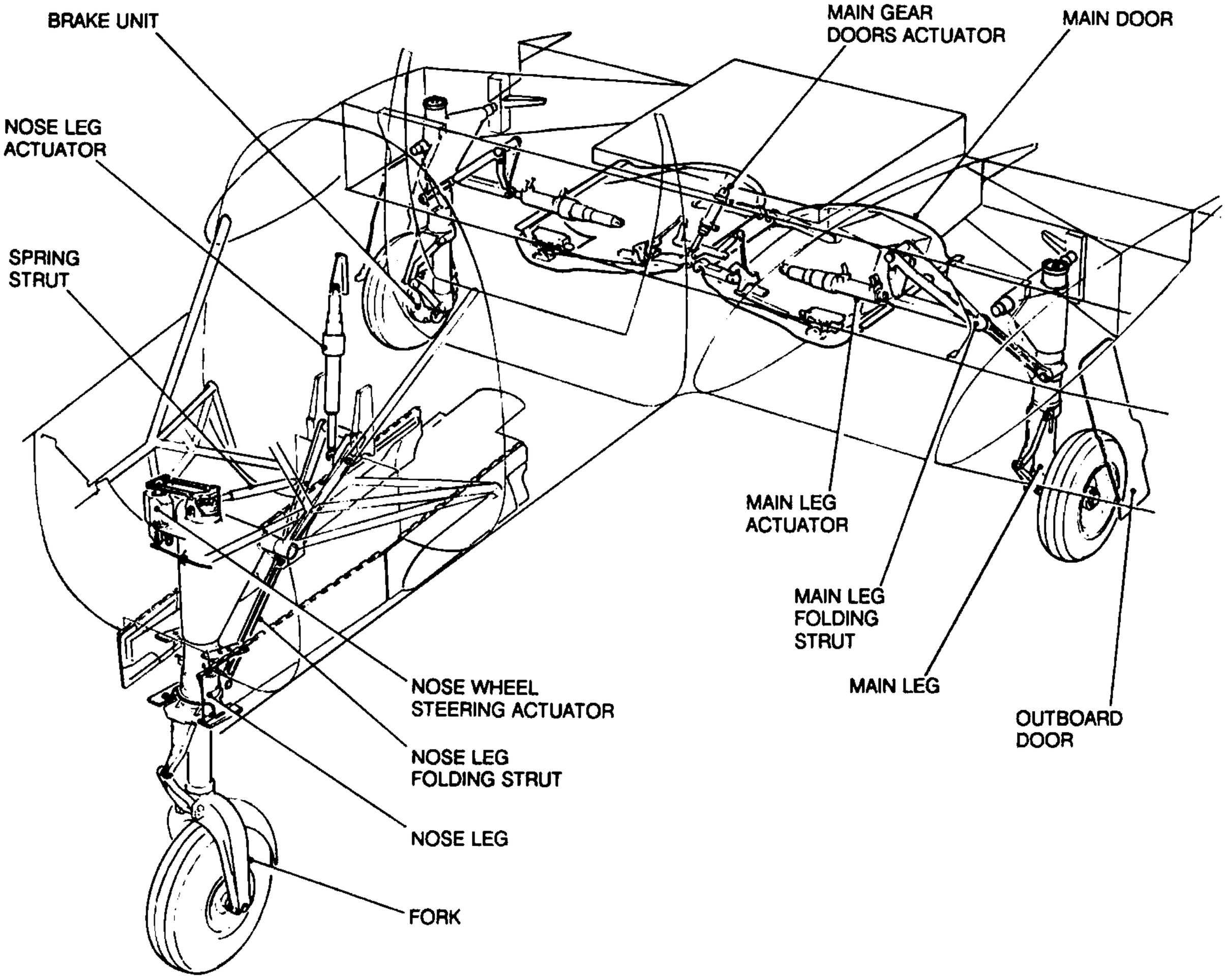



***Figure 1-16***

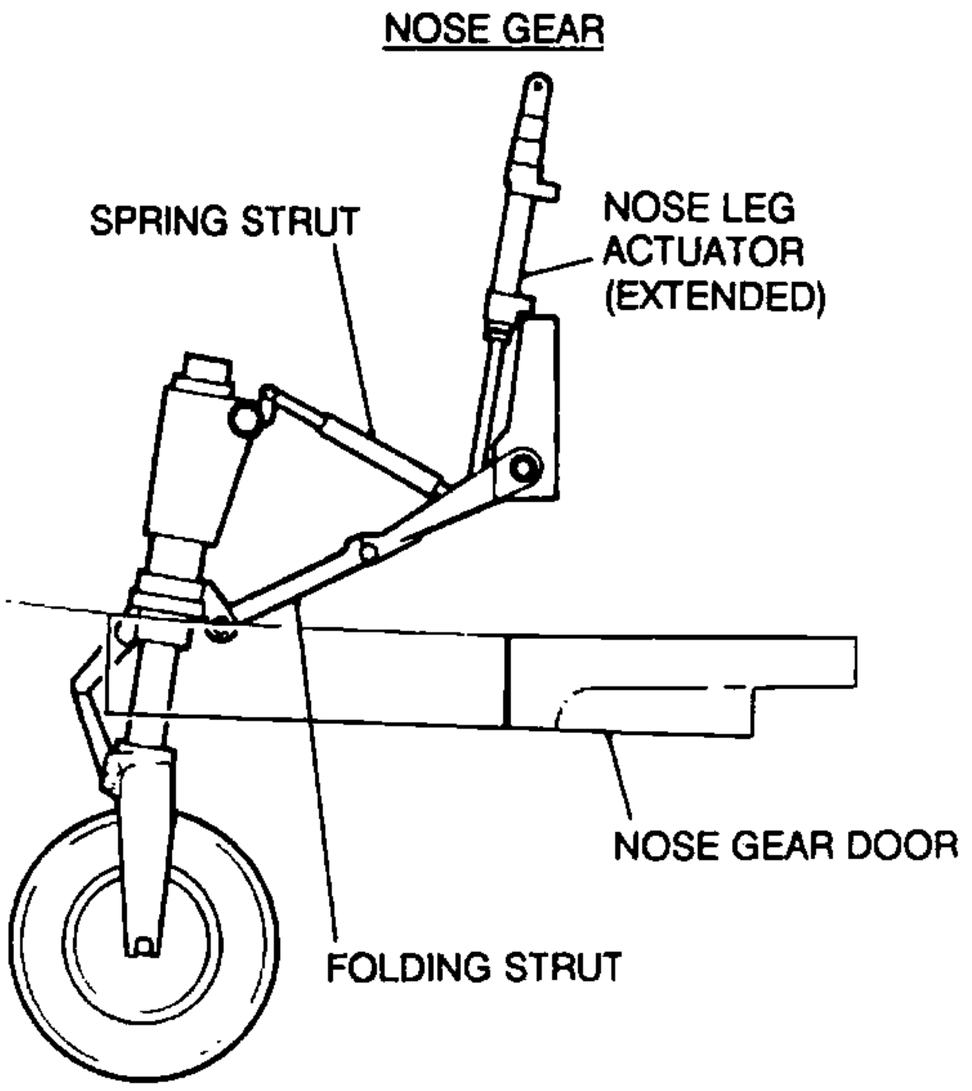
### Landing Gear Control Unit

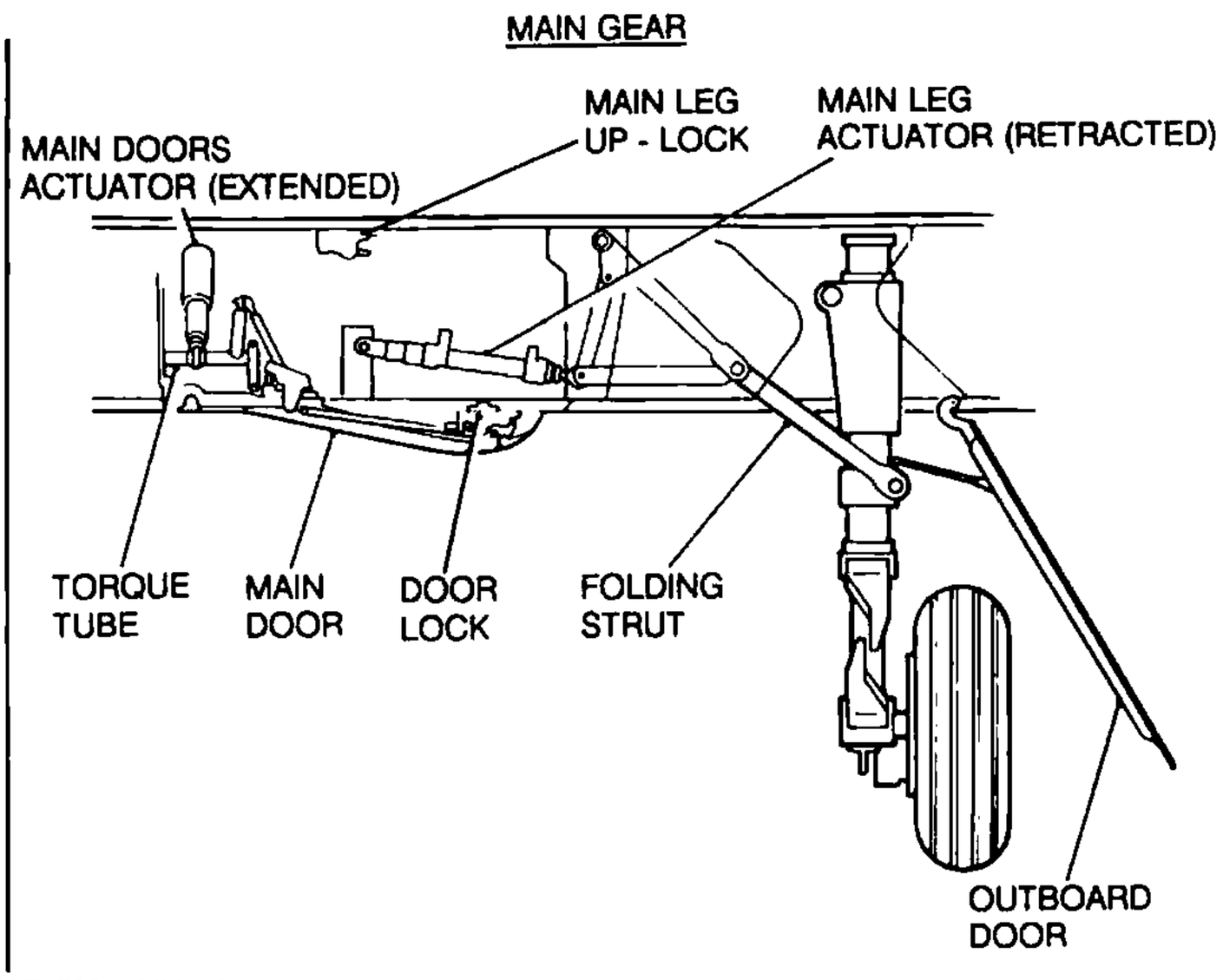
A landing gear control unit is located on the left side lower instrument panel in each cockpit. All control, indicating and warning circuits are routed through the rear control unit. The front and rear units are electrically linked for indication and aural warning silencer operation and are mechanically interconnected to allow landing gear selection to be made from either cockpit. The face of each unit forms a landing gear control and indication panel and comprises the LG selector, LG position indication lights, an aural warning silencer button and a landing gear down check button. The landing gear controls and indicators are shown in Fig 1-18.

### Landing Gear Locks

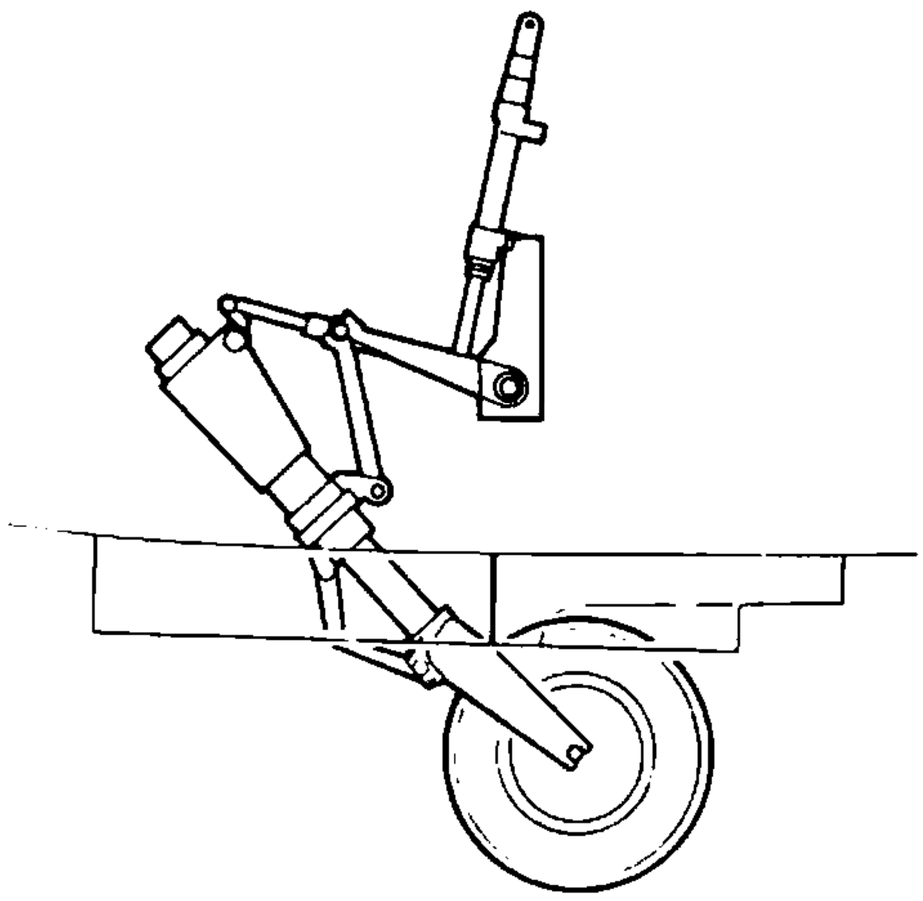
Each main leg up-lock comprises a latch mechanism which is actuated from the LG doors actuator to engage a latch pawl under the towing hook which is mounted on the bottom of each main leg. The main leg up-locks engage during the closing of the main doors and disengage during the opening of the doors. The nose gear is held up by a claw lock in the actuator. The MLG legs are locked down by an overcentre link and a claw lock in the actuator. The nose leg is locked down by the overcentering of the leg folding strut. A spring strut is connected between the nose leg and the upper link of the folding strut to maintain a strong over-centering force on the folding strut. The nose leg actuator counters the over-centering force during landing gear retraction. Each main door lock is a door-mounted sprung pawl which closes over a lock pin on the wing structure as the door closes. The pawl is opened by a push-rod operated by the initial movement of the door opening mechanism.

# EXTENSION/RETRACTION SEQUENCE

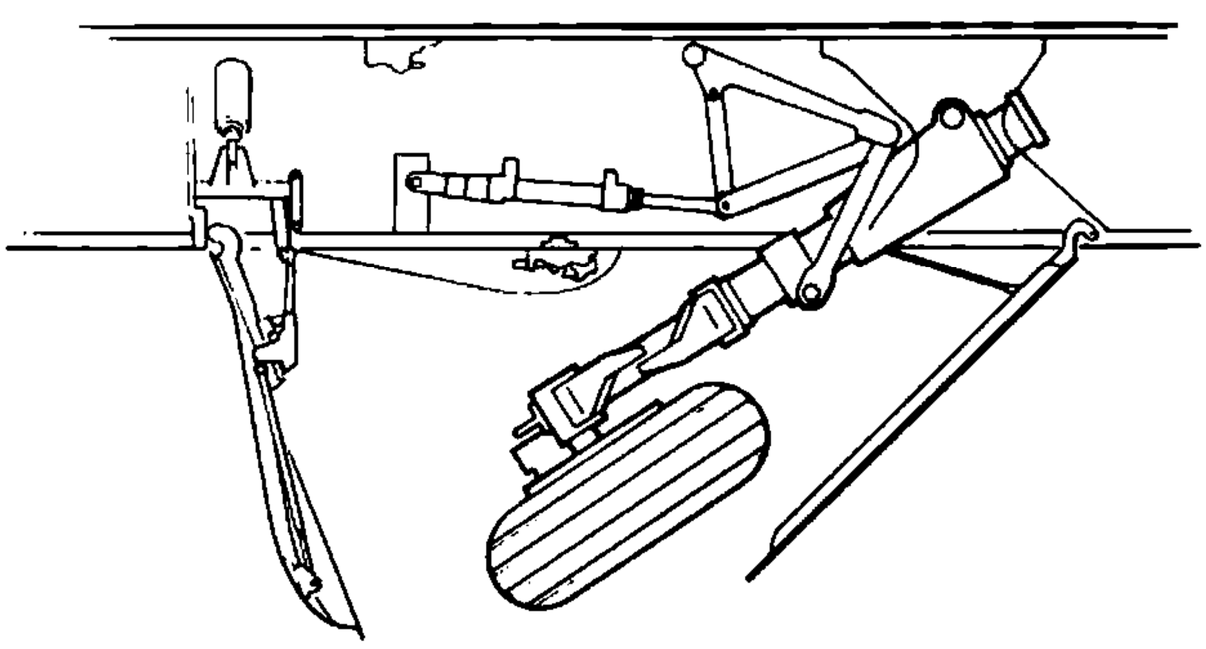


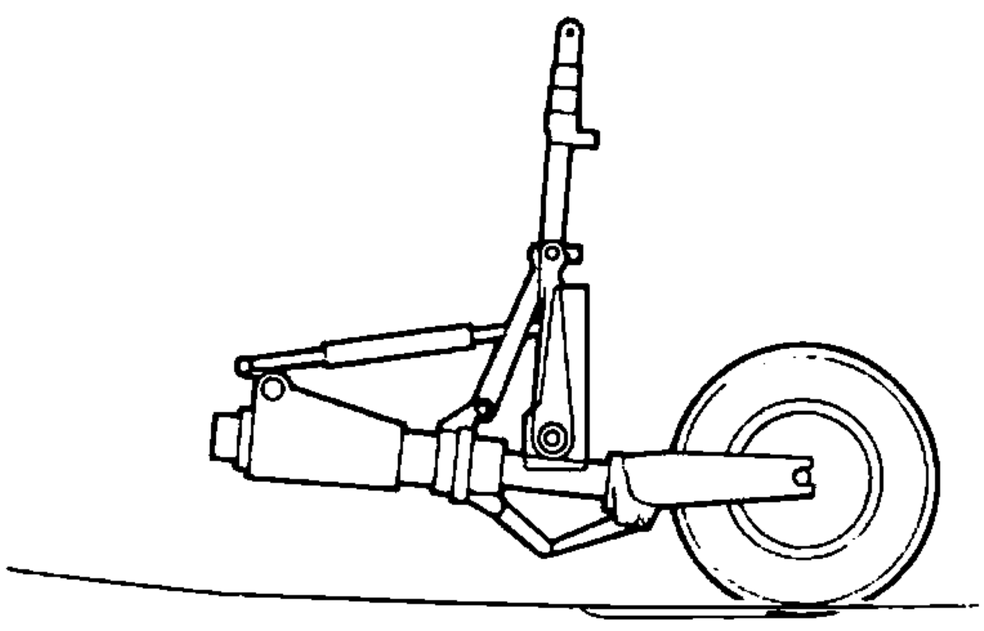




LANDING GEAR EXTENDED



LANDING GEAR IN TRANSISTION





LANDING GEAR RETRACTED

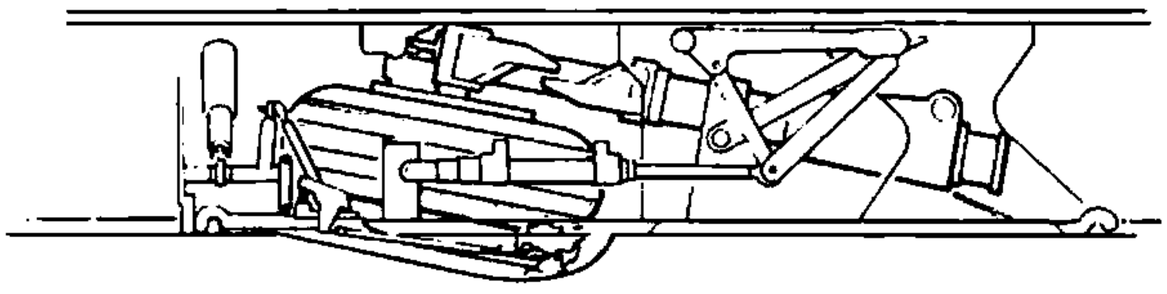


***Figure 1-17***

# LANDING GEAR CONTROLS AND INDICATORS

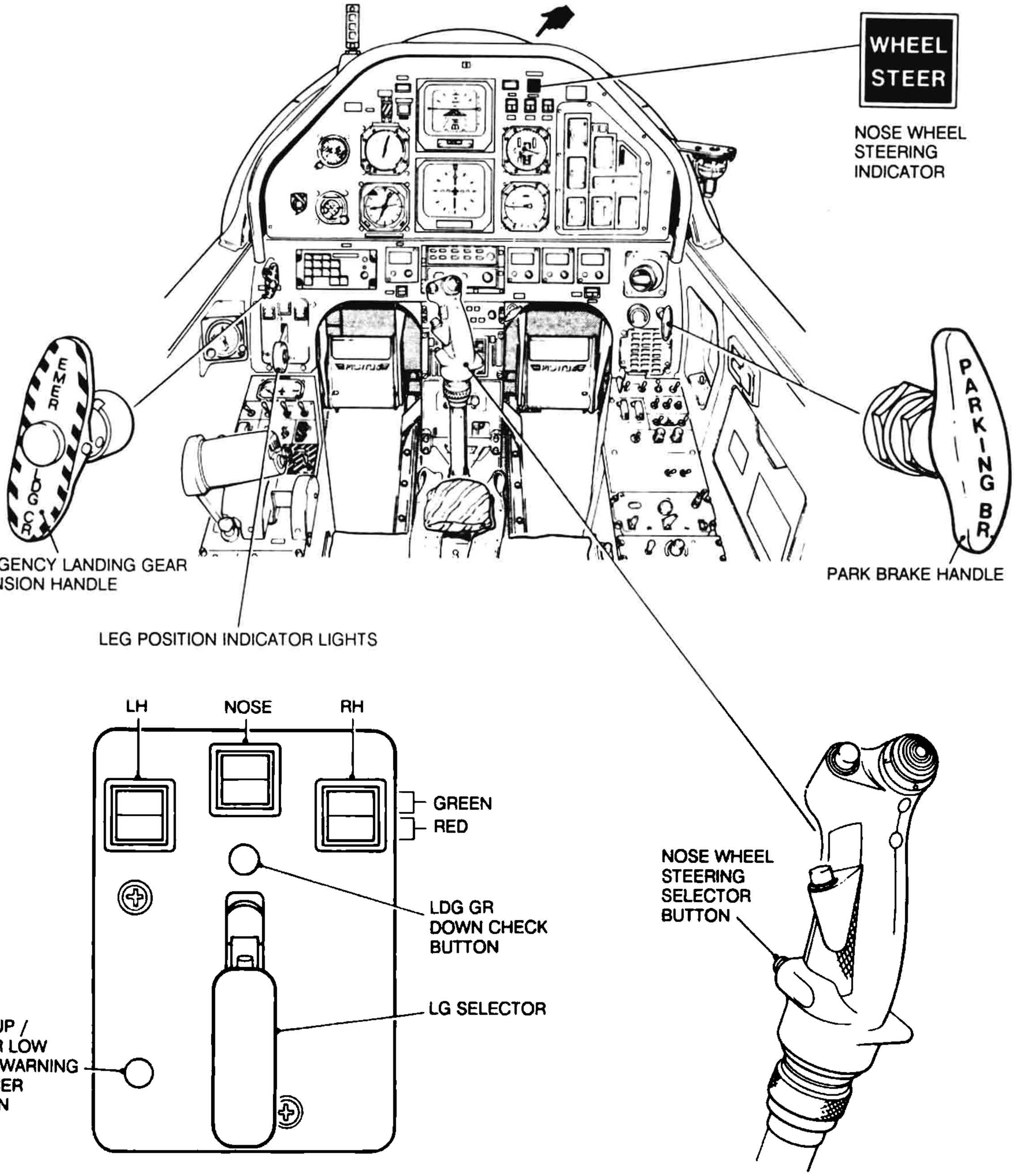



*Figure 1-18*

### Sequencing Circuits

The landing gear sequencing circuits utilise microswitches to provide sequencing of landing gear extension and retraction. The circuits are also used for landing gear indication. A 'weight-on' safety microswitch is installed on both the nose leg and the left main leg. The microswitches are operated, when the aircraft weight is on the legs, to prevent inadvertent retraction on the ground. Microswitches, operated by the legs, leg up-locks, doors and door lock ensure the correct sequence of operation and indication. Normal indication sequence for a landing gear retraction and extension cycle would be:

a. 3 greens - legs down, doors closed.
b. 3 greens/2 reds (main) - legs down, doors opening.
c. 3 reds - legs retracting.
d. 2 reds (main) - legs up, doors closing.
e. No lights - legs up, doors closed.
f. 2 reds (main) - legs up, doors opening.
g. 3 reds - legs extending.
h. 3 greens/2 reds (main) - legs down, doors closing.
i. 3 greens - legs down, doors closed.

The lights can be tested by pressing the SYSTEM TEST LIGHT switch on the left console. When the INSTR LIGHTS switch is selected to OFF the indicating lights are supplied with 28 V DC from the generator busbar. With the switch ON, the generator busbar supply is routed through a voltage regulator in the LG control panel and the brightness is reduced (night operations).

### Emergency Extension

In the event of a mechanical or electrical malfunction resulting in a failure of the hydraulic system or of the normal landing gear selection/actuation system, the landing gear can be extended by operating the EMER LDG GR handle on the left side of the front cockpit console. The handle has a centre button which is depressed and the handle pulled out. Operation of the handle connects pressure fluid from the emergency package accumulator to all four landing gear actuators. In each actuator, the pressure fluid is routed to an emergency shuttle valve. The actuators move to open the main gear doors and extend all three legs. After operation of the emergency extension system the main gear doors remain in the open position. The doors cannot be closed or the landing gear retracted until the emergency shuttle valves have been reset to the normal position. This is achieved by pushing each reset plunger into the actuator body until the detent re-engages.

**WARNING**

Main gear doors will close immediately electrical power and hydraulic pressure are applied to the main hydraulic system following emergency shuttle valve reset.

### Landing Gear Aural Warnings

Three aural warnings are associated with the landing gear:

a. A gear up/power low/low airspeed warning.
b. A gear up/flaps LAND warning.
c. An aircraft on ground/gear selected up warning.

The first two warnings are incorporated to prevent a 'wheels up' landing. The third is to warn of a possible 'ground' retraction. The warning signal (1100 Hz continuous tone) is produced by a tone generator and transmitted through the audio system. The warnings are activated when the landing gear selector is selected to LG UP and activated by the completion of a ground path via any one of the microswitches. A warning, when activated, is terminated when the landing gear selector is selected to LG DOWN or when the activating microswitch is opened. The 'gear up/power low/IAS low' aural warning can be cancelled by pressing a WARNING SILENCER button located on each LG control panel. This allows the pilot to make a power-off descent without the distraction of the aural warning. The WARNING SILENCER selection is cancelled when the PCL is opened beyond approximately one third of its travel or IAS increases past 120 kts. A subsequent retardation of the PCL or reduction of IAS below 120 KIAS with the LG selected UP, will reactivate the 'gear up/power low/low IAS' aural warning.

## NOSEWHEEL STEERING SYSTEM

### Description

The nose wheel steering system is controlled by movement of the rudder pedals which are mechanically linked to a hydraulically operated actuator installed on the nose leg housing. Hydraulic pressure at 3,000 psi is utilised from the aircraft hydraulic power system. The unit comprises a two-stage solenoid valve, a rotary control valve and a rotary actuator. The solenoid valve is controlled electrically by a selector switch on the lower front face of the control column handle and selection is indicated by an advisory light on the top of each instrument panel. Power is via the NLG STEER C/B on the front generator busbar. The nosewheel steering actuator is controlled by movement of the rudder pedals mechanically linked to the rotary control valve. When nose wheel steering is not selected, the nose leg is free to castor and aircraft steering is achieved by differential wheel braking or aerodynamic forces from the vertical stabiliser. During landing gear retraction the nose wheel is automatically centred. When illuminated, the selector lights indicate that nose wheel steering has been selected ON.

## WHEEL BRAKE SYSTEM

### Description

The aircraft wheel brake system operates independant of the aircraft hydraulic power system. The wheel brakes system comprises a left and right wheel brake

system supplied with hydraulic fluid from a common reservoir. Each main wheel is independently braked by multi-piston, disk brake units which are mechanically actuated and hydraulically operated by toe pedals mounted on the rudder pedal assembly in each cockpit. The front and rear cockpit toe pedals are interconnected by rods. The rear cockpit brake pedals are equipped with heel pedals to allow the pilot in the rear seat to oppose any excessive brake application by the front seat pilot. Hydraulic fluid is gravity fed to each wheel brake system from the brake fluid reservoir. A brake master cylinder in each system is directly operated from the toe pedals to apply pressure fluid via a common park brake valve to the brake unit.

Idle power on the ground can generate high taxi speeds under conditions of a tailwind and/or downslope. Judicious use of brakes is required as continuous light braking will lead to break unit overheating. The recommended technique is to allow the aircraft to reach a higher than normal taxi speed then apply smooth braking to bring the aircraft almost to rest before releasing the brakes completely. Each main wheel is equipped with fusible plugs which will allow gradual deflation of the tyre should excessive wheel or tyre temperatures be encountered. Fig 1-19 shows the layout of the aircraft wheel brake system.

### Park Brake

The park brake system is a bowden cable operated by a PARKING BRK handle located in the front cockpit on the right side. The park brake is applied by pulling and turning the PARKING BRK handle to the right and applying pressure to the toe brakes. When the park brake is applied, the bowden cable moves a park brake (shuttle) to lock hydraulic pressure in the brake units on release of the brake pedals. The park brakes are released by turning the handle to the left and releasing in.

## *Aircraft Wheel Brake System*

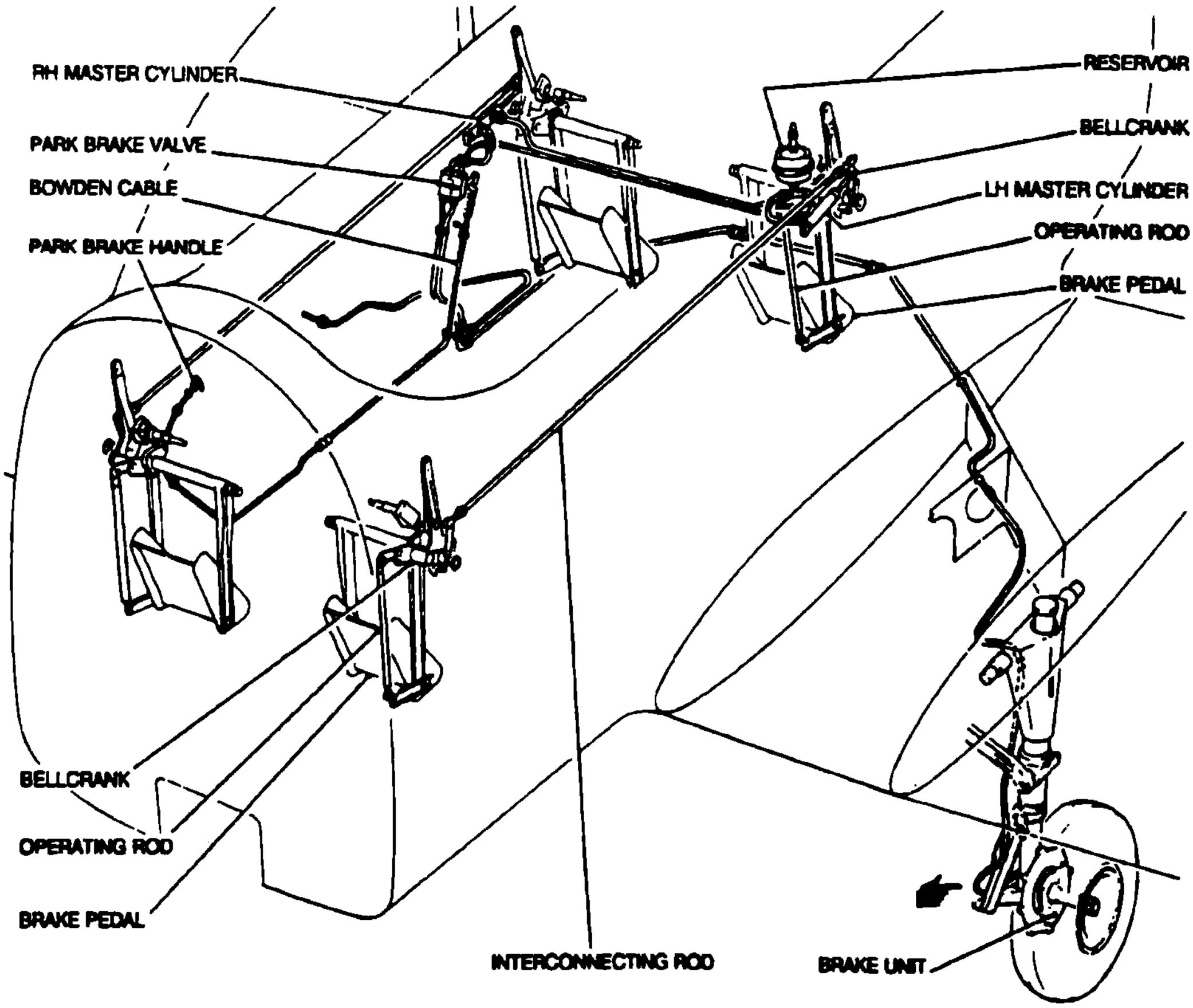



Figure 1-19

# INSTRUMENTS

## Description

The following flight instruments are fitted to both cockpits:

a. Two Tube Electronic Flight Instrumentation System (EFIS), incorporating an Electronic Attitude and Direction Indicator (EADI), an Electronic Horizontal Situation Indicator (EHSI) and a slip indicator.

b. Pitot Static System.

c. Standby Attitude Indicator.

d. Angle of Attack Indexer and Indicator.

e. Combined Mach/Airspeed Indicator.

f. Accelerometer.

g. Altimeter.

h. Vertical Speed Indicator.

i. Radio Magnetic Indicator (RMI).

# ELECTRONIC FLIGHT INSTRUMENT SYSTEMS

## Description

The Electronic Flight Instrumentation System (EFIS), visually displays selected flight information in a multi-colour graphic form on the Electronic Attitude and Director Indicator (EADI) and the Electronic Horizontal Situation Indicator (EHSI). The EFIS has one Symbol Generator (SG) which drives four multigraphic Display Units (DU). The EADI and EHSI displays are shown at Figure 1-20.

A Control Panel (CP) and a CRS/HDG Select Panel are also mounted in the front and rear cockpits. The SG receives data from the attitude and heading reference system (AHRS). Display imagery of the EFIS is organised so that the front cockpit displays are the same as the rear cockpit. All controls for the EFIS operate in tandem so that any format, mode or Nav source selections made in either cockpit are 'mirrored' in both cockpits without the use of external switching. System DC power is supplied from the battery and generator busbars. The system is designed so that a single loss of the battery or generator busbar will not result in a total loss of EFIS power. System AC power is supplied from one of two switchable static inverters.

## Display Units

The two DUs in each cockpit receive video data from the SG. A brightness 'BRT' control on the CP is operated to set the brightness level of DUs individually. A photo cell on each DU provides a small amount of brightness compensation depending on the ambient lighting in the cockpit. Any failure flags are displayed as red lines or words.

**CAUTION**

Turning the 'BRT' control to the 'OFF' position does not isolate electrical power to the DU. Electrical power can only be isolated from a DU by pulling the applicable EADI/EHSI circuit breaker on the appropriate circuit breaker panel.

## Electronic Attitude and Direction Indicators

The EADI displays primary flight attitude pitch and roll and secondary heading information. An aircraft symbol is used as a reference point and is located in the centre of the picture. The left side of the unit shows the AOA on the FAST/SLOW (F/S) scale and the right side shows the glideslope scale whenever an ILS frequency is selected and the CRS pointer is within 105 degrees of the aircraft heading. A turn indicator on the bottom of the unit displays the turn direction and an uncalibrated rate of turn. A runway symbol is shown just above the ROT scale and is displayed whenever an ILS frequency is selected. The runway symbol is the expanded localiser and shows left/right deviation of the LOC. The Marker Beacons are annunciated on the lower left hand corner of the unit when the MKR TEST is pressed or when a Marker is received by the Nav unit. An inclinometer on the bottom of the unit displays aircraft slip/skid.

## Electronic Horizontal Situation Indicators

The EHSI display presents the heading and navigation data that has been selected. The aircraft heading is indicated below the lubber line. The selected Navigation mode (VOR/LOC/TCN/ADF) is displayed on the left side of the tube. VOR, TAC or LOC are selected by the NAV/TAC frequency selector. When ADF mode is selected, the CDI bar can also be used similarly to the VOR and TAC, however the green distance and ground speed readouts in the upper right corner of the EHSI will be absent. A valid signal will be indicated by the presence of the selected needle on the EHSI face (excluding ILS), the absence of CDI ambiguity indicator when the CDI is matched to the selected navigation aid and, in the case of TAC or paired VOR, the presence of distance information in white in the lower left hand corner. The selected HDG (heading) is indicated by the orange bug that rides on the compass ring. A digital HDG bug readout is present whenever the 120° sectored format is displayed. The selected CRS (course) is indicated by the CRS needle and the digital readout in the upper left corner. The DME/TAC distance is shown in the upper right corner and the ground speed is shown below the distance. The ground speed will be replaced by the NAV frequency, in white, if DME Hold is selected. The right side will show the glideslope scale whenever an ILS frequency is selected and the CRS pointer is

within 105° of the aircraft heading. The single pointer will sequence between VOR or TAC and two nil selections. The double pointer will sequence between ADF and two nil selections. For VOR and TAC the distance will be shown for a valid signal.

### EFIS Control Panel

The EFIS Control Panel (CP), shown at Fig 1-21, is centrally located beneath the EHSI.

This panel is used to select the desired NAV system, the format and the mode for the EHSI. The HSI button switches the standard HSI format to the 120° sectored compass as shown at Fig 1-22.

The NAV/MAP button switches the standard compass card format to the NAV/MAP format. The MAP format shows the VOR/DME and TACAN stations as symbols with a selectable range ring. The RNG buttons increase or decrease the displayed range shown in the NAV/MAP or 120° sectored format. The 120° sectored map display is shown at Fig 1-23.

The NAV and ADF buttons select the CDI mode to the currently selected VOR/LOC/TAC or the ADF. The single and double needle RMI buttons are used to select the needles for RMI bearing. The CP also contains BRT (brightness) control for the EADI and EHSI. A detent is provided to switch the brightness OFF when turned fully CCW.

### Course Heading Select Panel

A Course Heading Select Panel is located beneath the CP. The CRS and HDG knobs are used to set the CRS needle and HDG bug on the EHSI. A momentary pull of the CRS knob will cause the CDI needle to centre on a valid VOR or TAC station. A momentary pull of the HDG knob will slew the HDG bug to beneath the lubber line. The Course Heading Select Panel is shown at Fig 1-24.

### EFIS Fault Annunciators

Any red lines or words on the displays are the flags that indicate faults may be present. The following is a summary of the faults:

a. A red ATTITUDE FAIL - the displayed attitude is unreliable.

b. A red HDG - the displayed heading information is unreliable. The standby compass and the 120° sectored map mode for VOR/TAC navigation is available. LOC and G/S still available on standard EHSI format.

c. A red SG - the system is unreliable.

d. Parallel lines through the glideslope scale, the CDI bar, DME distance, bearing pointer runway symbol - that indication is unreliable.

e. A red X on the ambiguity indicator or the bearing pointers - that information is unusable.

f. Parallel lines drawn through the ROT or F/S scale - information is invalid.

g. A yellow SG - the SG fan has failed (it must be repaired before the next flight).

h. A yellow DU1 - the EADI fan or unit has failed. Similarly, a DU2 indicates that the EHSI fan or unit has failed. (It should be repaired before the next flight).

I. A yellow CP OFF - the SG is sensing a CP failure (it should be repaired before the next flight).

System self-test is provided by depressing a self-test switch located on the front cockpit upper right wall, aft of the seat. System test displays all flags and indications after which the NAV button must be pressed to revert to normal displays. The system test is a maintenance function only and should not be otherwise operated. The EFIS is continually subjected to Built-In-Testing (BIT) during normal use, to alert the pilot of detected failures.

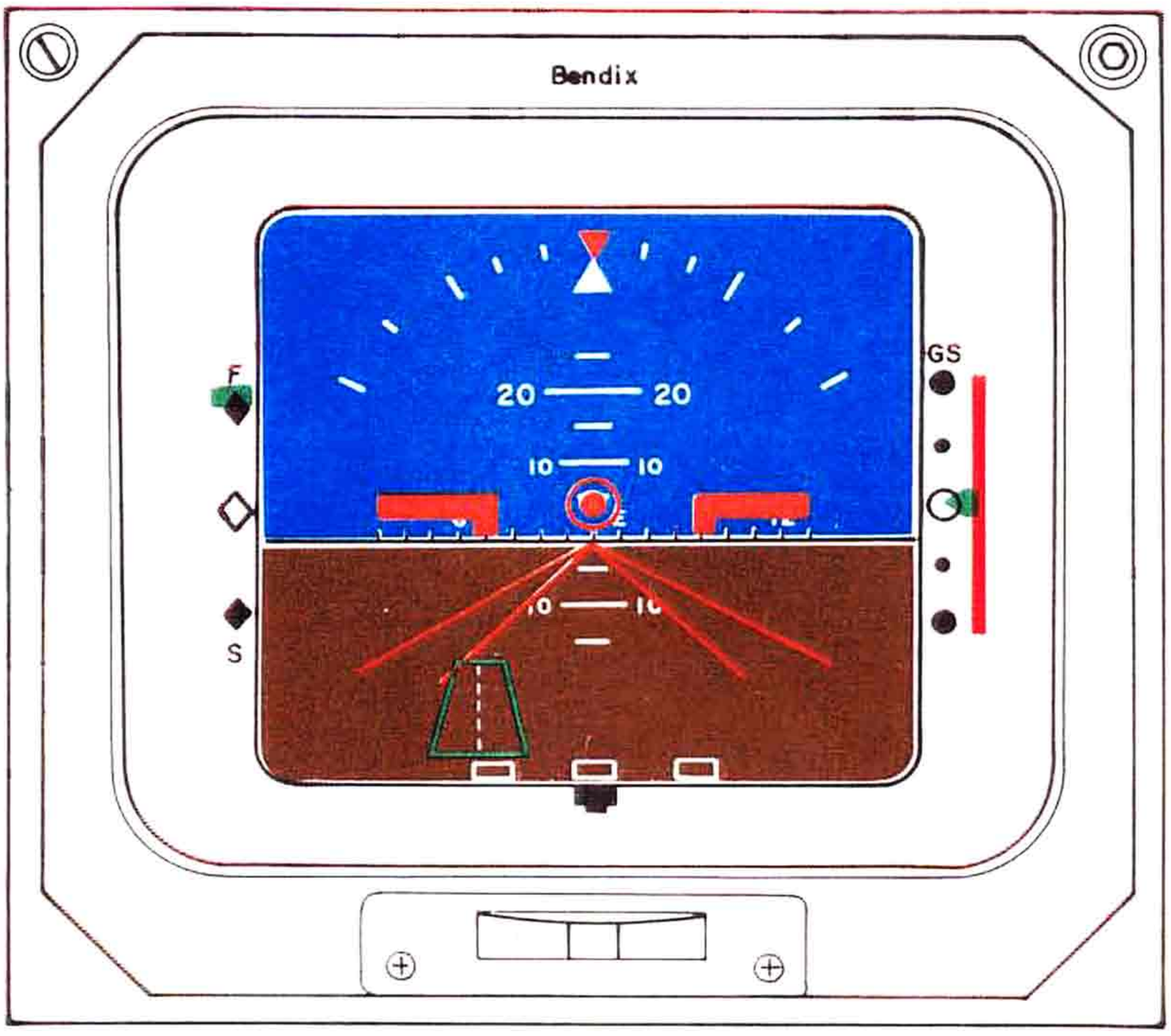
**WARNING**

AHRS information can be unreliable without a corresponding warning indicator. Cross-reference to the standby attitude indicator should be made, particularly in IMC, to verify correct operation.

### Reversionary Modes

If the EADI picture is unusable, depress the EFIS SWAP switch located under the VHF Communication panel. The EADI and EHSI will swap so that the attitude will be presented on the bottom display. The EADI and EHSI information can be monitored sequentially, if necessary. The EFIS SWAP switch has an integral indicating light which illuminates to show the swap process has been selected. The RMI can be used for heading and navigation information. If either CP fails, the SG will automatically set the EHSI picture to the standard HSI format. A yellow CP OFF will be displayed in the upper left corner of the EHSI. The SG will select NAV for the Nav mode. VOR, LOC or TAC continue to be selected by the NAV frequency selector. The SG will set the single pointer to nil and the double pointer to nil. The HDG bug will slew, initially to the lubber line and then follow the compass card. The CRS pointer will slew direct to the station and remain there. If an ILS frequency is selected, the pointer will slew to the lubber line and remain there. Failure of a red gun in any display tube inhibits the display of any red warnings.

# EADI AND EHSI DISPLAYS



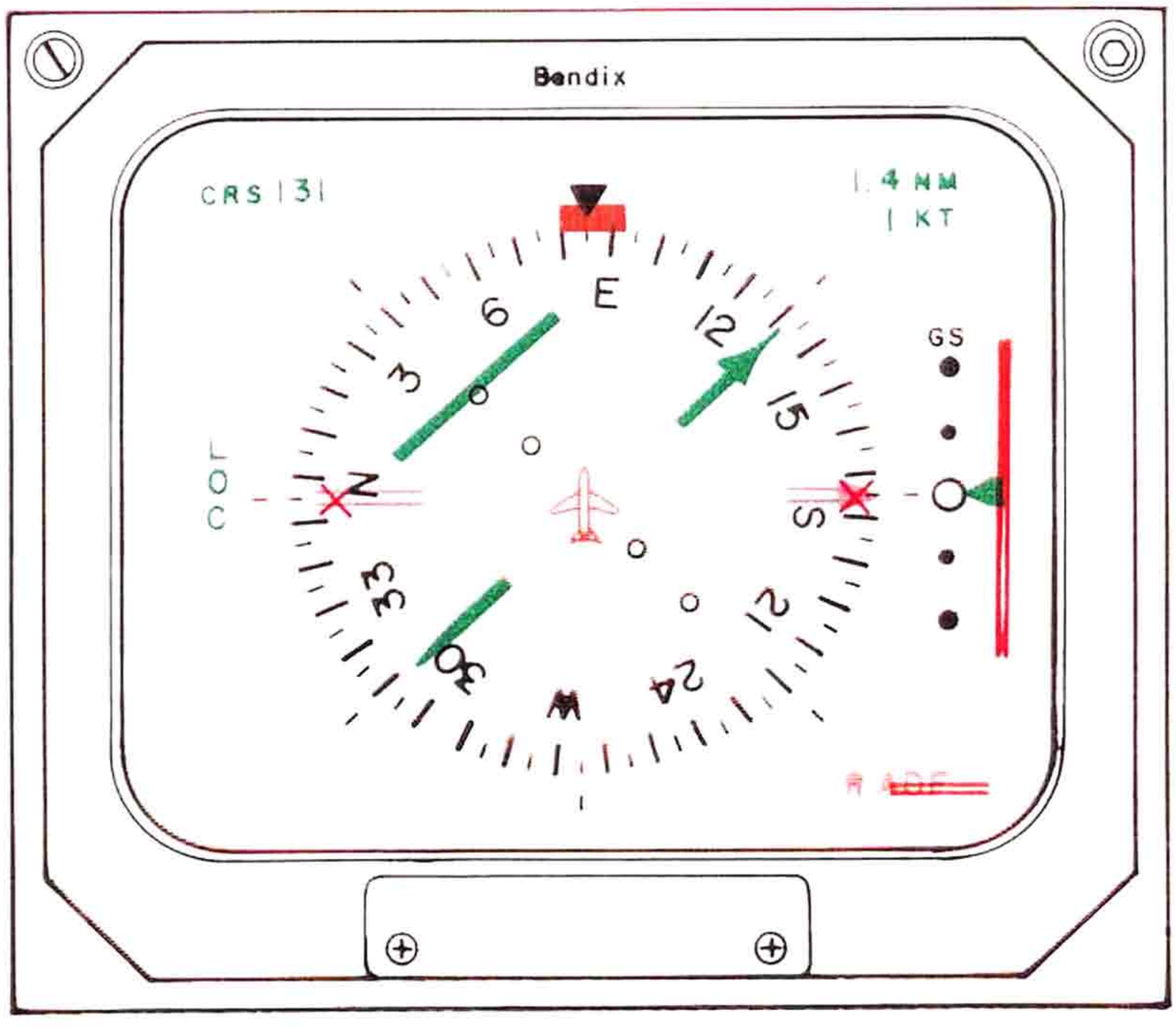




*Figure 1-20*

# EFIS CONTROL PANEL

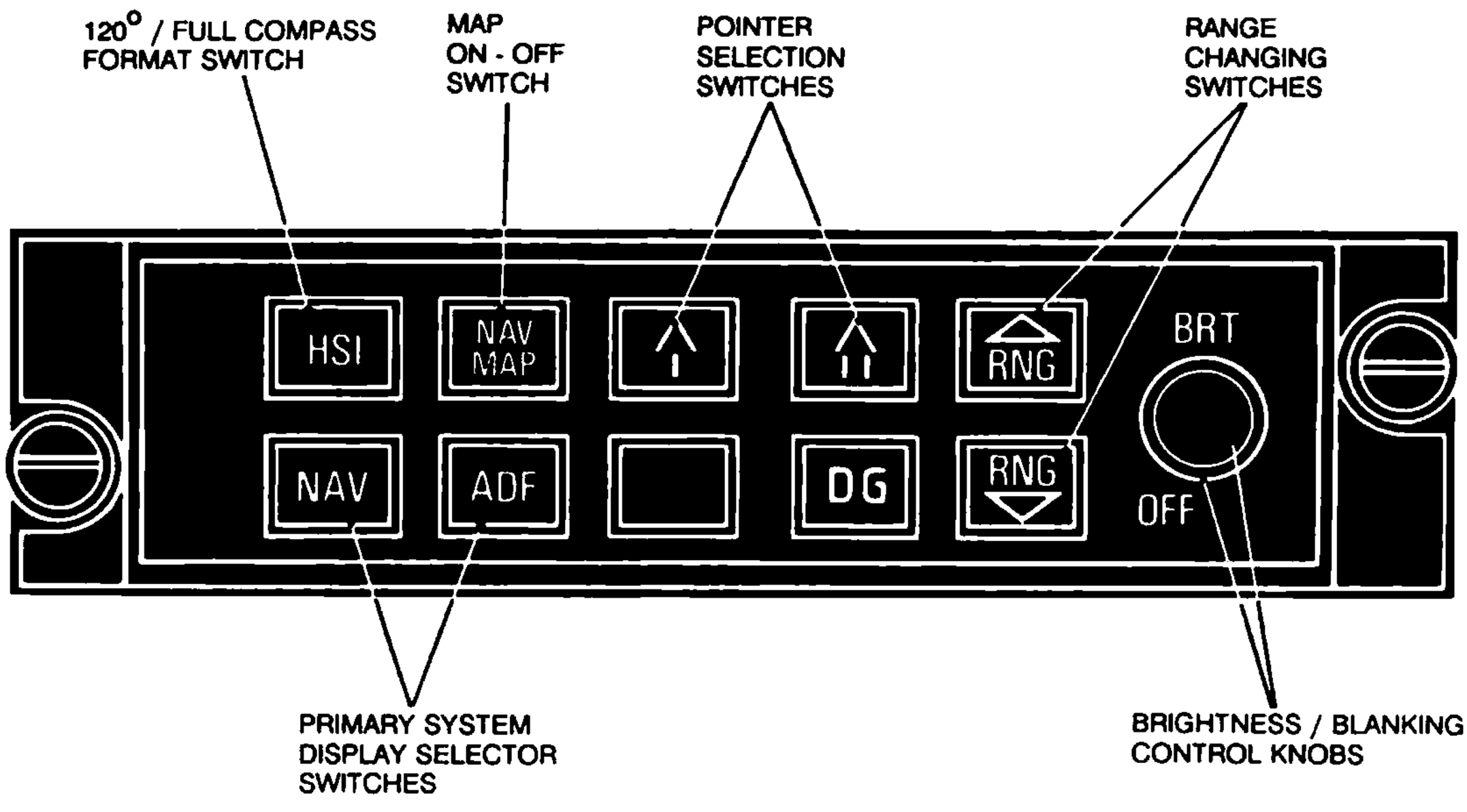



*Figure 1-21*

# HSI SECTORED COMPASS

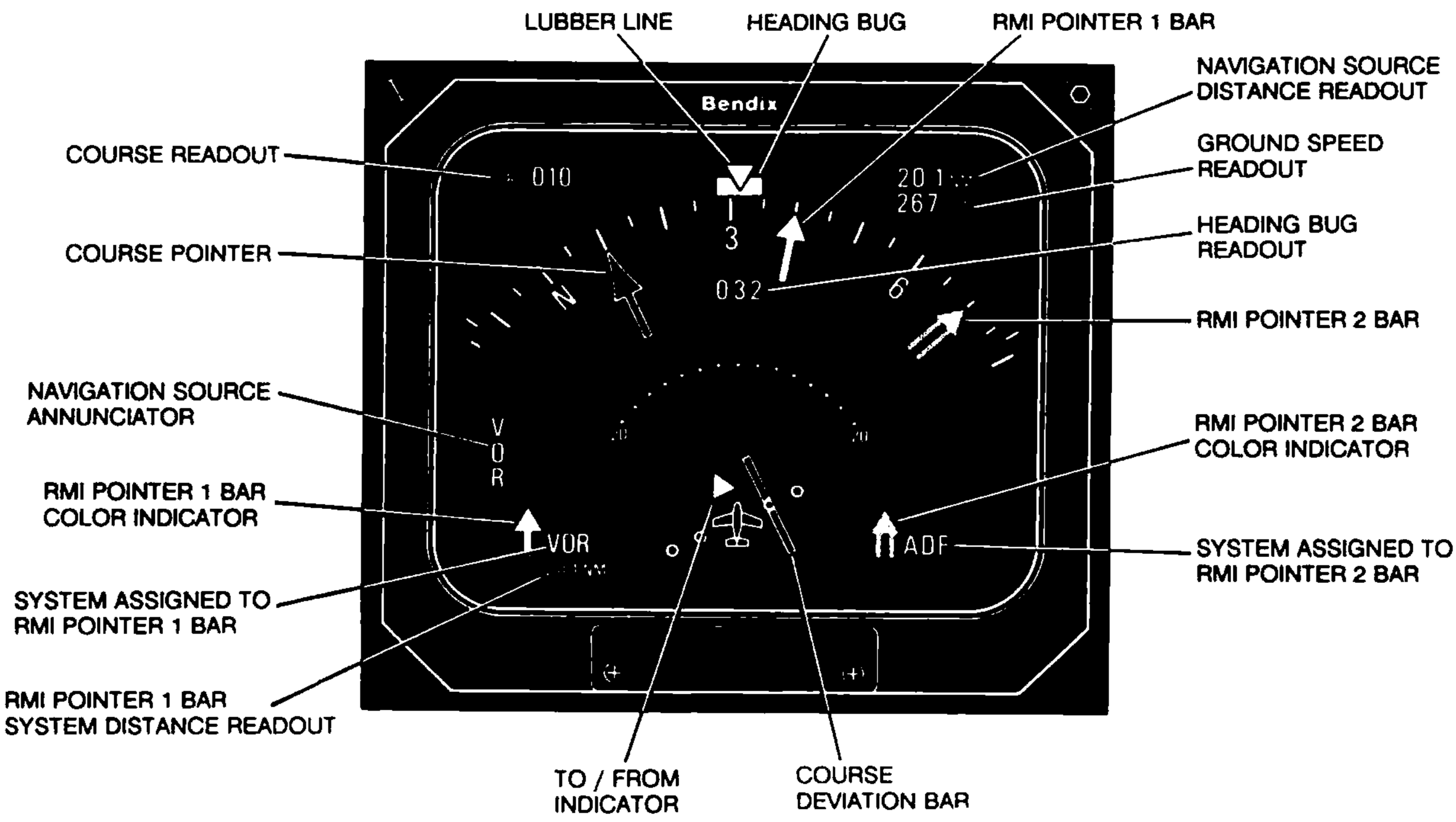



*Figure 1-22*

# SECTORED MAP DISPLAY

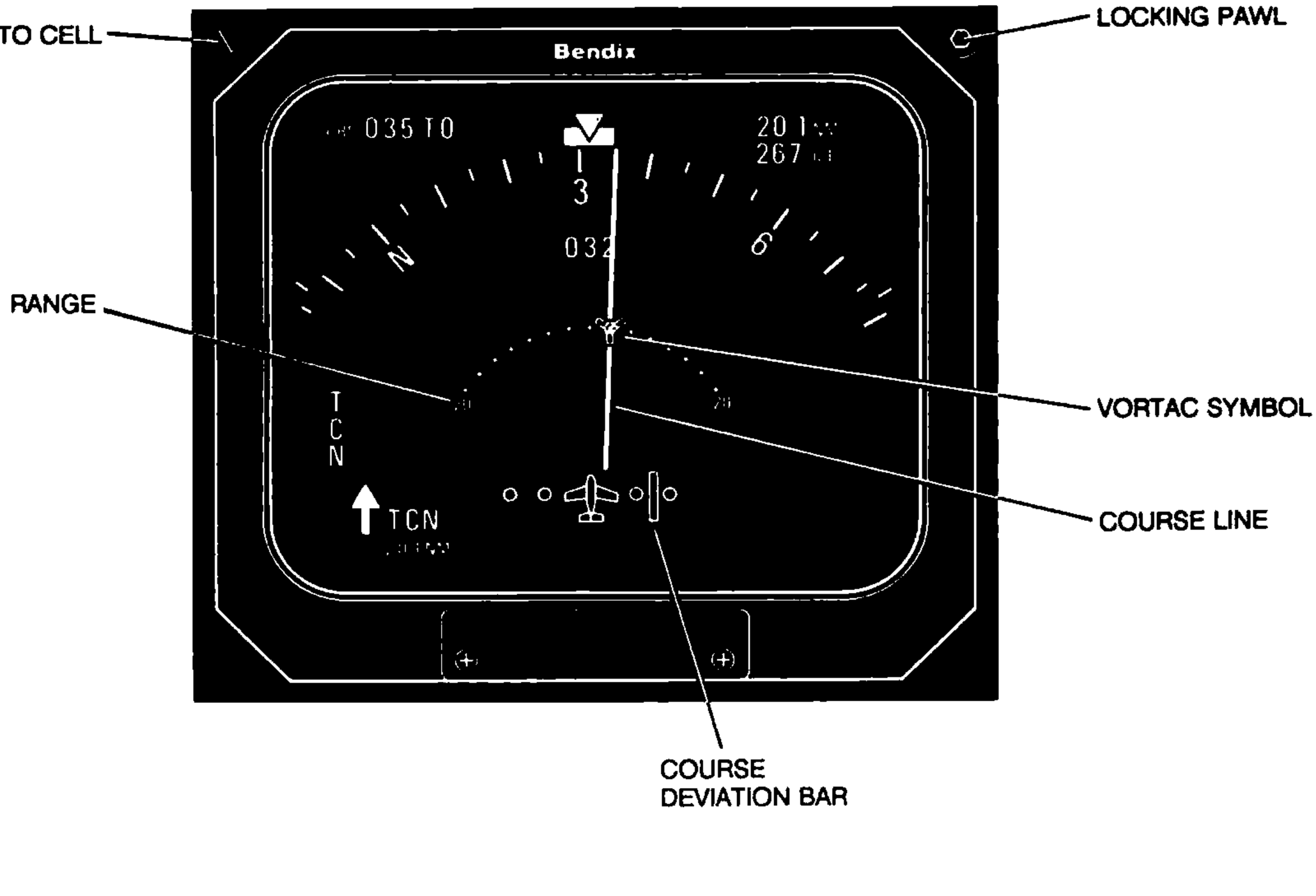



*Figure 1-23*

# COURSE HEADING SELECT PANEL




*Figure 1-24*

## PITOT STATIC SYSTEM

The Pitot Static system comprises static vents on each side of the rear fuselage and the pitot head under the left wing. The pitot static system is shown at Fig 1-25.

The static vents are interconnected (to reduce static pressure errors by aircraft yawing) and provide static pressure to the VSIs, Altimeters and the Mach/ Airspeed indicator with the pitot head providing pitot pressure to the Mach/Airspeed indicator. The pitot tube and static ports are equipped with heating elements for ice prevention which are controlled by the anti-ice PROBES switch in the front cockpit and a weight-off wheels microswitch on the left main landing gear leg. The ANTI ICE C/Bs are located on the front generator busbar. The microswitch selects reduced heating when the aircraft is on the ground.

### STANDBY ATTITUDE INDICATOR

Each cockpit is fitted with a standby attitude indicator on the left side of each instrument panel. The indicator contains a 28 VDC gyro, which is maintained in the vertical attitude by an integral mechanical erection system. The gyro is powered from the battery busbar (front and rear cockpits), via the SEC ATTD C/Bs. The gyro is mechanically linked to a display drum assembly which has 360° freedom in roll, 92° in climb and 78° in dive. The gyro will allow a minimum of 9 minutes reliable flight attitude information after failure of electrical power. An OFF flag provides indication of:

a. Electrical power lost.

b. Gyro is caged.

c. Failure of the OFF flag motor.

A knob on the lower right side provides adjustment of the miniature aircraft pitch attitude. The knob also provides a PULL TO CAGE function for inflight reset of the gyro position and for parking the gyro after flight. The standby attitude indicator is shown at Fig 1-26.

# PITOT STATIC SYSTEM

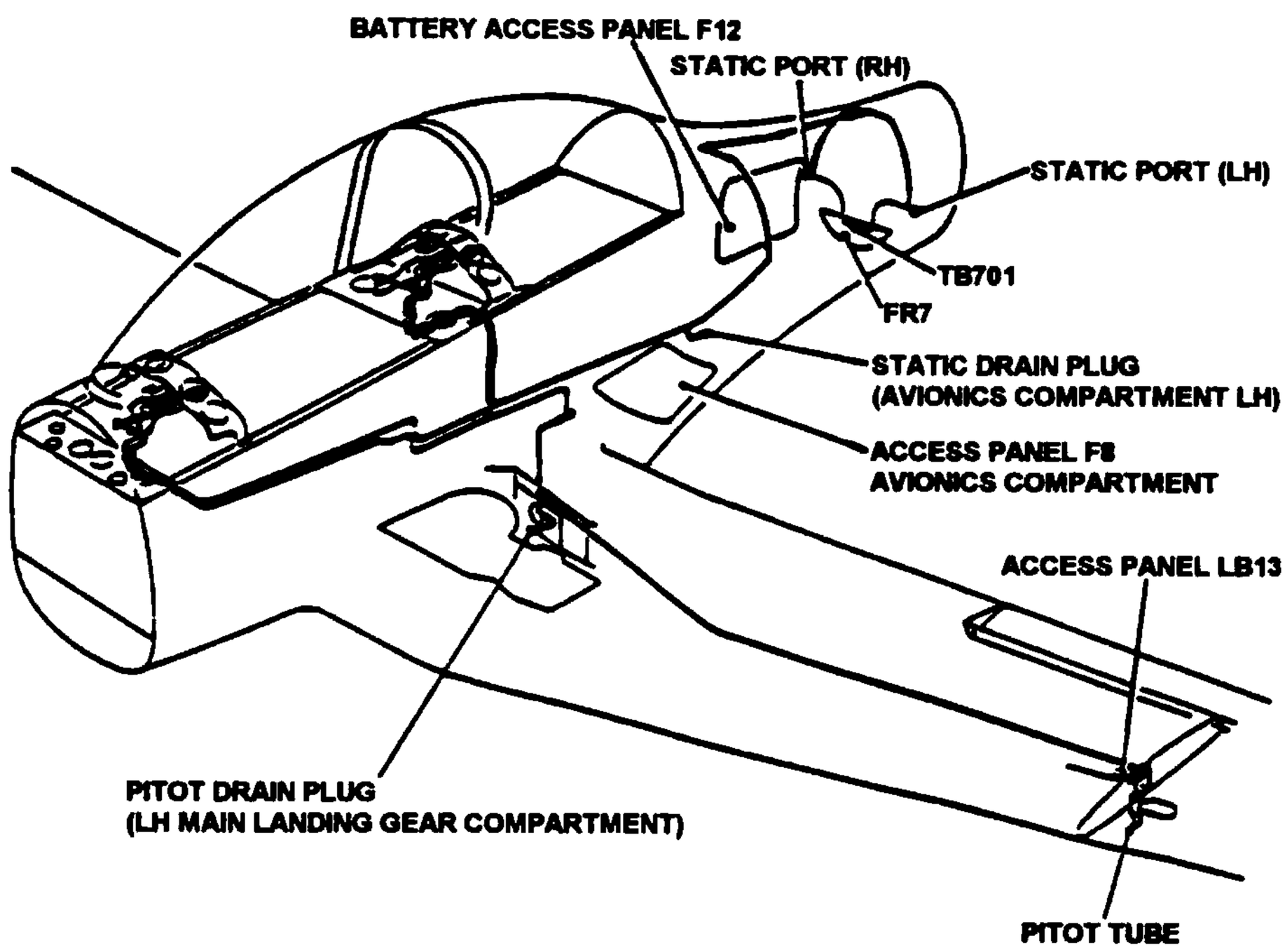



**Figure 1-25**

## ANGLE OF ATTACK INDEXER AND INDICATOR

The angle of attack (AOA), of the aircraft wing to the airflow, is detected by a vane on the leading edge of the left wing. A combined AOA electronics unit/aural warning (located on the upper rear face of the firewall) processes and converts the transmitter information and other inputs into appropriate output signals to drive the AOA indexers, the EFIS symbol generator and the aural stall warning system. The information is displayed on an angle of attack indexer on the left side of the instrument coaming in both cockpits, and to the EADI displays. The signal is also used to activate the aural stall warning (when the aircraft is off the ground). The indexer provides a head-up display of approach speed by illuminating a single light or two adjacent lights as per Fig 1-27. The AOA indexer is automatically switched off when the landing gear is retracted. The AOA system control and test functions are supplied with DC power from the front generator busbar via the AOA SYS C/B. Indexer lighting is supplied from the generator busbar via the INSTR LIGHT C/B, the INSTR LIGHTS switch and the LG control unit. Two switches in the front cockpit are used during AOA system testing. An AOA SYSTEM TEST switch on the left console (behind the PCL), placarded HIGH - OFF - LOW, is used to check the indicator displays, indexer lights and the stall warning system. A maintenance switch, placarded AIR - NORM - GROUND (rear left side of cockpit), is used to enable the stall warning circuit when the aircraft is on the ground. The maintenance switch is a maintenance function only and should not be otherwise operated.

## COMBINED MACH/AIRSPEED INDICATOR

Both cockpits are fitted with combined mach/airspeed indicators which are supplied with pitot and static pressure. The instruments provide speed indication from 60 to 410 KIAS and 0.3 to 1.0 Mach. A red-colored radial marked on the airspeed scale at 320 and at 0.68 on the Mach scale corresponds to the maximum operating speed of the aircraft. A yellow bug can be manually set for pilot's reference by rotation of the indicator knob on the lower left corner. A relay operated overspeed warning switch in the front indicator closes at overspeed conditions (320 KIAS or 0.68 Mach) or power supply failure, and activates a 1600 Hz (5 Hz interrupt) aural warning tone. A relay operated airspeed switch in the rear indicator is connected in series to a PCL low power switch to give a continuous 1100 Hz aural warning tone with low power settings below 120 KIAS. This tone can be cancelled by depressing the WARNING SLENCER push-switch on the landing gear control unit. A power supply failure causes the airspeed switch to be inoperative. Both indicators are supplied with power from the front generator busbar, via circuit breaker AURAL WARN (front indicator) and A/S LOW WARN (rear indicator). Each indicator has integral lighting. The mach/airspeed indicator is shown at Fig 1-28.

## STANDBY ATTITUDE INDICATOR

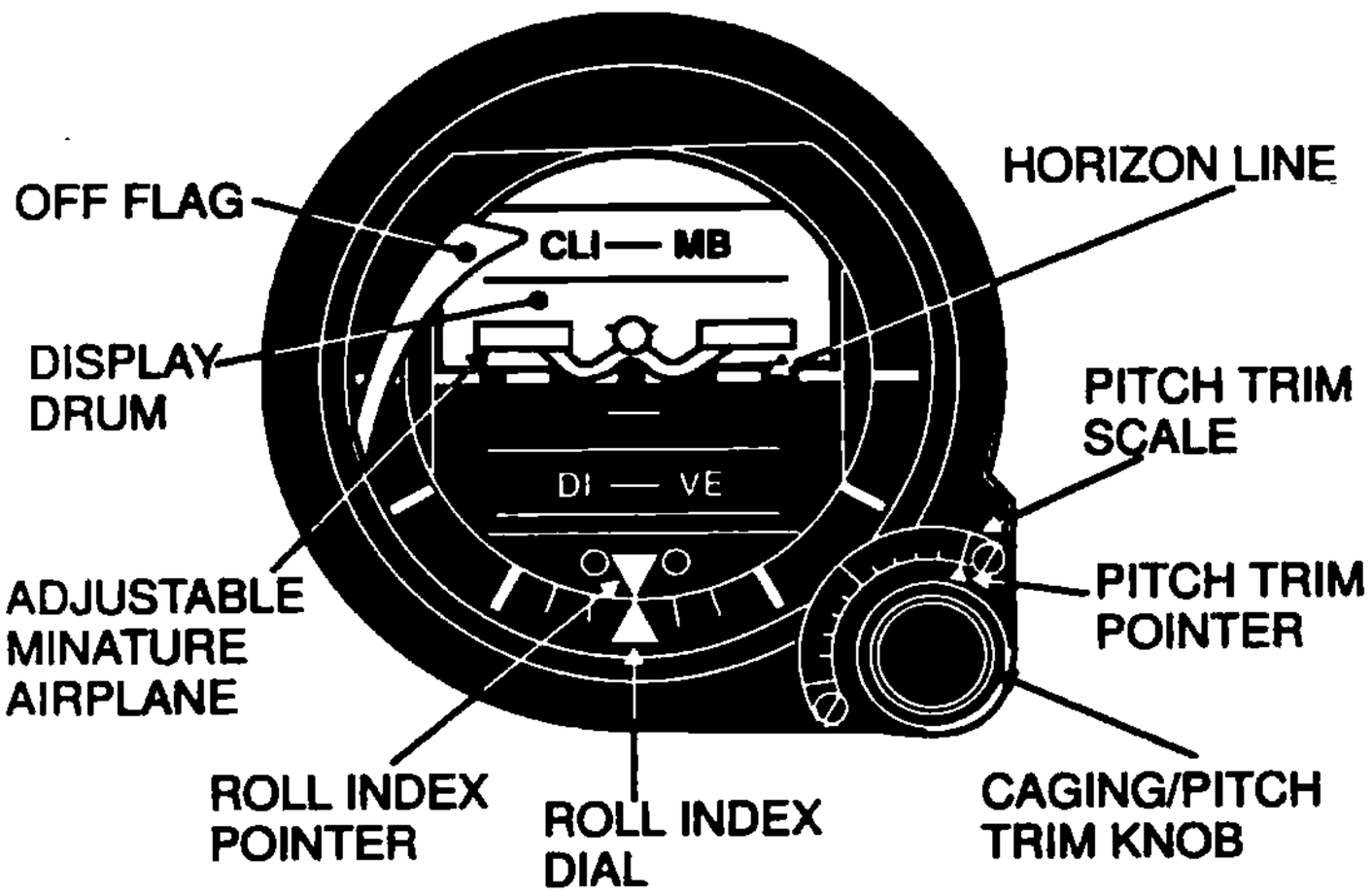



Figure 1-26

## ANGLE OF ATTACK

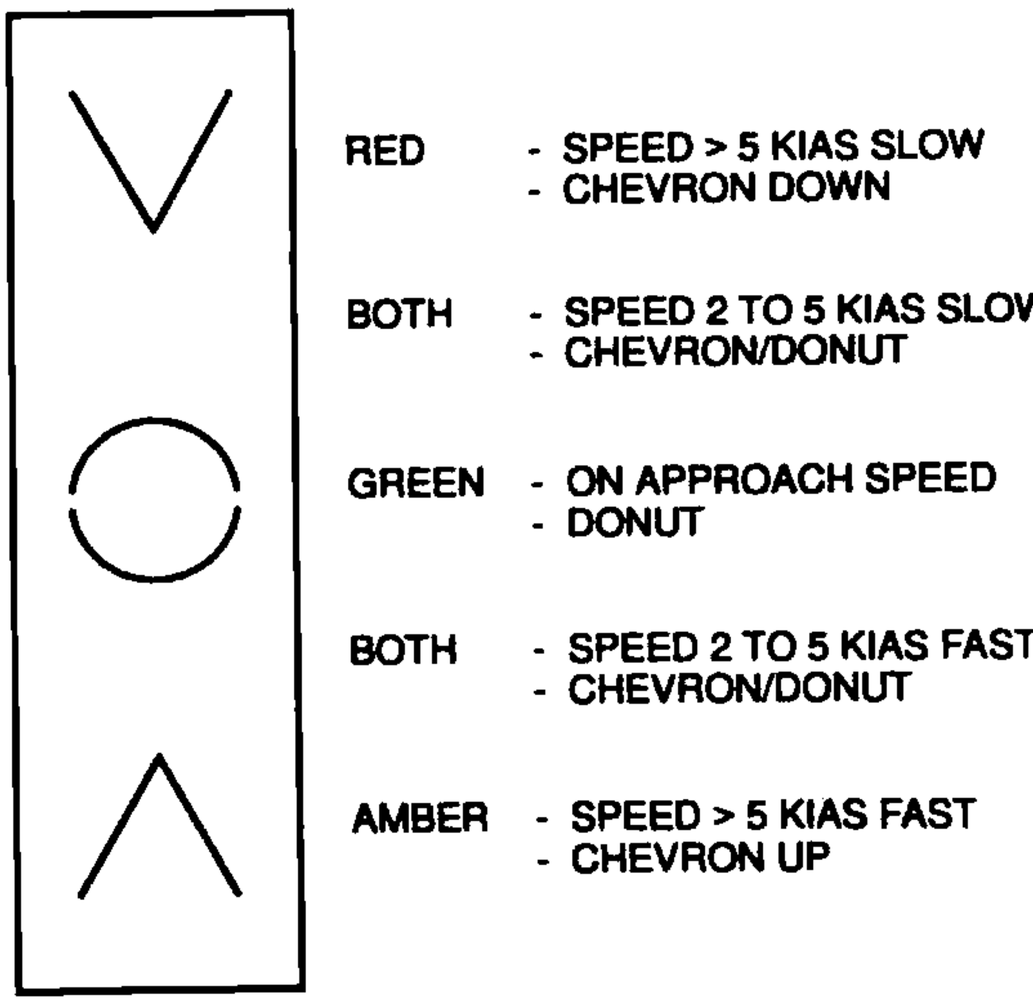



Figure 1-27

## ACCELEROMETER

The G indicating system comprises two accelerometers, one mounted on the instrument panel in each cockpit. The accelerometers provide instantaneous indication of load in the vertical axis (G) and a recording function, where the maximum positive and negative G is recorded by two secondary pointers. These secondary pointers can be reset after the readings have been noted. The accelerometers have integral instrument lighting and are shown at Fig 1-29.

The aircraft is fitted with a fatigue accelerometer, located on the main spar, and a fatigue recorder, which is located in the avionics compartment. The readings are visible on the left side of the rear ejection seat and are recorded after each flight. The recorder is inhibited on touch-down (LG micro-switch), and is powered from the rear generator busbar via the FATIGUE METER C/B.

## ALTIMETER

Pointer-drum counter type pneumatic altimeters are installed in the aircraft with the front instrument having an encoding facility for Mode 3C IFF. They are connected to the static pressure system and measure the difference in barometric pressure between the static pressure and a manually set datum. The difference in barometric pressure is displayed as altitude from the datum in terms of feet. The encoding altimeter provides coded digital altitude (referenced to 1013 HPa) for the transponder. A CODE OFF flag moves from view when electrical power is applied to indicate coded height information is available. The ranges of the altimeters are:

**a.** Altitude: -1,000 to 35,000 ft.

**b.** Barometric: 950 to 1050 HPa (mbar).

**c.** Barometric: 28.1 to 31.0 in.Hg

# AIRSPEED INDICATOR

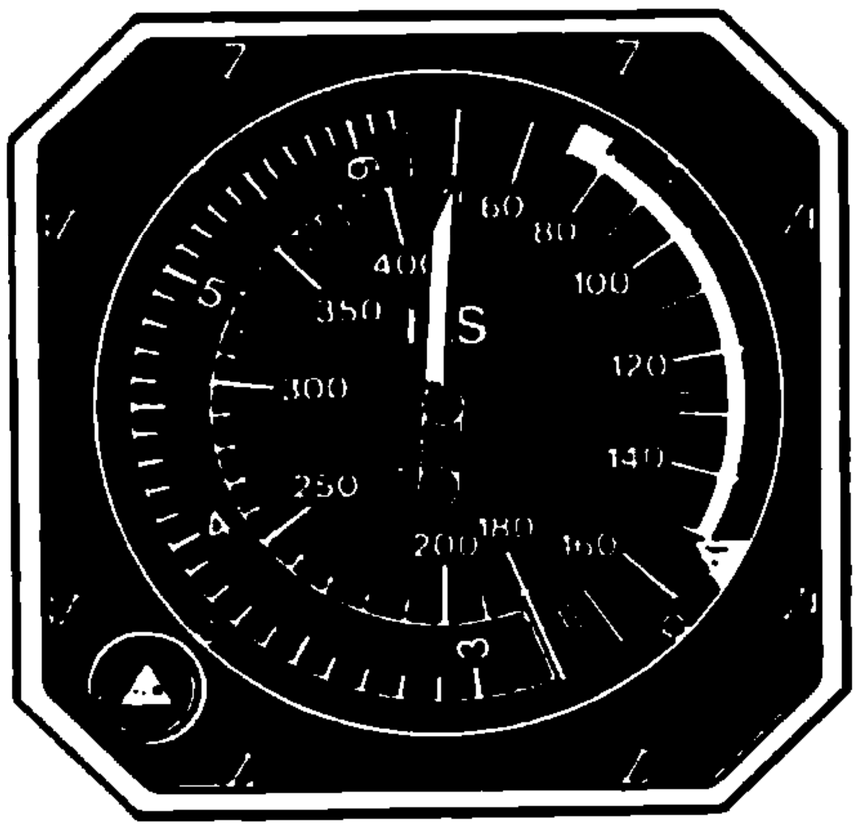


**Figure 1-28**

# ACCELEROMETER

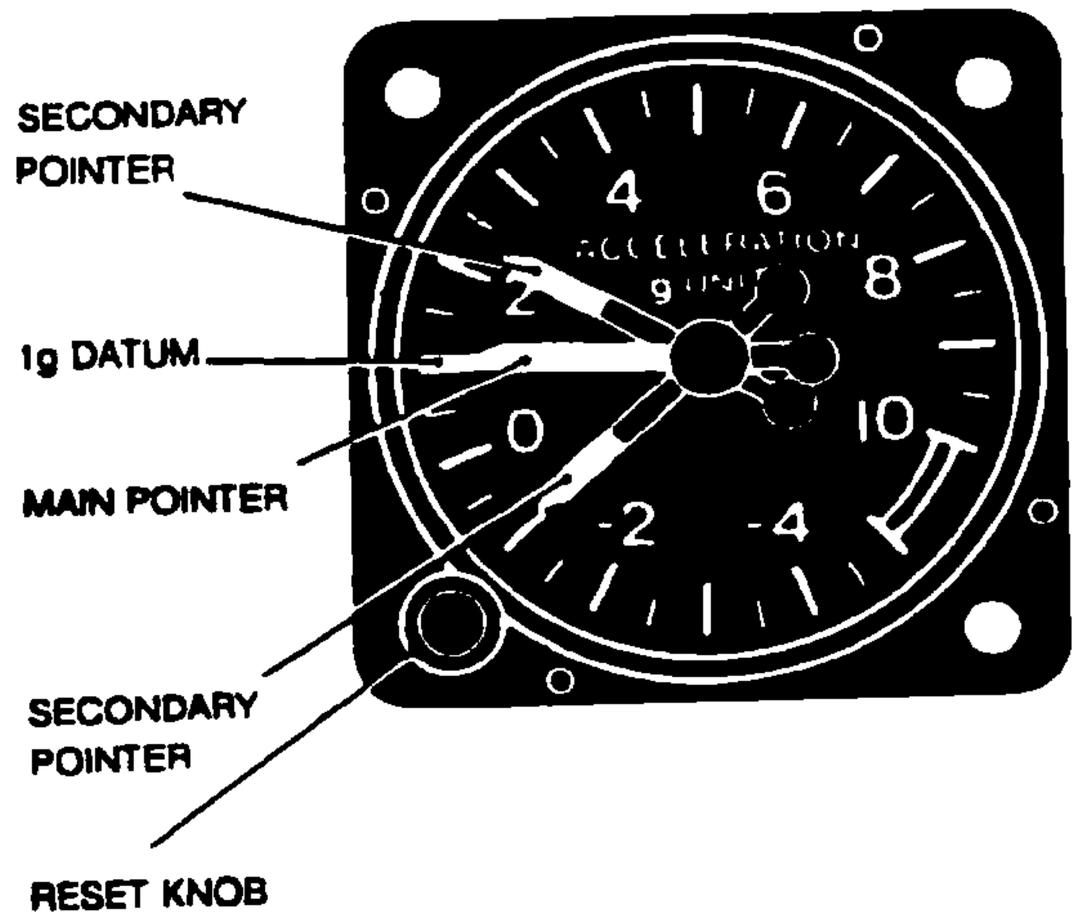



**Figure 1-29**

The dial faces are graduated in increments of 20 ft, divided into ten main divisions. Each of these main divisions represents 100 ft. One revolution of the pointer indicates an altitude change of 1,000 ft. The three drum counter totalises the pointer revolutions and displays the altitude difference in 10,000 / 1,000 / 100 ft units. A barometric scale adjusting knob on the lower left side of the instrument is used to set the barometric pressure datum (QNH / QFE) from which the altitude is measured.

The vibrator in the front (encoding) altimeter is supplied with 28V DC from the front Battery Busbar. The vibrator in the rear altimeter is supplied with 28V DC from the front Generator Busbar. 5V intergral lighting is supplied to both altimeters from the aircraft instrument lighting system. The 'CODE OFF' flag in the front (encoding) altimeter indicates a loss of power to both the encoding device and the vibrator. In the rear altimeter the 'CODE OFF' flag indicates a loss of power to the vibrator only. The altimeter is shown at Figure 1-30.

**NOTE**

The aircraft static pressure system exhibits a large pressure error variation with airspeed. During a deceleration from 260 KIAS to 150 KIAS, the aircraft will descend approximately 100 ft while indicating a constant altitude .

## VERTICAL SPEED INDICATOR

Vertical speed indicators (VSI) are fitted to both cockpits and are connected to the static pressure system. The VSIs measure rate of change of static pressure with changes in altitude. The rate of change produces pointer deflection to indicate speed of ascent or descent to a maximum of 6,000 feet per minute. The instruments have integral lighting from the aircraft instrument lighting system. The VSI is shown at Figure 1-31.

## RADIO MAGNETIC INDICATOR

A King KNI 582 Radio Magnetic Indicator (RMI) (left side of each instrument panel), provides aircraft heading and bearing information to selected stations. The RMI is shown at Figure 1-32.

The bearing information displayed is determined by the aircraft navigational system selected and the position of the ADF-NAV switches on the lower front of the RMI. Integral lighting is provided via the INSTR

## *ALTIMETER*

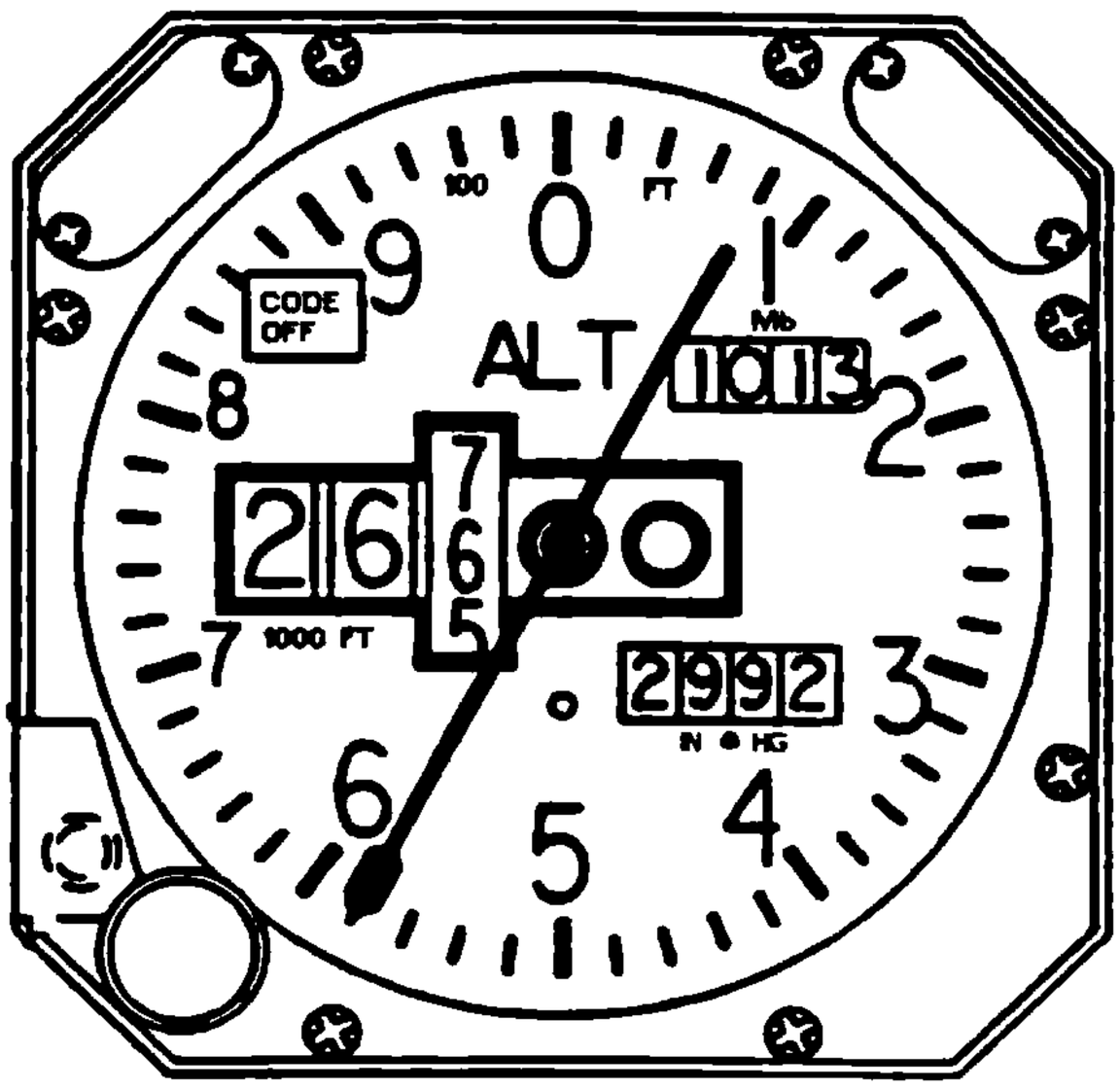



*Figure 1-30*

# VERTICAL SPEED INDICATOR

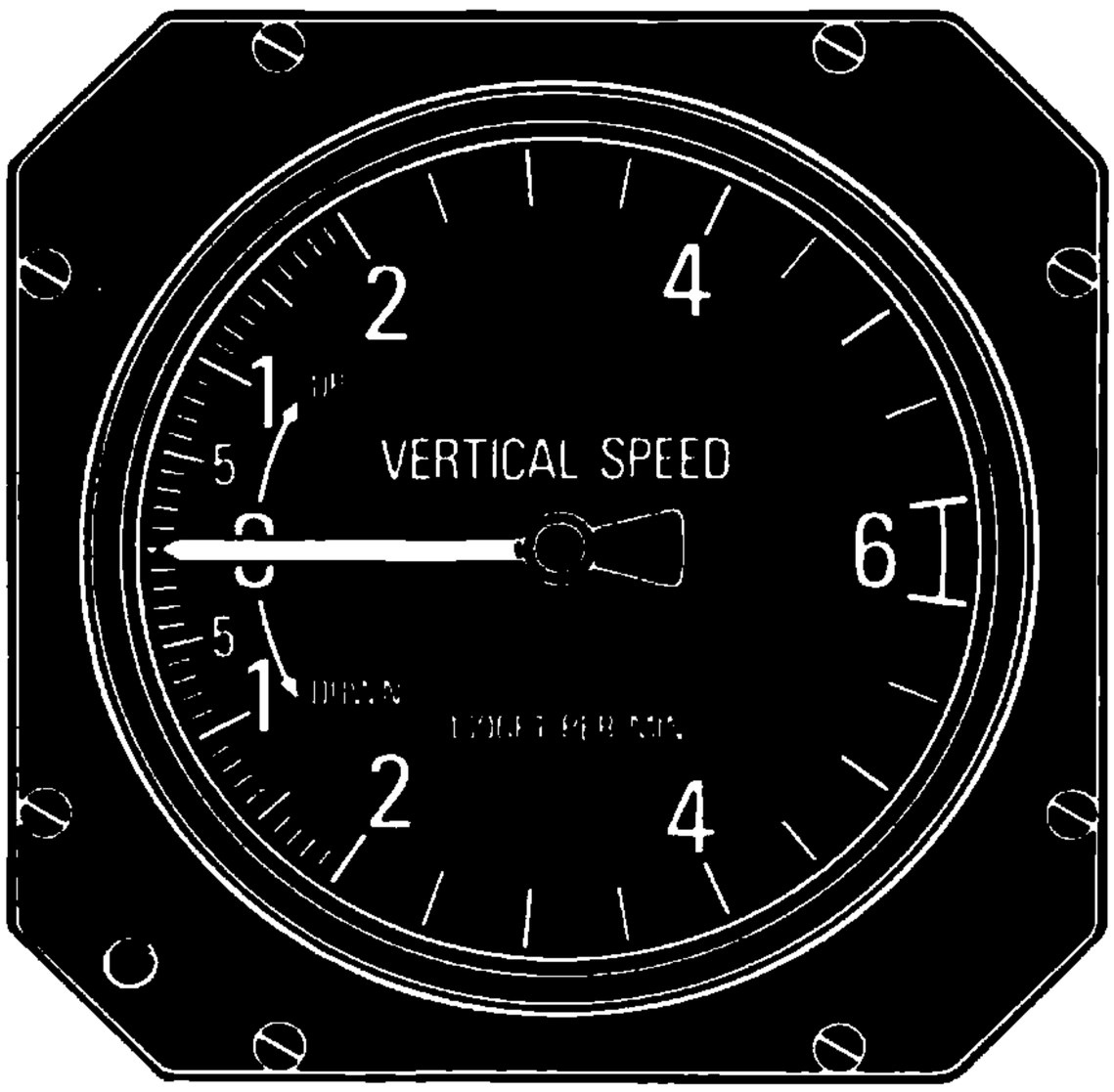



*Figure 1-31*

LIGHTS switch. Aircraft heading is indicated by a fixed lubber line. A HDG (heading) flag comes into view when:

a. Servo error detected.

b. Invalid heading or compass signal.

c. Loss of electrical power.

The navigation system provided to the single bar pointer is controlled by the left side push-switch. With the switch IN the ADF is selected, and the NAV/TAC system is selected in the OUT position. Operation of this switch also causes a small single bar pointer on the lower left to point to ADF or NAV, to provide a visual indication of selected system. If the navigational signal input is too weak, or if NAV is selected with an ILS frequency, the pointer will park at the three o'clock position. The double pointer is used for ADF information only. The double bar push-switch and indication operation is similar to the single bar system. However, since there is no NAV input to the double bar system, the double bar pointer will park when NAV is selected. The RMIs are powered via two RMI C/Bs on each cockpit battery busbar. As the RMIs are part of an integrated navigational system, the INVERTER switch must be selected to BAT or GEN and the AVIONICS BAT and GEN switches selected to ON, before full operation of the RMIs is possible.

### OUTSIDE AIR TEMPERATURE

The outside air temperature (OAT) is measured by a temperature sensitive bulb which is located on the lower surface of the right wing. The signal is sent to

# RADIO MAGNETIC INDICATOR

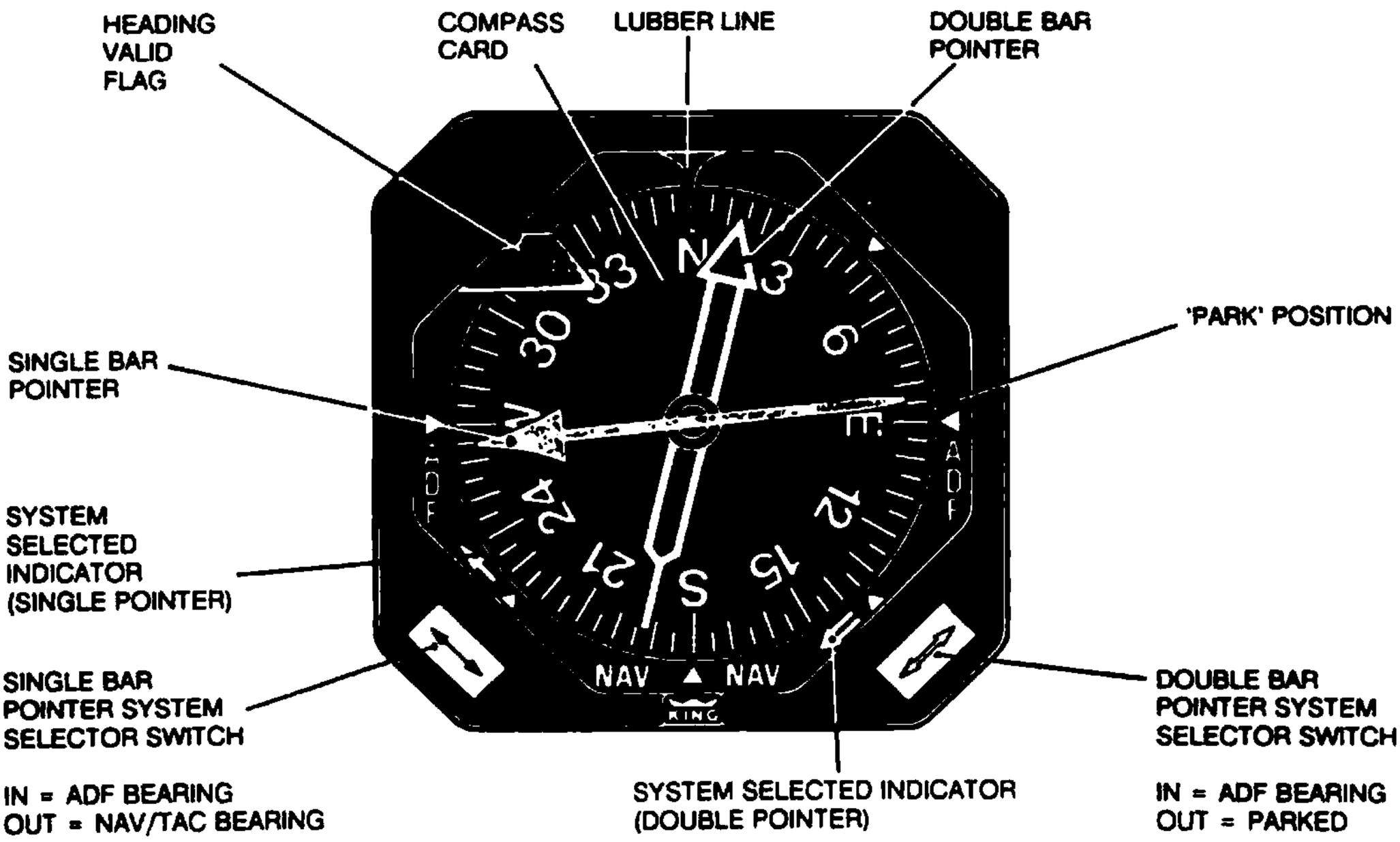



*Figure 1-32*

# OUTSIDE AIR TEMPERATURE

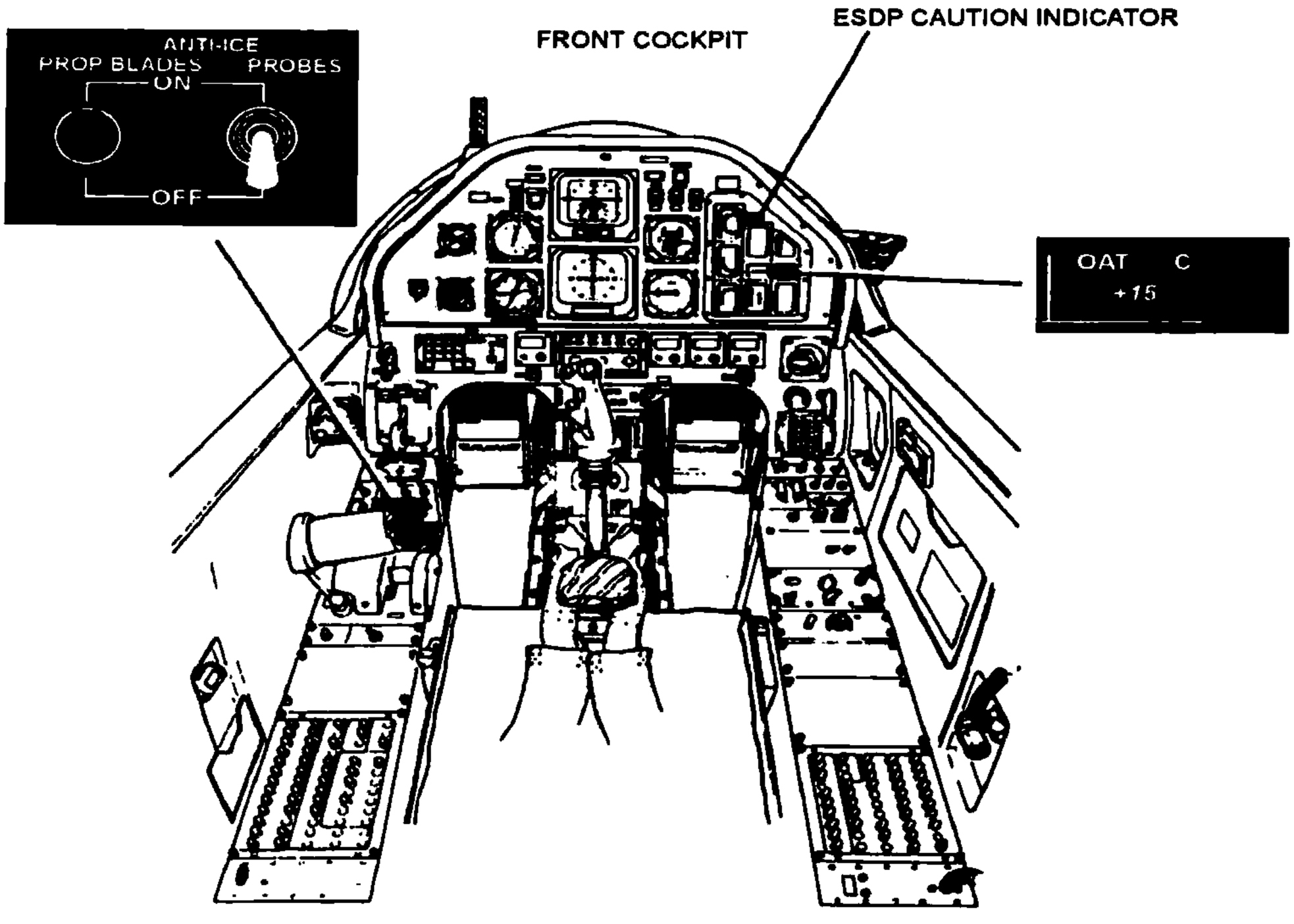



**Figure 1-33**

the SICU where it is processed and displayed on the ESDPs in degrees centigrade. When the OAT falls to +4C° the ESDP caution indicator illuminates and the OAT display commences to flash, to advise the pilot to turn the ANTI-ICE PROBES on. Selection of the probes to ON subsequently resets the caution indicator and the OAT display returns to steady. The OAT indicator is shown at Fig 1-33.

## WARNING SYSTEMS

### Description

Aircraft warning systems comprise the following:

a. A Central Warning system (CWS), which utilises illuminating warning, caution and advisory captions and an aural signal to alert and inform the pilot of aircraft system faults.

b. An Aural Warning System (AWS), which utilises a tone generator to warn the pilot of an impending dangerous or abnormal situation.

c. An Engine and Secondary Instruments Warning System (ESIWS), which utilises visual indicators, located immediately above the ESDP panel, to inform the pilot of a system parameter exceeded, a sensor fault or an integral system fault.

### Central Warning System

The Central Warning System (CWS), shown at Fig 1-34, comprises the following components:

a. An annunciator assembly (combined panel and control box), located on the right side of the lower instrument panel in each cockpit.

b. Master WARNING PRESS TO RESET and master CAUTION PRESS TO RESET indicators are located on the upper instrument panel in each cockpit.

### System Operation

Detection circuits in the front CWS alert the pilots to a warning or caution condition by switching on an electronic 'gong' and illuminating the appropriate WARNING or CAUTION light. The annunciator panel caption identifying the faulty system is illuminated at the same time. The volume of the gong automatically diminishes and switches off within a few seconds. The WARNING or CAUTION light remains illuminated until it is manually cancelled (PUSH TO RESET). The annunciator caption remains illuminated until the fault condition is removed. The annunciator panel advisory captions illuminate when the pilot switches on certain systems which are not normally operating (as a reminder). The pilot can de-activate the system and extinguish the caption at any time with the exception

# CENTRAL WARNING SYSTEM

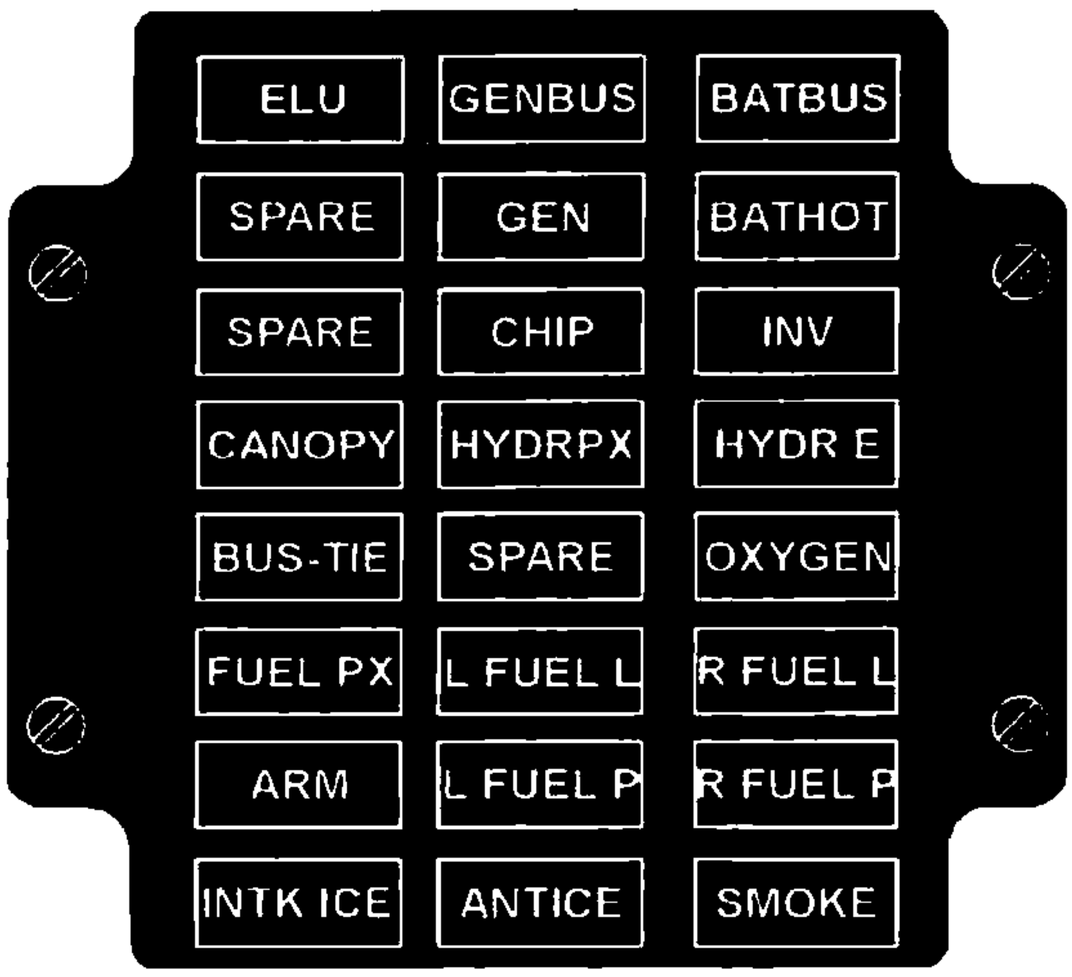



**Figure 1-34**

of the intake ice system, which can be reset only when the aircraft is on the ground. The gong and the master WARNING and CAUTION lights are not activated/illuminated for advisory captions. Master WARNING and CAUTION light and annunciator caption illumination intensity is reduced when the instrument lights are selected on. The lamps and electronic gong are tested by pressing the SYSTEM TEST push-switch, located on the left side console, behind the PCL, in each cockpit. The gong signal is connected to the audio system from the rear annunciator assembly for transmission to the pilots headsets. The gong will operate only on detection of the first warning and the first caution condition unless the system is reset by pressing and cancelling the active WARNING or CAUTION light. An inhibiting circuit in the rear annunciator assembly prevents distracting gong operation during engine starting. The inhibiting circuit is connected to the engine starter switch in the front and rear cockpits, and is activated when either starter switch is selected to ON. The SYSTEM TEST push-switch in each cockpit is connected to the front annunciator assembly. Pressing either push-switch completes a grounding circuit which causes all captions on the front and rear cockpit annunciator, master WARNING and master CAUTION lights to illuminate with the exception of the LD GR red lights which are only illuminated by the switch in the actuating cockpit. The electronic gong is activated at the same time.

## Annunciator Panel

Each annunciator panel has provision for 24 captions. The captions are coloured red, yellow or green when illuminated to indicate:

a. Warning (red): an emergency situation.

b. Caution (yellow): a dangerous situation.

c. Advisory (green): a system not normally in use is operating.

To ensure continued operation in the event of a single busbar failure, independent system supply lines are connected to the front control box/annunciator panel from the front battery busbar and front generator busbar. The system supply circuit breakers are placarded CAUTION PANEL and are located on the battery and generator busbar C/B panels in the front cockpit. Each control box contains its own dimming circuit which is activated when the individual cockpit instrument lighting switch is selected to ON. The warning (red) captions illuminate under the following conditions:

a. ELU: engine electronic limiting unit inoperative.

b. GEN BUS: generator busbar voltage below 14 VDC.

c. BAT BUS: battery busbar voltage below 14VDC.

The caution (yellow) captions illuminate under the following conditions:

a. GEN: generator off-line or below 23.6 VDC for more than 1.5 seconds.

b. BAT HOT: battery overheating or battery hot system plug disconnected.

c. CHIP: metal particles present in engine oil system.

d. INV: inverter 26 VAC output below 13 VAC.

e. CANOPY: canopy not closed and locked.

f. HYDR PX: main hydraulic system depressurised.

g. HYDR E: emergency hydraulic system nitrogen pressure low.

h. FUEL PX: fuel pressure low.

i. L FUEL L: left wing tank fuel level low or, with left external fuel transfer pump selected on, left underwing tank fuel level low.

j. R FUEL L: right wing tank fuel level low, or with right external fuel transfer pump selected on, right underwing tank fuel level low.

k. OXYGEN: caption illuminates when pressure falls to 300 psi.

l. BUS-TIE: bus-tie C/B popped.

The advisory (green) captions illuminate to indicate:

a. L FUEL P: booster pump energised in left wing tank.

b. R FUEL P: booster pump energised in light wing tank.

c. INTK ICE: engine intake ramp and bypass door actuated.

d. ANT ICE: anti ice system heating elements activated.

e. ARM: smoke arm switch ON (where fitted).

f. SMOKE: smoke trigger switch activated (where fitted).

### Aural Warning System

The main component of the Aural Warning system (AWS) is the tone generator component of the combined AOA electronic unit/tone generator assembly. The assembly is located in the front instrument compartment on the upper rear face of the firewall. The tone generator circuit is supplied with 28 VDC from the front generator busbar via the AURAL WARN C/B. The tone generator aural warning signals, when activated, are transmitted to the pilots' headsets via the aircraft audio system. The aural warning signals generated are detailed in this section.

### Landing Gear Warnings

A continuous 1100 Hz tone warning is activated to alert the pilot when the landing gear (LG) is UP under certain conditions. This warning is activated when the LG control unit is selected to LG UP and any of the following three conditions exist. In all cases, de-activation of this warning can be achieved by selecting the landing gear down.

a. The Power Control Lever (PCL) is retarded to a low power setting and airspeed is below 120 KIAS. This 'gear up/power low/low IAS warning is intended to prevent an inadvertent wheels-up landing.

**NOTE**

The gear up/low power/low airspeed warning can be cancelled by pressing the WARNING SILENCER push switch on the LG control panel in either cockpit.

b. Flaps are in the LAND position. The warning is de-activated when flaps are operated to the TO (take-off) or UP position. This 'gear up/flaps down' warning is also intended to prevent an inadvertent wheels-up landing.

c. Weight is on the main landing gear. This 'gear up/ aircraft down' warning is intended to prevent inadvertent LG retraction on the ground.

In all of the above situations the red lights in the landing gear control panel will also illuminate.

### Stall Warning

An intermittent 1100 Hz tone (2 Hz interrupt) is provided to warn of approach to the stall at high angles of attack. This warning is only provided when weight is off the main landing gear. The warning circuit remains inactive until a stall condition is detected by the AOA (angle of attack) transmitter. The stall warning circuit is routed through the LG safety switch, to prevent the pilot from being distracted during taxiing by spurious warnings, caused by AOA transmitter 'bounce'.

### G Warning

A continuous 600 Hz tone is provided to warn of wing loading in excess of 6 G. This warning is activated by an inertia switch, located on the upper rear face of the firewall. The switch closes when more than 6 G is applied to warn the pilot that the aircraft is approaching its maximum positive load limit.

### Overspeed Warning

An intermittent 1600 Hz tone is provided to warn the pilot when 0.68 Mach ($M_{MO}$) is exceeded. Speed information for the warning is taken from a 0.68 Mach switch on the front combined airspeed/mach indicator.

## EJECTION SEATS

### EJECTION SEAT SYSTEM

### Introduction

The MK AU 11-A Martin Baker ejection seat is a fully automatic, cartridge operated ejection seat. It provides safe escape for most combinations of aircraft altitude speed, attitude and flight path, within the envelope of zero altitude and 40,000 ft. The -1 and -2 seats are fitted to the front and rear cockpit respectively. Ejection is initiated by pulling the seat firing handle situated between the thighs on the front edge of the seat pan. The parachute pack is fitted with canopy breakers and the seat is ejected through the canopy. After ejection, man/seat separation and parachute deployment are automatic. A manual separation (override) system provides for possible failures of the automatic system. The seat is ejected by action of the gas pressure developed within a telescopic ejection gun when the cartridges are ignited. A command firing system is fitted enabling the occupant of the rear seat to initiate the ejection sequence for both seats. A command delay breech unit in the front cockpit ensures that on command ejection, separation is maintained to prevent a possible collision. The front-left view of the ejection seat is shown at Fig 1-36, left-rear view at Fig 1-37, and the right-rear view at Fig 1-38.

**WARNING**

The ejection seat is a potential source of injury to personnel and damage to aircraft if inadvertently operated. Before entering the cockpits, ensure that the safety pin in front of each seat pan is correctly fitted.

# AURAL WARNING SYSTEM

*Figure 1-35*

**(Deleted)**

# EJECTION SEAT FRONT LEFT

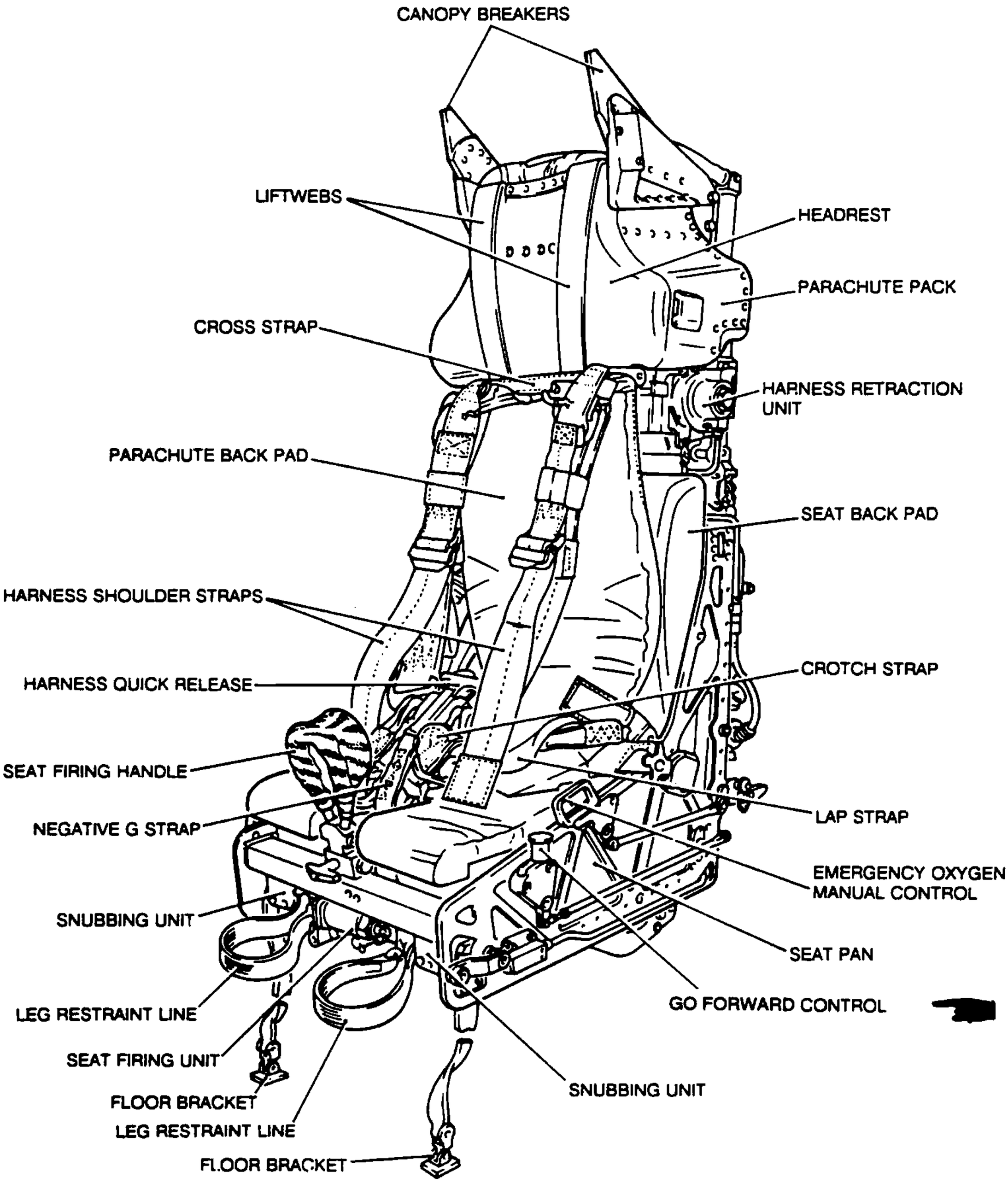



*Figure 1-36*

# EJECTION SEAT LEFT REAR

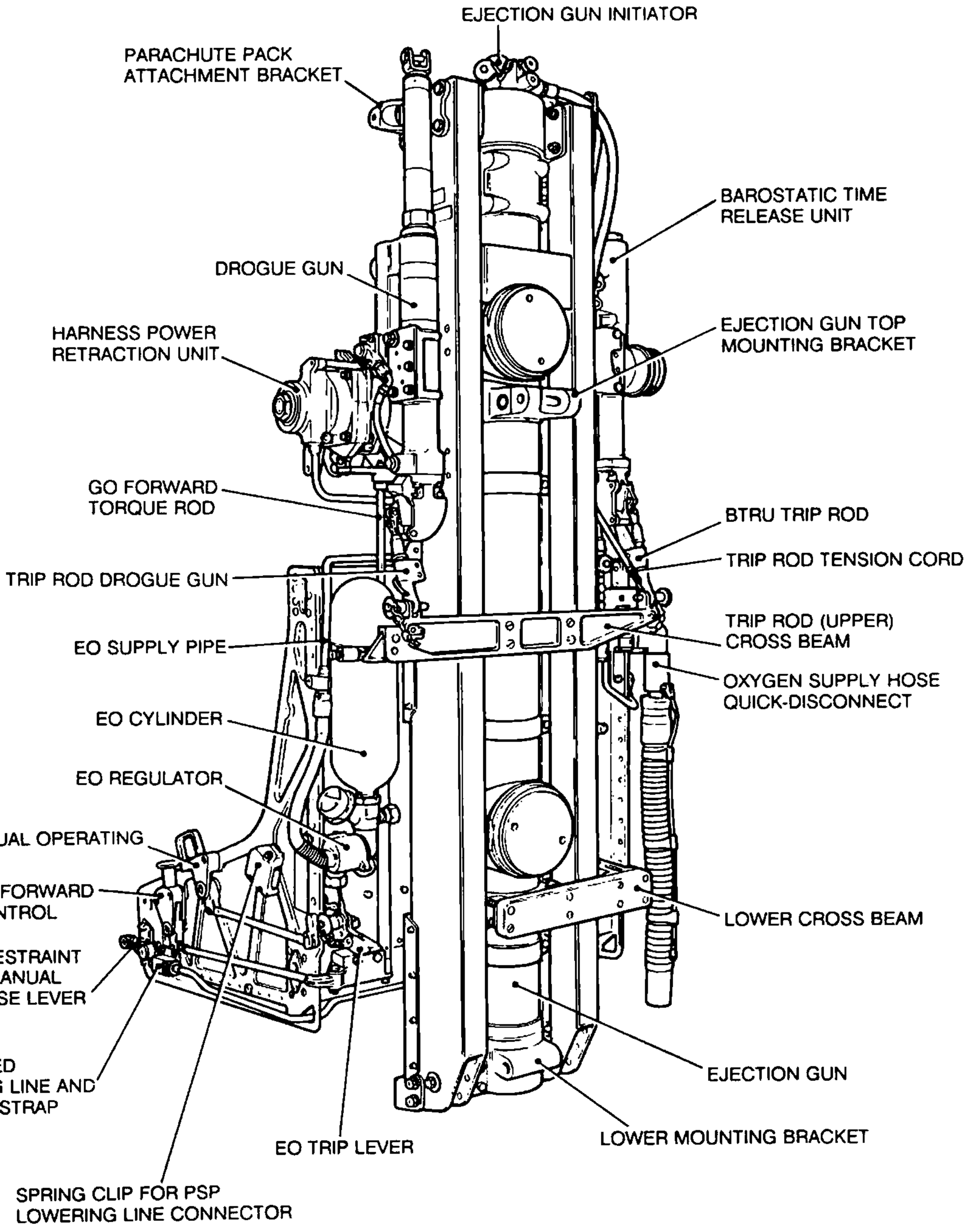



***Figure 1-37***

# EJECTION SEAT RIGHT REAR

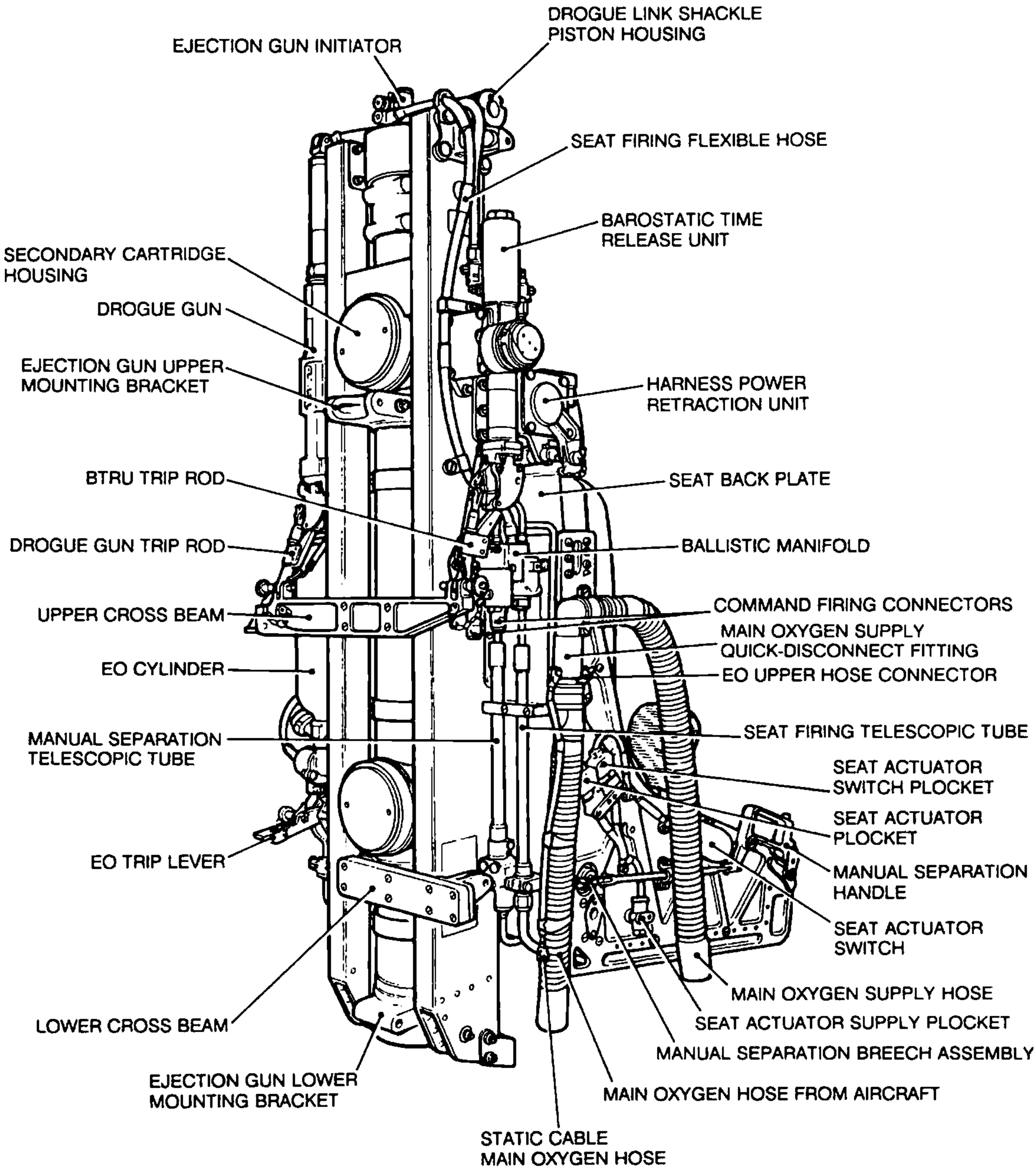



*Figure 1-38*

## General Characteristics

The MK AU 11A seat has the following characteristics:

a. ***Envelope.*** Zero altitude at 60 KIAS (no downwards vector) minimum and 40,000 ft at 400 KIAS maximum.

b. ***Seat Weight.*** Approximately 115 lbs including harness and excluding survival equipment.

c. ***Cartridges.*** Total of 9 cartridges per eat (2 in the command ejection system) providing maximum acceleration of 14 to 16 G.

d. ***Harness Retraction.*** The shoulder harness is automatically retracted when ejection is initiated. The harness is normally locked for flight, but may be unlocked with a go-forward lever on the left side of the seat.

e. ***Leg Restraint.*** Legs are correctly positioned and prevented from flailing. Leg garters are automatically pulled into the seat by cords connected to shear pins on the cabin floor. The cords pass through automatic release units which also allow for manual quick release. The legs must be left in the stretched-out position when initiating ejection.

f. ***Height Adjustment.*** An electric actuator is operated by a spring loaded toggle switch on the right side of the seat for vertical adjustment. The pilot positions the seat using the small sight and crosshairs on the top of each instrument panel.

**CAUTION**

The seat adjusting actuator is not to be operated for more than 1 minute in any 8 minute period. It is most important that the seat pan height is adjusted to the normal inflight position before strapping in, to ensure correct adjustment of the harness.

g. ***Emergency Oxygen.*** Emergency oxygen with 5 to 10 mins supply is provided by a small 1800 psi bottle mounted on the left side of the seat. The bottle is automatically activated upon ejection and feeds into the pilot's oxygen mask. The oxygen may be manually discharged inflight by pulling a manual release handle on the left side of the seat.

h. ***Survival Pack.*** The survival pack is located in a hardened container which forms the basis of the seat. The pack is connected to the harness and may be released before landing on a lowering line.

I ***Seat Pins.*** Five seat pins are located on the seat. Four of the pins are connected to a red lanyard and are removed prior to strapping in. These pins are stored in a small case on the upper, rear leftside of each cockpit. The fifth pin (seat pan firing handle) is removed in the after start checks and is stored in a tube on the canopy handle (left sill). The seat pan firing pin is visible by the start crewman. Safety pin location is shown at Fig 1-39.

j. ***Command Ejection.*** A control valve with an ON/ OFF selector is located in rear cockpit right side against the back bulkhead. The handle knob is on a threaded shaft and must be turned anti-clockwise to enable movement of the valve handle. It is rotated clockwise to lock the valve after selection. When the rear seat handle is pulled the rear seat ejects immediately and the front seat ejects 1.25 sec later. The command ejection system is shown at Fig 1-40.

## Ejection Performance

The pilot must be aware of the requirements for a successful ejection. The best chance of a successful ejection is when the aircraft has an upward vector and is upright and wings level. If the aircraft has any downward vector, the seat will require extra height to safely deploy the chute and retard the occupant. Ejection seat performance is summarised in Fig 1-41 and operating limitations are listed in Section 5.

## Manual Separation

The Manual Override (MOR) handle, located on the right side of the seat, is locked until the seat firing handle is pulled. The pilot may release himself from the seat after ejection, if automatic separation fails, by simultaneously depressing the button on the MOR handle and pulling it upward. This action will fire the drogue gun (if drogue chute has not already deployed) and override the BTRU, allowing deployment of the main parachute and separation of the occupant from the seat.

**WARNING**

If the command ejection system is used, the occupant of the front seat must first pull the seat firing handle to unlock the Manual Override (MOR) handle. The MOR handle will be inoperative whilst the seat firing handle remains in place.

# SAFETY PIN LOCATION

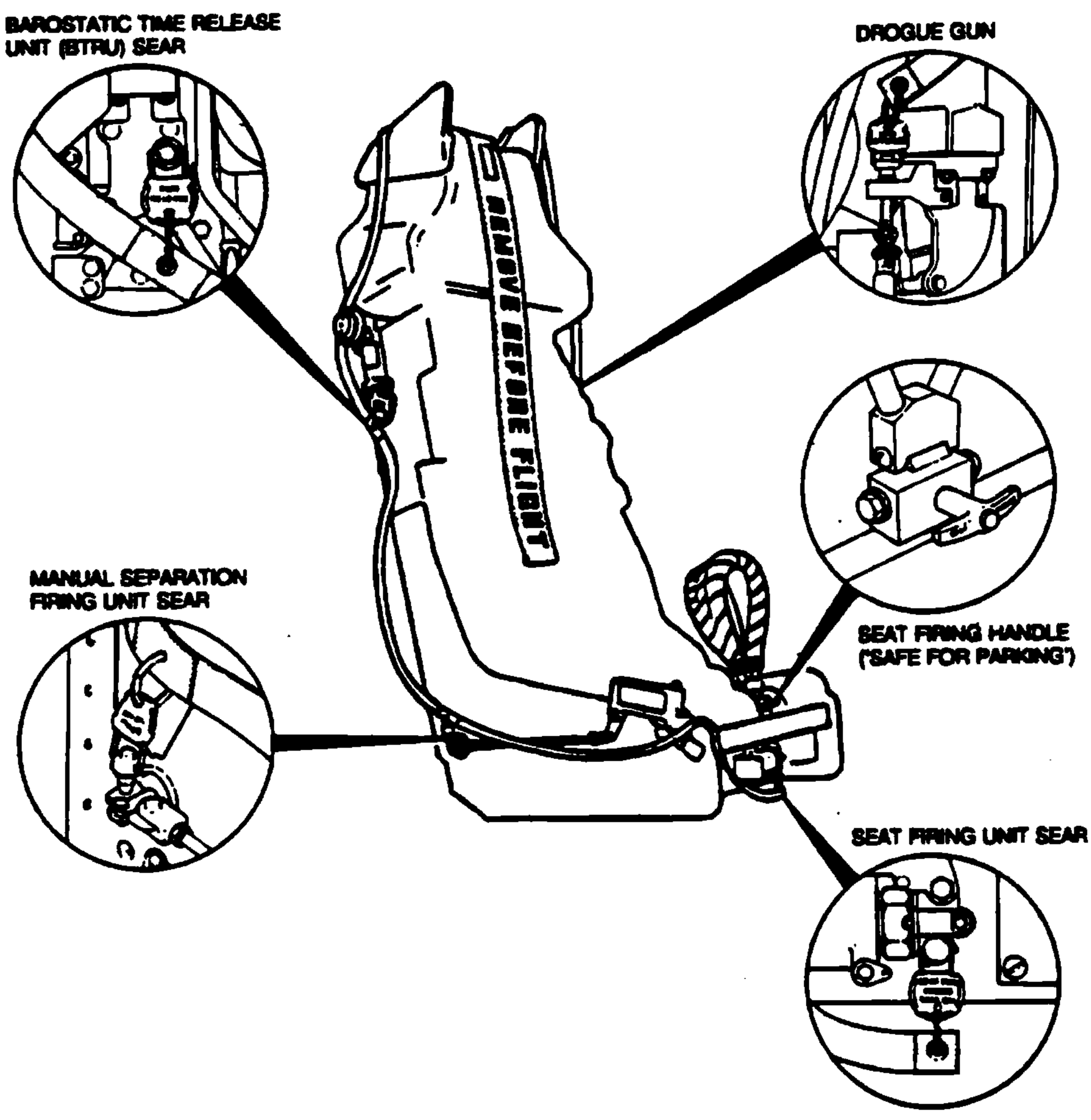
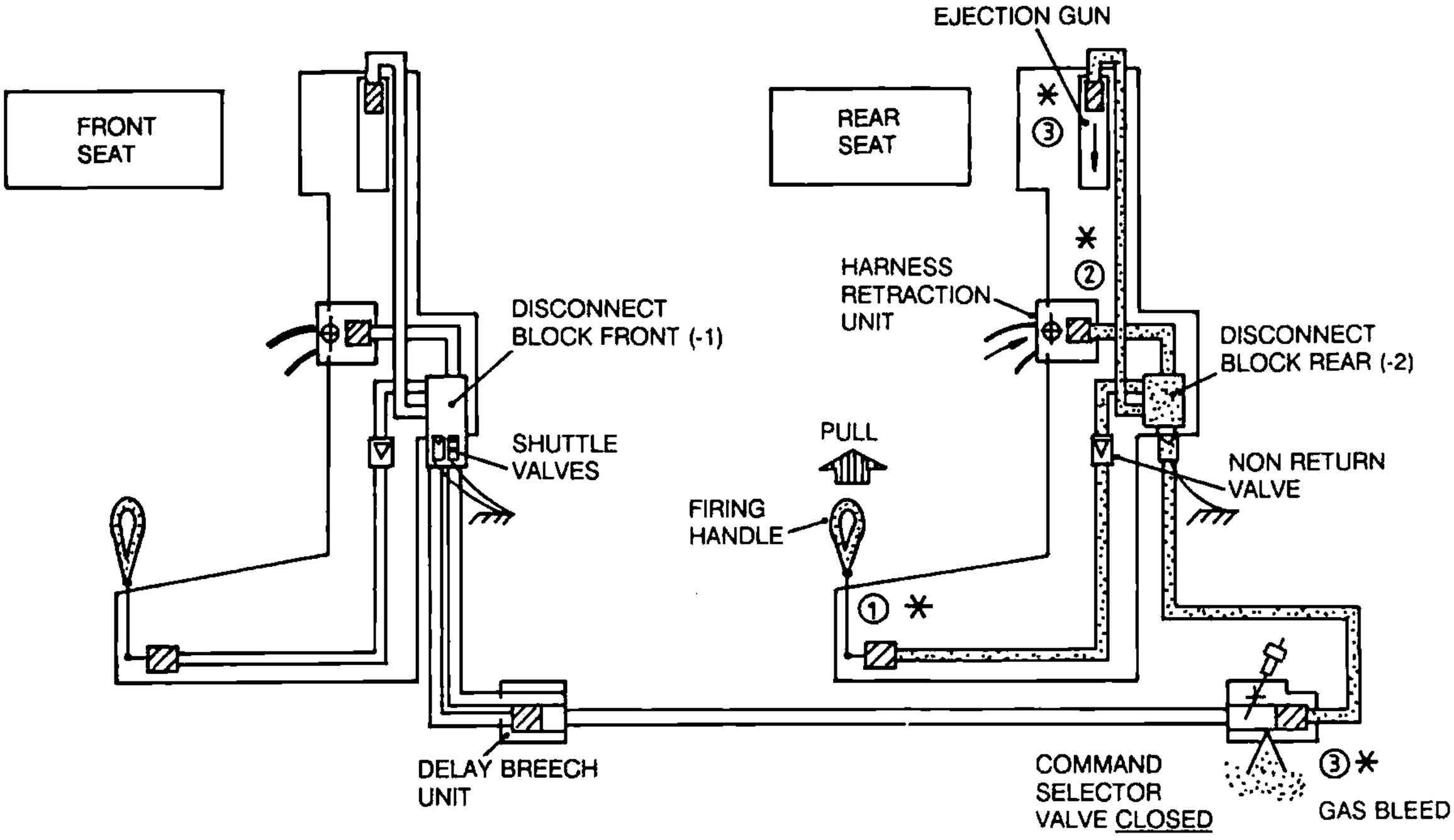



*Figure 1-39*

# COMMAND EJECTION SYSTEM

## REAR SEAT EJECTION ONLY




## REAR SEAT INITIATED COMMAND EJECTION

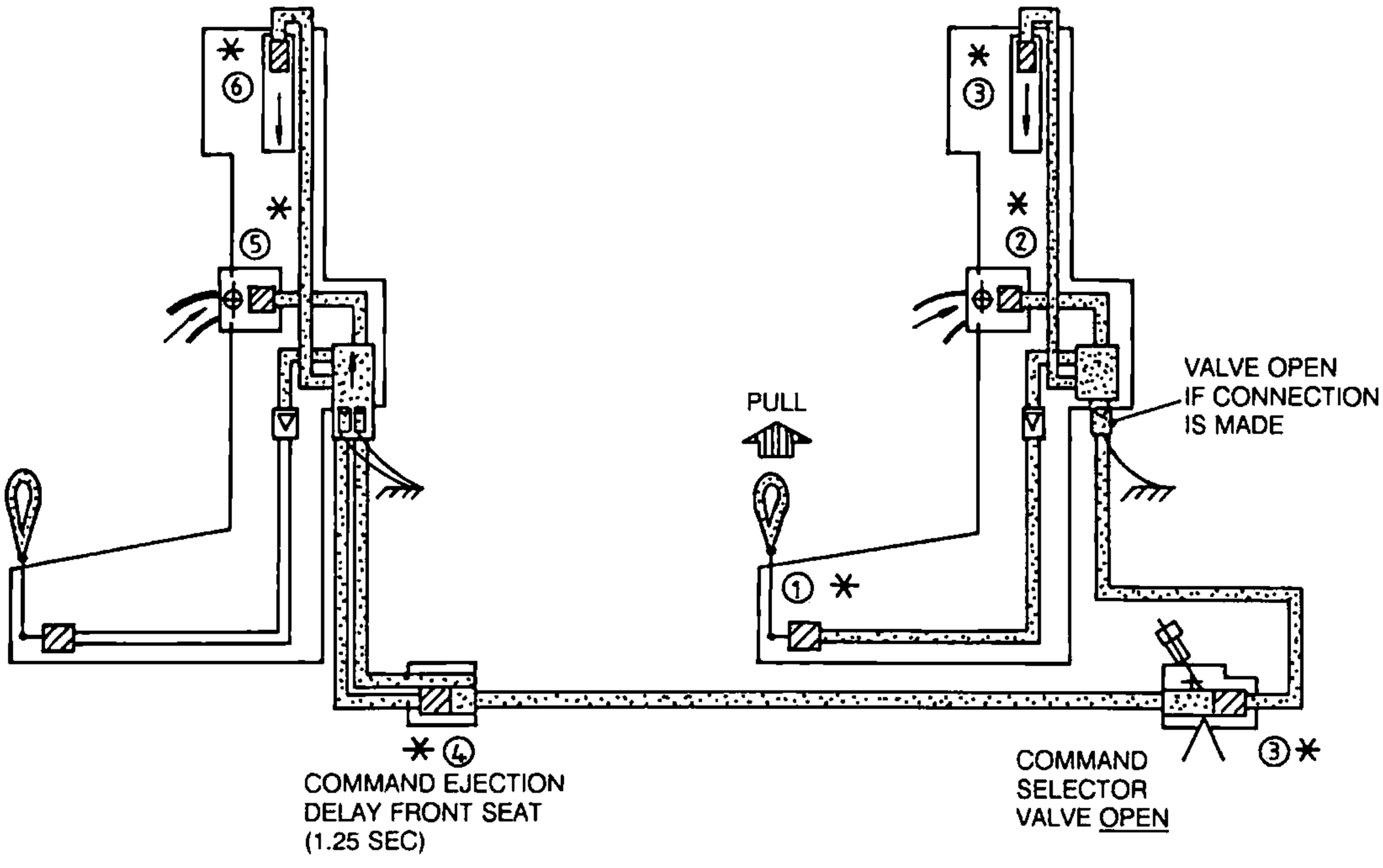
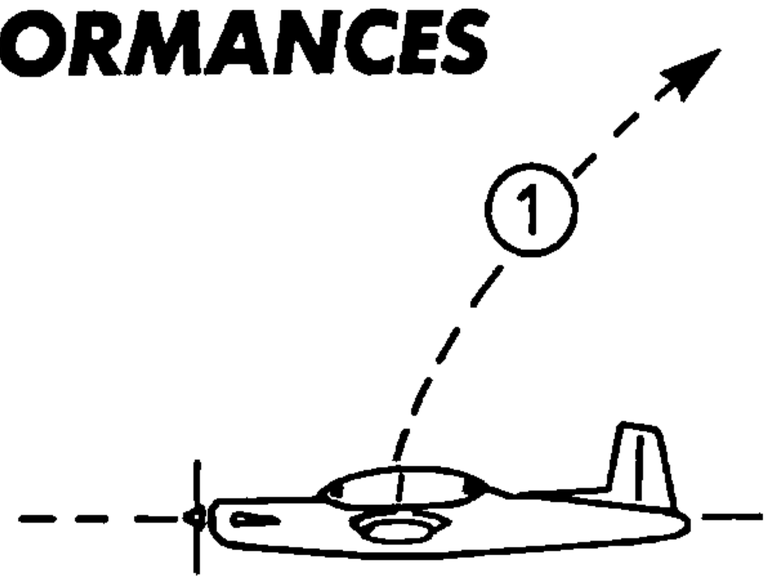



***Figure 1-40***

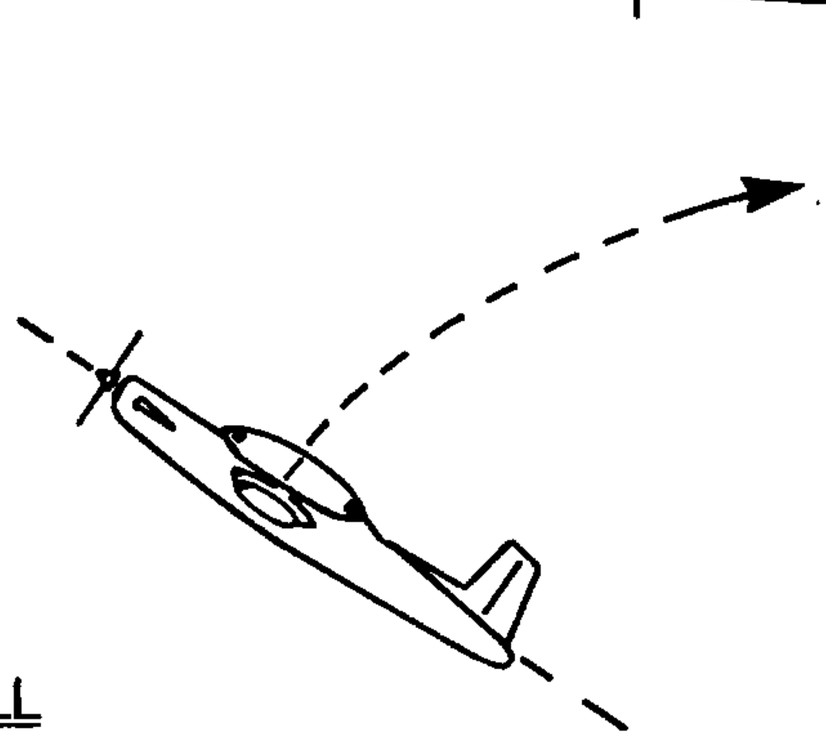
# EJECTION PERFORMANCES



1

CLIMB OR
LEVEL FLIGHT
(IDEAL)

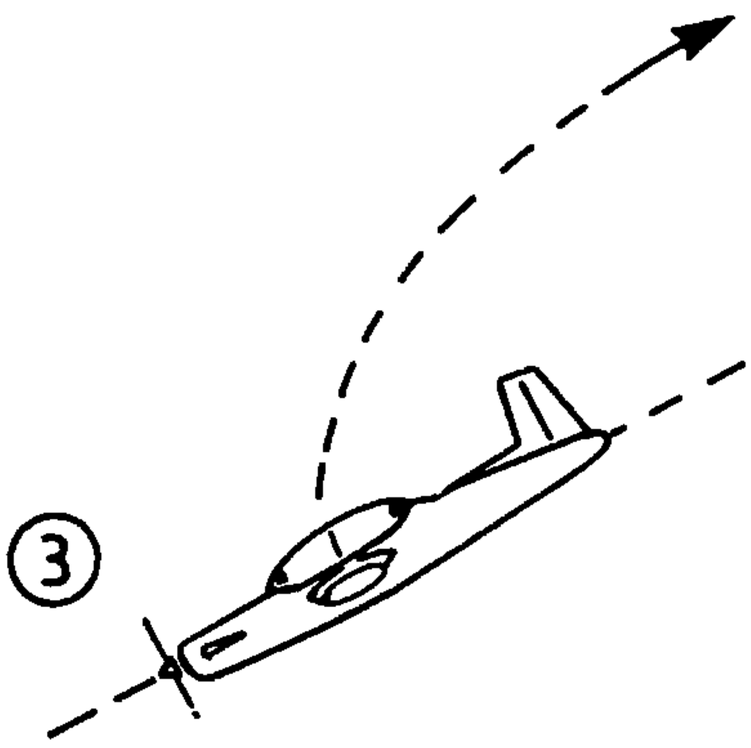
BEST CHANCE OF SURVIVAL



2

STALL

EXAMPLE:
60 FT/SEC ROD -
THE SEAT HAS NO SPEED AFTER
LEAVING AIRCRAFT AND MUST FALL
FURTHER TO DEPLOY PARACHUTE



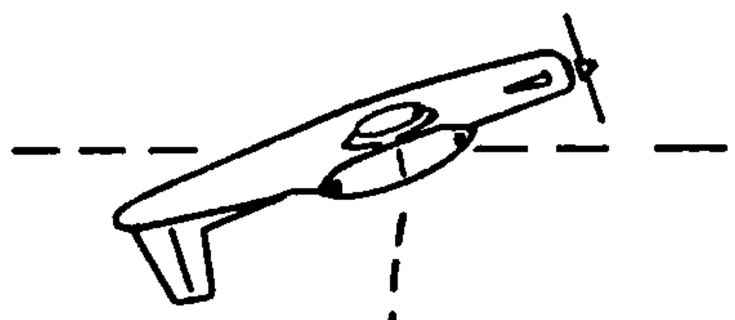
3

DIVE

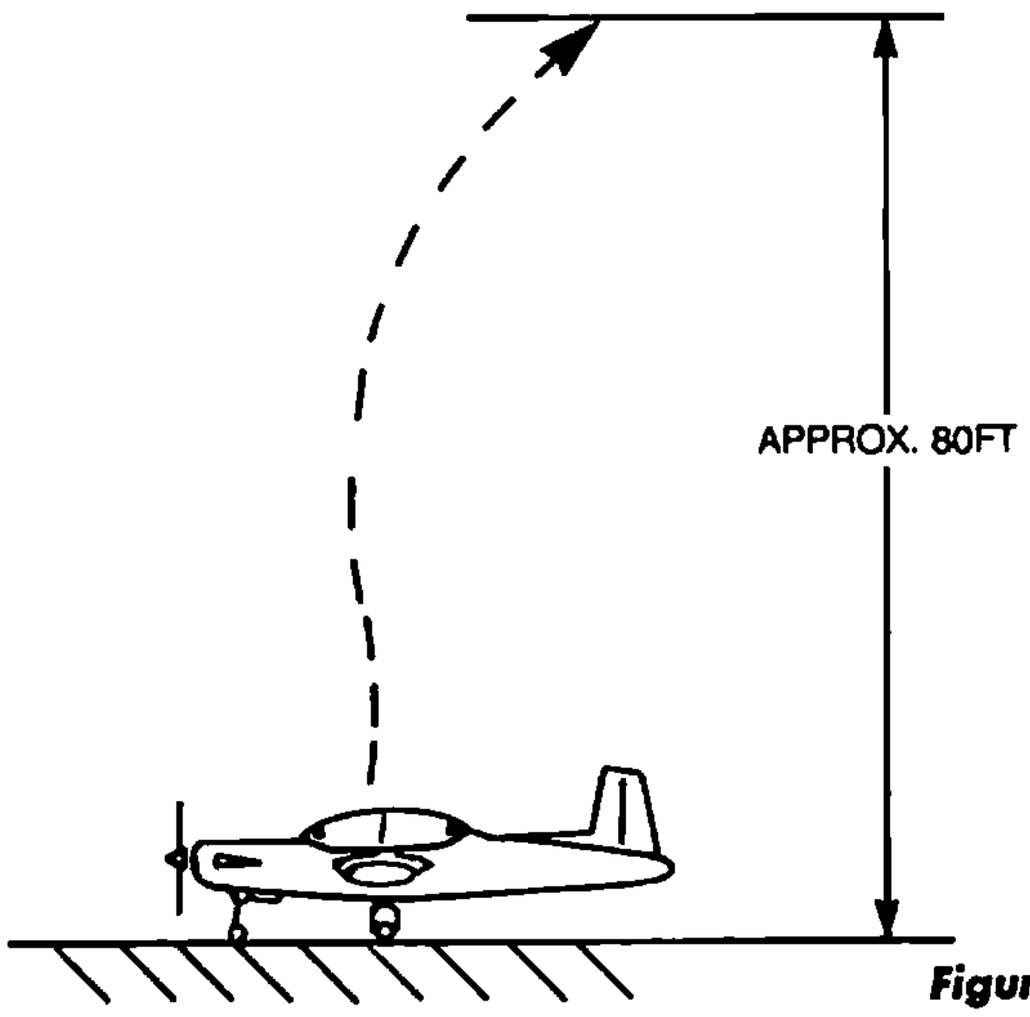
FORWARD SPEED OF THE SEAT
INCREASES HEIGHT NEEDED
FOR SURVIVAL

INVERTED FLIGHT

MINIMUM SAFE ALTITUDE AND SPEED:
* 240FT ABOVE GROUND AT 125 KIAS
* 140FT ABOVE GROUND AT 400 KIAS



4




5

ON GROUND
ZERO SPEED

POOR CHANCE OF SURVIVAL

| CHANCE OF SURVIVAL ON GROUND |
| --- |
| MINIMUM SPEED - 60 KIAS |

*Figure 1-41*

### Solo Flight

An apron must be fitted over the rear seat and harness prior to solo flight to prevent flailing of harnesses, buckles. leads and restraint lines.

## EJECTION SEQUENCE

The following seat ejection sequence is as follows:

a. **0** ***sec*** - Ejection.

b. **0.5** ***sec*** - Drogue gun fires.

c. **1** ***sec*** - Drogue and seat is stabilised.

d. **1.25** ***sec*** - Command ejection (if used) delay for front seat ejection.

e. **1.5** ***sec*** - Provided the seat is below 5000 m (as detected by the Barometric Time Release Unit), the main parachute is deployed.

f. **2.6 sec** - Parachute is fully deployed (unless delayed by BTRU).

The ejection sequence (steps 1 to 8) is shown at Fig FO-10.

### Command Firing System

The ejection sequence will depend on the selection of the command firing system control valve. With the selector lever in the ON position, if the rear seat occupant (normally the instructor) initiates ejection, the rear seat is ejected followed, after 1.25 sec, by the front seat. Initiation of ejection by the front seat ejects only the front seat regardless of the command ejection control valve position. If the aircraft is being flown from the front seat with a passenger in the rear seat the control valve is set to the OFF position prior to flight so that initiation of ejection by either occupant will eject only the one seat.

### Cartridge Sequence

On pulling the rear seat firing handle the seat firing unit cartridge fires, piping gas to the gas manifold assembly. From the gas manifold, the gas passes simultaneously to:

a. The harness power retraction unit firing the cartridge in the breech to operate the unit, drawing the occupant back to the correct posture for ejection.

b. The ejector gun gas initiator firing the primary cartridge (the two secondary cartridges fire as the seat ascends the rail).

c. The command ejection control valve, firing a cartridge in the valve breech.

If the command ejection control valve is closed the gas is vented to atmosphere and the front seat remains unaffected by the rear seat ejection. With the control valve in the open position (ON) the gas from the control valve then passes to the front seat harness power retraction unit, via the command delay breech unit, retracting the occupant to the correct position. A time delay mechanism delays the firing of a cartridge in the command delay breech unit which passes gas to the front seat manifold valve and onto the ejection gun primary cartridge. The two secondary cartridges fire as the seat ascends the rails.

### Seat Ascent

As the seat ascends the rail, canopy breakers shatter the aircraft canopy. The following actions also occur automatically:

a. The drogue gun delay mechanism is armed.

b. The Barometric Time Release Unit (BTRU) is armed.

c. The main oxygen supply hose and the seat pan actuator electrical supply are disconnected.

d. The emergency oxygen supply is operated.

e. The command ejection quick disconnects are disconnected.

f. The leg restraint lines are tightened to draw the occupants legs back against the seat.

After the delay elapses the drogue gun cartridge fires and the drogue gun piston is ejected, drawing out the drogue parachute.

### Parachute Deployment

The BTRU has a delay to allow the drogue to stabilise the seat. Once this delay has elapsed and if the seat is below 5000 m, the BTRU cartridge fires to allow the following:

a. The drogue shackle link is freed, operating the parachute mechanical lock which enables the parachute to withdraw from the pack.

b. The upper and lower harness release locks are released, freeing the upper and lower harness lugs, negative G strap and leg restraint lines.

The parachute lifts the occupant and survival pack from the seat. A normal parachute descent follows. On descent the pilot must manually release one side of the single handed release strap to enable the Personal Survival Pack (PSP) to fall on the lowering line, automatically inflating the life-raft. Subsequent operation of the quick release connector on the Secumar life vest will enable complete separation from the PSP. Once the occupant has landed the harness may be removed by operation of the Quick Release Fitting (QRF).

## STRAPPING-IN PROCEDURE

### Seat Pre-flight

**WARNING**

Ensure the MOR linkages on the right of the seat pan are free of obstruction prior to seat height adjustment. Loose objects may jam and bend the linkages, causing the drogue to fire.

### Quick Release Fitting

**WARNING**

If the ejection seat is lowered with the QRF below the seat, the gas initiator sear may be fouled, causing the seat to fire. Whenever the seat height is adjusted, the QRF and straps must be placed on the seat or the occupant is to be strapped in.

To place the QRF in the 'DON' position, depress the thumb button, lift the T-handle and allow the handle to rotate forward approximately 45° under spring pressure. To fit a lug into the QRF, rotate the T-handle to the 'DON' position, engage the swivel lug in the side aperture of the fitting and press home until it clicks. To lock the QRF, depress the thumb button and press the handle into the body with the other hand. An audible click will indicate correct locking. Without depressing the button, attempt to pull the T handle forward to ensure locked. Ensure that the groove from the thumb button is visible above the housing. The QRF is operated by depressing the thumb button, lifting the T handle and rotating it forward past the 'DON' position refer Figure 1-42.

### Strap-In

Enter the cockpit taking care not to stand on the seat firing handle. Sit back in the seat ensuring that the back is well supported and proceed as follows:

a. Adjust the height of the seat pan to the normal in-flight position using the visual sight provided.

b. Attach the leg restraint garters ensuring that the D-ring is at the rear and the quick release connector is on the inside of the legs. The restraint lines must not be crossed.

c. Attach the G-suit inflation connector (left side) and the PSP quick-release connector to the Secumar life vest.

d. Ensure that the go-forward control lever is in the rear (locked) position and bring down the shoulder harness.

e. Bring the harness quick-release fitting (QRF), mounted on the negative G strap, and the crotch straps up between the legs ensuring that they are not crossed or twisted.

f. Place the lap (side) straps over the thighs, ensuring that the left strap passes over the PSP lowering line attachment.

g. Route the right crotch strap through the D-ring on the lap strap, fold the strap over to the left, using a forward 90° twist, pass the right shoulder strap lug down through the end loop of the crotch strap and secure the swivel lug into the QRF. Carry out a similar operation for the left strap.

h. Lock the QRF and tighten the lap straps, ensuring the QRF stays central. Secure the excess strap.

i Fully tighten the shoulder straps with the QRF central. Pull the excess lift webs (top brown strap on shoulder) to the rear and tuck away behind the shoulders.

j. Move the go-forward lever forward and check for freedom to move forward. Lock the go-forward lever, sit back hard and re-tighten the shoulder straps.

k. Fit helmet, connect the oxygen fitting to the mask connector and plug in the mic-tel lead.

l. Fit mask and check oxygen flow and intercom system.

The seat firing handle pin is removed in the after-start checks.

### Unstrapping

The seat firing handle pin is fitted in the after-landing checks. When leaving the aircraft after landing proceed as follows:

a. Disconnect the oxygen mask connector and switch the oxygen system to OFF and 100% OXYGEN. Disconnect the mic-tel lead.

b. Operate the harness QRF, ensure the lugs are released and then return the QRF to the locked position.

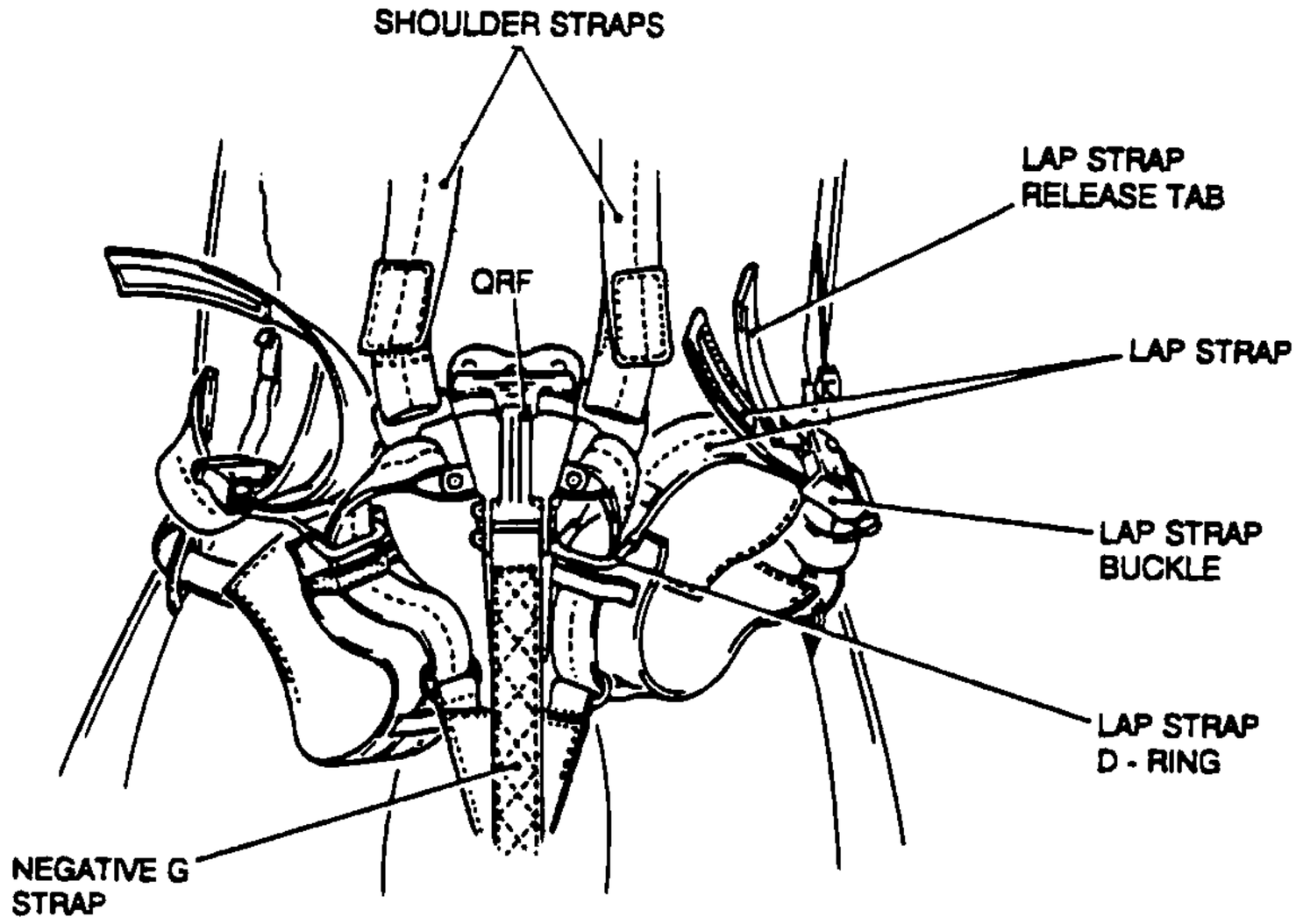
c. Release the PSP/Secumar quick release connector and the G-suit hose.

d. Disconnect the leg garters by the quick release connector, reconnect and stow over the control column.

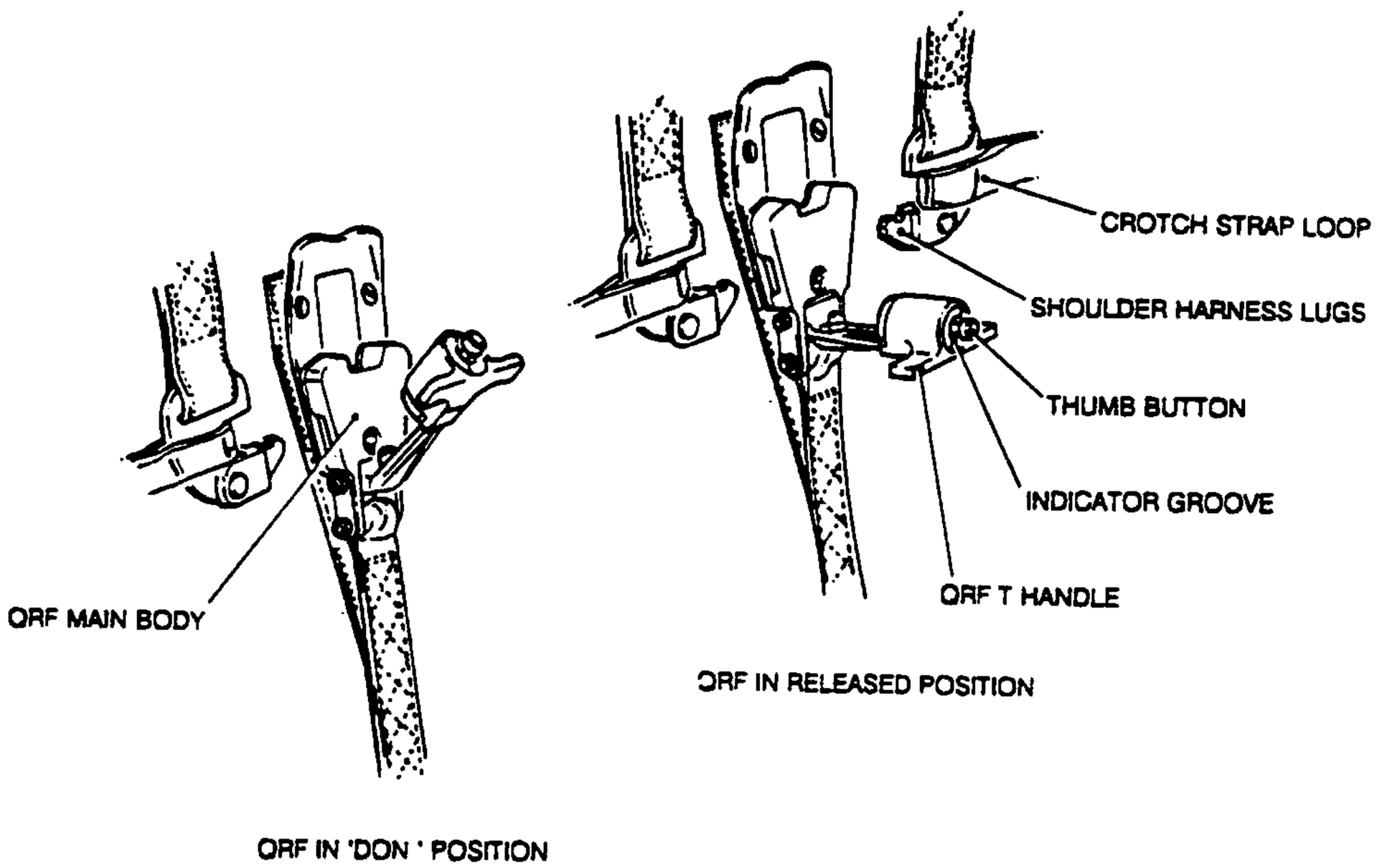
e. Lengthen all straps and stow shoulder harness at upper side stowage points.

f. Vacate the cockpit, taking care not to step on the seat firing handle.

# QUICK RELEASE FITTING




HARNESS STRAP ROUTING




QRF IN RELEASED POSITION

QRF IN 'DON ' POSITION

**Figure 1-42**

## AIRCRAFT ABANDONMENT

### General

The ejection facility is available from ground level at 60 KIAS in a level attitude and throughout the flight envelope. In-flight the optimum speed for ejection is in the range 200 to 250 KIAS. To avoid excessive loading on the parachute and the seat occupant, the aircraft speed should, wherever possible, be reduced to the optimum range before ejecting. In all ejections, before pulling the firing handle, warn the other occupant (if appropriate), lower both visors and sit erect. Grasp the seat firing handle with both hands when possible, stretch the legs out forward of the seat, keep the back as straight as possible and the head located hard back against the headrest. Close the eyes, pull the handle smartly upwards to its full extent withdrawing the sear from the seat firing unit to initiate ejection. Retain the grip on the handle until the harness release mechanism functions.

### Aircraft Abandonment In-Flight

At low altitude the optimum ejection conditions will be achieved when the aircraft is as close to level flight as possible. If the aircraft has an excessive bank angle, pitch attitude or rate of descent, additional terrain clearance is necessary. Chances of a successful escape in marginal ejection conditions are improved by initiating a zoom manoeuvre where the aircraft speed is converted to height. At higher altitude, aircraft attitude is less important, but, in controlled ejection conditions, speed and height should be adjusted before ejection. Ideally the aircraft should be positioned over an unpopulated area or over the sea.

### Ejection On The Ground

On the ground, an ejection option exists providing the occupants are properly strapped-in, the seat safety pins are removed, the canopy is closed and locked, the aircraft IAS is not less than 60 KIAS and the aircraft is essentially level. Ejection must not normally be attempted unless each of these conditions is satisfied.

### Drogue Gun Failure

If the drogue gun fails to operate above barostat height operating the manual separation handle will fire the cartridge in the manual separation breech which initiates the automatic release sequence and fires the drogue gun through the gas-operated firing unit. If drogue gun failure occurs below barostat height, gas pressure from the barostat time-release unit passes to the drogue gun to fire the gas-operated firing unit, therefore no action is required. Operation of the manual separation handle in this case is of no consequence.

## EMERGENCY EQUIPMENT

### Electronic Location Transmitter

An Electronic Locator Transmitter (ELT) is located in the battery bay with a remote operation switch on the left console of the front cockpit. The ELT is designed to automatically transmit on 121.5 MHz and 243.0 MHz when an internal G switch operates (as experienced by an aircraft crash). The battery operates independent of the aircraft electrical system and will enable transmission for 52 hrs. The ELT may be removed from the aircraft by a crash survivor and transported while it is operating. The pilot may manually select the ELT ON, by operating the guarded switch on the front cockpit left console. After removal the ELT may also be selected OFF or ON by the switch on the ELT unit.

### Personal Survival Pack

The Personal Survival Pack (PSP) forms part of the Ejection Seat and remains attached to the pilot after ejection. Operation of the single-handed release strap allows the pack to fall on the Secumar line and the dinghy inflates. The PSP contains the following:

a. LRU-16P single-man dinghy which is orange in colour and contains leak stoppers and a baler.

b. $CO_2$ gas bottle for dinghy inflation.

c. 250 ml of drinking water.

d. First aid kit with antiseptic ointment, two dressings and a safety pin.

e. Knife.

f. Compass.

g. Sea survival book.

h. Backpack (adapted from dinghy case).

### Canopy Breaker

Each cockpit is fitted with a canopy breaking knife which is located on the right canopy side frame. The knife may be used, on the ground, for exiting the aircraft where normal operation is not possible. To remove the knife, remove the locking pin on the end of the handle and withdraw the knife from the protective sheath. A two-handed backhand grip is recommended, striking the canopy at a point above the canopy frame on one side. Gloves should be worn to minimise risk of personal injury.

## CANOPY

### Description

The aircraft is fitted with a side opening, non-jettisonable, canopy/windscreen assembly comprising two acrylic perspex sections mounted in a stiffened frame. The front perspex transparency is an 8 mm thick acrylic sheet, the rear is 4 mm thick. The canopy will not provide protection from bird-strikes at high speeds. A rubber seal around the edges of the canopy provides weatherproofing. The canopy is shown at Fig 1-43.

The manually operated canopy is hinged on the right side and is counterbalanced by a spring strut to ensure ease of operation. The spring strut has a latch device to hold the canopy in the open position. The latch is released by a handle in either cockpit to allow the canopy to be lowered. When lowered, an internal lever in each cockpit is moved from rear to forward operating four canopy latches which pull the canopy down into the locked position, compressing the canopy seal and closing a microswitch to extinguish the CANOPY caption on the CWS annunciator panel. The CANOPY caption and the CAUTION PRESS TO RESET light illuminate when the canopy is unlocked. The canopy levers are held in the closed position by spring locking latches. The canopy also has an external opening and closing handle which is connected to the internal handles. To open the canopy the external lock release button (to the left of the external lever) must be depressed and the lever turned clockwise. The external lock release button provides a means of locking the canopy when leaving the aircraft unattended. Fig 1-44 shows internal and external canopy operating levers.

**CAUTION**

To avoid excessive loads on the canopy mounting frame, care must be exercised when operating the canopy in strong wind conditions. The canopy must not be left open in strong wind conditions.

## ENVIRONMENTAL CONTROL

### Description

The environmental control system (ECS) is designed to provide an acceptable cockpit environment under a wide range of temperature and humidity conditions. The ECS comprises an air conditioning unit, a distribution system and a control system. The air conditioning supply is provided by engine bleed air. The unit operates by cooling a portion of the bleed air and then mixing it with hot bleed air to provide the correct temperature selected by the crew. This enables the system to quickly respond to a change of temperature selection, as both hot and cold air are immediately available. Ram-air cooling is used in-flight and an electric fan is used on the ground (the nose landing gear down). A firewall shutoff valve (located in hydraulic services compartment) is incorporated in the inlet to the distribution system. If a fire occurs in the engine compartment the valve can be closed (with the firewall shutoff handle in the front cockpit) to isolate the conditioned air supply from the cockpit. The ECS is not available if:

a. the front generator busbar is not powered,

b. ram air lever is ON,

c. either STARTER switch is ON, or

d. the overheat switch is open circuited.

The cooling fan is only available if external power is connected or the generator is on-line.

### Air Conditioning Unit

The air-conditioning unit comprises a ground cooling fan, a two-level pressure regulator shutoff valve, a two stage heat exchanger, a cooling turbine air cycle unit, a temperature control valve and a water separator. The ram air intake is flush mounted on the left side of the engine compartment fixed section. The electric cooling fan is located in the lower left side of the engine compartment.

Pressure regulated air from the engine compressor P3 tapping is pre-cooled through a primary heat exchanger located in the nose of the aircraft, behind the engine. The P3 connection is restricted with a high pressure venturi to prevent unrestricted bleed air (and a possible flameout) if the air supply lines fail. The air is compressed by the compressor in the air cycle unit (compression/turbine) and re-cooled in the secondary heat exchanger. To provide additional cooling, water, which collects in the water separator, is sprayed into the heat exchanger second stage cooling air inlet. From there the air is directed over the air cycle unit turbine which in turn drives the air cycle unit compressor. Passing through the air cycle unit turbine the air temperature is reduced as the air expands. The resultant cold air passes into the mixing section of the air cycle unit to be mixed with hot P3 air, as required by the temperature control valve, and then to the cockpit. When cold air only is selected, the cold air from the air cycle unit turbine is directed to the cockpit. A low pressure flow limiting venturi prevents unrestricted hot air supply if the temperature control valve fails in the open position. An overheat switch is installed at the exchanger bleed air second stage inlet to close the pressure regulator shutoff valve if an overtemperature occurs. An ECS interrupt switch is incorporated in the ECS control box, which automatically switches the ECS to OFF for a period of

# CANOPY

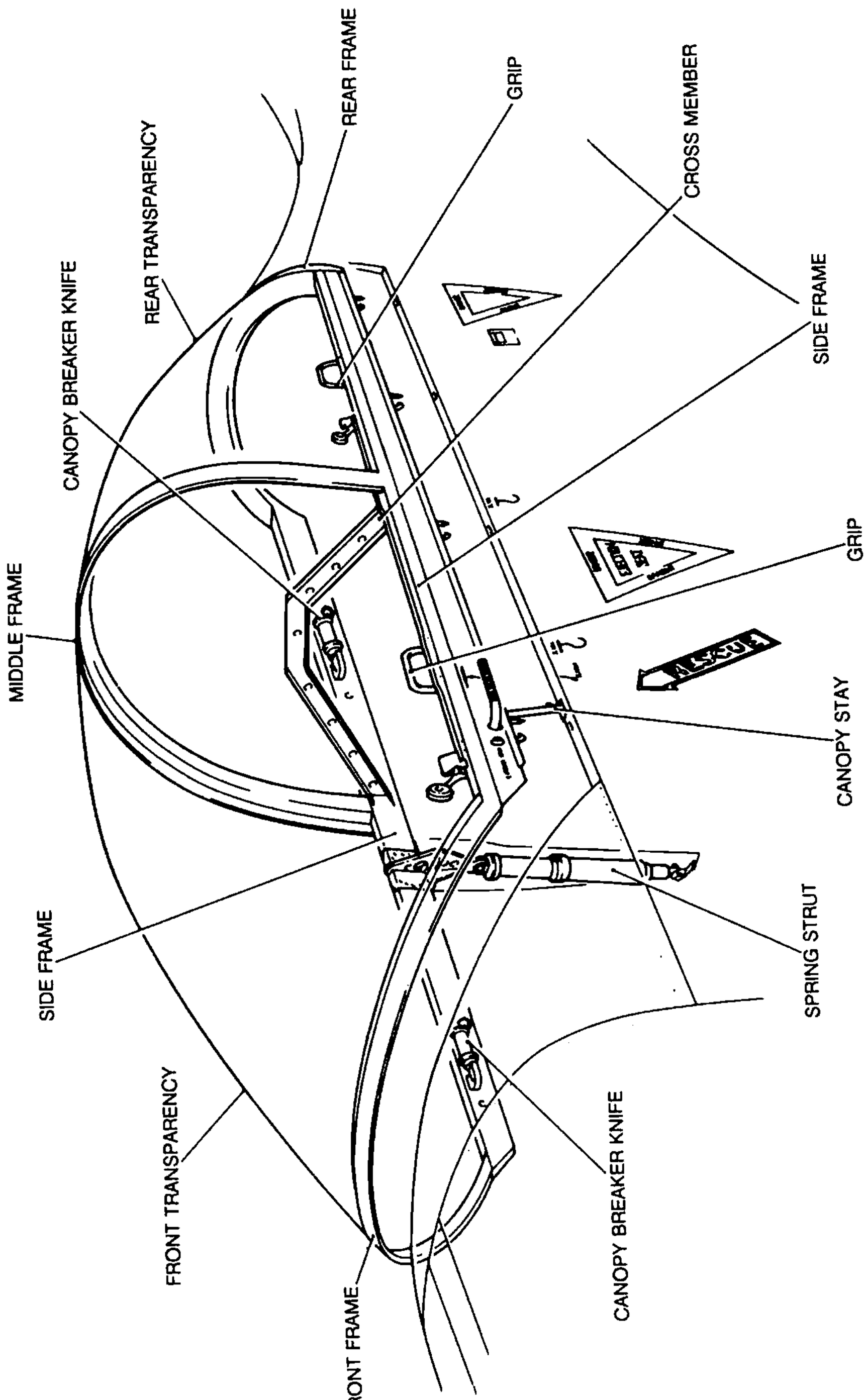
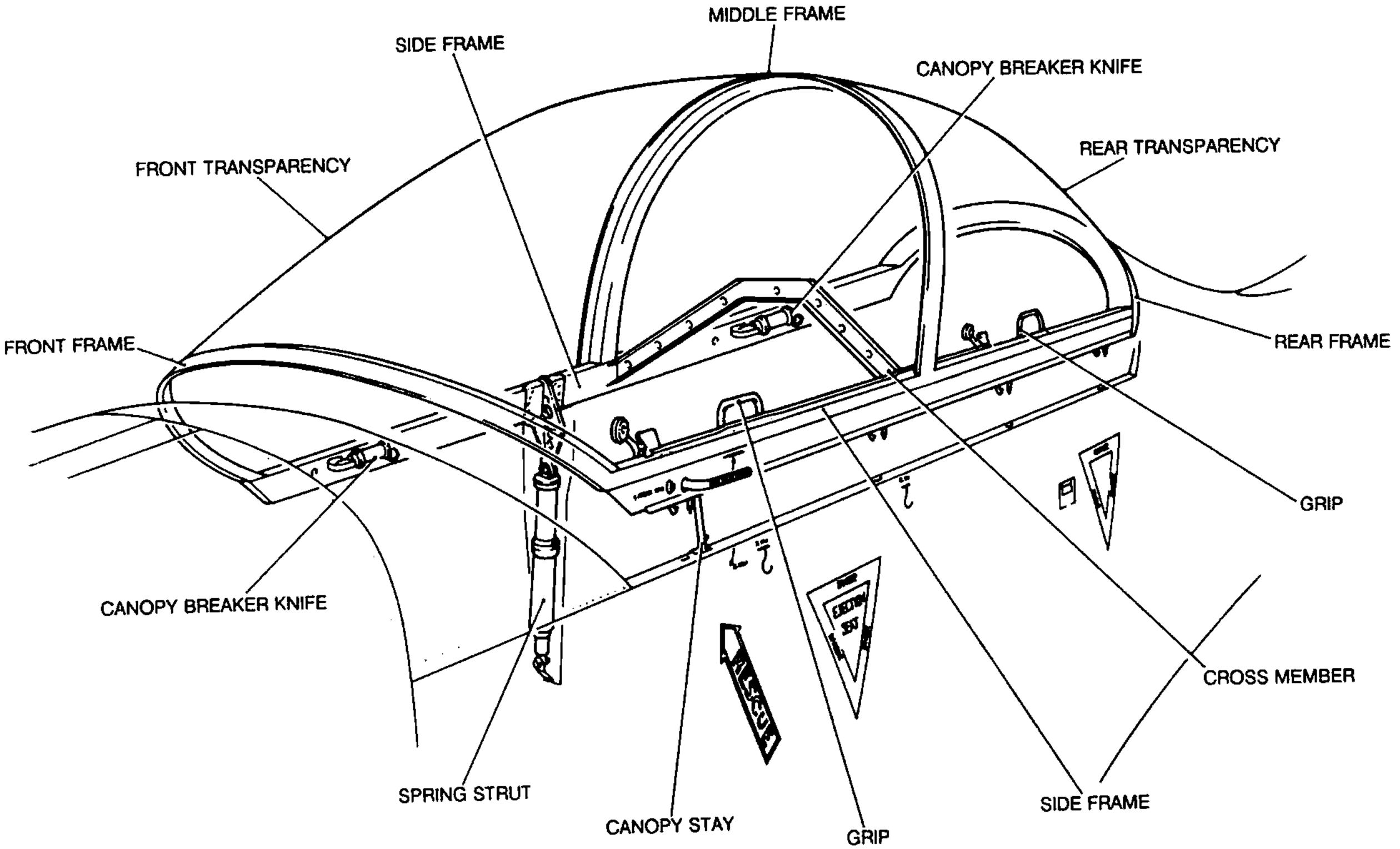



***Figure 1-43***


MIDDLE FRAME
SIDE FRAME
CANOPY BREAKER KNIFE
REAR TRANSPARENCY
FRONT TRANSPARENCY
FRONT FRAME
REAR FRAME
GRIP
CANOPY BREAKER KNIFE
SPRING STRUT
CANOPY STAY
GRIP
CROSS MEMBER
SIDE FRAME

# CANOPY OPERATING LEVERS

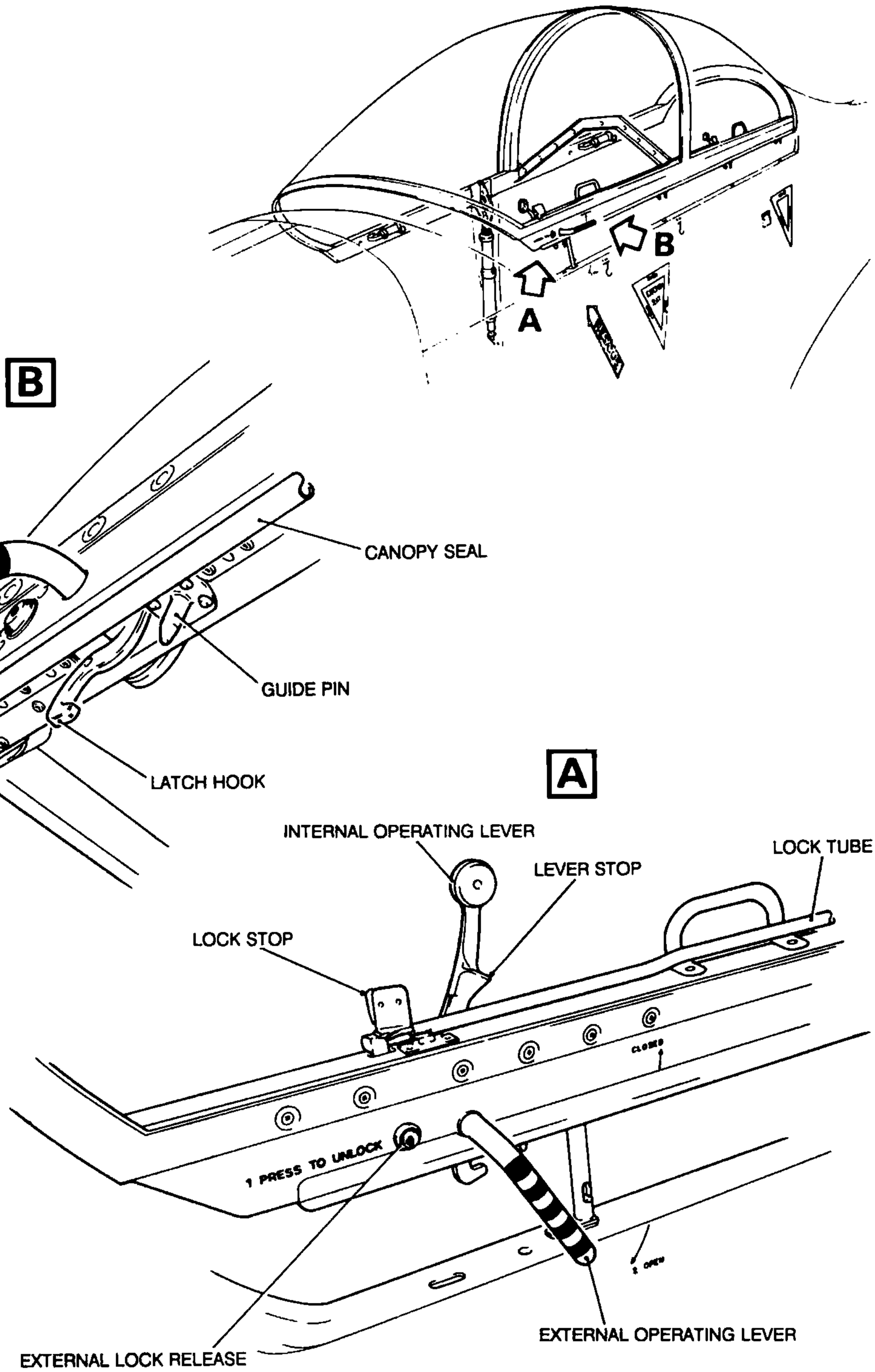



*Figure 1-44*

6 seconds (+1 second) after the PCL is selected to MAX power with the landing gear down and locked.

### Temperature Control

System delivery air temperature is determined by controlling the proportion of hot P3 air which is allowed to by-pass the cooling system. This is accomplished by the temperature control valve operating in response to pilot selection and electric signals from the temperature sensor located in the front cockpit ducting. The temperature selection signal is compared with a temperature sensor signal and if different, the temperature control valve adjusts the by-pass flow rate actuator and equalises selected temperatures.

### Air Distribution

Cockpit conditioning air is distributed around each pilot through louvres in the right side of the cockpit, centre console and outlets at foot level. The ECS distribution and ventilation system airflow is shown at Fig 1-47.

### ECS Control Panel

An ECS control panel, for ECS mode control and temperature selection, is located in the front cockpit on the right console. Control positions are as follows:

**a.** ECS Mode Switch OFF - P3 air supply shut-off.

**b.** LOW - P3 air supply regulated to approximately 25 psi.

**c.** HIGH - P3 air supply regulated to approximately 35 psi.

**d.** Temperature Selection - A rheostat type control with a range from COLD to HOT to give a continuous variation in selected cockpit temperatures.

### Air Distribution Panel

An air distribution panel is located in the lower centre console. Three vertical type slide controls are provided as follows:

**a.** Air Distribution Lever

▲ Full hot demist function, with high pressure. P3 air, irrespective of ECS control panel selection.

■ Air directed to face and torso outlets.

▼ Air directed to foot outlets only.

▲
■ Air directed to all three outlets.
▼

**b.** Flow Rate Control

Varies the outlet flow rate (volume of air).

**c.** Ram Air

OPEN - Automatically shuts off the ECS and opens a vent on the forward underside of the fuselage allowing ambient ram air to enter the distribution ducting.

CLOSED - Closes external vent.

## OXYGEN SYSTEM

### Description

The aircraft oxygen system provides each pilot with an automatically regulated oxygen supply and a manual selection for 100% oxygen, emergency and test. If failure of the aircraft system occurs, an independent emergency oxygen system, mounted on each ejection seat, can be selected by a seat mounted handle. During normal operation oxygen is fed from a storage cylinder to a regulator in each cockpit. Oxygen demand is set by the pilot breathing depth and rate and the regulator output is fed to the face mask. Replenishment of the oxygen system is done at a service panel, on the left rear side of the fuselage. Normal maximum pressure is 1,800 psi and the system is regarded as empty when the pressure falls below 50 psi. The OXYGEN light on the CWS illuminates when system pressure falls to 300 psi. The system must be placed unserviceable when the total pressure falls below 50 psi.

**WARNING**

If the oxygen pressure falls below 100 psi the aircraft must be descended below 10,000 ft AMSL.

The oxygen system is protected from excessive pressures by an overpressure relief system. Fig 1-48 shows the layout of oxygen system components.

### Oxygen Storage and Distribution

The single oxygen cylinder is a lightweight type with a shutoff valve and a cylinder pressure gauge. The cylinder, when full, contains oxygen at a charge pressure of 1,800 psi. The oxygen cylinder is mounted longitudinally in the lower rear fuselage left side, with provision for a second bottle.

# ECS POWER SUPPLIES

*Figure 1-45 (Deleted)*

# AIR CONDITIONING UNIT

*Figure 1-46*

**(Deleted)**

# ECS DISTRIBUTION AND VENTILATION SYSTEM AIRFLOW

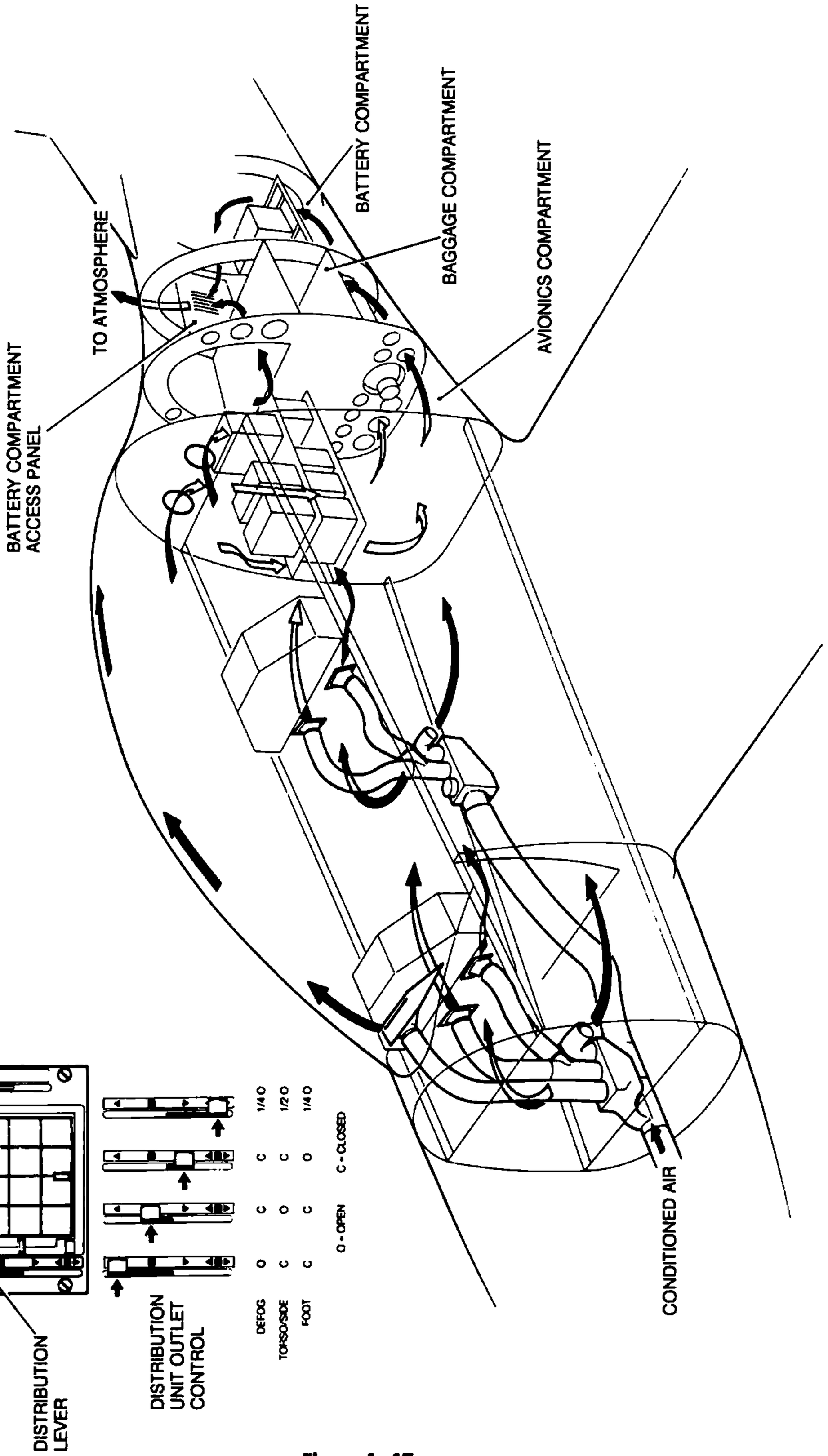



*Figure 1-47*

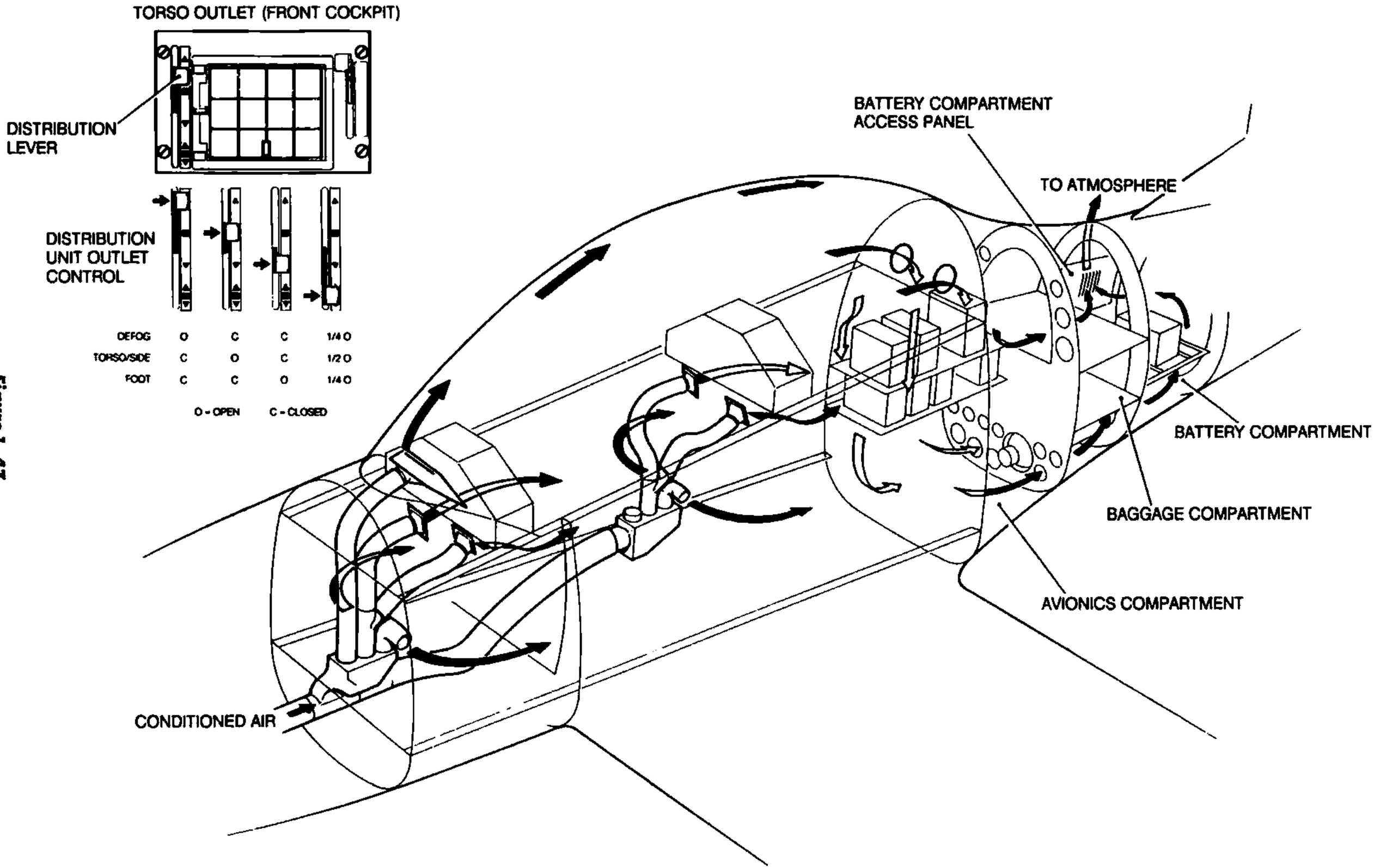

TORSO OUTLET (FRONT COCKPIT)
DISTRIBUTION LEVER
DISTRIBUTION UNIT OUTLET CONTROL
DEFOG O C C 1/4 O
TORSO/SIDE C O C 1/2 O
FOOT C C O 1/4 O
O - OPEN C - CLOSED
BATTERY COMPARTMENT ACCESS PANEL
TO ATMOSPHERE
BATTERY COMPARTMENT
BAGGAGE COMPARTMENT
AVIONICS COMPARTMENT
CONDITIONED AIR

Figure 1-47

## Service Panel

The service panel is located on the left fuselage aft of the wing trailing edge. The service panel is fitted with an oxygen replenishment valve and a system pressure gauge, which is graduated from 0 to 2,000 psi in increments of 500 psi with a red line at 1,800 psi and a FULL marking.

## Oxygen Regulators

An oxygen regulator is mounted in the right-hand console of each cockpit. Each regulator is of the diluter/demand type capable of automatically supplying the correct oxygen/air ratio for a given altitude. The regulator has manual selection for EMERGENCY, NORMAL or TEST MASK flow, 100% OXYGEN or NORMAL dilution, and an ON/OFF SUPPLY lever. During NORMAL operation the regulator senses pilot demand and automatically supplies oxygen, diluted with cockpit air, to the face mask. The oxygen content is in proportion to altitude up to 30,000 ft, where 98 to 100% oxygen is supplied. A 'blinker' display indicates flow during the breathing cycle.

## Supply ON/OFF

The green SUPPLY lever on the oxygen regulator panel permits the supply of oxygen from the regulator and must be ON for breathing. The SUPPLY lever is normally left in the OFF position when vacating the aircraft.

## Supply Diluter Lever

The white OXYGEN supply diluter lever on the regulator panel is placed in the NORMAL OXYGEN position for normal (air/oxygen mixed) use or in the 100% OXYGEN position when cockpit air is excluded and 100% oxygen is required.

## Emergency Lever

The red emergency lever on the oxygen regulator is not moved from the centre NORMAL position unless an oxygen pressure increase is necessary. The EMERGENCY position excludes cockpit air and provides 100% oxygen continuously, at a slight positive pressure, at all altitudes. It is used as a safeguard, when required, to prevent inward leakage of ambient air into the mask. The TEST mode provides a continuous supply at an increased positive pressure to test the system operation and fit of oxygen mask.

**WARNING**

As positive pressures may be required, the oxygen mask must be well fitting on the face. Unless special precautions are taken to ensure a leakproof fit, continued use of positive pressure may result in rapid depletion of the oxygen supply and possibly an extremely cold oxygen flow to the mask. Therefore, the mask emergency toggle should be clipped down when using positive pressure for other than a short period.

## Pressure Gauge (Cockpit)

An oxygen supply pressure gauge is located above the CWS in both cockpits. The gauge dial is graduated from 0-2500 psi in 100 psi increments with red radial lines at 90 psi and 2210 psi and a green arc between 265 psi and 1850 psi. The pressure transducer, supplying system pressure, and both gauges are powered from the front generator busbar via the OXYGEN IND C/B.

**Note**

As the aircraft ascends to high altitudes and low temperatures, the oxygen cylinders become chilled and oxygen gauge pressure indication is reduced, sometimes rather rapidly. With a 50° C decrease in cylinder temperature the gauge pressure can be expected to drop by over 20%. As the aircraft descends to warmer altitudes, the pressure rises again and the usage rate of oxygen may appear to be slower than normal. A rapid fall in pressure while in level flight, or while descending, is not due to falling temperatures and leakage must be suspected.

## Overpressure Relief System

The overpressure relief system comprises a safety assembly (with a rupture disc set to 2,500 to 2,755 psi) installed in the oxygen cylinder shutoff valve, a stainless steel discharge pipe and a discharge indicator The discharge indicator is mounted flush to the aircraft skin, adjacent to the service panel, and has a green nylon snap-in discharge disc. If overpressurisation occurs the safety relief assembly disc will rupture. The oxygen will pass through the discharge pipe to the discharge indicator and displace the indicator disc. Absence of the indicator disc provides a visual indication that discharge has occurred.

# OXYGEN SYSTEM

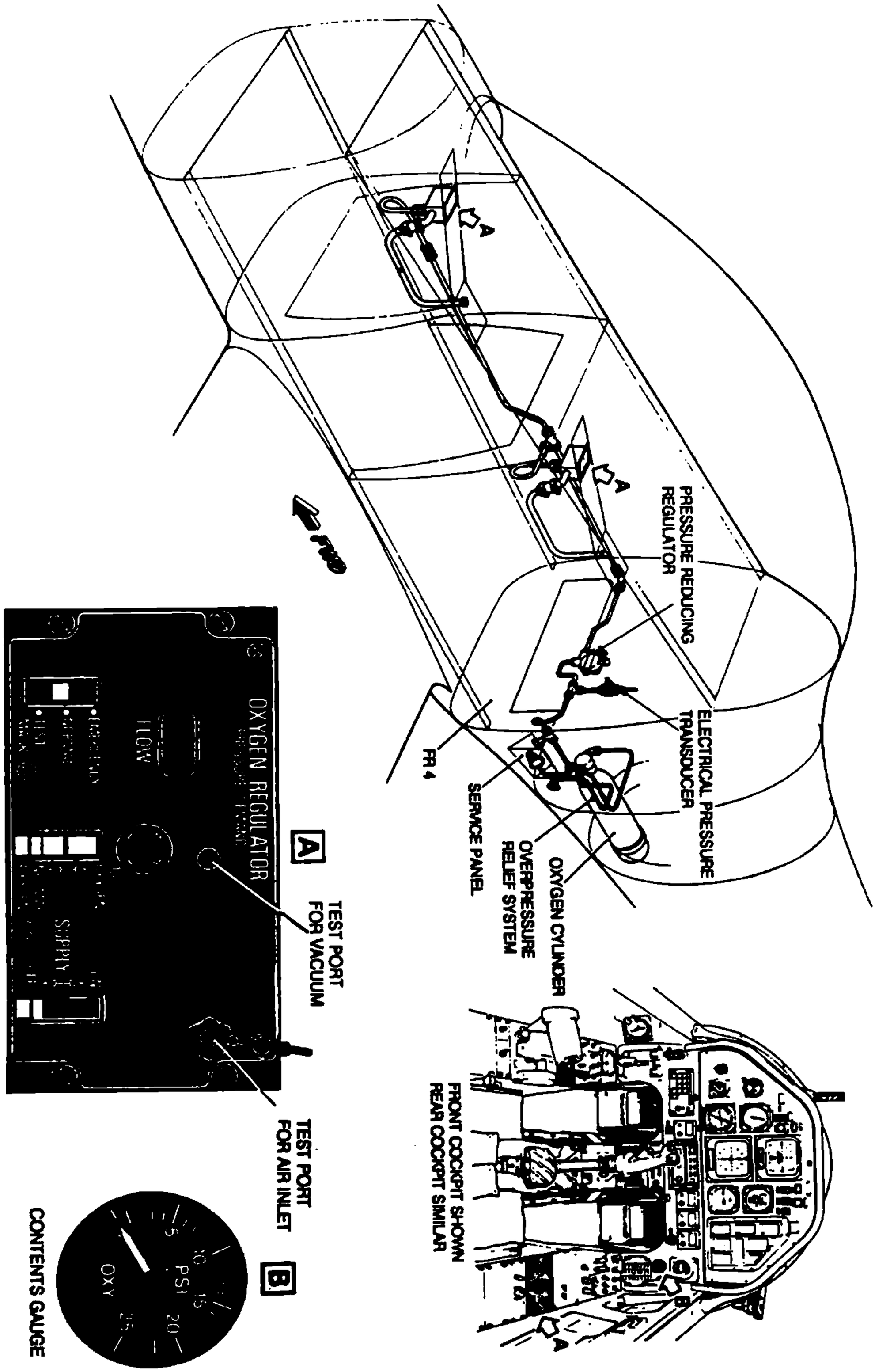
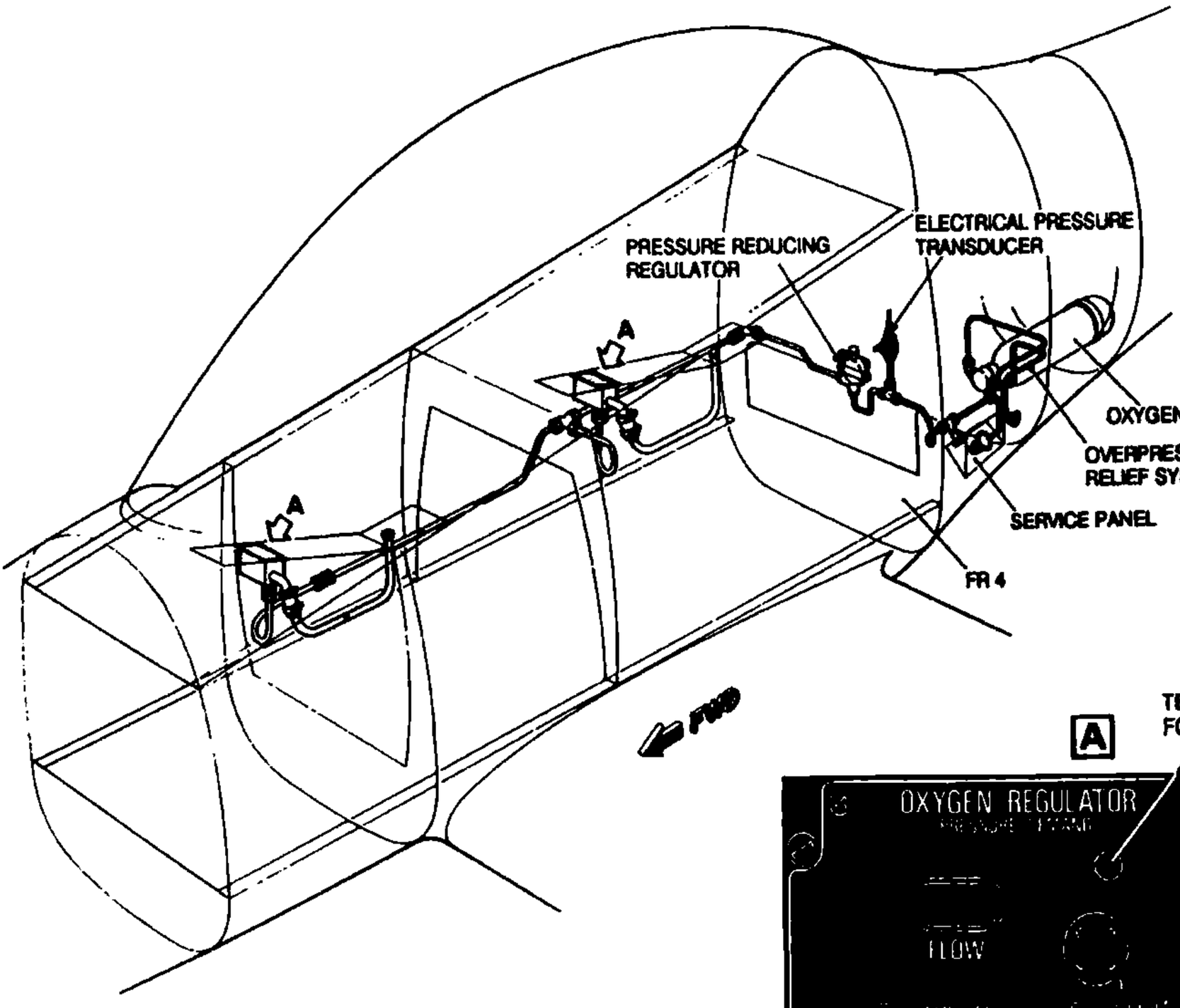
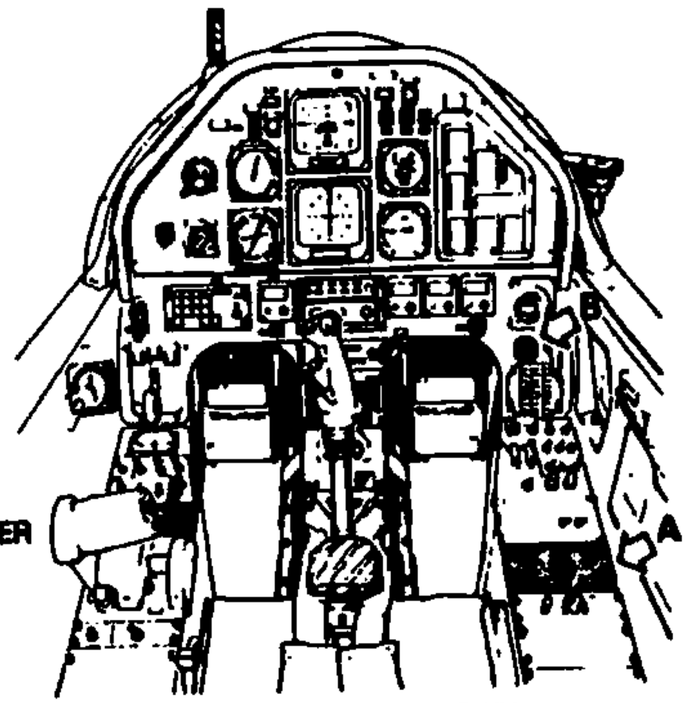
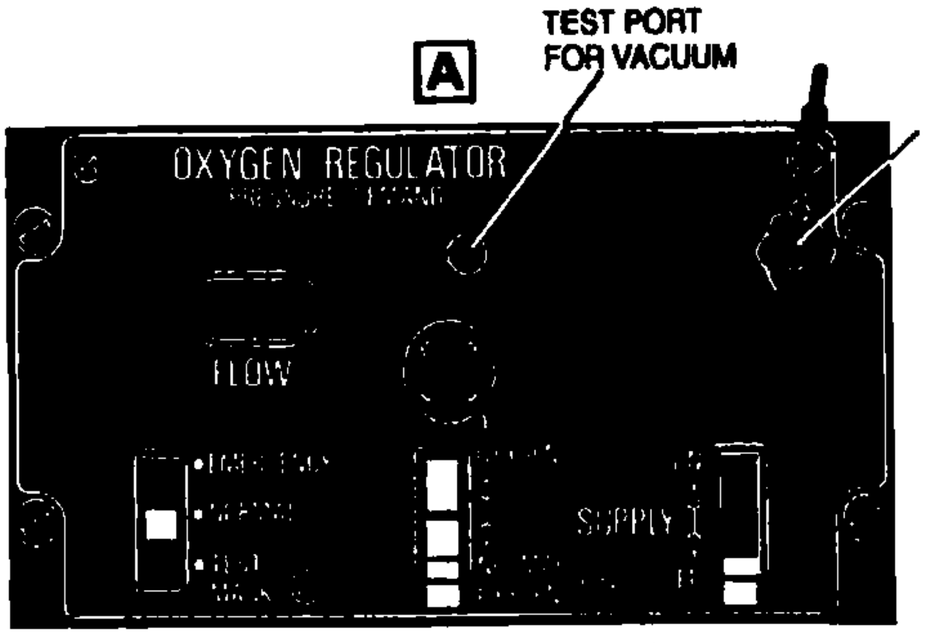



*Figure 1-48*




FRONT COCKPIT SHOWN
REAR COCKPIT SIMILAR




REGULATOR CONTROL PANEL




CONTENTS GAUGE

Figure 1-48

# ANTI-ICING SYSTEMS

## General

The aircraft anti-icing systems comprise:

a. an engine intake inertial separation system, and

b. a pitot static and AOA transmitter heating system.

There are no other protection systems for the engine, and therefore the aircraft cannot operate in icing conditions.

## Engine Intake Inertial Separation System

The inertial separation system is operated during icing conditions where the intake temperature is below +4° C with visible moisture in the air. The system is designed to prevent blockage of the engine compressor air inlet screen with ice and snow. The system comprises:

a. A ramp on the upper surface of the engine intake duct.

b. A bypass door on the left side of the fuselage (below the engine cowling), which is opened when the ramp is lowered.

c. An actuation system, comprising a control handle, actuation cables, levers and springs.

When the inertial separation system is operated, the ramp is lowered into the air intake passage and the bypass door is opened. In the lowered position the ramp reduces the area of that part of the passage causing a venturi effect. The resultant acceleration of the airstream through the air intake passage causes the heavier-than-air particles to bypass the engine air intake screen, and exit through the bypass door. Once the inertial separation ramp has been lowered it can only be reset when the aircraft is on the ground. The reset handle is stored on the door of the baggage compartment. Operation of the system causes illumination of the INTK ICE advisory caption on the CWS panel. The system is shown at Fig 1-49.

## Anti Ice System Heating

The Pitot Static system static vents (on each side of the rear fuselage) and the pitot head (under the left wing) and the AOA probe are equipped with heating elements for ice prevention. The heating elements are controlled by the ANTI-ICE PROBES switch in the front cockpit and a weight-off wheels microswitch on the left main landing gear leg. Activation of the heating system is indicated by the advisory ANT ICE caption on the CWS panel. The microswitch selects reduced heating when the aircraft is on the ground.

## Ice Warning Condition

An ice warning condition is activated when the outside air temperature falls to +4° C. If the PROBES switch is OFF the ESDP yellow caution light illuminates and the OAT gauge flashes. Selecting the PROBES switch ON cancels the caution and stops the OAT gauge flashing.

# COMMUNICATION EQUIPMENT

## General

Avionic system tandem control is achieved by using identical front and rear system control/display units interconnected by a 'cross-talk' databus. The utilisation of the 'cross-talk' databus, combined with digital displays in the control/ display units, means that frequency/channel/mode selection can be made from either control unit and that the selection made will be displayed simultaneously on both display units.

## Audio Integration System

The tandem control Pilatus/Becker Audio Integration System has been designed to provide maximum system protection against single component or power supply failures. Each audio control panel contains two pairs of microphone and headphone amplifiers, a main audio pair powered from the battery busbar and an emergency audio pair powered from the generator busbar. Two helmet connectors are installed on each seat harness, one connector to one pair of amplifiers. If one section of amplification or the power supply is lost the pilot can connect his helmet to the emergency audio connector (banded yellow), stored on the right side of the ejection seat as shown in Fig 1-50.

Both pilots must use the emergency connector to use the aircraft IC facility after amplifier failure or partial loss of power supply. In this mode with both cockpits having IC selected (and AVIONICS GEN switch ON) the ICS is on hot mic. Located on both the control column and the PCL in each cockpit is a Press-to-Transmit push-switch and an intercom/mute push-switch. If using the normal MIC/TEL lead, with IC selected in both cockpits, the intercom/mute push-swtich must be pressed to use the ICS. A voice activated hot-mic facility is incorporated and this is the normal operating mode. In addition to the communications and navigation systems receiver audio inputs, each audio control panel receives warning tone generator and electronic alarm inputs from the Central Warning System (CWS). A landing gear tone generator provides the means to transmit a 1 kHz tone on the selected communications transmitter. The tone can only be transmitted when the landing gear is down and locked and the DOWN TONE push-switch on the landing gear control unit is pressed. The audio integrating system switches are shown at Fig 1-51.

## Audio Control Panel

The audio control panel is located on the centre instrument panel in each cockpit, as shown in Fig 1-52.

Each audio control panel receives its main audio power from the battery busbar, via an AUDIO circuit breaker. Emergency audio power is supplied by the generator busbar, via an AUDIO circuit breaker, whenever the AVIONICS GEN switch in the front cockpit is ON. Integral lighting is provided from the cockpit instrument lighting system. Audio reception is available from the communications and navigation systems push-switches. Volume is controlled by both the outer knob of the volume control and the selected

# INERTIAL SEPARATION SYSTEM

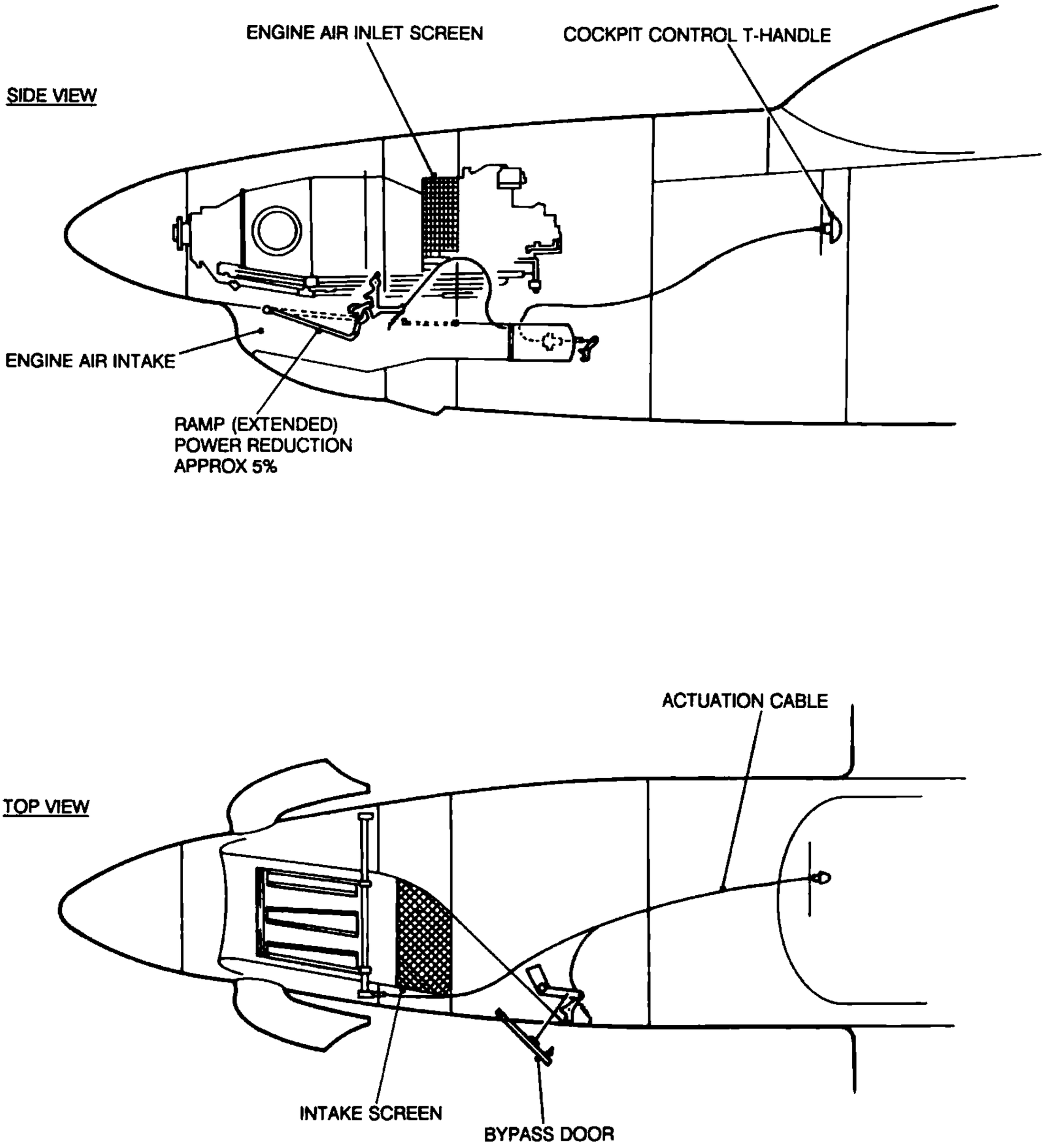



*Figure 1-49*

# HEADSET CONNECTORS

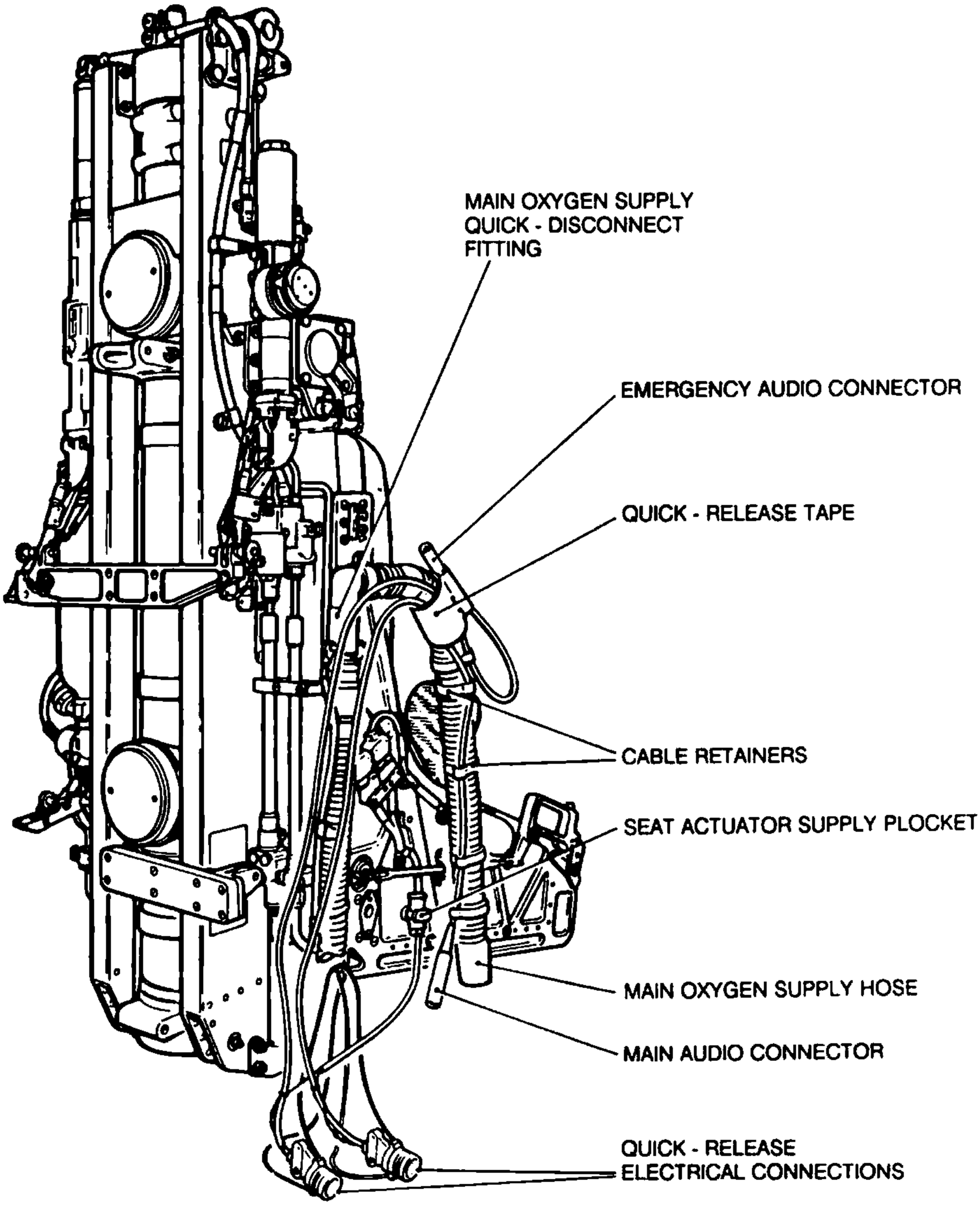
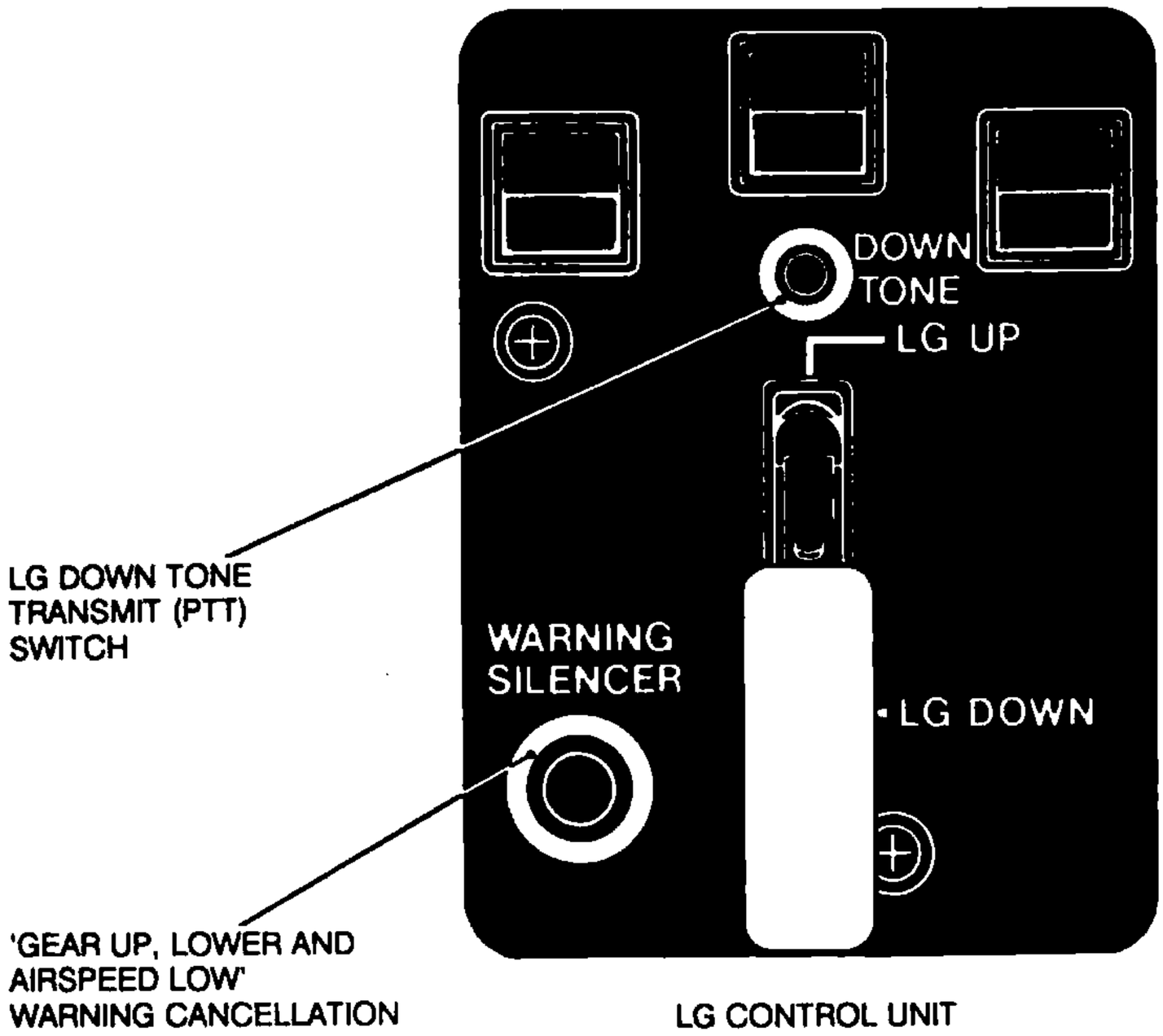
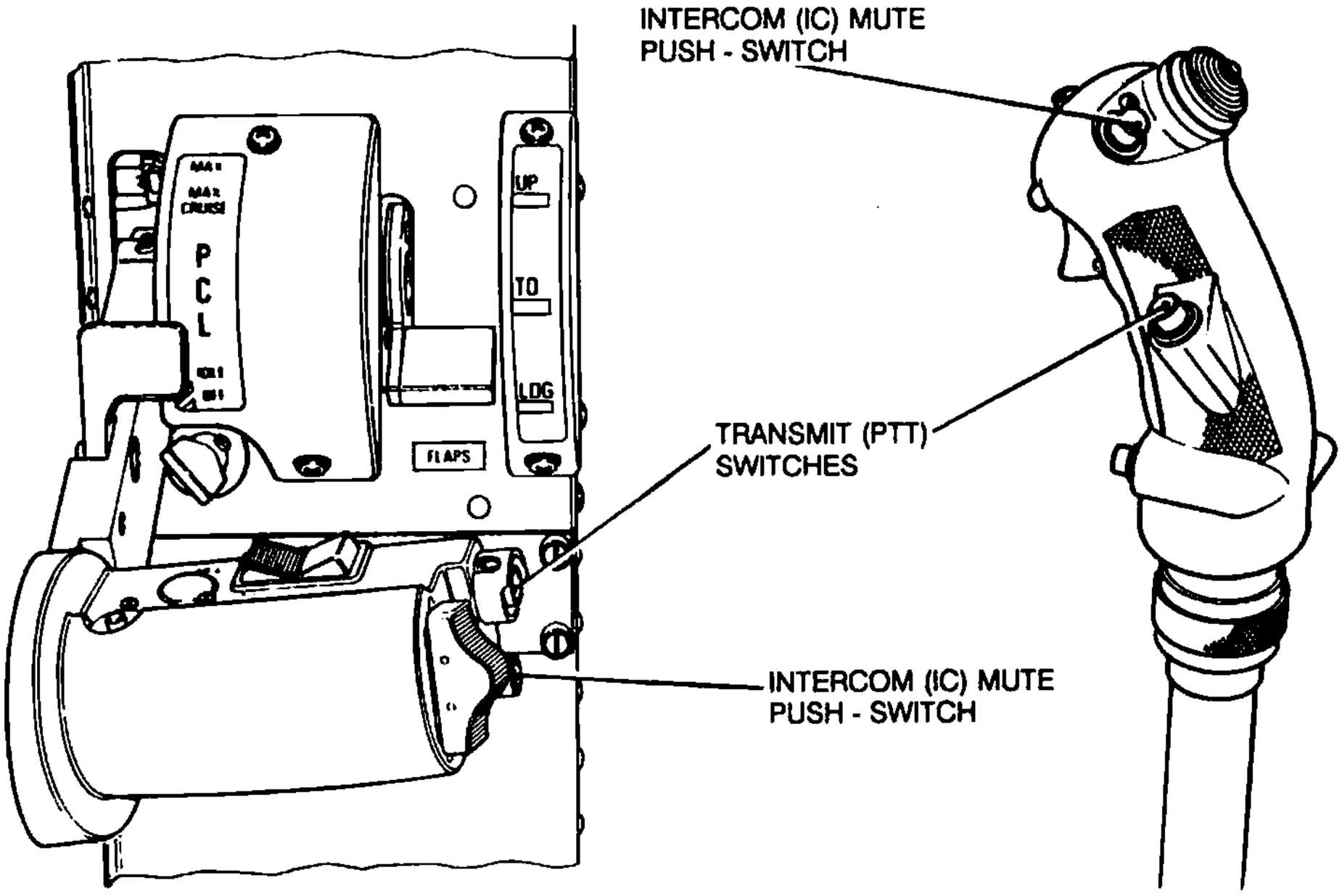



*Figure 1-50*

# TRANSMIT AND IC/MUTE SWITCHES




LG CONTROL UNIT




POWER CONTROL
LEVER (PCL)

CONTROL COLUMN

***Figure 1-51***

# AUDIO CONTROL PANEL LOCATION

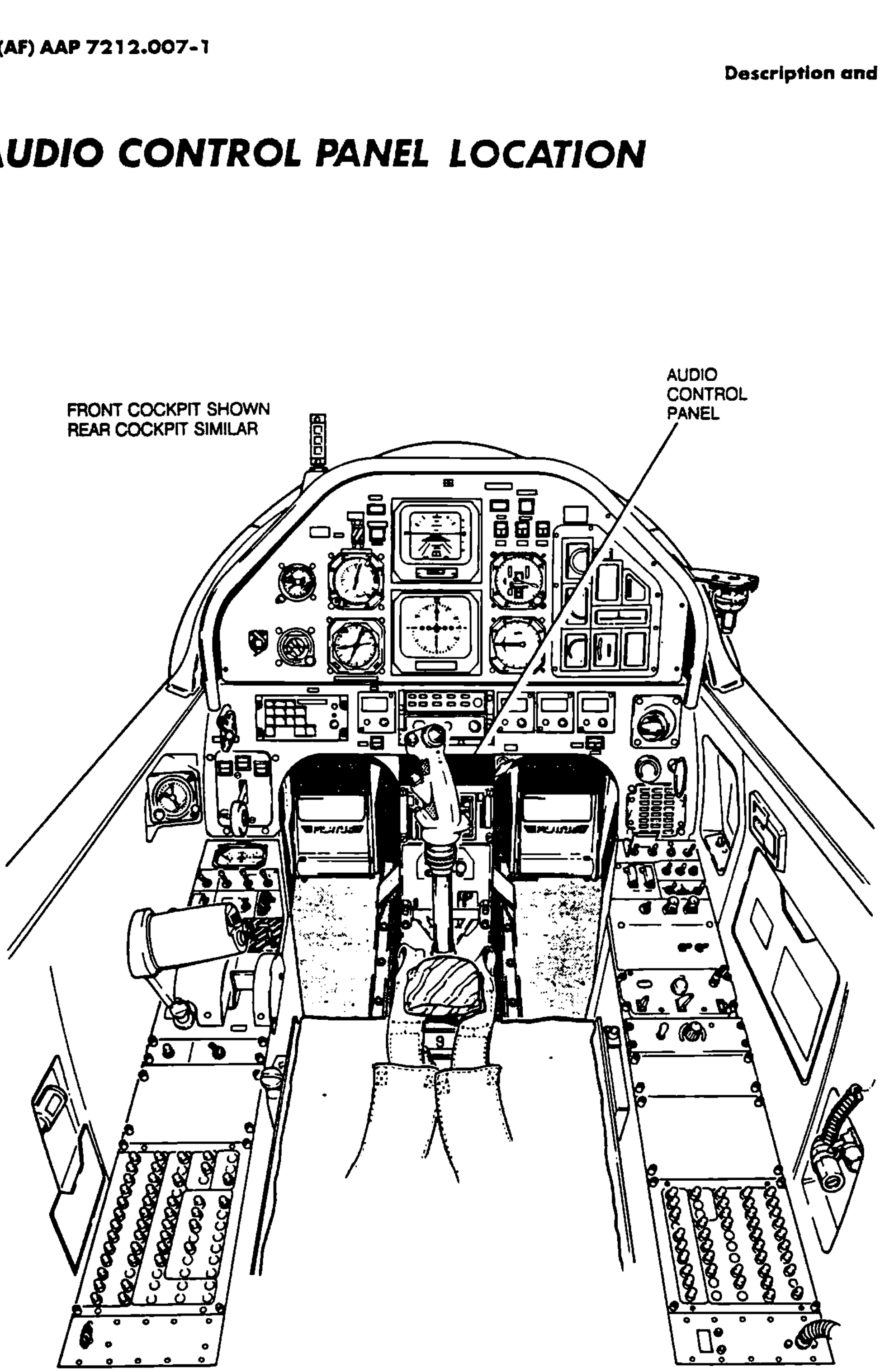



*Figure 1-52*

system's integral control unit in the selected cockpit. IC audio is controlled by the inner knob of the VOL control. The audio control panel is shown at Fig 1-53.

Receiver audio is muted during IC operation. The rotary selector selects the Bofors VHF/UHF transceiver at position 1 and the VHF transceiver at position 2. When selected to one of the first four positions:

a. Receiver audio is heard except when muted by either occupant.

b. Voice transmission is enabled.

c. The LG DOWN TONE transmission is enabled.

d. A selected system indicator light (top of panel) illuminates and side tone is heard when transmitting.

e A voice activated IC facility is enabled.

The VOICE push switch provides masking of the navigation system's identification signal to allow for clearer voice broadcasts. When selected to the IC position the hot-mic facility is disabled in the selecting cockpit. IC operation is available when the IC/mute switch is pressed. Receiver audio is muted during IC operation. After failure of the main audio amplifier, both pilots must select IC when connected to the emergency audio connector and transmission is available only when moving the rotary selector back to the selected radio. Audio reception is still available from the communications and navigation systems push-switches.

**UHF**

**Description**

The aircraft is fitted with a Pilatus/Bofors AMR 345 Communication System, designated COMM 1, providing selection of VHF frequencies and UHF frequencies from the same control panel. The combined display/control unit is located on the lower left section of each cockpit instrument panel, as shown at Fig 1-54.

The COMM 1 is an equal priority tandem control unit. The system contains a memory storage capacity for 500 preset channels. Frequency ranges are 104.0 to 161.975 MHz (VHF) and 223.0 to 407.975 MHz (UHF). The VHF band has a small area of reduced performance and the UHF band has two small areas of reduced performance. A logic unit provides a non-volatile and a volatile memory. The non-volatile memory holds the channel to frequency conversion table and the last channel and frequency entered by the operator. The volatile memory holds the current operating parameters of the transceiver. Automatic squelch is provided. The operating controls and display indications are:

a. ***Volume.*** Volume control for receiver audio is provided by rotation of the rotary selector.

b. ***CL Push-Switch.*** The current display is cancelled by pressing the CL (clear) push-switch.

# AUDIO CONTROL PANEL

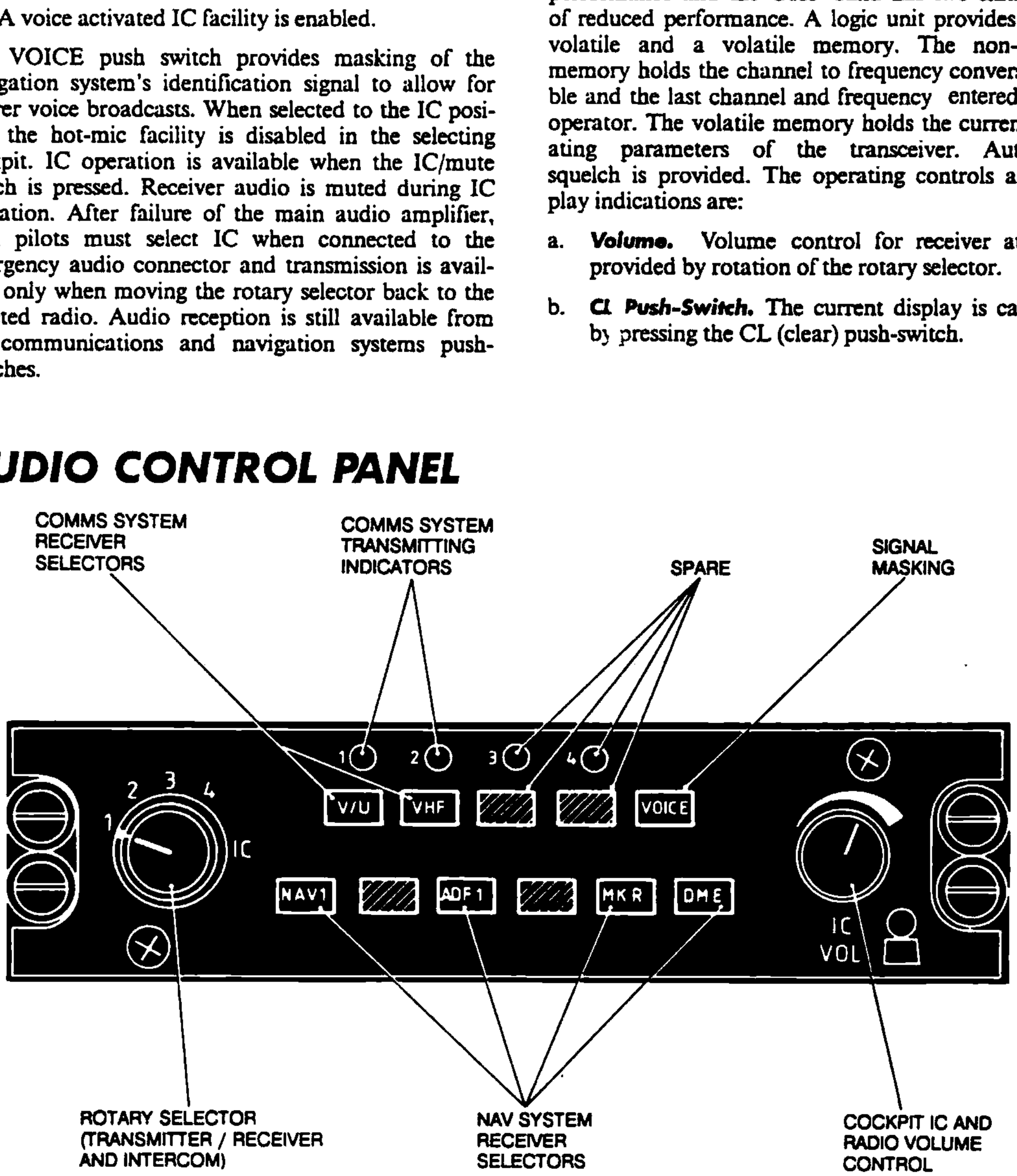



***Figure 1-53***

# COMM 1 LOCATION

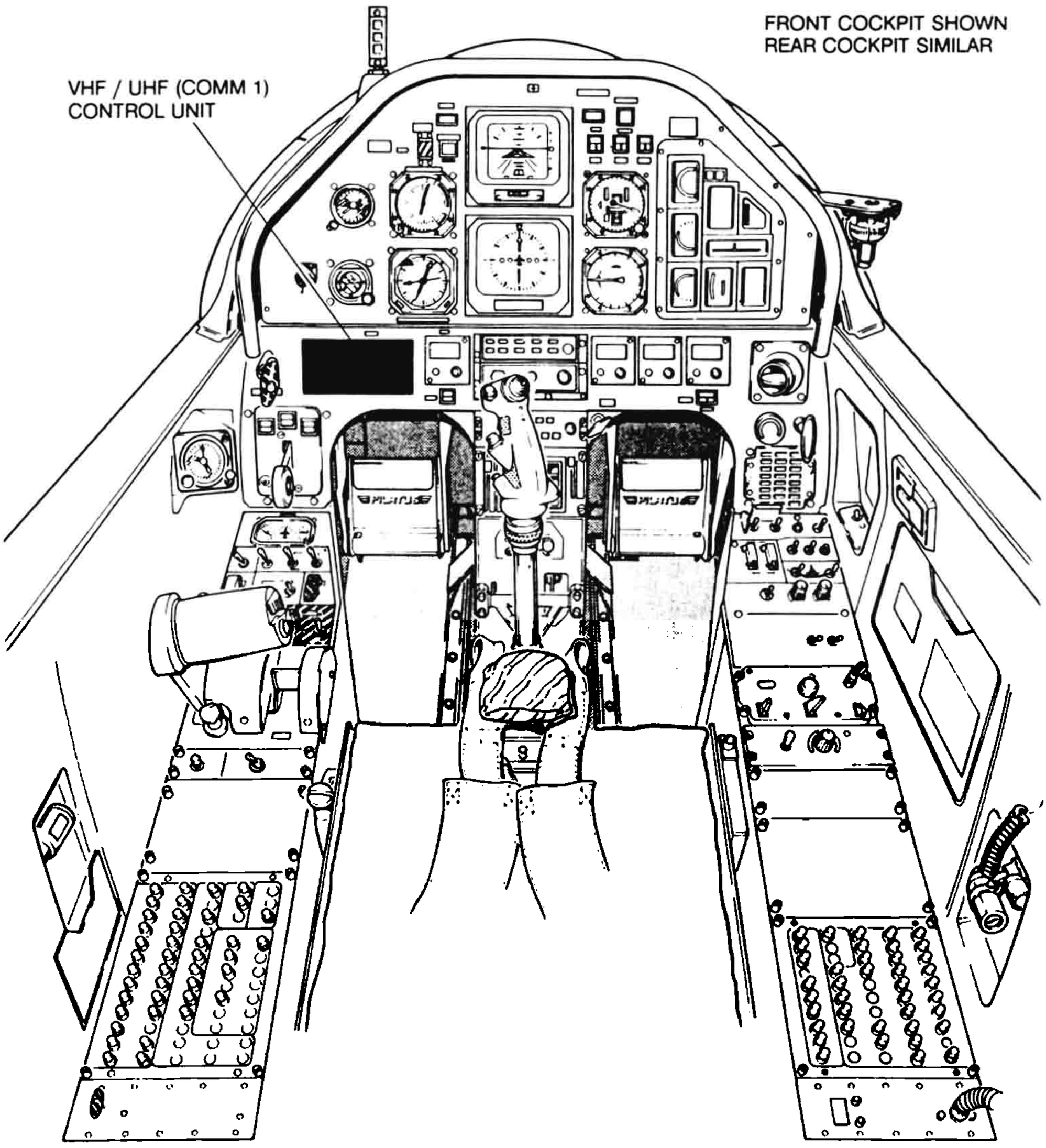



*Figure 1-54*

c. ***FM AM Push Switch.*** Amplitude modulation (AM) and frequency modulation (FM) is provided and is selected manually by means of an FM AM switch on the COMM 1 control units.

d. ***PRE (Preset Channel Mode) Key.*** Preset channel mode is enabled by pressing the PRE key. The preset channel is then selected on the keypad. Selection of a new preset channel is made by pushing the CL push-switch and entering the new channel on the keypad. Preset channel mode must be enabled before the system built-in-test can be initiated, or the VHF guard frequency (121.50 MHz) can be directly accessed.

e. ***MHz (Frequency Mode) Key.*** Frequency mode is enabled by pressing the MHz key. The last directly entered frequency will be displayed. To select a new frequency, the current display must first be cleared (CL switch) and the new frequency (5 digits) entered on the keypad. The transceiver tunes to the new frequency at completion of the fifth digit. Swapping between the new frequency and the previous frequency is possible by pressing the MHz key.

f. ***T/R Key.*** The T/R key enables the transceiver and transmits a carrier signal.

g. ***Keypad.*** Frequency or preset channel access is selected through a keyboard on either control unit. The preset guard frequency of 121.50 MHz is also available from the keypad.

h. **Display.** Invalid entries made in frequency or preset channel mode is indicated by the display flashing. The flashing display can be cancelled by pressing either the CL switch, the MHz or PRE keys. Direct access of the guard frequency is indicated on the display with only four digits (1215). After built-in-testing, 'FE1' is displayed if there is a function error.

The front control unit and the COMM 1 transceiver is protected by the V/UHF COM 1 C/B on the front battery radio busbar and the rear control unit is protected by the V/UHF COM 1 C/B on the rear battery radio busbar. Panel lighting is provided by the cockpit instrument lighting system. The COMM 1 VHF/UHF control unit is shown at Fig 1-55.

The COMM 1 antenna is located on the underside of the rear fuselage, rear of the avionics compartment access panel.

### Operation

COMM 1 receiver is enabled when the AVIONICS BAT switch is selected to ON and headphones are connected. The transmitter is enabled when the rotary switch, on the audio integration panel, is selected to position 1 and the press-to-transmit button on the control column or PCL is pressed in either cockpit. A sidetone output is supplied to the audio integrating system when the COMM 1 is transmitting. Operation of COMM 1 is as follows:

## COMM 1 CONTROL UNIT

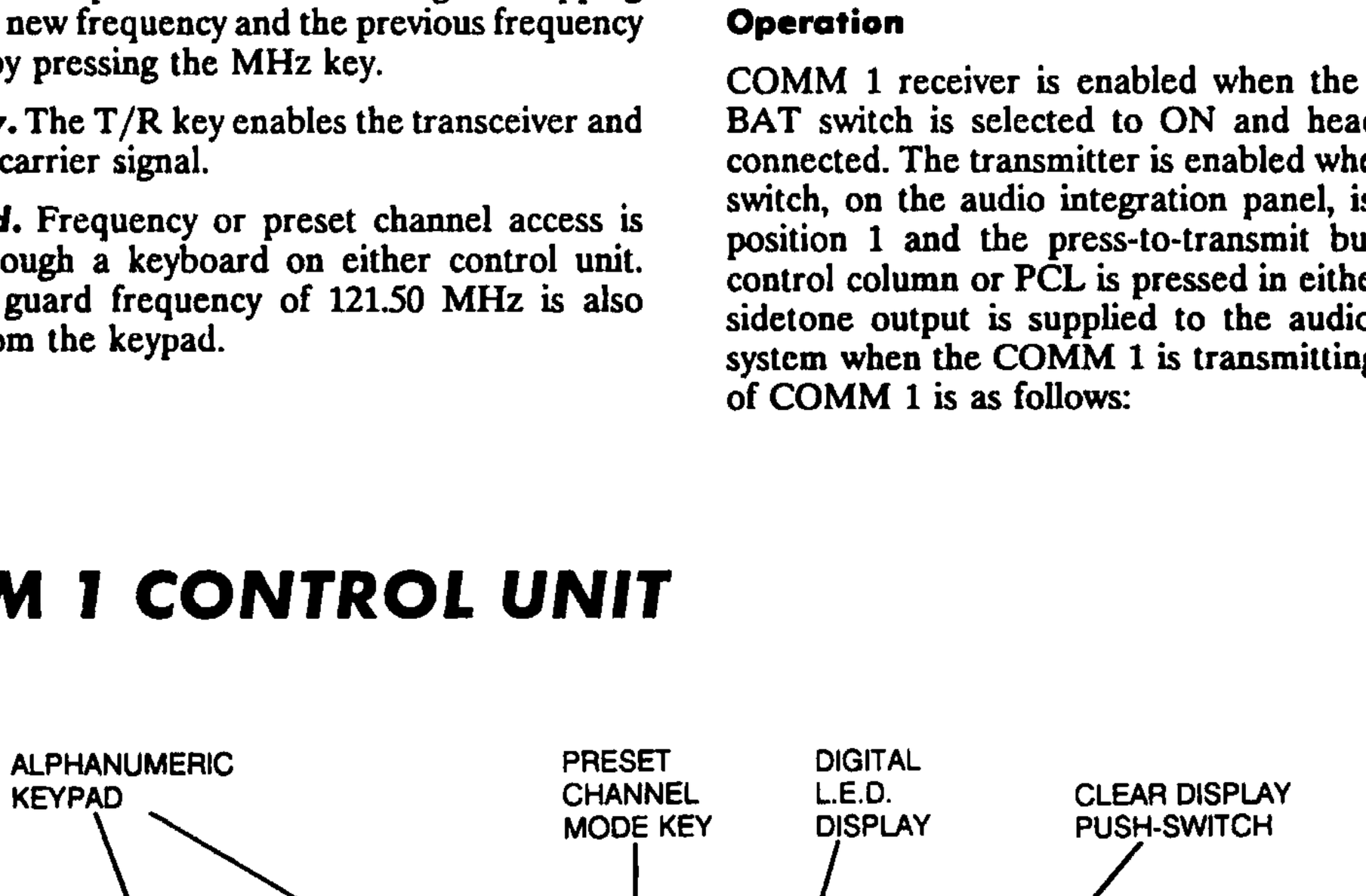



***Figure 1-55***

a. ***Directly Entered Frequency.***

(1) Press the MHz key (last directly entered frequency is displayed).

(2) Press the CL switch (display cleared but frequency still tuned).

(3) Enter the frequency on the keypad.

(4) Press the MHz key to swap between the new and old frequency.

b. ***Preset Channel Selection.***

(1) Press the PRE key (the last preset channel appears on the display).

(2) To select a new channel in the same group (5 groups each of 100 channels available), enter the two digits of the channel.

(3) To select a new channel in a different group, enter the new group number (1 to 5) and then enter the two digits of the channel.

(4) Press the PRE key to toggle between the new and old frequency.

c. ***Guard Channel Selection.***

(1) Press the PRE key.

(2) Press the CL switch.

(3) Press key 9 (digits 1215 are displayed).

(4) To cancel the guard channel selection, press PRE or MHz.

## VHF

### Description

The aircraft is fitted with a King KTR 908 VHF Communication System, designated the COMM 2 system. The COMM 2 is used in a tandem control configuration. The combined display/control unit is located on the lower left section of each cockpit instrument panel, as shown at Fig 1-56.

The COMM 2 system may be operated in a frequency mode or a preset channel mode, with 9 non-volatile channels available. When operated in the frequency mode the COMM 2 control units display 2 freqencies, active and standby. The upper display is the active frequency to which the transceiver is tuned. The lower display shows the selected standby frequency. The two frequencies can be inter-changed by operation of a transfer switch on the COMM 2 control units or a remote switch on the PCL. Automatic squelch control with manual disable is provided. The frequency range is 118.0 to 151.975 MHz and power is supplied from the front generator radio busbar via the VHF COM 2 C/B. Lighting is provided from the cockpit instrument lighting system. The controls perform the following functions:

a. ***OFF VOL Knob.*** COMM 2 is switched on by rotating the knob clockwise out of the OFF detent. Further rotation increases the audio volume level for the selected cockpit.

b. ***PUSH TST Knob.*** The automatic squelch control is disabled by pressing and releasing the OFF VOL knob. Re-pressing enables the squelch.

c. ***CHAN Push-switch.*** The CHAN switch switches the transceiver from FREQUENCY mode to PRESET CHANNEL mode and vice versa. Holding the switch down for greater than 2 seconds places the system in PROGRAM mode. The system reverts from program mode to frequency mode if no programming activity has occured for 20 seconds.

d. ***Tuning Knobs.*** The inner and outer tuning knobs operate under different modes as follows:

(1) **Frequency Mode - Standby Entry.** The outer knob increases or decreases the MHz portion of the displayed frequency in 1 MHz steps, with rollover at 151 MHz (increasing) and 118 MHz (decreasing). The inner knob increases or decreases the kHz portion of the displayed frequency in 50 kHz steps with the knob pushed in, or 25 kHz steps with the knob pulled out. Rollover occurs at 975 kHz (increasing) and 000 MHz (decreasing).

(2) **Frequency Mode - Active Entry.** Both knobs operate as in d.1, but the displayed active frequency will be tuned. The transceiver is tuned to the displayed active frequency.

(3) **Preset Channel Mode.** Rotation of the inner or outer tuning knob changes the displayed channel number and its corresponding preset channel frequency. The display will only change to those channel numbers which are programmed with preset frequencies. The transceiver is tuned to the preset channel frequency (shown in the SBY display window).

(4) **Program Mode.** Rotation of the tuning knobs changes the channel number or the preset channel frequency, whichever is flashing at the time. The flashing selection is made by the transfer push-switch.

e. ***Transfer Push-switch.*** The transfer push-switch is used in the following operations:

(1) **Frequency Mode - Standby Entry.** Pressing and releasing the transfer switch interchanges the displayed active and standby frequencies. The transceiver is tuned to the new active frequency.

(2) **Frequency Mode and Preset Channel Mode - Active Entry.** Holding the transfer switch depressed for longer than two seconds causes the system to enter FREQUENCY MODE - ACTIVE ENTRY. The transfer switch is operated momentarily to return the system to its original

# COMM 2 LOCATION

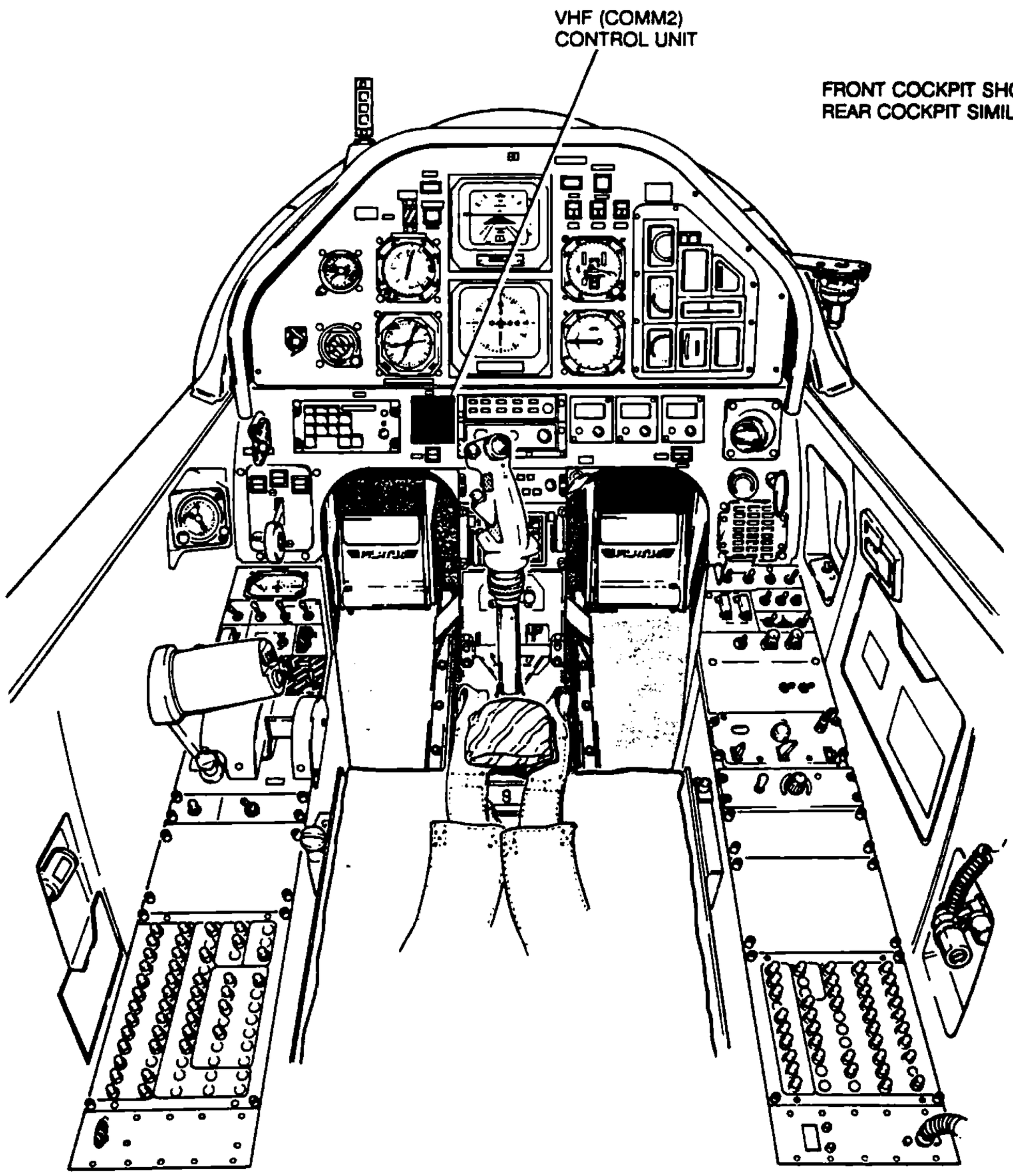



*Figure 1-56*

mode. This does not apply to the PCL transfer switch.

(3) **Program Mode.** Pressing and releasing the transfer switch causes a flashing channel display to change to a flashing frequency display, or vice versa.

(4) **Default Mode.** Turning the unit on while holding the transfer switch down causes the unit to come on in active entry mode with 120.0 MHz active frequency displayed.

The remote transfer switch operates in the same way as the control unit transfer switch, with the exception that pressing the remote transfer switch for longer than two seconds does not place the system in the active entry mode. The COMM 2 control unit is shown at Fig 1-57.

## Operation

The transmitter is enabled when the rotary selector on the audio integration panel is selected to COMM 2, the OFF VOL knob on the control panel is rotated clockwise out of the detent, and a press-to-transmit (PTT) is operated . A `TX' message is annunciated on the COMM 2 control unit displays when the PTT is operated. A sidetone output is supplied to the audio integrating system when the COMM 2 transmitter is operating. Automatic squelch is enabled at turn-on. Selection of the standby frequency is made with a rotary channel selector with concentric knobs. The pilot then transfers the standby frequency to the main by pressing either transfer push-switch. Preset frequencies can be called up for use by pressing the CHAN switch and selecting the channel (the frequency will also be displayed). If the PTT is operated for more than 90 seconds the TX message flashes and the transmitter is automatically disabled. Releasing and operating the PTT switch again will restore operation. Preset frequencies are programmed as follows:

a. Enter program mode by holding the CHAN switch for longer than two seconds.

b. With the channel number flashing, rotate either tuning knob until the required channel number is displayed.

c. Press the transfer switch. The selected channel number will remain displayed and the frequency will flash. Select desired frequency.

d. To continue channel programming, press the transfer switch and repeat b. and c.

e. Leave programming mode by pressing and releasing the CHAN switch. Automatic return will occur after 20 seconds of program mode inactivity.

# COMM 2 CONTROL UNIT

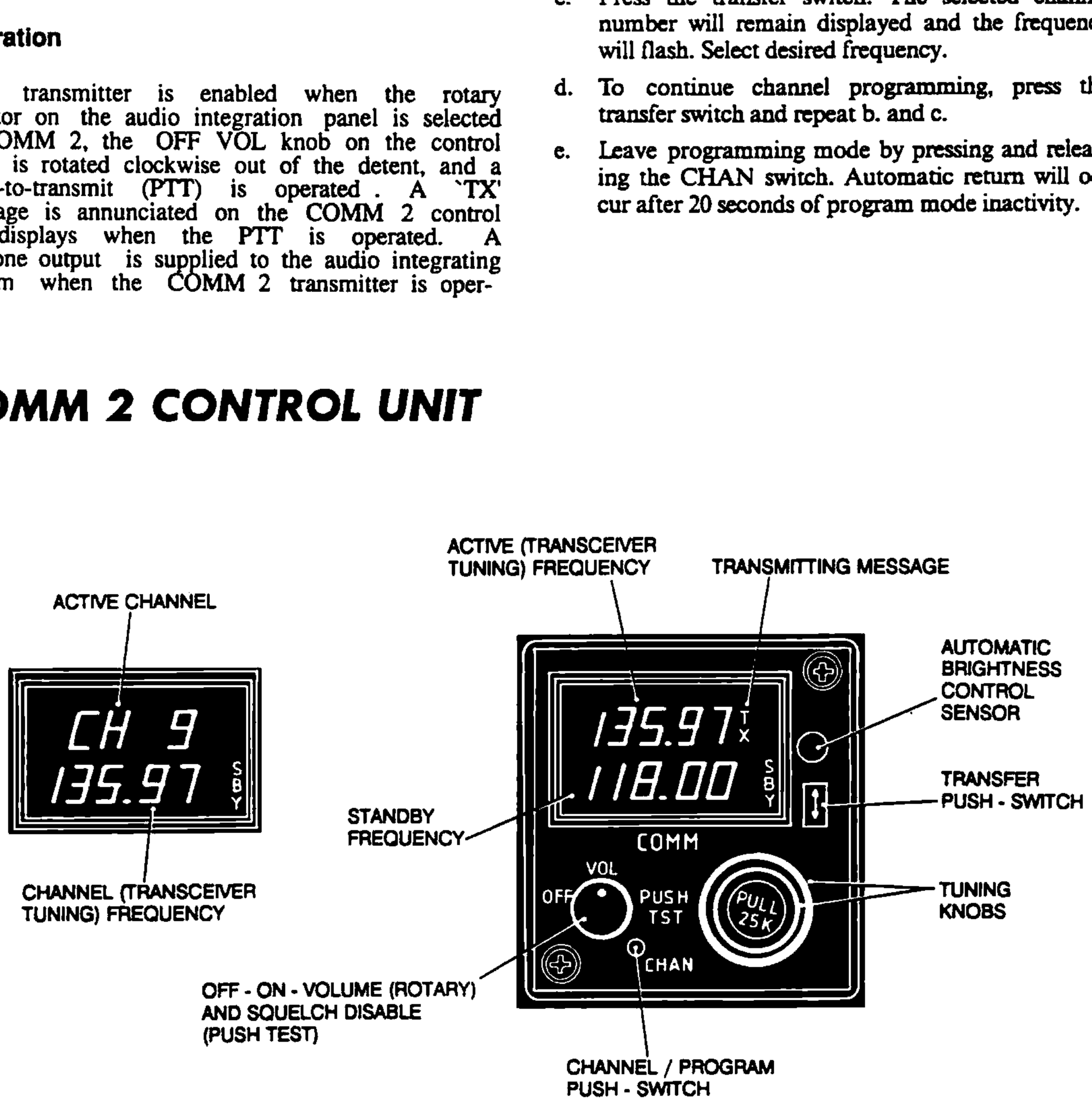



*Figure 1-57*

# NAVIGATION EQUIPMENT

## NAVIGATION/TACAN SYSTEM

### Description

The Navigation/Tacan (NAV/TAC) system comprises composite management for the VHF Navigation System and the Tactical Air Navigation System (TACAN). The NAV/TAC system is used in a tandem control configuration. The combined display/control unit is located on the lower right section of each cockpit instrument panel. The control unit has two frequencies, or two channels, or a frequency and a channel displayed. The upper display is the active frequency or channel to which the transceiver is tuned. The lower display shows the selected standby frequency or channel. The two frequencies can be interchanged by operation of a transfer switch on the NAV/ TAC control units. The standby frequency may be swapped from the NAV mode to the TAC mode by depressing a small red MODE button on the control panel. The control panel performs the following:

a. ***OFF VOL Knob.*** NAV/TAC is switched on by rotating the knob clockwise out of the OFF detent. Further rotation increases the audio volume level for the selected cockpit.

b. ***PUSH TST Knob.*** The PUSH TST knob activates a built-in-test function and will give the following display on the EHSI if pressed:

(1) DME reads 0.1 NM,

(2) G/S reads 1 KT, and

(3) Course bar reads 180° ± 2°.

c. ***MODE Push-switch.*** The MODE switch switches the unit from TACAN channel entry to NAVigation frequency entry.

d. ***Tuning Knobs.*** The inner and outer tuning knobs operate under different modes as follows:

(1) **TACAN Channel - Standby Entry.** Both knobs are used to change the TACAN channel, with rollover occurring at 129 (increasing) and 0 (decreasing). Changing TACAN channels from X to Y is accomplished by pulling out and rotating the inner knob.

(2) **TACAN Channel - Active Entry.** In the active entry mode the outer and inner knobs act as in d.(1), but the displayed active frequency will be tuned.

(3) **Navigation Frequency - Standby Entry.** Rotation of the inner and outer tuning knobs change the displayed standby frequency.

(4) **Navigation Frequency - Active Entry.** Rotation of the inner and outer tuning knobs change the displayed active frequency.

e. ***Transfer Push-switch.*** The transfer push-switch is used in the following operations:

(1) **Frequency or Channel Mode - Standby Entry.** Pressing and releasing the transfer switch interchanges the displayed active and standby frequencies/channels. The unit is tuned to the new active frequency/ channel.

(2) **Frequency or Channel Mode - Active Entry.** Holding the transfer switch depressed for longer than two seconds causes the system to enter frequency or channel mode - active entry. The standby window will blank-out and the active frequency/channel will be directly controlled. The transfer switch is operated momentarily to return the system to its original mode.

The NAV/TAC control panel is shown at Fig 1-58.

## VHF NAVIGATION SYSTEM

### Description

The King KNR 634 VHF Navigation system provides the facility to monitor 200 VOR navigation or localiser (LOC) stations and 40 glideslope (G/S) channels. The frequency range is 108.00 to 117.95 MHz with automatic selection of glideslope with a valid LOC frequency. Fast 'within range' checking of next frequency is possible with the transfer push-switch. System information is displayed on the EHSI's and RMI's. Marker beacon lights are installed on each cockpit instrument panel, incorporating a press-to-test capability. Accuracy is approximately 0.5° for VOR and 2.5° for RMI.

### Operation

Audio is available with the NAV button on the audio integration panel. Selections on either cockpit NAV/ TAC control unit automatically changes the display selections in the other cockpit.

## NAV/TAC CONTROL PANEL




***Figure 1-58***

## TACTICAL AIR NAVIGATION SYSTEM

### Description

The King KTU 709 TACAN unit provides slant-range distance, groundspeed and bearing information to the EHSI. Bearing information is also available to the RMI. The bearing signal can be used to drive the Course Deviation Indicator (CDI) on the EHSI. Frequency selection is possible in the X and Y bands on the NAV/TAC control unit.

### Operation

Audio is available with the DME button on the audio integration panel. Selections on either cockpit NAV/TAC control unit automatically changes the display selections in the other cockpit. A feature of the system is the ability to display the DME output to the EHSI while having an active NAV frequency. This is done by first tuning the DME station and depressing the DME button at the top of the instrument panel to HOLD the frequency. The NAV frequency may then be transferred to active and the distance will still be available (evidenced by a white readout on the EHSI and the letter H displayed at the end).

## ADF SYSTEM

### Description

The King KDF 806 ADF System is a solid state system with a combined control/display unit located on the lower section of each cockpit instrument panel. The upper display is the active frequency to which the receiver is tuned. The lower display shows the selected standby frequency. The two frequencies can be interchanged by operation of a transfer switch on the ADF control units. The ADF has channel and frequency modes of operation. Frequency range is from 190 to 1799 kHz in 1 kHz increments. Bearing accuracy is $\pm 3°$. The audio is selected by depressing the ADF button on the Audio Integration Panel. Automatic dimming circuits adjust the display brightness according to changes in ambient light conditions. The control panel performs the following:

a. ***OFF VOL Knob.*** The ADF is switched on by rotating the knob clockwise out of the OFF detent. Further rotation increases the audio volume level for the selected cockpit.

b. ***PUSH MODE Knob.*** The PUSH MODE knob cycles the ADF through the following modes:

(1) ANT (sense antenna only with bearing pointer parked at 90° position),

(2) ADF (sense and loop),

(3) ANT/BFO (beat frequency oscillator mode with audio for a carrier wave station), and

(4) ADF/BFO.

c. ***CHAN Push-switch.*** The CHAN switch switches the unit from frequency mode to channel mode.

d. ***Tuning Knobs.*** The inner and outer tuning knobs operate under different modes as follows:

(1) **Frequency Mode.** The tuning knobs change the ADF frequency of the standby display. The X character to the left of the active frequency informs the operator that the bearing indicator directional information is not valid.

(2) **Channel Mode.** Momentarily pressing the CHAN button puts the unit in the channel mode. The upper display contains a CH and a number 1 to 9. The lower frequency displays a frequency. The channel number is changed with the tuning knobs, provided there is a valid frequency tuned into any other channel. Momentarily depressing the transfer switch returns the unit to the frequency mode and places the channel frequency in active/CHAN display with the previous active frequency in the standby window.

(3) **Program Mode.** Pressing the CHAN button for 2 seconds places the unit in the program mode. The unit displays a P and a channel number in the active/CHAN display, with a frequency or dashes in the standby display. The channel number will flash and may be changed with the tuning knobs. Momentarily pressing the transfer switch causes the frequency to flash, which may then be changed with the tuning knobs. Momentarily pressing the CHAN button, or no button activity for 20 seconds, returns the unit to frequency mode.

e. ***Transfer Push-switch.*** The transfer push-switch is used in the following operations:

(1) **Frequency or Channel Mode - Standby Entry.** Pressing and releasing the transfer switch interchanges the displayed active and standby frequencies/channels. The unit is tuned to the new active frequency/channel.

(2) **Frequency Mode - Active Entry.** Holding the transfer switch depressed for longer than two seconds causes the system to enter frequency mode - active entry. The standby window will blank-out and the active frequency will be directly controlled. The transfer switch is operated momentarily to return the system to its original mode.

The ADF control unit is shown at Fig 1-59.

## TRANSPONDER (IFF)

### Description

The King KXP 756 Transponder is of a reliable, solid state design with minimal warm up time. The control unit is located on the lower right of the front cockpit instrument panel. The unit transmits on 1090 MHz and receives on 1030 MHz with 4096 codes possible. The control unit performs the following functions:

# ADF CONTROL UNIT




***Figure 1-59***

a. ***OFF Knob.*** The transponder is switched on by rotating the knob clockwise out of the OFF detent.

b. ***Function Switch.*** The function switch is the outer knob of the rotary selector and is continuously rotatable in both directions. The function switch selects the mode of operation to SBY (standby), ON (mode 3), ALT (mode 3C) and TST (test). The selected code is displayed in all modes except TST where it is replaced by the altitude input from the system's encoding altimeter and the letters FL (Flight Level).

c. ***Code Selector Switch.*** The inner knob of the rotary selector is used to change the transponder reply code. Momentarily pressing the inner knob changes the position of a cursor indicating which code digit is selected for changing. If the code selector switch is held in for 3 seconds the transponder code will change to the emergency code of 7700 and the cursor shifts to the left position.

d. ***Ident Switch.*** The ident switch enables transmission of SPI (special position indicator) pulse for approximately 23 seconds. The IDT nomenclature will appear on the display.

e. ***Mode Switch.*** A push-button mode switch is located under the control panel enabling selection of mode A or B.

The transponder control unit and system schematic is shown at Fig 1-60.

## ATTITUDE HEADING REFERENCE SYSTEM

The Attitude and Heading Reference System (AHRS) comprises:

a. Gyro electronic unit (GEU).

b. Compass adjust unit.

c. Compass calibration memory unit.

d. Flux detector.

e. Compensator.

The GEU is located in the avionics compartment and contains two strapdown gyroscopes sensors, two level detector (bubble) sensors and a digital computer. The strapdown gyroscopes provide a highly accurate short term reference for aircraft movement in roll, pitch and yaw. Aircraft motion about these axes is converted into digital signals for processing by the computer and then passed to the user systems. Attitude and heading sensors are used as references to detect long term inaccuracies in the system and the error signals are used by the digital computer to correct the system outputs. The long term references are provided by:

a. the bubble sensors detecting off vertical (attitude) conditions, and

b. the external flux detector, providing magnetic north heading reference.

The GEU computer is programmed to compensate for sensor errors caused by aircraft turning and acceleration, earth rate and Coriolis. An AHRS FAST ERECT, located on the front cockpit right wall, switch can be used to increase the rate of correction,

# TRANSPONDER SYSTEM



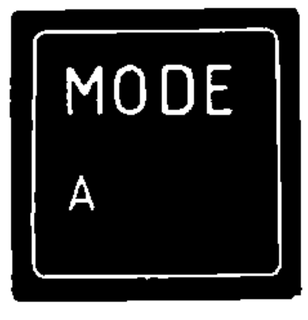




***Figure 1-60***

when required. Pressing the AHRS RESET switch for a few seconds causes the AHRS to initiate a new startup sequence, which takes about 30 secs. To avoid synchronisation to a false datum the aircraft must remain straight, level and unaccelerated when the system is going through its automatic startup sequence or when the FAST ERECT is used. The GEU also computes and automatically corrects for compass deviation, using data from the last compass swing. The GEU is provided with 28 VDC from the front generator and battery radio busbars via AHRS C/Bs, and 26 VAC 400 Hz from the front and rear 26 VAC busbars via AHRS C/Bs. DC power is connected to the GEU via the AHRS RESET switch. The GEU provides validity status control of the RMI's HDG flag and the EFIS HDG and ATTITUDE FAIL flags. The compass adjust unit, located in the avionics rack, provides the GEU with:

a. latitude information (required for computation of error compensations, eg, earth rate, coriolis), and

b. compass swing heading information.

The compass calibration memory unit is located on the GEU and is used to store compass calibration data. The flux detector and compensator is installed in the right wing tip and is used by the GEU as a long term heading reference. The flux detector uses a pendulous sensing element to detect the direction of the earth's magnetic field. The RMI and EFIS HDG/ ATTITUDE FAIL flag circuits are controlled by the AHRS valid circuits within the GEU. The flags will remain in view for approximately 30 secs when electrical power is connected, until the GEU heading valid circuits are completed. Accuracy of the GEU's roll, pitch, yaw and heading signals is dependant upon the quality of the external references used by the GEU and the integral GEU references. Certain flight conditions might produce small, noticeable errors in the system due to the correction rate being too slow. These errors can be rectified by selecting the AHRS FAST ERECT switch to ON for a few seconds when flying straight, level and unaccelerated, until the system is synchronised. Straight and level is determined (as detected by the GEU) as:

a. roll less than 6° AOB, and

b. pitch less than 15°.

## STANDBY COMPASS

The type E2C standby magnetic compass is located on the right side of the front cockpit instrument coaming and provides compass heading information. Aircraft heading is read directly from the compass card against a vertical lubber line engraved on the inside of the compass bowl. The compass is a self-contained unit with integral compass error compensating magnets for coefficients B and C. Accuracy is ±10. The cockpit instrument lighting system provides internal lighting when the INSTR LIGHTS switch is ON. The standby compass is shown at Fig 1-61.

# *STANDBY COMPASS*

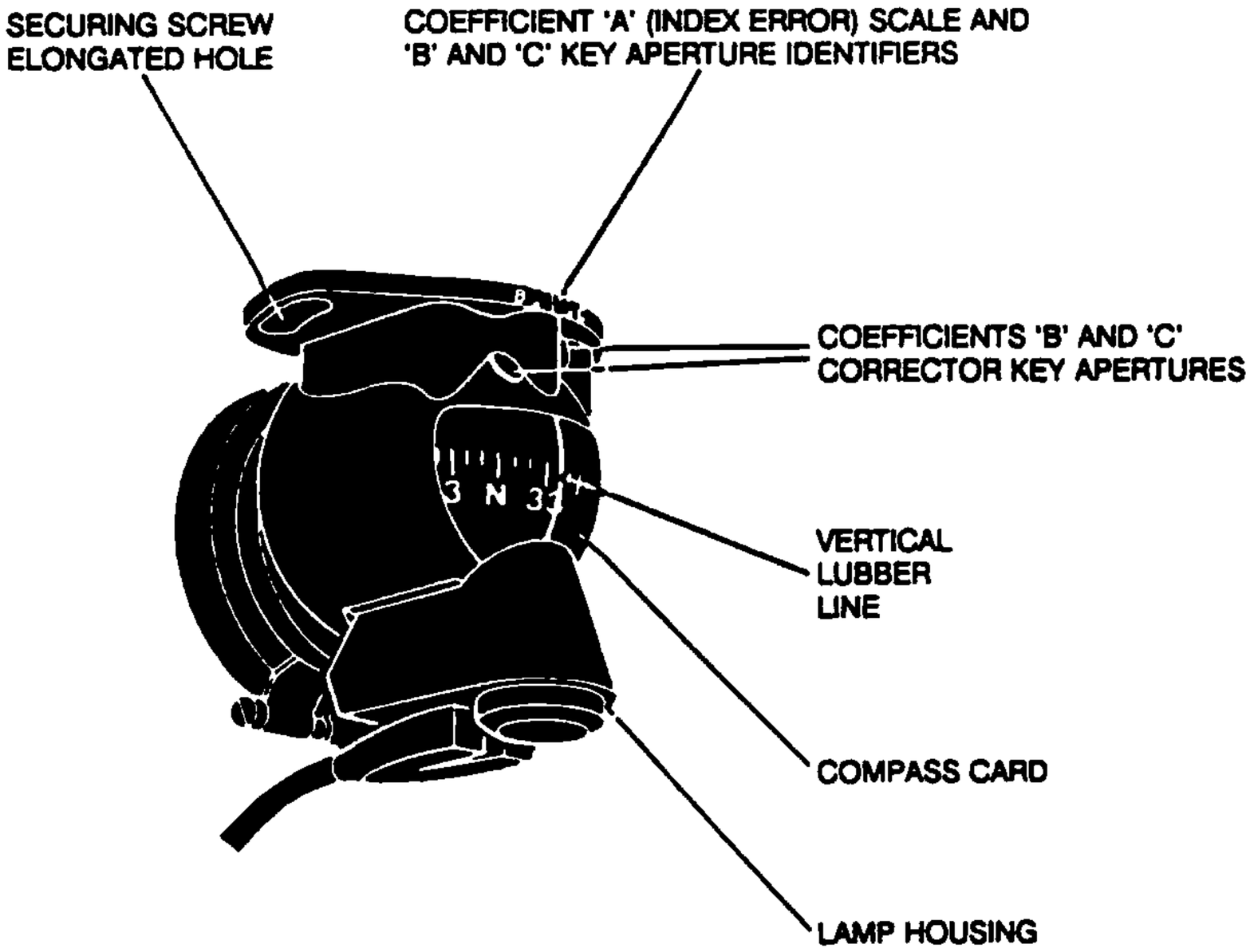



***Figure 1-61***

# MISCELLANEOUS EQUIPMENT

## ANTI-G SUIT PROVISIONS

### General

The physiological effect of high G loads on the pilot is counteracted by the wearing of the anti-G suit which is inflated during high G manoeuvres. The anti-G system utilises an anti-G valve, with an integral shutoff valve, and the aircraft ECS system.

### Operation

The anti-G valves are controlled by weights which are displaced by gravity during the application of normal positive loads and which increase the amount of pressure permitted to enter the trousers in direct proportion to the severity of the G. A push-to-test button on the top of the valve allows manual testing of the system. The trousers are connected to the valves through quick-disconnect couplings on the rear left side panel. During ejection, the quick disconnect coupling disengages, separating man and aircraft. The system schematic is shown at Fig 1-62.

## AIRCRAFT CLOCKS

An aircraft clock is located on the left side wall of each cockpit. The clock is powered from the battery direct busbar via the CLOCKS COMP LIGHT C/B. Instrument lighting is supplied from the cockpit instrument lighting system. An elapsed time function is operated by a triple function start-stop-reset knob, located on the top right corner. The instrument incorporates a calibrated sixty segment outer scale, which is rotatable by operation of an adjust knob on the bottom right corner. The time is adjusted by pulling a knob, on the bottom right corner, and turning to set the minutes and hours pointers. The aircraft clock is shown at Figure 1-63.

## MAP CASES

Both cockpits are provided with map cases on the right wall capable of carrying small books or maps. The case has a hinged door which is secured with velcro strips. Caution should be exercised prior to carrying out negative G manoeuvres as the door may come open in-flight. An additional small case is provided on the left wall of each cockpit.

## BAGGAGE COMPARTMENT

An aicraft baggage compartment is located on the left, rear fuselage. Maximum cargo weight in the compartment is 25kg (55lb). Compartment dimensions are : 18 inches wide, 11.5 inches high and 29 inches deep.

## INSTRUMENT FLYING HOOD

The rear cockpit can be fitted with an instrument flying hood for simulated instrument flight. The hood stays in position for ejection.

## *ANTI-G SYSTEM*

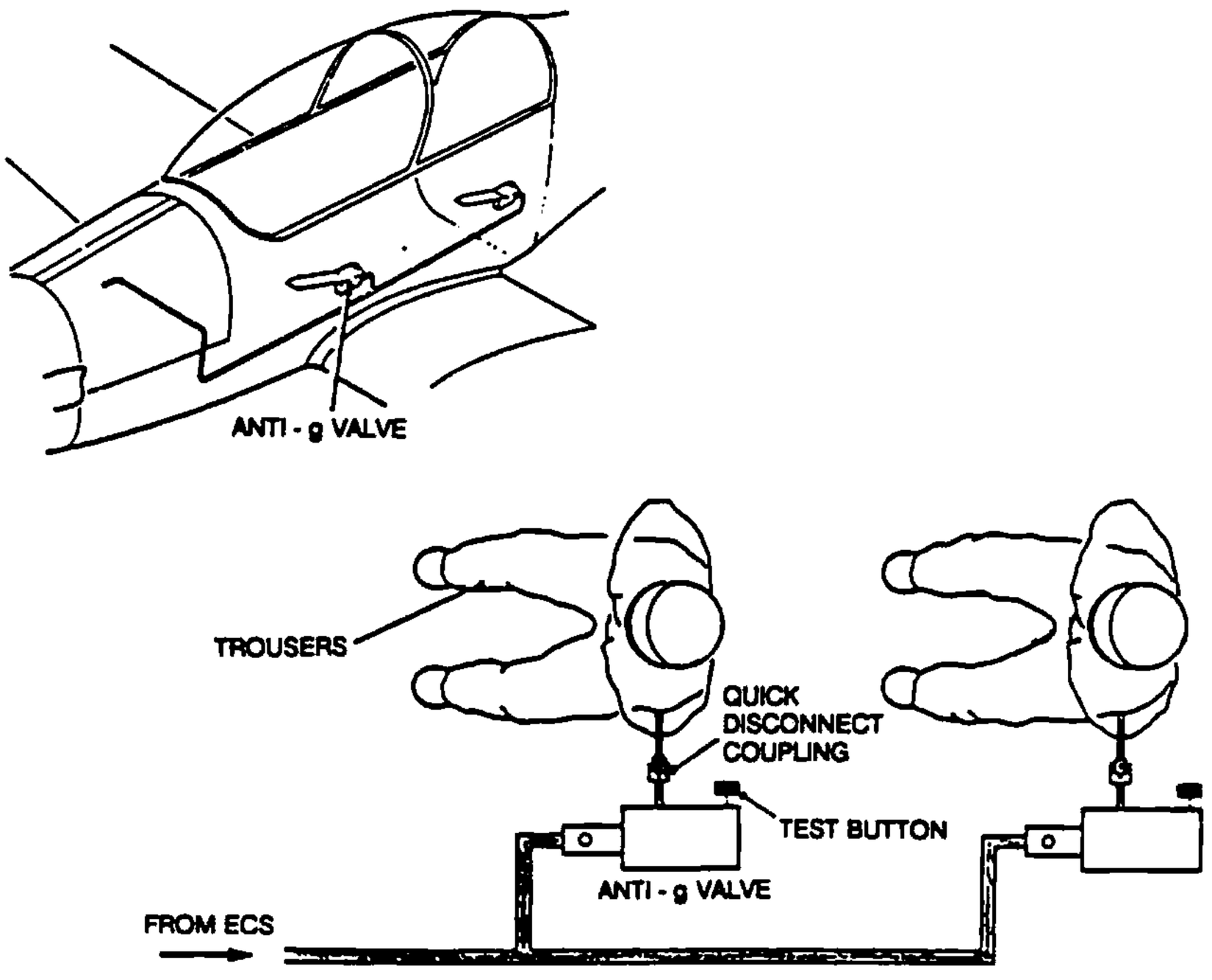



***Figure 1-62***

# AIRCRAFT CLOCK

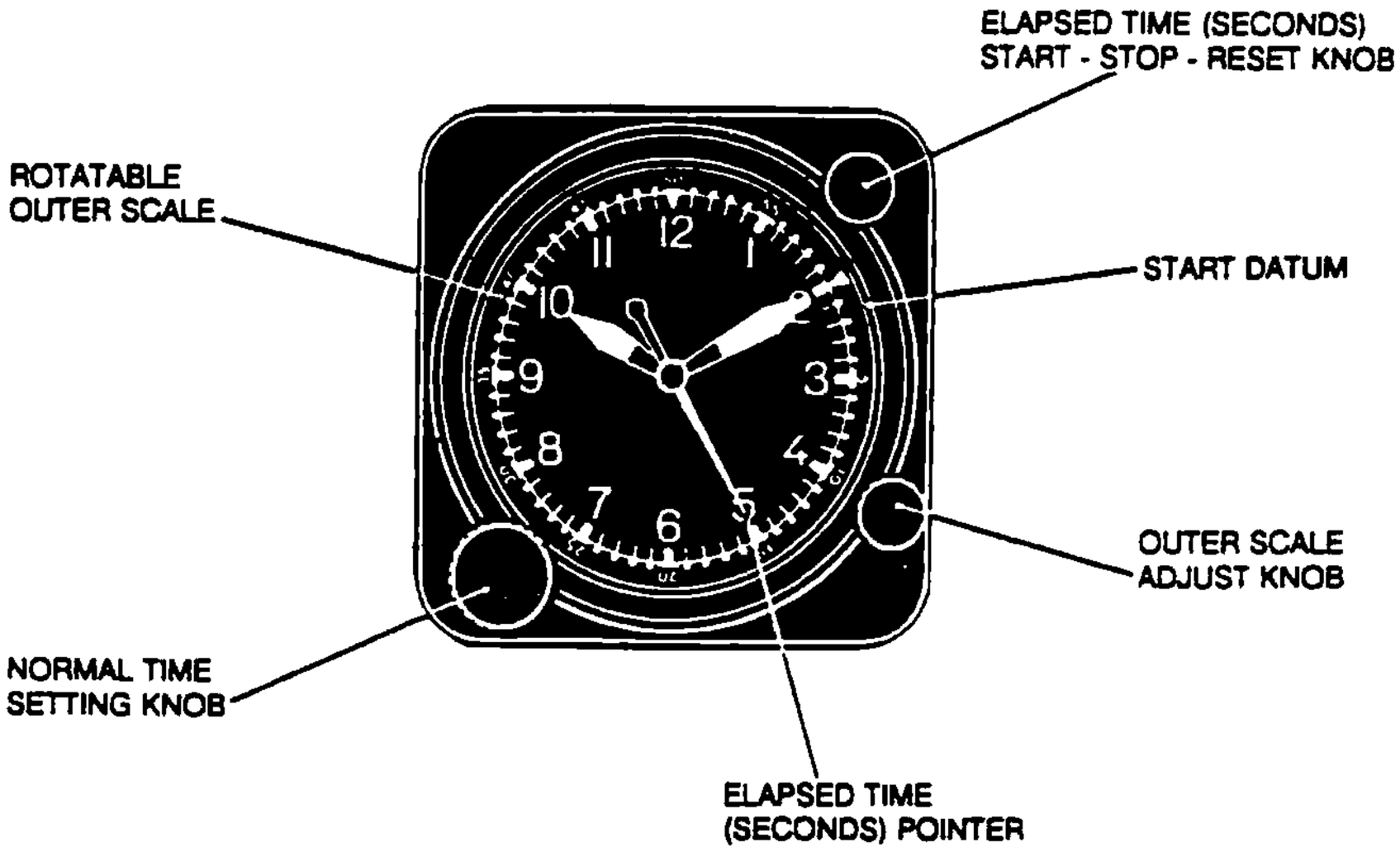



**Figure 1-63**

## ABBREVIATED CHECKLIST

An abbreviated checklist (DI(AF) AAP 7212.007-1CL) is provided for in-flight use by the pilot. The checklist is an abbreviated version of Section 2 (Normal Procedures) and Section 3 (Emergency Procedures) of the PC9/A Flight Manual.

## STATIC DISCHARGERS

Static electricity can cause noise interference in radio communications equipment and disturbances in other electrical systems. Aircraft pick up static electricity charges during high speed impacts of rain, snow, ice and dust particles. The resultant electric field is concentrated at the tips of the aircraft's extremities, with a high field intensity in the atmosphere at the tip. Static dischargers are installed on the aircraft's extremities to move the discharge points away from the aircraft's surface, minimising RF coupling. In this way, the static charge picked up by the aircraft is discharged in a regulated and electrically `quiet' way. The location of the static dischargers is shown at Figure 1-64.

# SERVICING

## PARKING AND MOORING

In normal weather conditions the aircraft can be parked on any firm level ground, with park brake applied, or wheel chocks in place, or both. Blanks and covers should be fitted at any time the aircraft is not being ready for flight. The aircraft should be moored if it is to be parked in the open for long periods and weather conditions are unfavourable. In extreme conditions the aircraft should be parked in a hangar, as structural damage can occur in high winds, even when moored correctly. The method of mooring and the aircraft mooring points are shown at Figure 1-65.

## OXYGEN SYSTEM

Oxygen system replenishment is carried out at a service panel located on the left side of the fuselage, to the rear of the wing trailing edge, (access panel F9). The service panel is fitted with an oxygen replenishment valve, an earthing point and system pressure gauge. The pressure gauge is marked from 0 to 2,000 psi with a red line at 1,850 psi and a legend FULL.

**CAUTION**

Replenishment of the oxygen system should only be carried out by qualified personnel.

**NOTE**

If the oxygen used to replenish the aircraft is of an unknown or suspect quality, the aircraft oxygen system is to be fully purged upon return to home base.

# STATIC DISCHARGERS

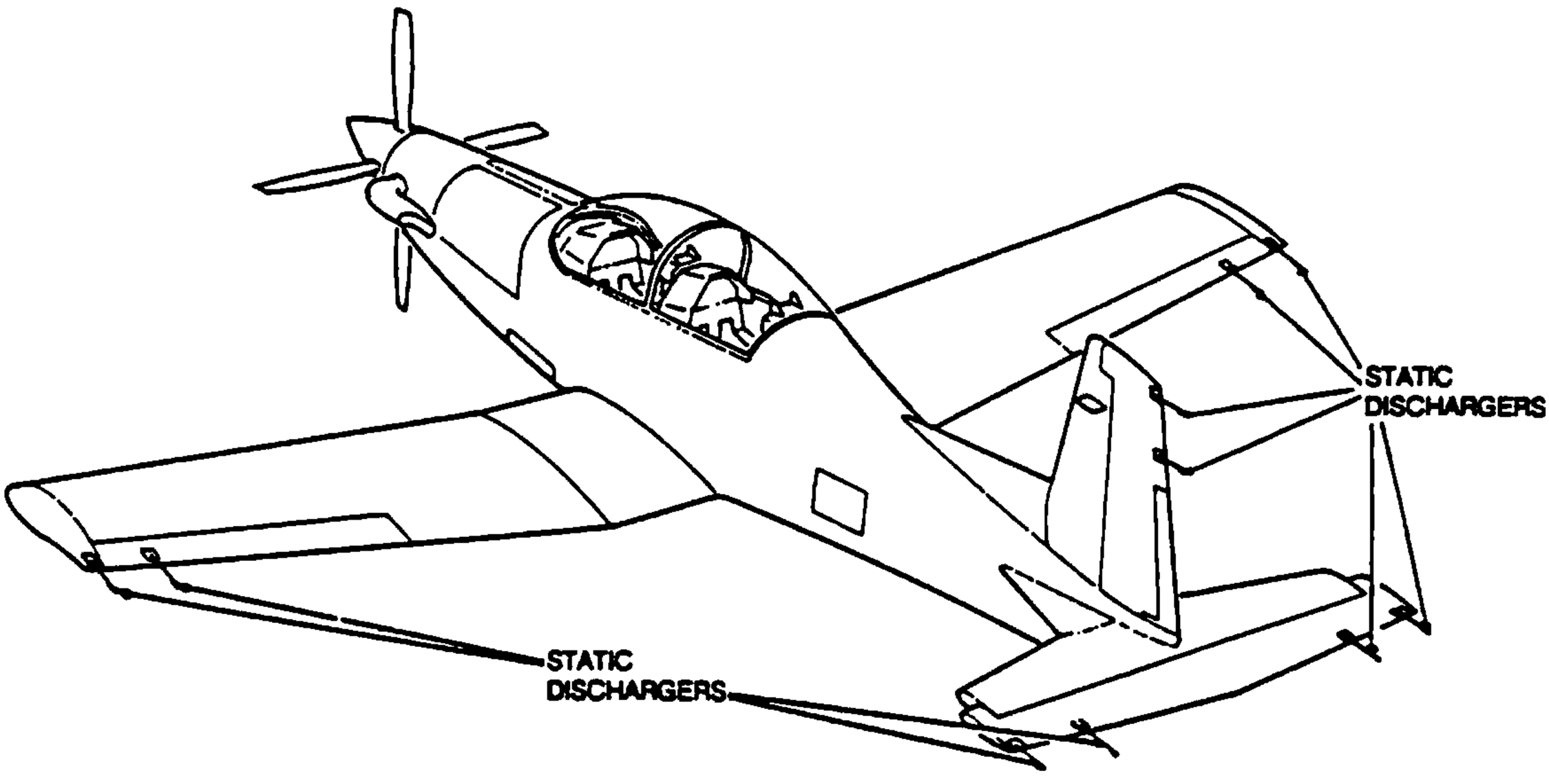



**Figure 1-64**

# STATIC DISCHARGERS

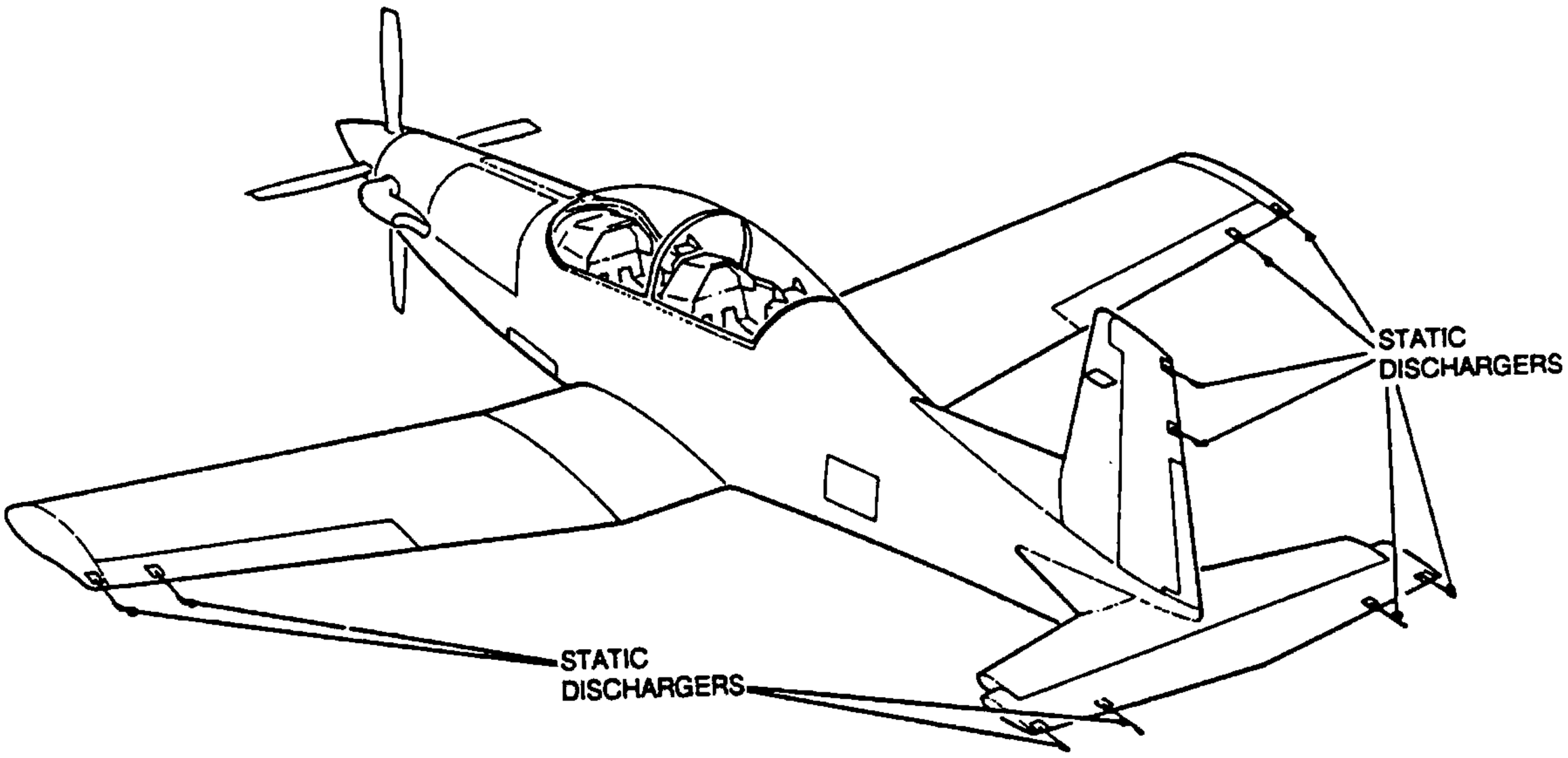



**Figure 1-64**

aa. Disconnect oxygen trolley static ground leads.

ab. Check service panel gauge, front and rear instrument panel gauges and oxygen cylinder gauges indicate the same pressure plus or minus 100 psi.

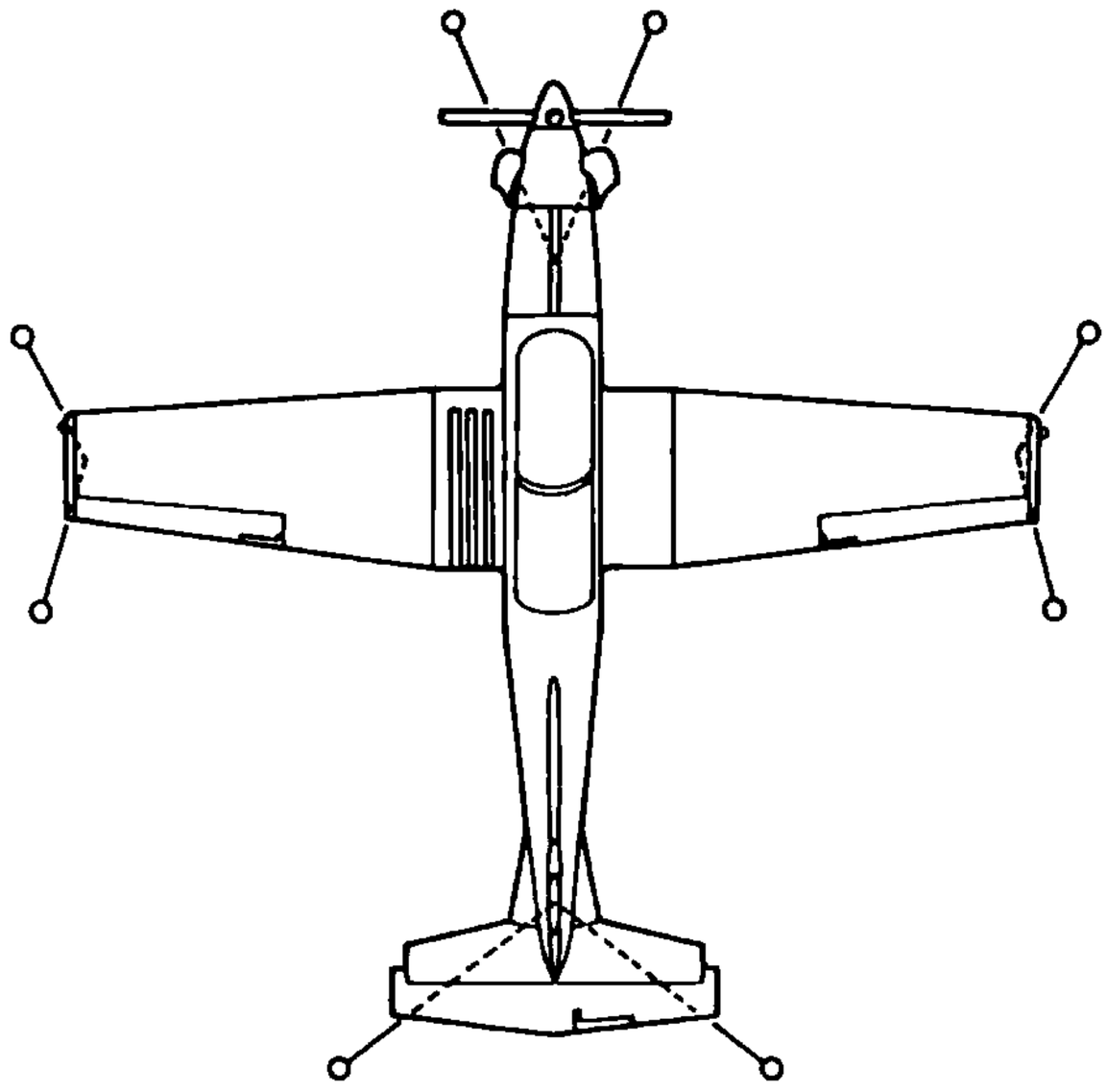
ac. Close the oxygen replenishment access panel.

ad. Close avionics bay access panel.

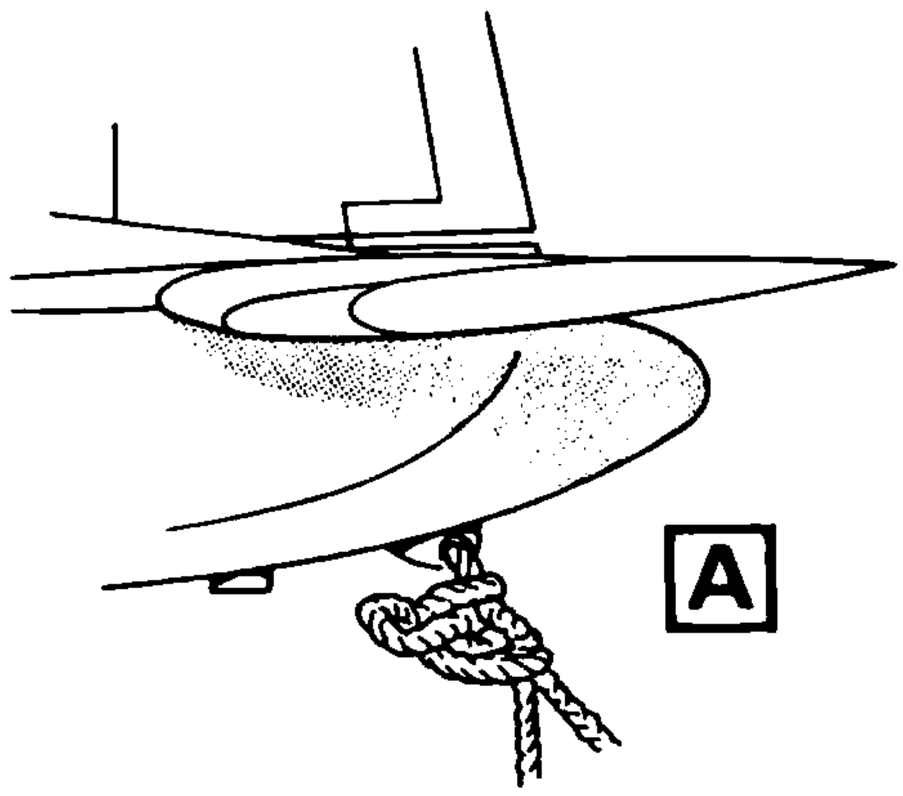
ae. Remove placard from external power unit receptacle.

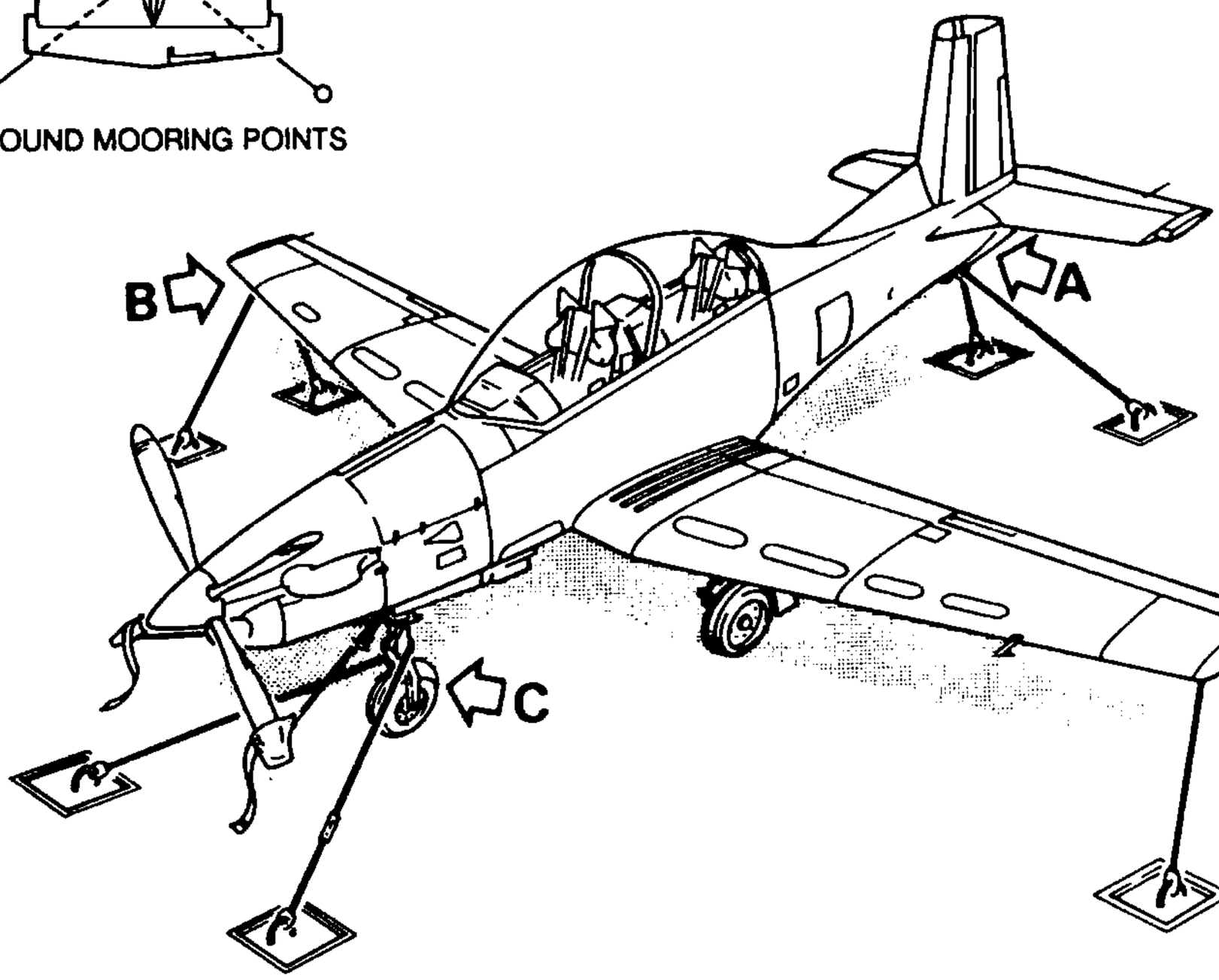
af. Make the necessary entry in the aircraft EE500.

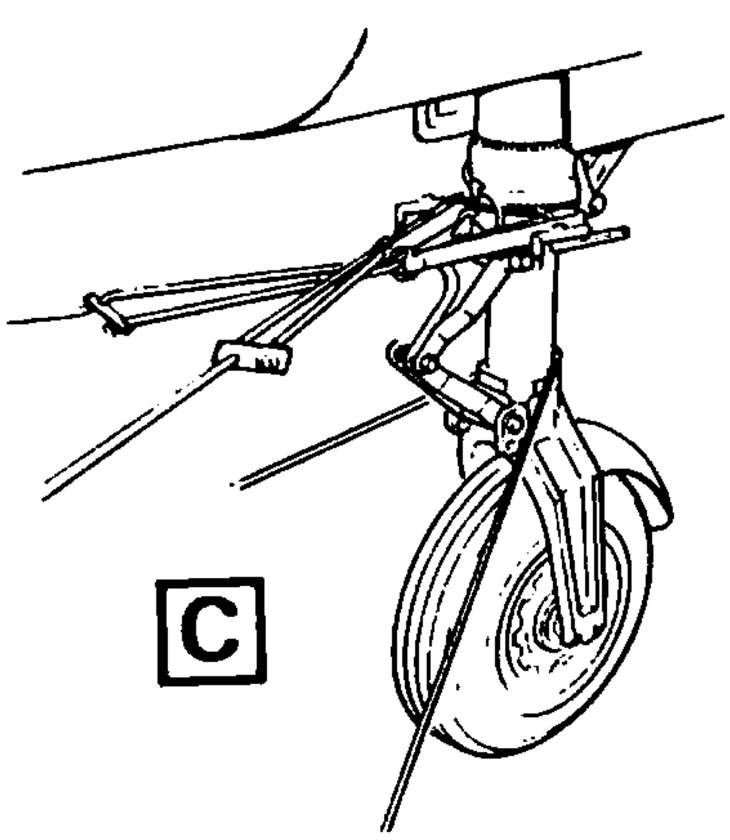
# MOORING POINTS

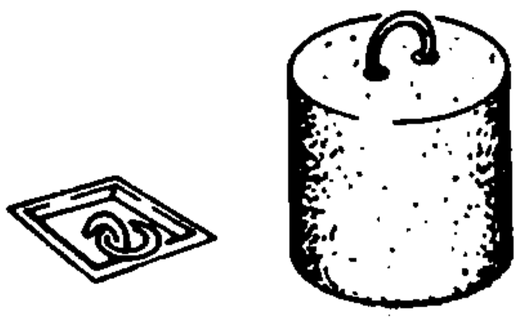


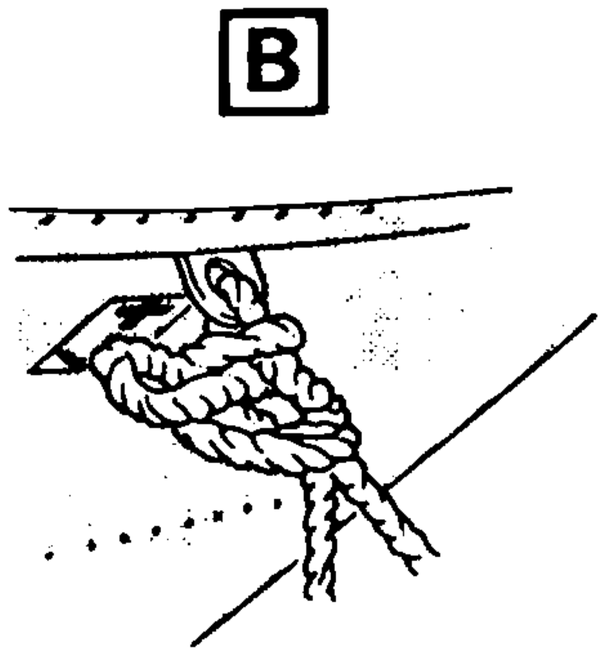
GROUND MOORING POINTS






RECESSED RING MOORING POINT
AND CONCRETE MOORING BLOCK

***Figure 1-65***

# OXYGEN CHARGING PRESSURES

| Temp(°C) | -10 | -5 | 0 | 5 | 10 | 15 | 20 | 25 | 30 | 35 | 40 | 45 | 50 | 55 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| Max Cylinder Pressure (PSI) | 1590 | 1630 | 1670 | 1720 | 1760 | 1800 | 1840 | 1880 | 1920 | 1960 | 2000 | 2050 | 2090 | 2130 |

***Figure 1-66***

# FUEL TANK AND SYSTEM CAPACITIES

| TANKS | | US Gal | IMP Gal | L | Kg | Lb |
|---|---|---|---|---|---|---|
| Internal Fuel | Unusable | 1.3 | 1.0 | 5 | 4 | 8 |
| | Usable | 141.3 | 117.7 | 535 | 420 | 925 |
| | Total | 142.6 | 118.7 | 540 | 424 | 933 |
| Ferry Underwing (total) | Unusable | 3.2 | 2.6 | 12 | 9 | 21 |
| | Usable | 126.8 | 105.6 | 480 | 377 | 831 |
| | Total | 130.0 | 108.2 | 492 | 386 | 852 |

***Figure 1-67***

**FUEL**

The left and right wing fuel tanks and the external underwing tanks are gravity filled through filler openings on the upper surface. The tanks should be kept full between flights to reduce explosive vapour space and condensation. Allowance should be made for expansion to minimise venting of fuel if ambient temperature is expected to rise markedly. The approved fuels are:

a. Jet A, Jet A-1 Aviation Turbine Kerosene (SG of 0.785 kg/L) or Jet B,

b. JP-4 or JP-5,

c. JP-8, and

d. Aviation Gasoline (MIL-G-5572), all grades, may be used for an accumulated time not to exceed 150 hrs in the period between engine overhauls.

Fig 1-67 lists the fuel tank capacities (calibrated in litres, with the other capacities being estimated equivalents).

**WARNING**

- Check fuel supply unit for correct fuel grade and type and for water contamination with an approved water detection kit.

- Ensure that the following earthing arrangements are complied with:

  a. Aircraft to ground.

  b. Vehicle to ground.

  c. Vehicle to aircraft.

  d. Filler nozzle to aircraft.

- Refuel and defuel in well ventilated areas only.

- Do not operate aircraft electrical or radio equipment whilst refueling.

- Do not replenish oxygen system during fueling or defueling.

- Suitable fire fighting equipment must be on immediate call.

- High frequency pulse transmissions in the vicinity of the aircraft represents a fire hazard.

The fueling procedure is as follows:

a. If refuelling at a non-RAAF establishment, ensure fuel supplied is checked for type, grade and freedom from contamination.

b. Earth the aircraft. Earth the fuel vehicle. Earth the fuel vehicle to the aircraft at the grounding point next to the filler cap.

c. Remove external power if connected.

d. Ensure all power is OFF.

e. Connect fueling nozzle grounding point to grounding facility adjacent to the filler cap.

f. Ensure that the nozzle is clean. Open the filler cap and fill the tank. Allow for the fuel to settle when topping off.

g. Remove fuel nozzle, install and secure filler cap and remove nozzle grounding point.

h. Repeat e. to g. for other tanks.

i. Remove fuel supply bonding lead from the grounding point next to the filler cap.

j. Clean up any fuel spillage (hose down if excessive).

k. Select MASTER BAT switch to ON and check fuel quantity gauges for correct indication. Reset the fuel totaliser with the F USED RESET switch and add external fuel with the F QTY SET switch. Select MASTER BAT to OFF.

### OIL

Oils specified for use in the PT6A-62 engine oil system are listed in Pratt and Whitney Aircraft of Canada Limited Service Bulletin No 13001, latest revision. Oils listed to conform to P&WC specification CPW 202 or PWA 521, Type II, are the only approved oils. The oil system capacities are as per Fig 1-68.

## OIL SYSTEM CAPACITIES

| | US Gall | IMP Gall | Litres |
|---|---|---|---|
| Engine Integral Tank | 3.3 | 2.75 | 12.49 |
| Tank Expansion Space | 0.7 | 0.58 | 2.65 |
| Total (including cooler) | 4.5 | 3.7 | 17.0 |

***Figure 1-68***

If operating conditions are such that the engine will be subjected to frequent cold soaking at ambient temperatures of -18° C or lower, use of 5 c.s (centistoke) oil (Type II) is recommended. The engine oil dipstick is marked MAX HOT, MAX COLD, ADD LITRES OR US QUARTS, 1,2,3,4,5,6. The term HOT refers to engine condition in the first ten minutes after shutdown. COLD refers to engine condition when the engine has been shutdown for 12 hours or more. Ideally the engine oil tank level should be checked and replenished, as required, within 10 minutes of shutdown.

**CAUTION**

- Never replenish oil in a cold engine as this can result in overfilling of the system. Always run the engine and recheck oil level before adding oil to the system.
- Ensure oil is of correct type. Do not mix brands, specifications or types of oils. If oils are accidentally mixed, drain and flush complete system and refill with approved oil.
- To prevent oil drips from dipstick contaminating equipment, hold a piece of absorbent lint free material under dipstick during removal.

The oil replenishment procedure is as follows:

a. Open the left engine access panel and secure open with struts provided.

b. Release locking mechanism and remove filler cap/dipstick assembly from the filler neck on accessory gearbox.

c. Replenish oil according to HOT/COLD condition of the engine.

d. Reinstall filler cap/dipstick assembly and engage locking mechanism.

e. Close engine access panel.

Complete oil system replenishment (after full draining) requires the following differences in procedure:

a. After draining, fill with 12.49 litres of oil.

b. Reinstall and lock filler cap/dipstick assembly and close engine access panel.

c. Start engine, idle for at least two minutes and feather the propeller twice during idling. Select PCL to OFF and shutdown the engine.

d. Recheck the oil level and fill to MAX HOT.

### BRAKE FLUID

Brake hydraulic system servicing consists primarily of maintaining the hydraulic fluid in the brake reservoir. The reservoir is mounted below the forward left corner of the rear cockpit instrument compartment with the fluid level indicator/filler plug protruding through the compartment base structure. Brake fluid is contained in the cylinder chamber below the piston and the chamber above the piston is vented to atmosphere. The reservoir is filled through the filler plug on top of the hollow piston shaft, the shaft being lifted as the fluid is poured in. The fluid level is indicated by the degree of exposure of the piston shaft. When the green (lower) band on the piston shaft is visible the fluid level is correct. If only the red (upper) band is visible the fluid level is low and the reservoir requires replenishing. The brake hydraulic fluid is to conform with specification MIL-H-5606E and total system capacity is 0.42 litres (0.89 US Pint or 0.74 IMP Pint). The following procedure is used for replenishing the brake fluid:

**CAUTION**

When removing the reservoir filler plug do not allow the piston shaft to rotate as this will damage the diaphragm.

a. Remove the lockwire from the reservoir filler plug and remove filler plug.

b. Discard O-ring.

c. Add hydraulic fluid until the green band is visible.

d. Smear hydraulic fluid on O-ring (P/N 946.91.27.103) and install on reservoir filler plug.

e. Install and tighten filler plug.

f. Lockwire filler plug.

## HYDRAULIC FLUID

The hydraulic reservoir is incorporated in the power package which is installed in the hydraulic services compartment. Replenishment of the system is through the pressure and return line servicing connections of the package. A fluid indicator rod with four coloured indicator bands is installed in the reservoir and protrudes forward from the package. The hydraulic fluid is to conform with specification MIL-H-5606E. Approximate system capacity is 4.0 litres (1.5 US Gall or 0.87 IMP Gall).

**CAUTION**

Replenishment of hydraulic fluid should only be carried out by qualified personnel.

## TYRE SERVICING

The main wheels are fitted with 6.50-8-8 TL low pressure tubeless tyres. The nose wheel is fitted with a 6.00-6-8 TL low pressure tubeless tyre. Inflation pressures are 55 psi for the main and nose wheels.

# ACCUMULATOR PRE-CHARGE PRESSURE

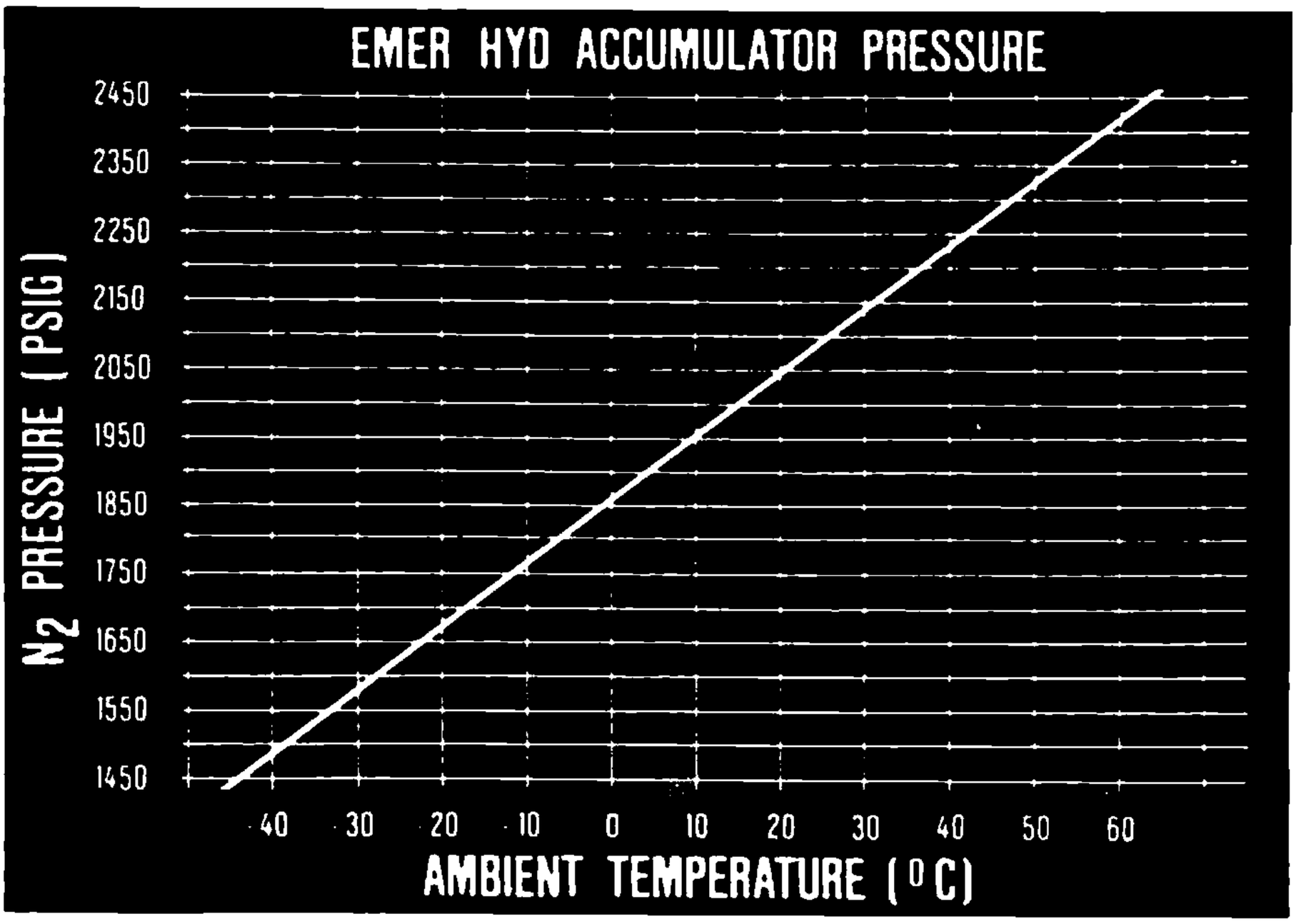



**Figure 1-69**

The procedure for replenishing the hydraulic accumulator is as follows:

a. Energise electrical system.

b. Open hydraulic services compartment and switch indication switch ON.

c. Pull manual release handle and ensure all hydraulic pressure in the main and emergency systems is released.

d. Obtain OAT and determine required accumulator pressure from the table.

e. Check pressure. On a turn-around it is permissable to ensure the pressure remains stable and there is no appreciable loss of pressure over a ten minute period.

f. The air brake must be extended and ground lock fitted to allow access to charging valve.

g. Remove dust cap from accumulator charging valve, attach charging cylinder to valve, using adapters as required.

h. Purge nitrogen charging hose and install hose on charging valve.

i. Charge system until pressure is reached. Allow 10 minutes to stabilise and re-charge if necessary to required pressure, plus 50 psi.

j. Disconnect charging cylinder and replace dust cap in charging valve.

k Select hydraulic services compartment indication switch to OFF.

l. Pressurise hydraulic system to 3,000 psi.

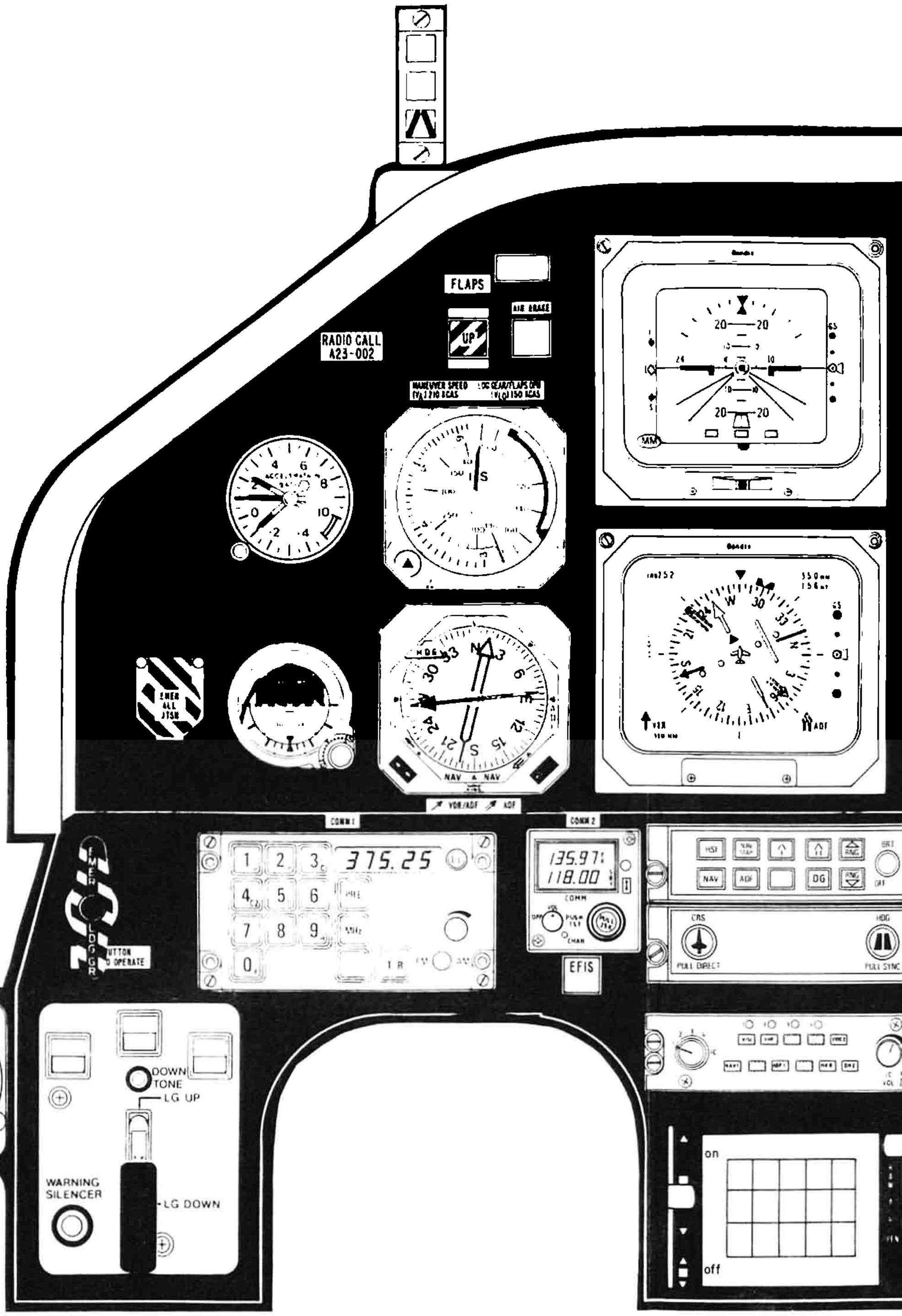
m. Close and secure access panel.

n. De-energise aircraft electrical system.

## TYRE SERVICING

The main wheels are fitted with 6.50-8-8 TL low pressure tubeless tyres. The nose wheel is fitted with a 6.00-6-8 TL low pressure tubeless tyre. Inflation pressures are 55 psi for the main and nose wheels.

# FRO


FLAPS
UP
AIR BRAKE
RADIO CALL
A23-002
MANOEUVER SPEED
(VA) 210 KCAS
LDG GEAR/FLAPS DWN
(VLO) 150 KCAS
NAV
ADF
DG
COMM 1
COMM 2
375.25
135.97
118.00
COMM
EFIS
CRS
HDG
PULL DIRECT
PULL SYNC
DOWN
TONE
LG UP
LG DOWN
WARNING
SILENCER
on
off

# FRONT COCKPIT INSTRUMENT PANEL

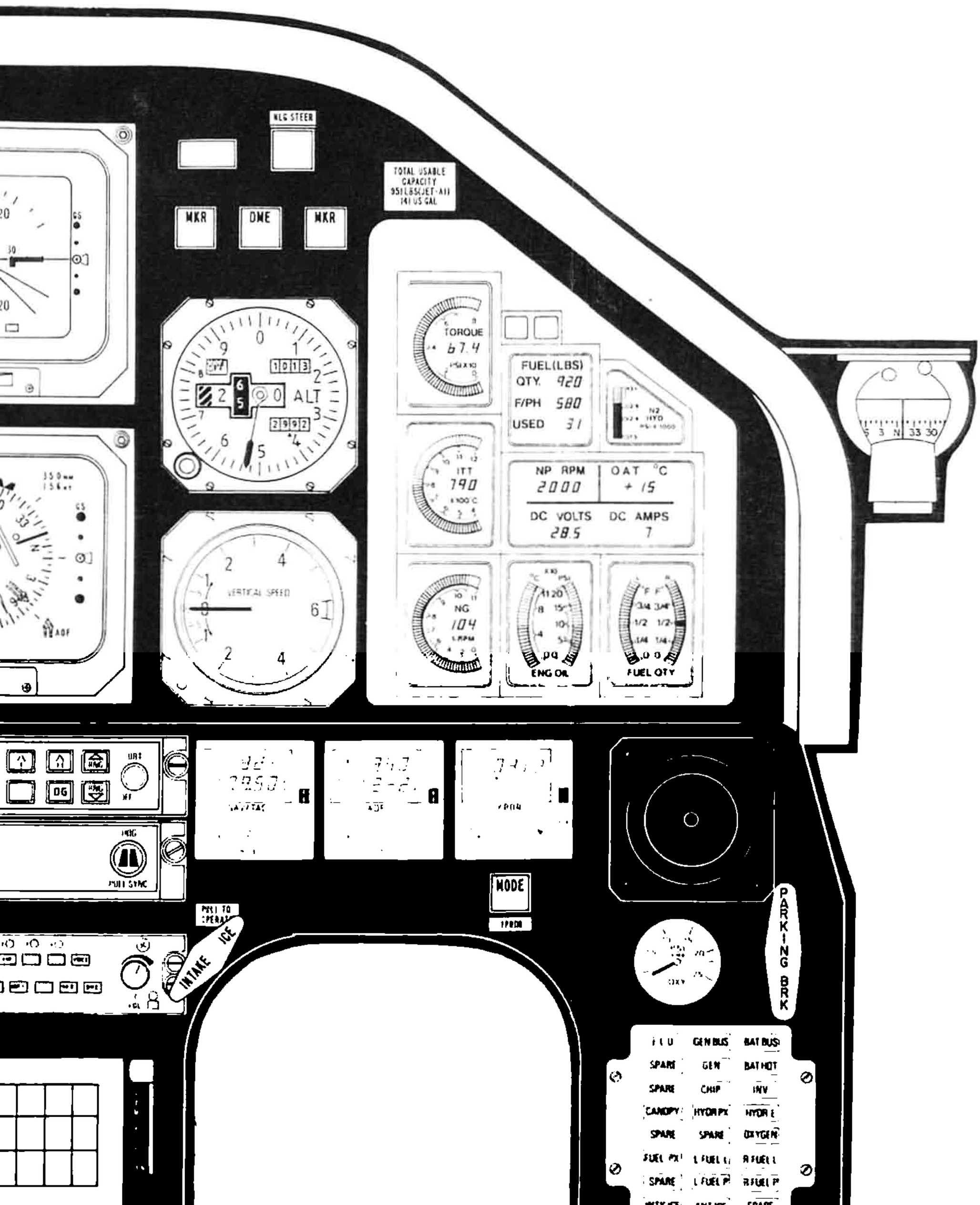



*Fig FO 1-1*

***Figure FO 1-1 (Sheet 1 of 2)***

# FRONT COCKPIT CONSOLES

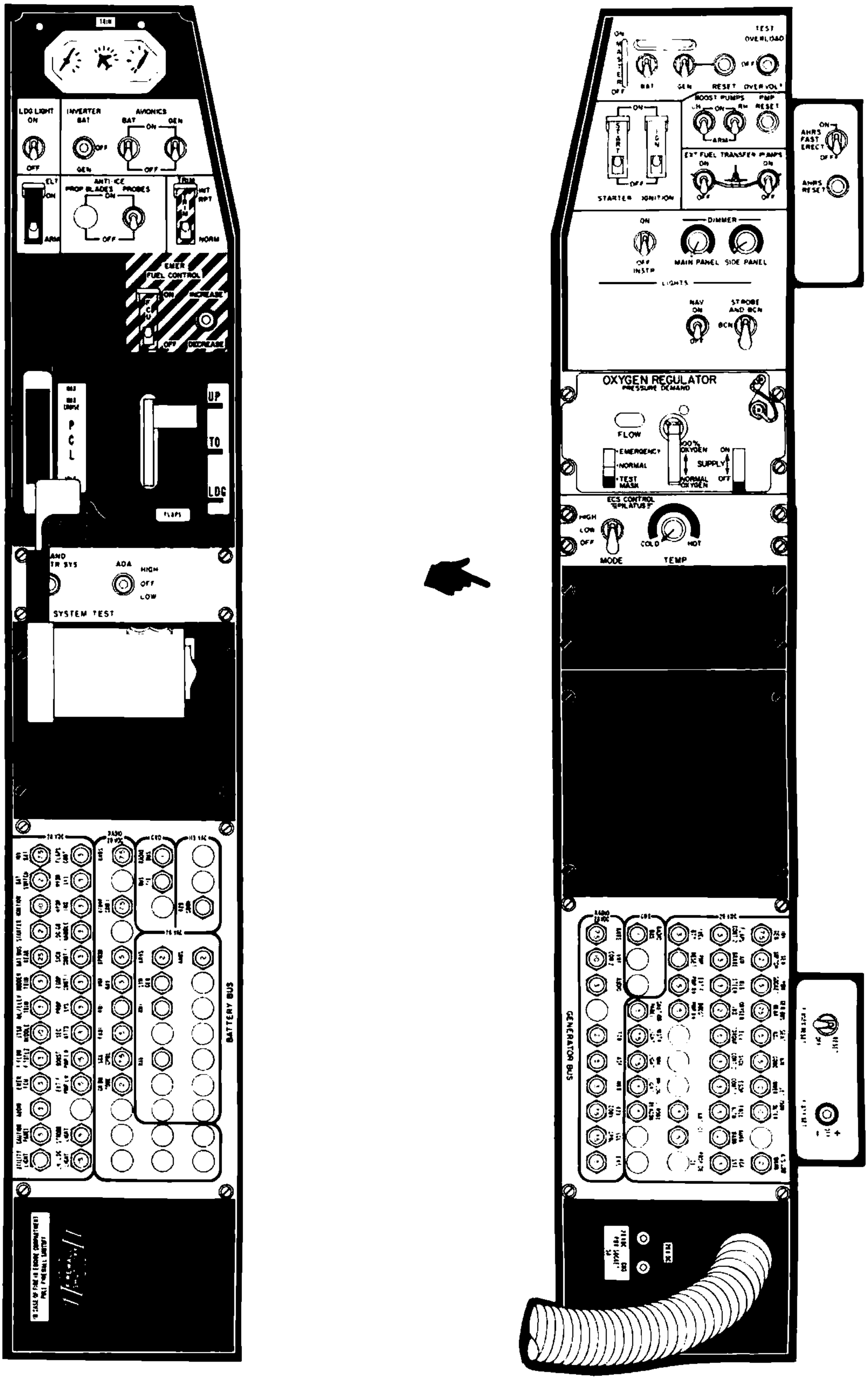
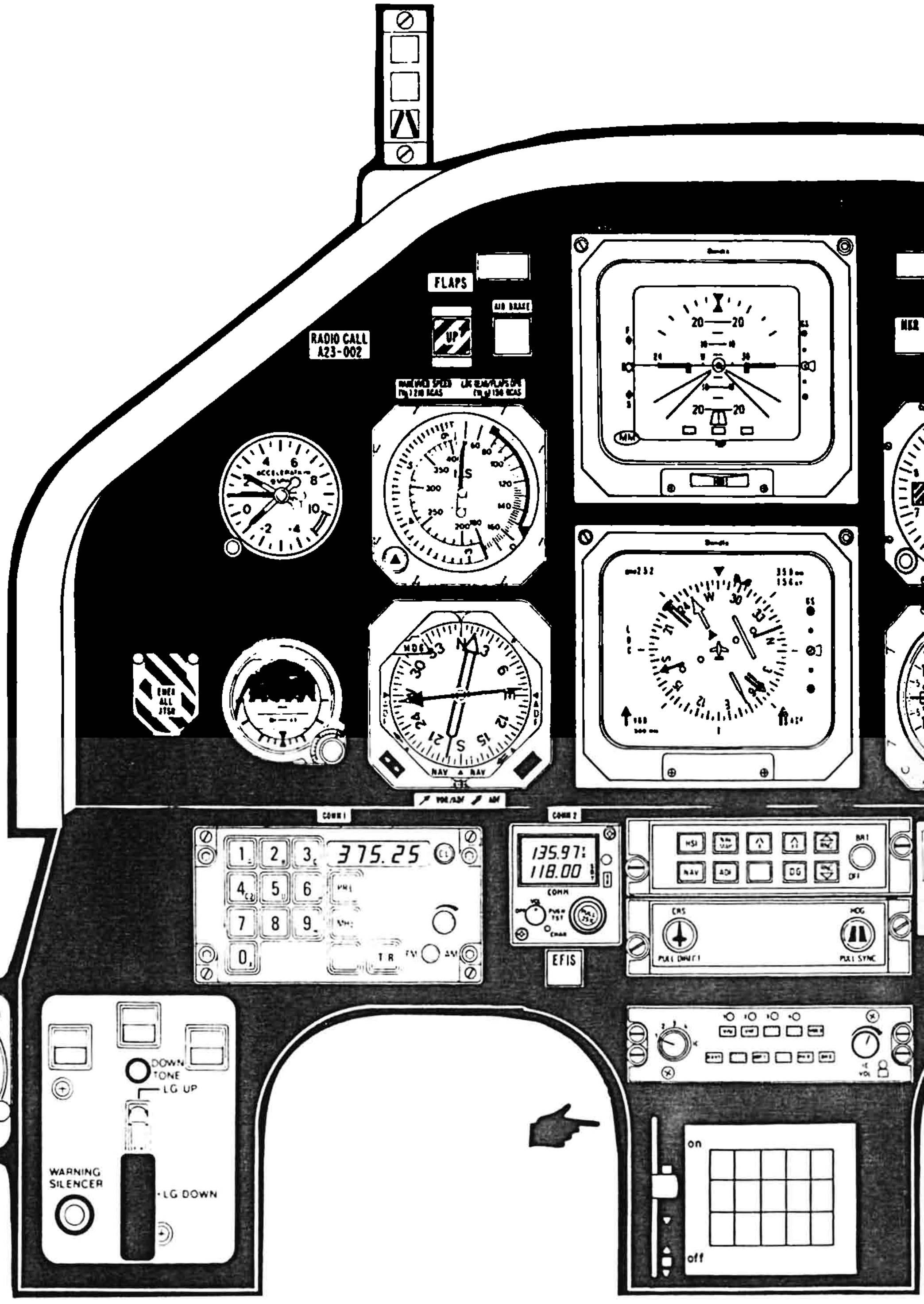



Fig FO 1-1

**Figure FO 1-1 (Sheet 2 of 2)**

# REAR COC


FLAPS
UP
AIR BRAKE
RADIO CALL
A23-002
ACCELERATION
COMM1
COMM2
375.25
135.97
118.00
COMM
EFIS
HSI
NAV
ADI
HDG
BRT
DIM
CRS
PULL DIRECT
PULL SYNC
DOWN
TONE
LG UP
WARNING
SILENCER
LG DOWN
on
off

# REAR COCKPIT INSTRUMENT PANEL

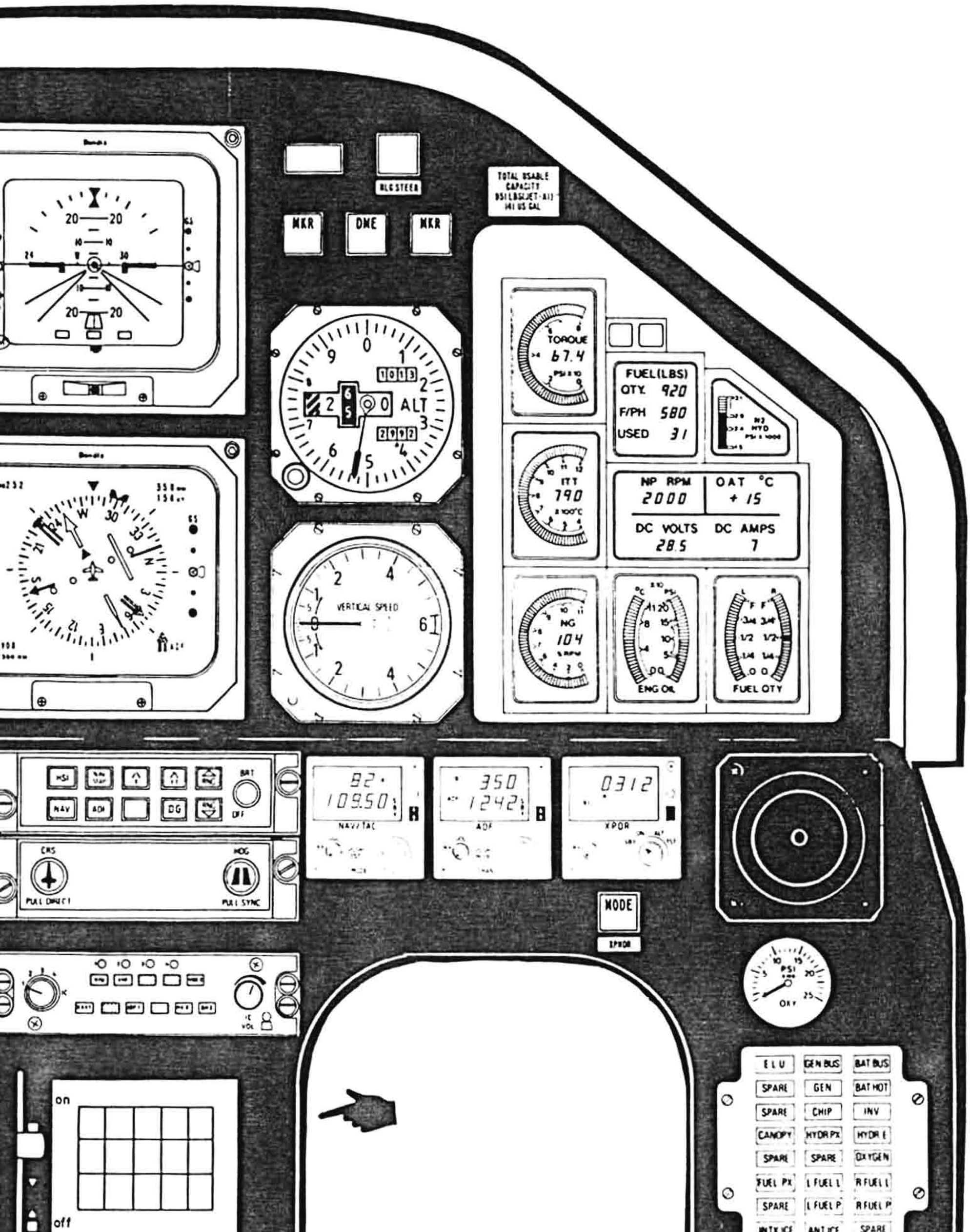



Fig FO 1-2

Figure FO 1-2 (Sheet 1 of 2)

# REAR COCKPIT CONSOLES

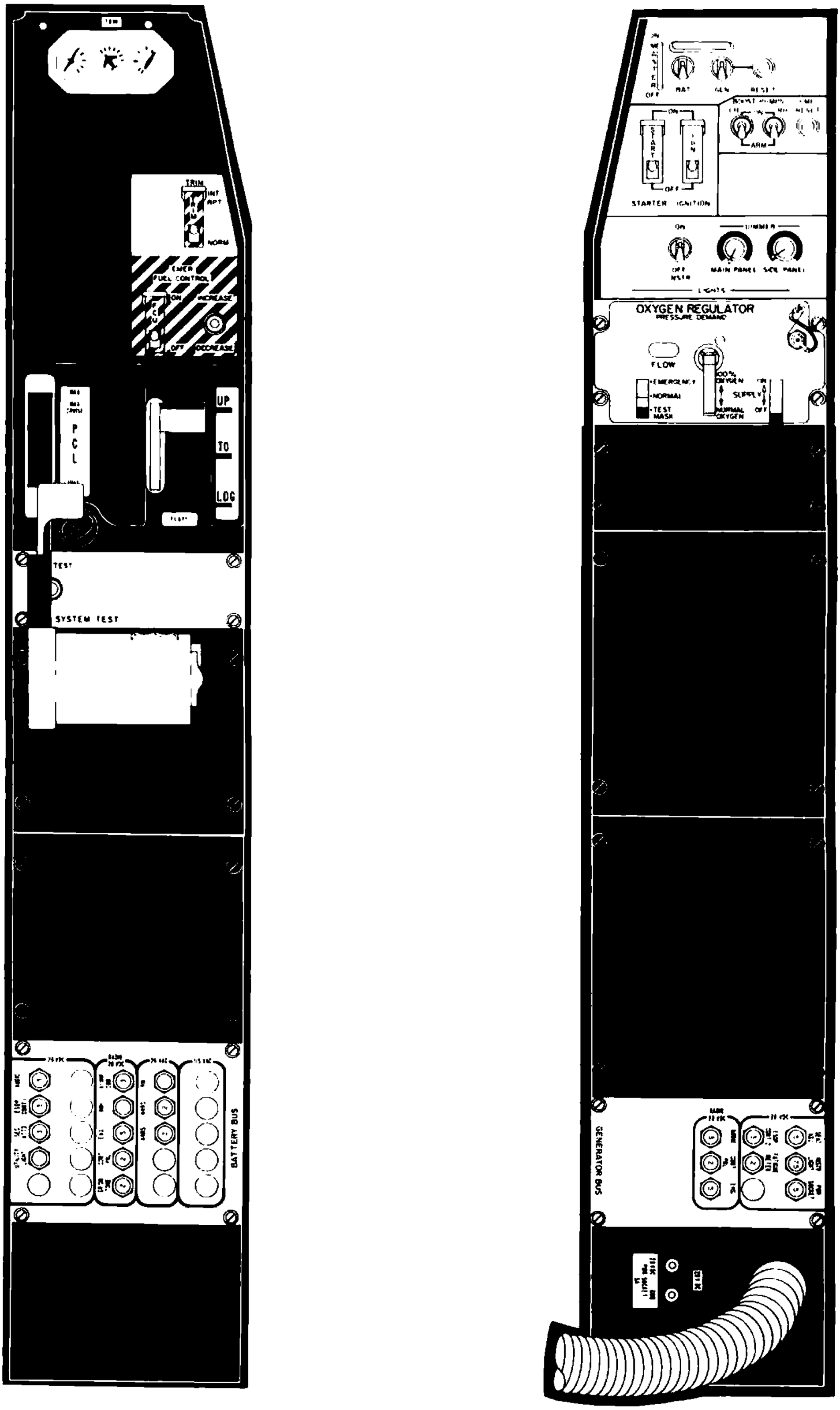



Fig FO 1-2

**Figure FO 1-2 (Sheet 2 of 2)**

FUEL CONTROL UNIT

MINIMUM
PRESSURIZING
VALVE

AIRBLEED TO ATMOSPHERE

MANUAL OVERRIDE
ACTUATING SHAFT

OPERATING PIN

GOVERNOR BELLOWS

PROPELLER
GOVERNOR

INTERFACE
VALVE

ISOLATING
SOLENOID
VALVE

ACCELERATION
BELLOWS
(EVACUATED)

Np

Ng

TORQUE

ITT

AIRFRAME

TRIM PLUG
OR PDIU

ELECTRONIC
LIMITER UNIT

ACCELERATION PRESSURE

GOVERNOR SERVO PRESSURE

BYPASS FUEL PRESSURE

UNMETERED FUEL PRESSURE

METERED FUEL PRESSURE

$N_g$ GOVE

ME

COMPRESSOR
PRESSURE AIR

FILTE

# FUEL CONTROL SYSTEM SCHEMATIC

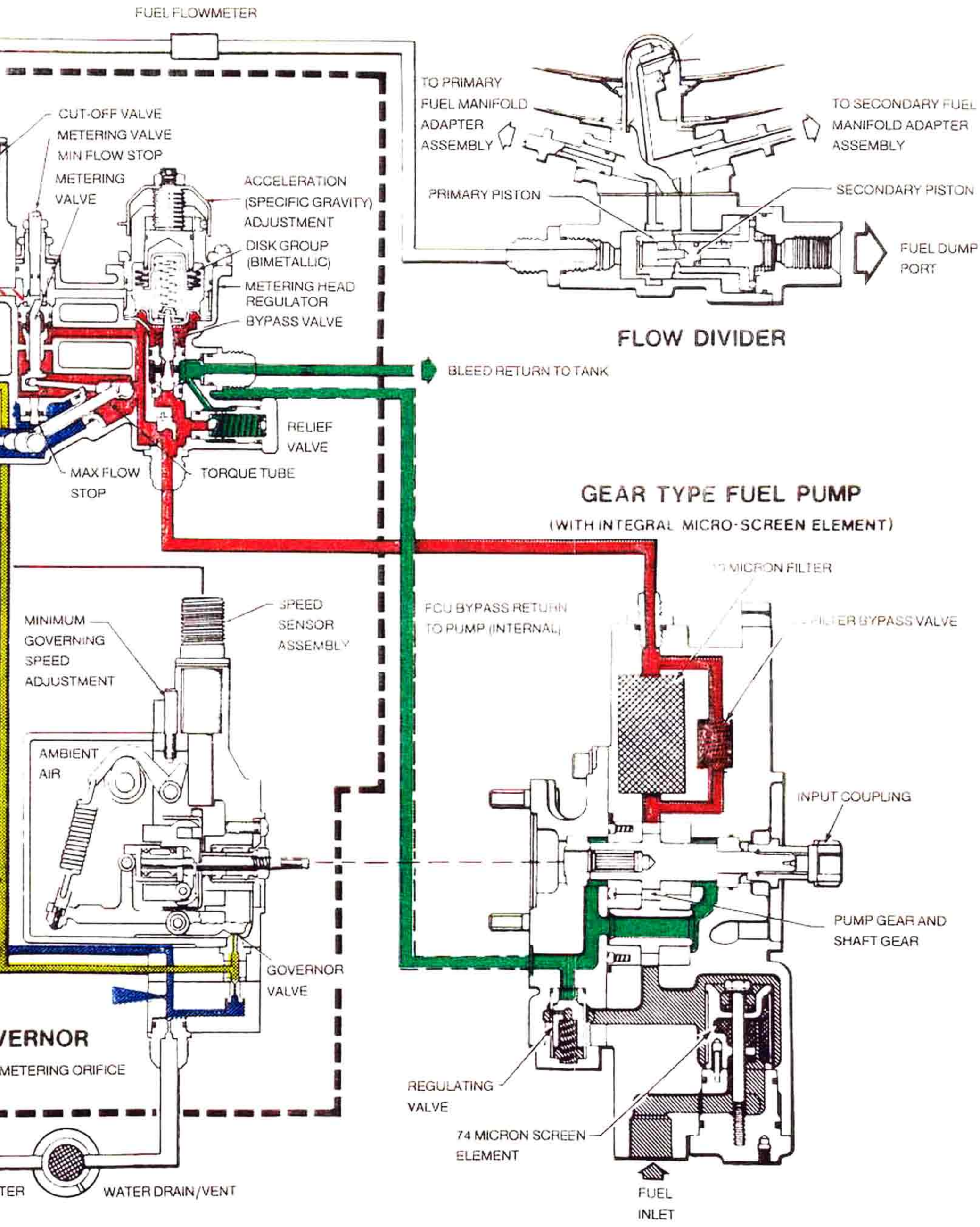
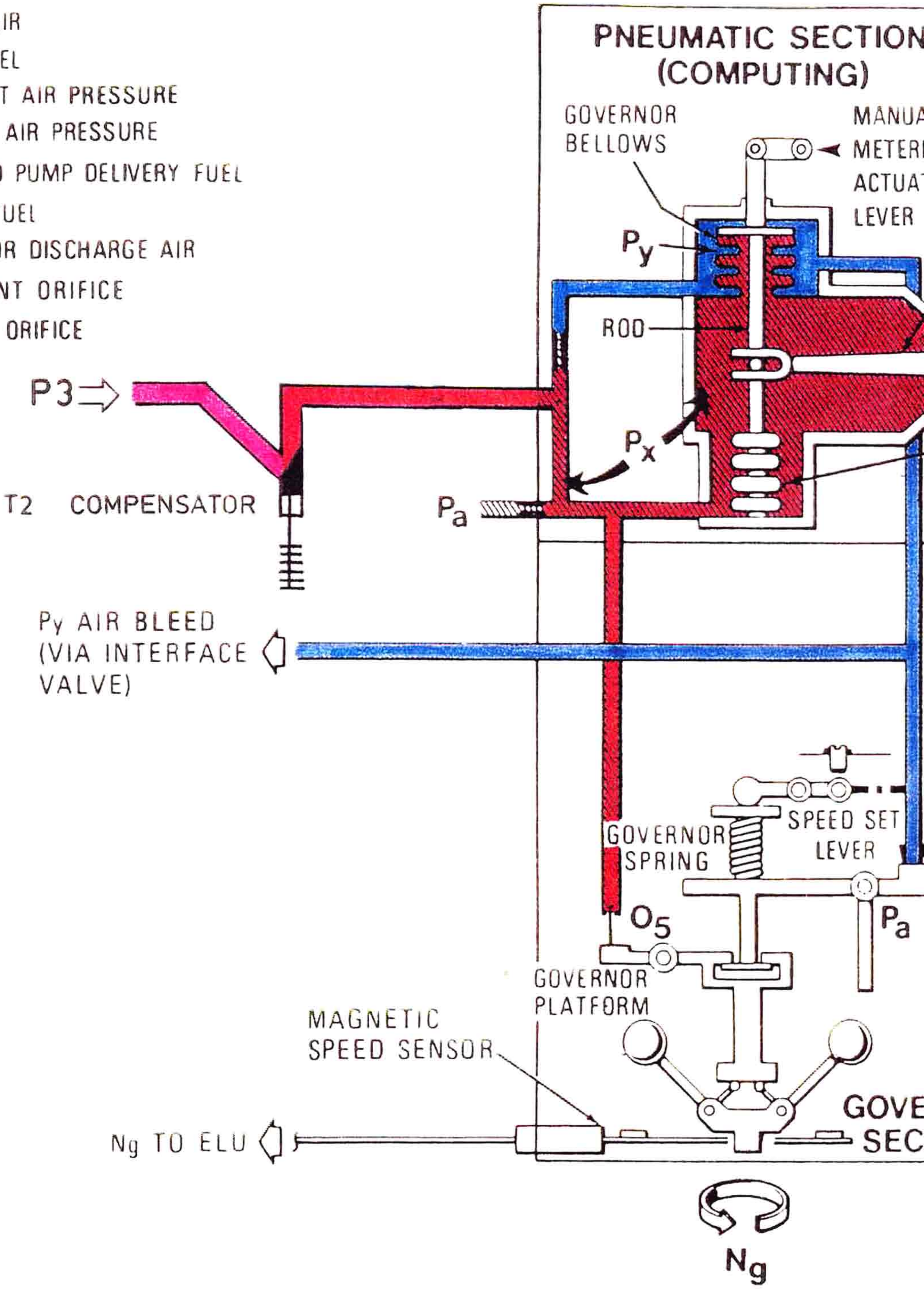



*Fig FO 1-3*

***Figure FO 1-3***

## LEGEND

| | |
|---|---|
| $P_a$ | AMBIENT AIR |
| $P_o$ | BYPASS FUEL |
| $P_x$ | ENRICHMENT AIR PRESSURE |
| $P_y$ | GOVERNING AIR PRESSURE |
| $P_1$ | UNMETERED PUMP DELIVERY FUEL |
| $P_2$ | METERED FUEL |
| $P_3$ | COMPRESSOR DISCHARGE AIR |
| $O_5$ | ENRICHMENT ORIFICE |
| $O_6$ | GOVERNING ORIFICE |



# 6A–62 FUEL CONTROL SCHEMATIC

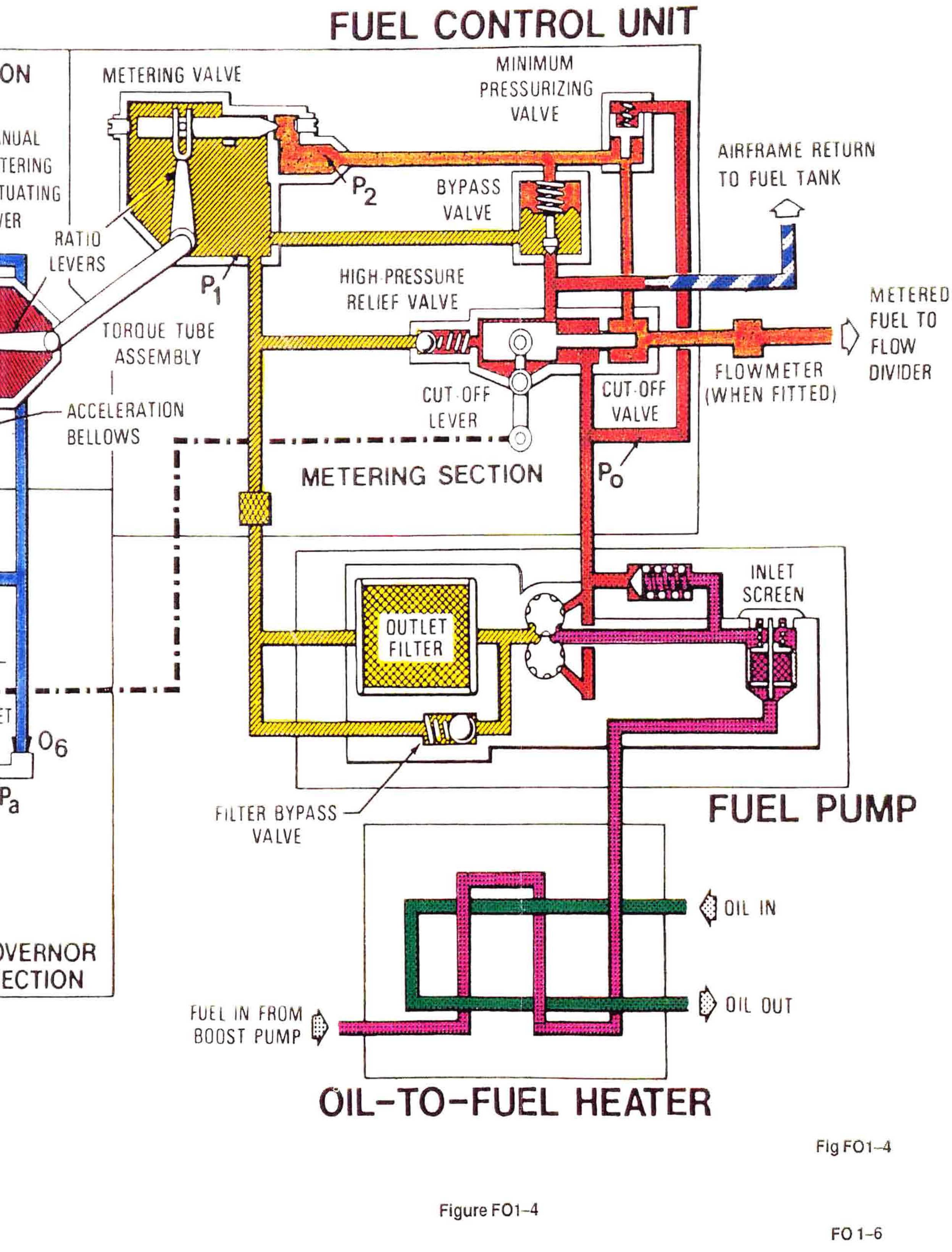
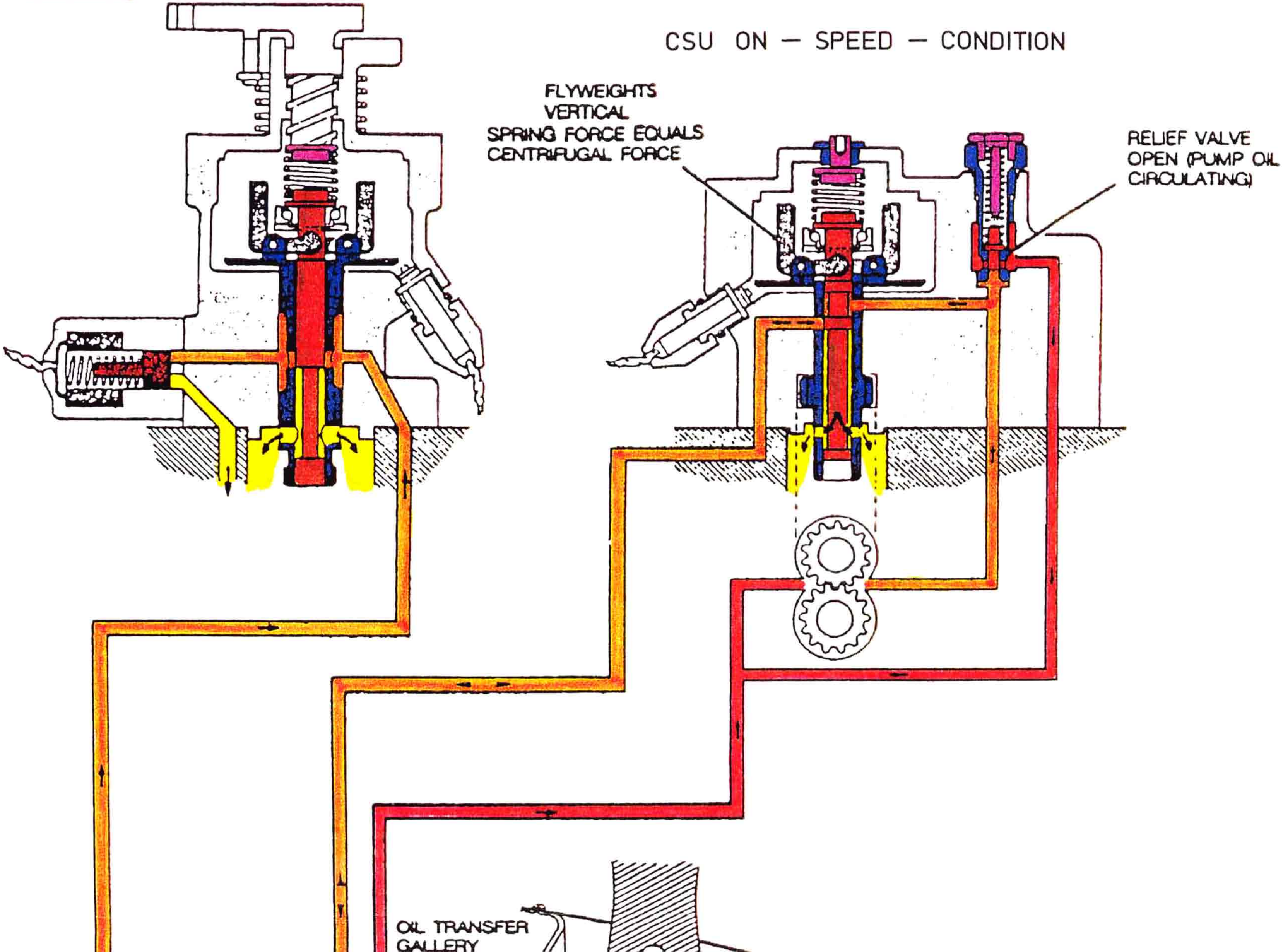



Fig FO1–4

Figure FO1–4

OVERSPEED GOVERNOR

CSU ON — SPEED — CONDITION



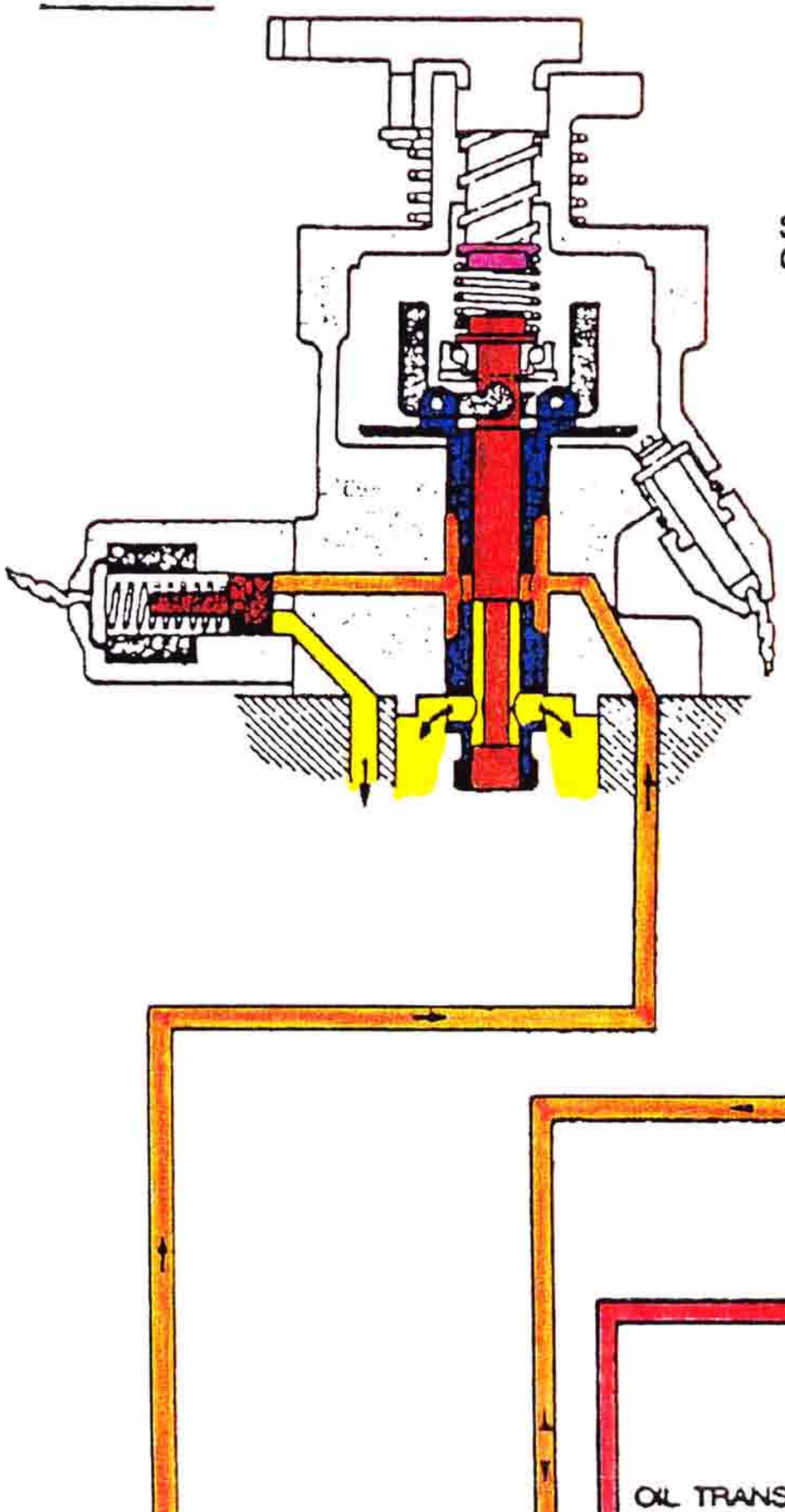
PC9/A PROPELLER SYSTEM SCHEMATIC

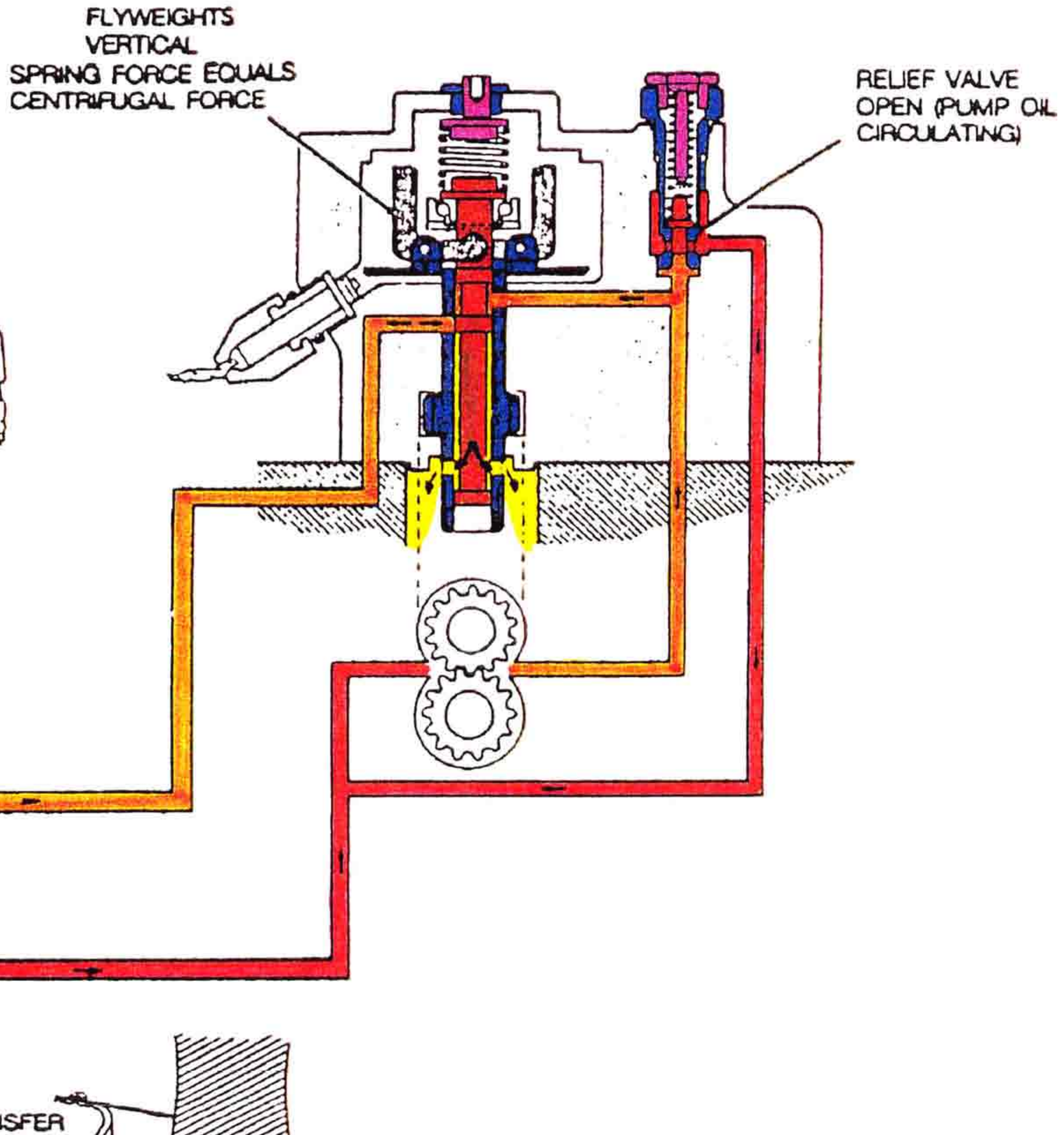
Figure FO1–5

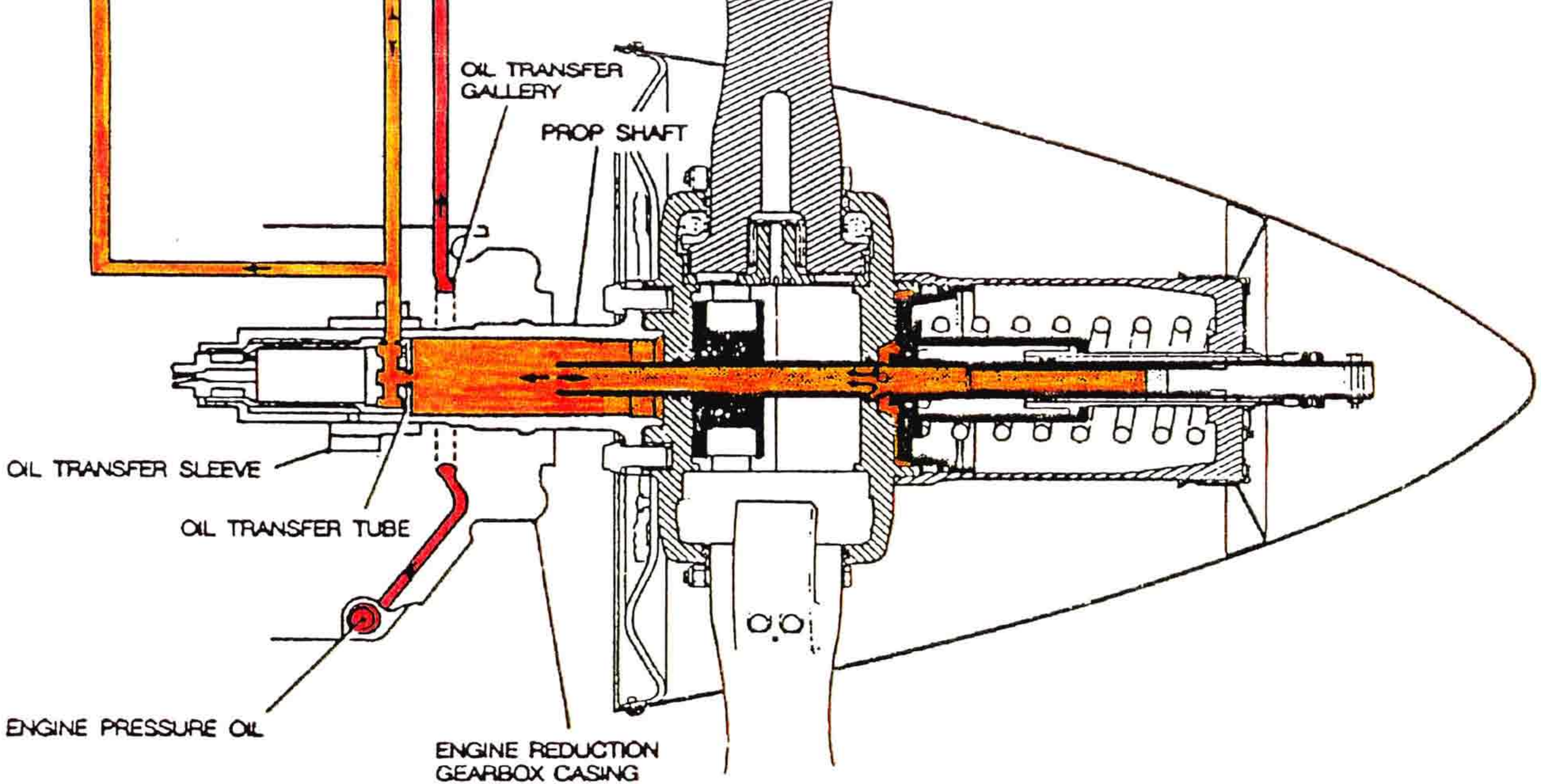
Fig FO 1–5

OVERSPEED GOVERNOR

CSU ON — SPEED — CONDITION

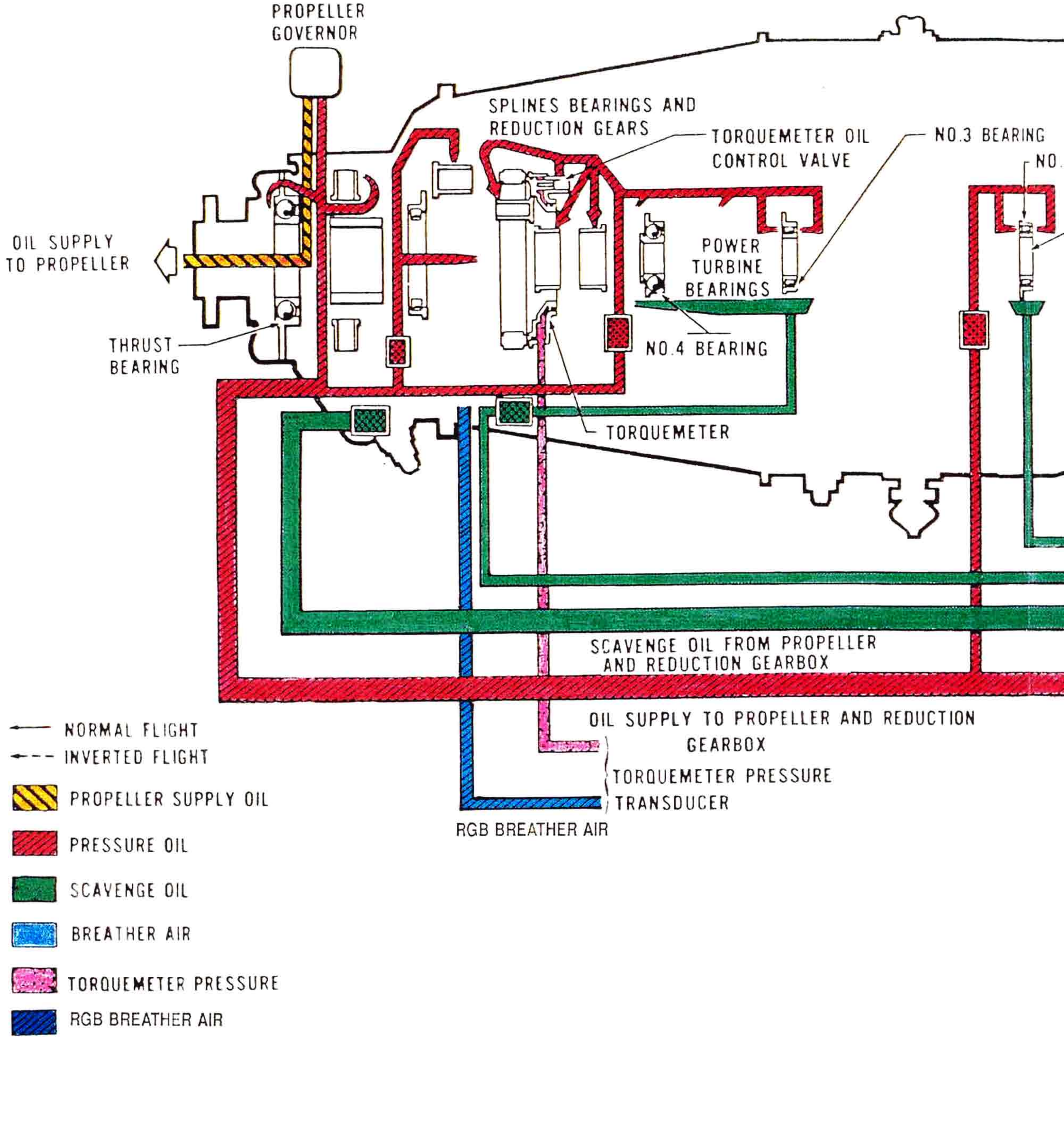







# PC9/A PROPELLER SYSTEM SCHEMATIC

# PT6A-62 ENGINE L



# NE LUBRICATION SCHEMATIC

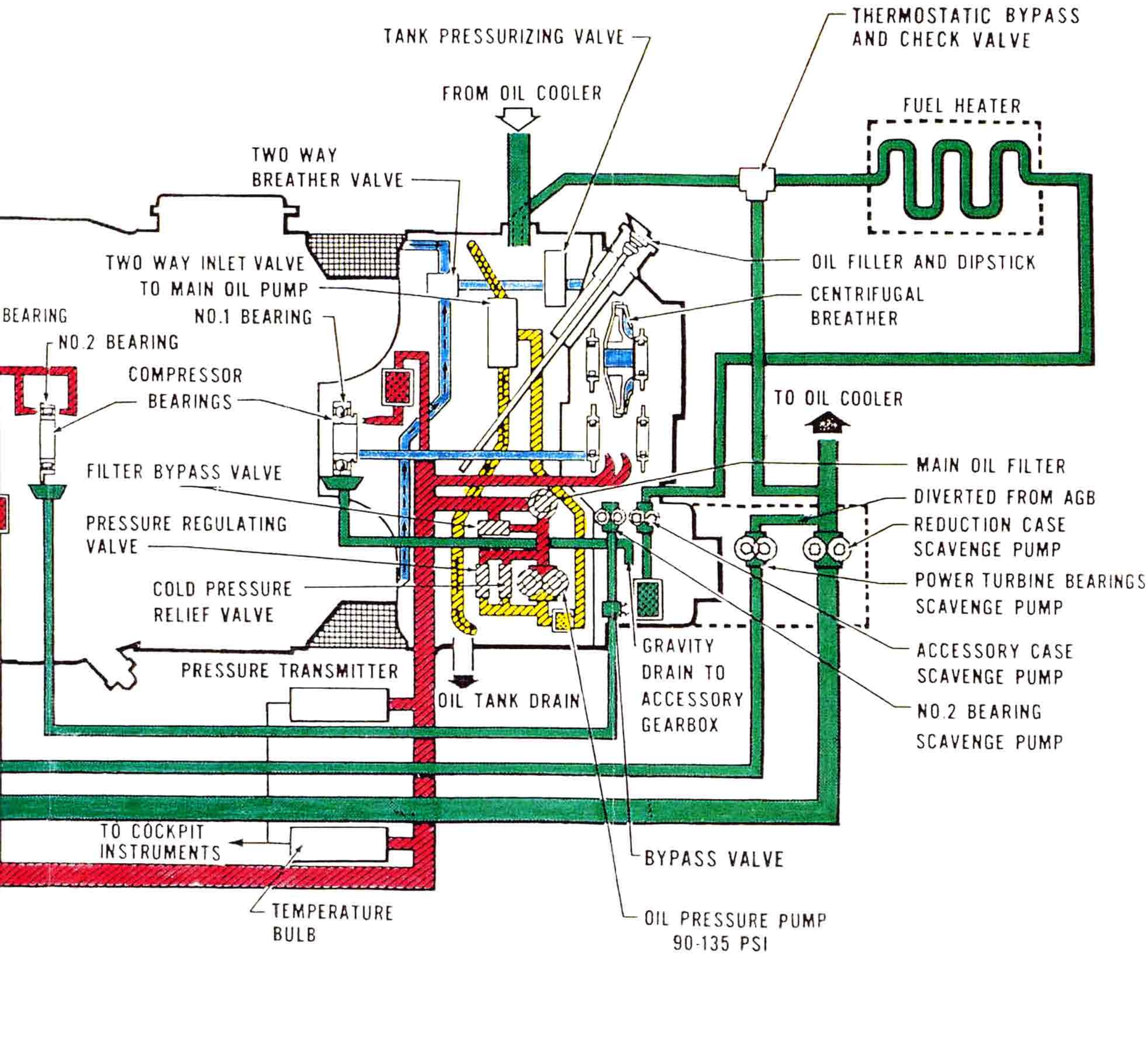
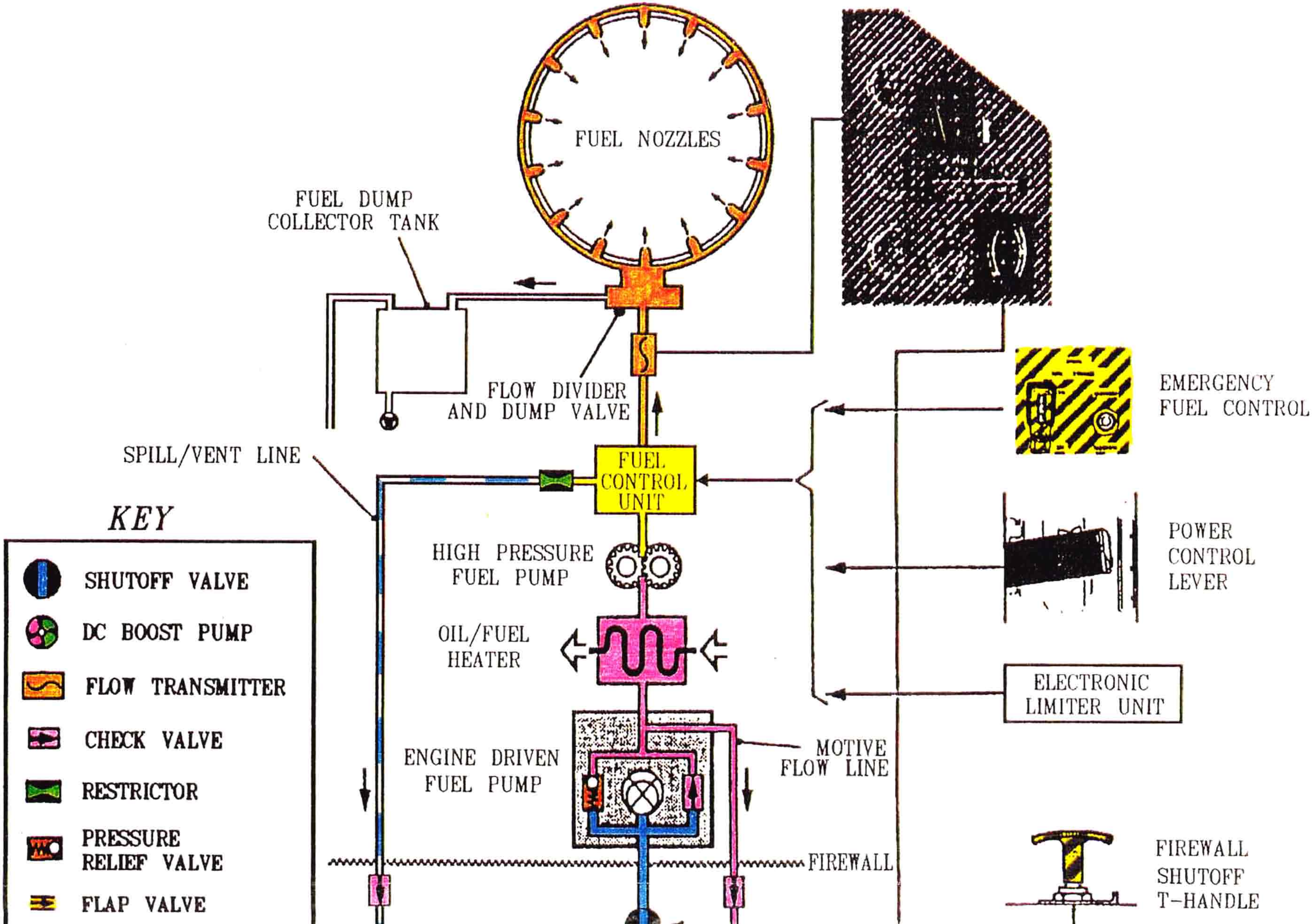



Fig FO 1–6

Figure FO 1–6


FUEL NOZZLES
FUEL DUMP
COLLECTOR TANK
FLOW DIVIDER
AND DUMP VALVE
SPILL/VENT LINE
FUEL
CONTROL
UNIT
HIGH PRESSURE
FUEL PUMP
OIL/FUEL
HEATER
ENGINE DRIVEN
FUEL PUMP
MOTIVE
FLOW LINE
FIREWALL
EMERGENCY
FUEL CONTROL
POWER
CONTROL
LEVER
ELECTRONIC
LIMITER UNIT
FIREWALL
SHUTOFF
T-HANDLE
KEY
SHUTOFF VALVE
DC BOOST PUMP
FLOW TRANSMITTER
CHECK VALVE
RESTRICTOR
PRESSURE
RELIEF VALVE
FLAP VALVE

FUEL NOZZLES

FUEL DUMP
COLLECTOR TANK

FLOW DIVIDER
AND DUMP VALVE

SPILL/VENT LINE

FUEL
CONTROL
UNIT

HIGH PRESSURE
FUEL PUMP

OIL/FUEL
HEATER

ENGINE DRIVEN
FUEL PUMP

MOTIVE
FLOW LINE

FIREWALL

EMERGENCY
FUEL CONTROL

POWER
CONTROL
LEVER

ELECTRONIC
LIMITER UNIT

FIREWALL
SHUTOFF
T-HANDLE

*KEY*

- SHUTOFF VALVE
- DC BOOST PUMP
- FLOW TRANSMITTER
- CHECK VALVE
- RESTRICTOR
- PRESSURE RELIEF VALVE
- FLAP VALVE

PRESSURE RELIEF VALVE

FLAP VALVE

DILERVERY JET PUMP

TRANSFER JET PUMP

LOW PRESSURE SWITCH

DRAIN VALVE

T FUEL LEVEL TRANSMITTER

F FUEL FILLER CAP

FIREWALL

FIREWALL SHUTOFF T-HANDLE

BOOST PUMP SWITCHES

AEROBATIC TANK

FILTER BYPASS

FUEL FILTER

FUEL PX

ANNUNCIATOR PANEL

L FUEL L

L FUEL P

R FUEL L

R FUEL P

ANNUNCIATOR PANEL

AIR SPACE

LEFT WING MAIN TANK

INWARD VENT LINE

OUTWARD VENT LINE

COLLECTOR TANK

CROSS VENT

COLLECTOR TANK

AIR SPACE

RIGHT WING MAIN TANK

INWARD VENT LINE

OUTWARD VENT LINE

# *PC9/A AIRCRAFT – FUEL SYSTEM SCHEMATIC*

# AC/DC ELECTRICAL SYSTEM SCHEMATIC

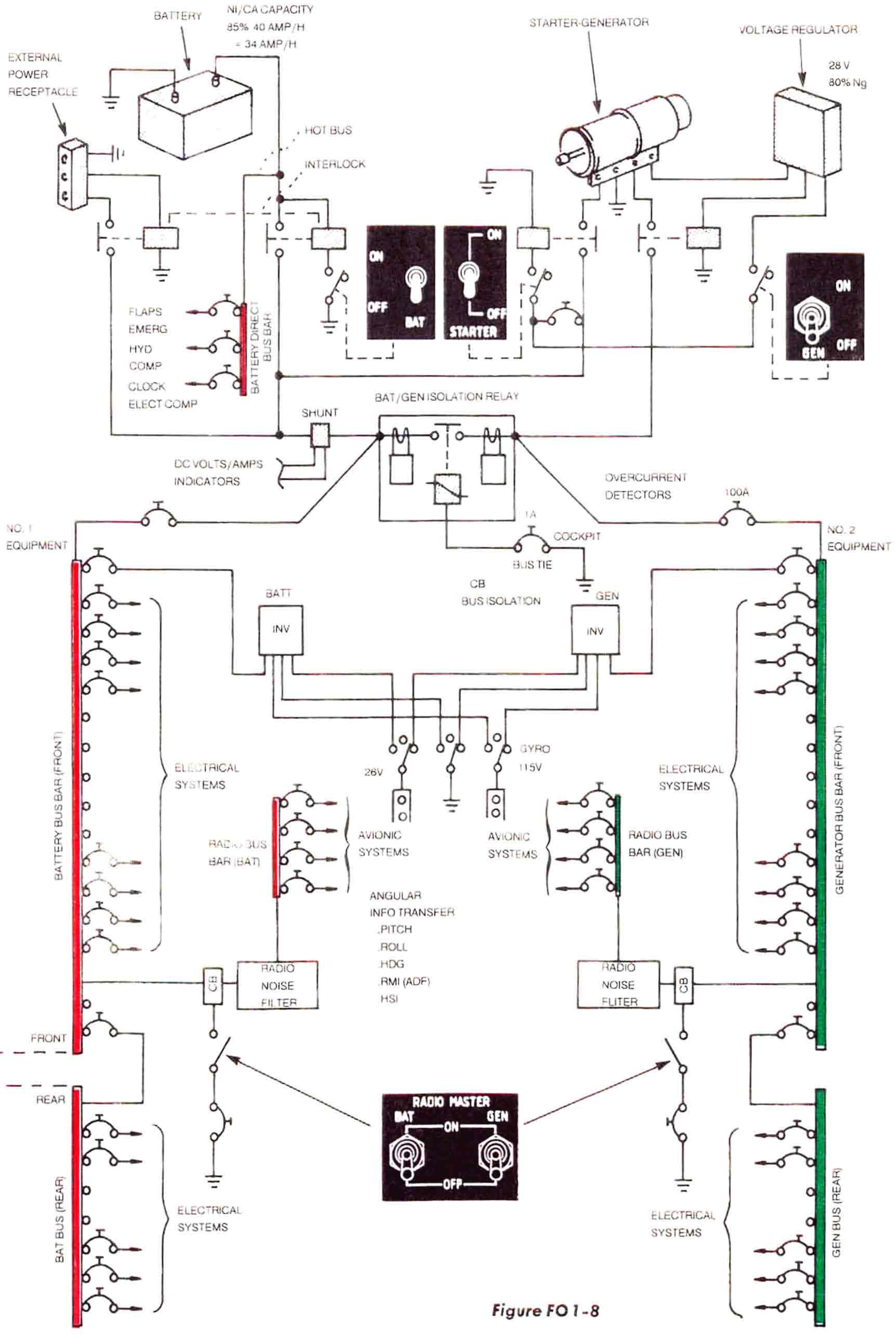



Figure FO 1-8

Fig FO 1-8

# PC9/A — Electronic Li

# ic Limiting System

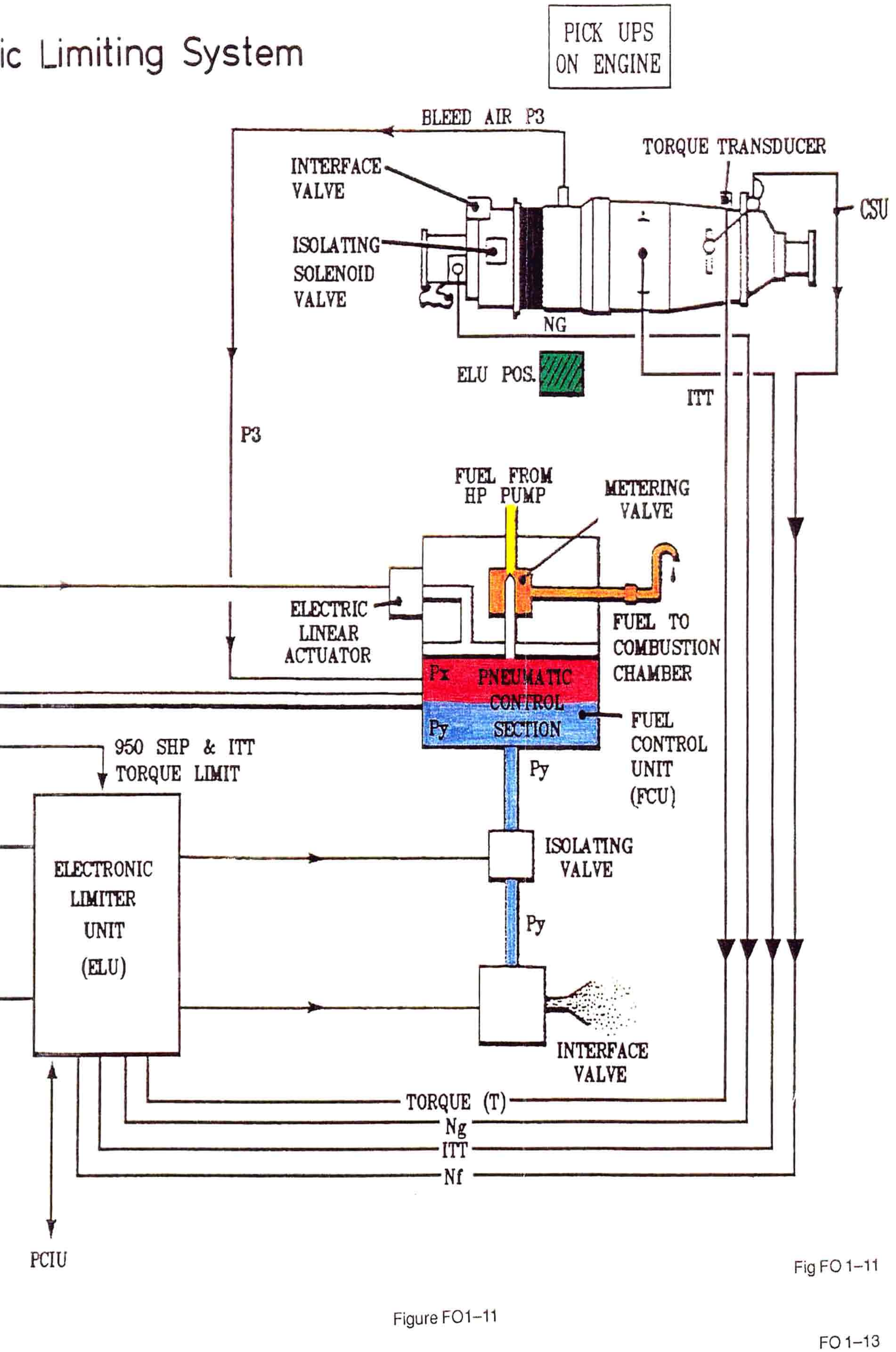
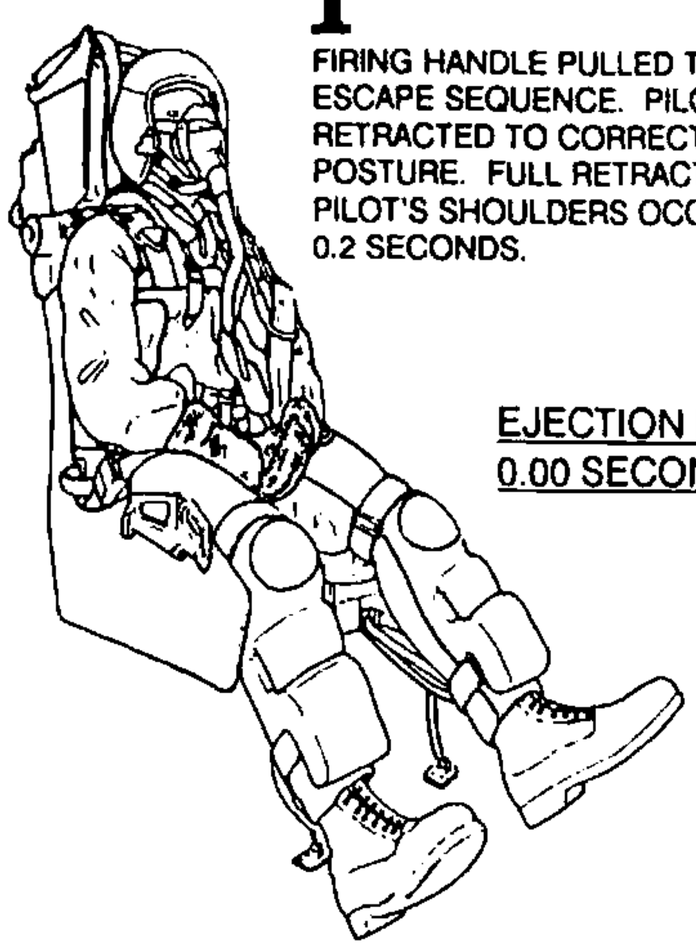



Fig FO 1–11

Figure FO1–11

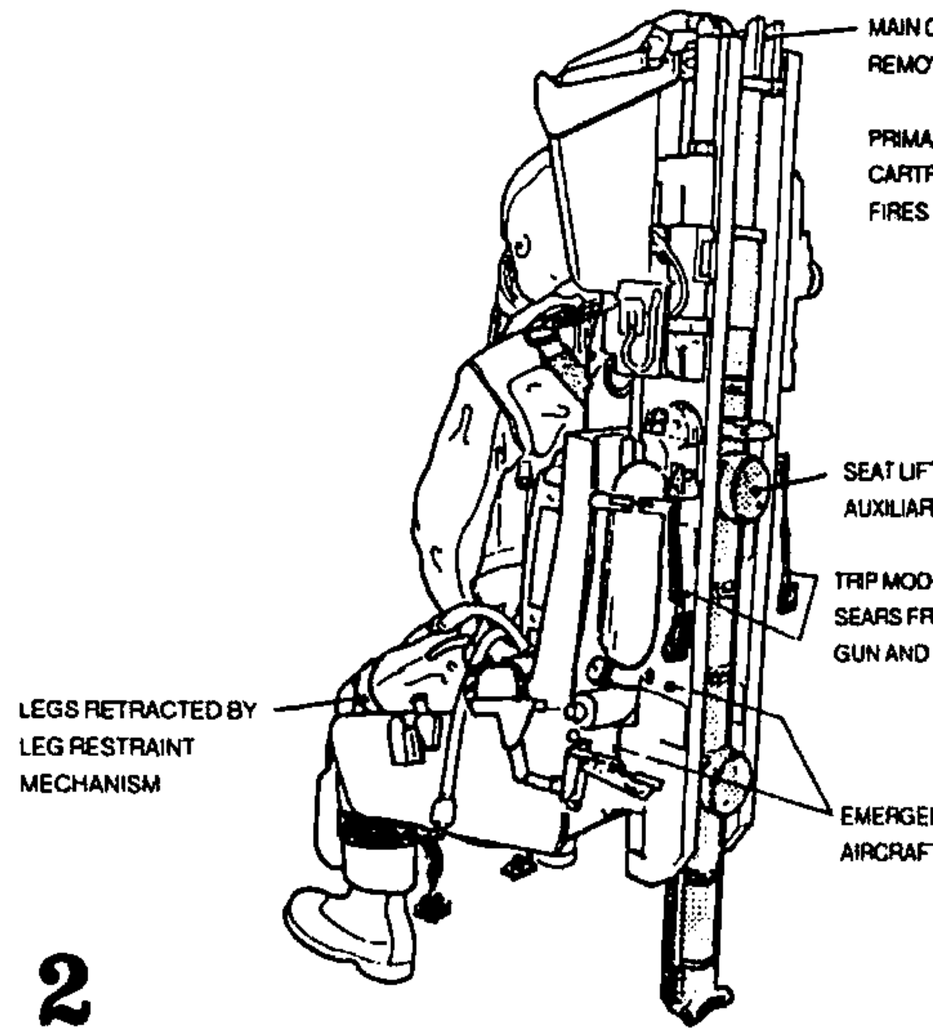


**1**

FIRING HANDLE PULLED TO INITIATE ESCAPE SEQUENCE. PILOT'S BODY RETRACTED TO CORRECT EJECTION POSTURE. FULL RETRACTION OF THE PILOT'S SHOULDERS OCCURS WITHIN 0.2 SECONDS.

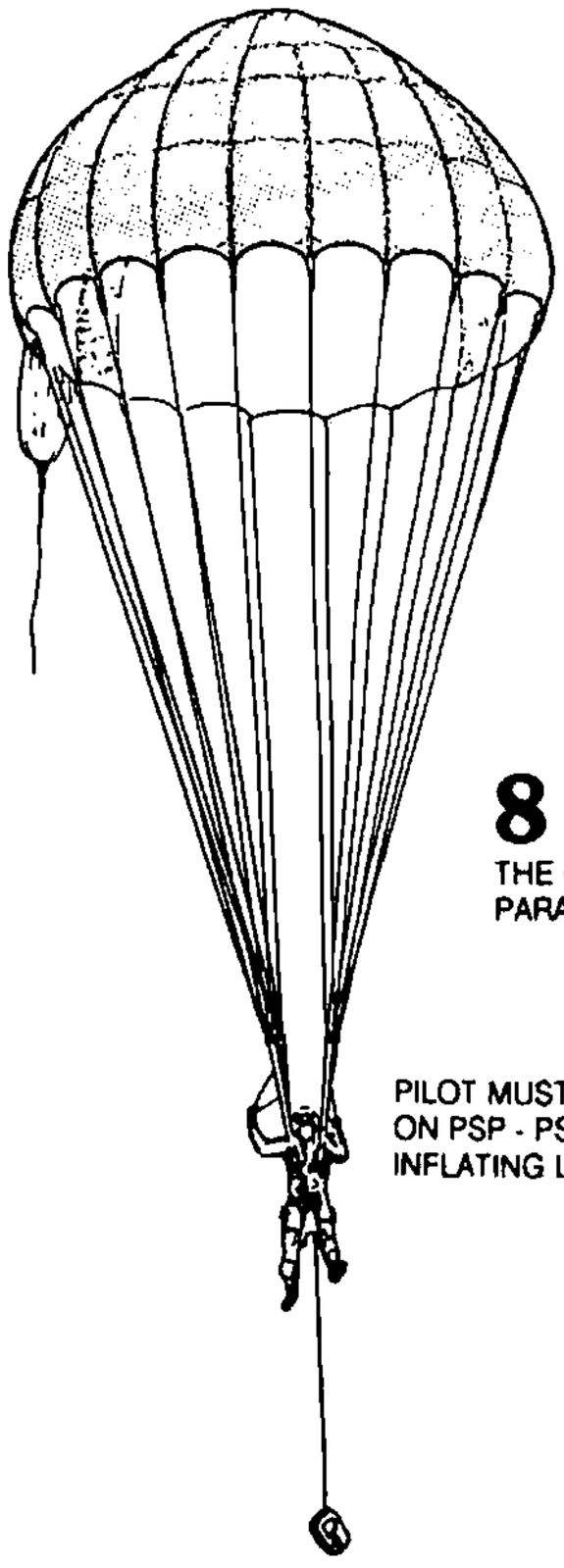
EJECTION INITIATION
0.00 SECONDS

NOTE: THE CANOPY IS PENETRATED
GUN ON THE EJECTING SEAT.




**2**

EJECTION SEAT FIRES, UNLOCKING SEAT FROM AIRCRAFT AND EJECTING SEAT. SEAT MOVEMENT OPERATES LEG RESTRAINT AND ARM RESTRAINT (IF FITTED), DISCONNECTS PERSONAL SERVICES, ACTUATES EMERGENCY OXYGEN AND TRIPS DROGUE GUN AND TIME RELEASE MECHANISM.



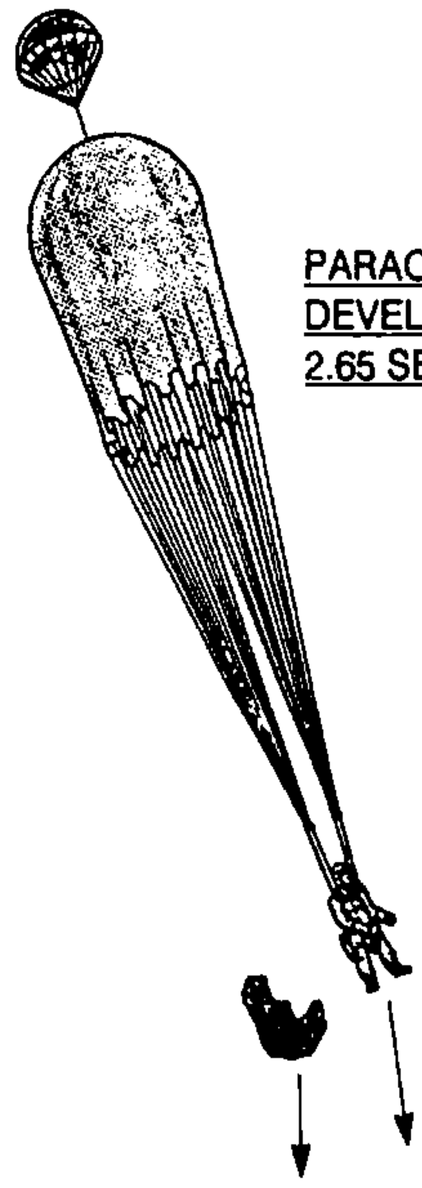
**8**

THE OCCUPANT MAKES A NORMAL PARACHUTE DESCENT.

PILOT MUST MANUALLY RELEASE ONE SIDE ONLY ON PSP - PSP WILL FALL ON LOWERING LINE - AUTO INFLATING LIFE RAFT.

**7**

THE DROGUE DEPLOYS THE PARACHUTE AND THE OCCUPANT IS LIFTED OUT OF THE SEAT, ALLOWING THE SEAT TO FALL FREE.



PARACHUTE FULLY
DEVELOPED
2.65 SECONDS

# EJECTION SEQUENCE

PENETRATED BY
SEAT.

MAIN GUN SEAR
REMOVED

PRIMARY
CARTRIDGE
FIRES

SEAT LIFTS, FIRING
AUXILIARY CARTRIDGE

TRIP MODE, WITHRAW
SEARS FROM DROGUE
GUN AND TIME RELEASE

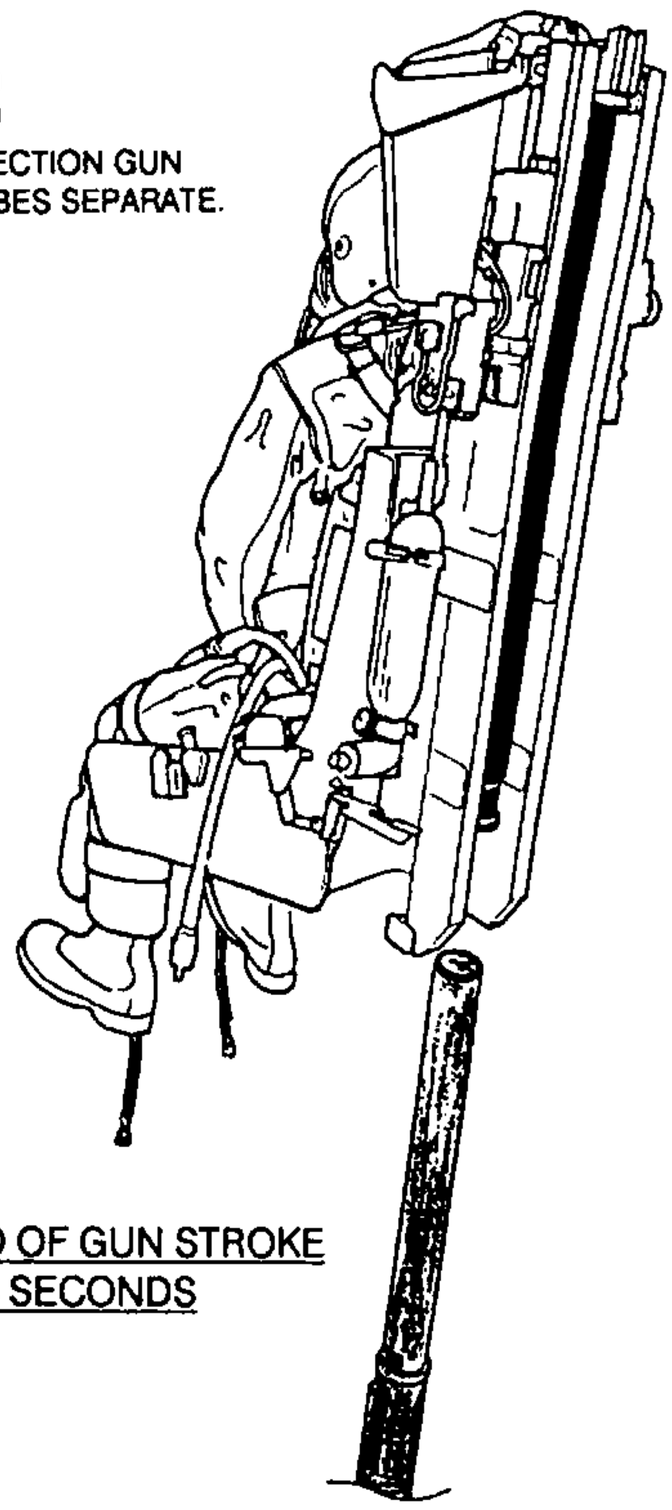
EMERGENCY OXYGEN TRIPPED.
AIRCRAFT SUPPLIES DISCONNECT

NT
T
S,
ROGUE

CCUPANT
O

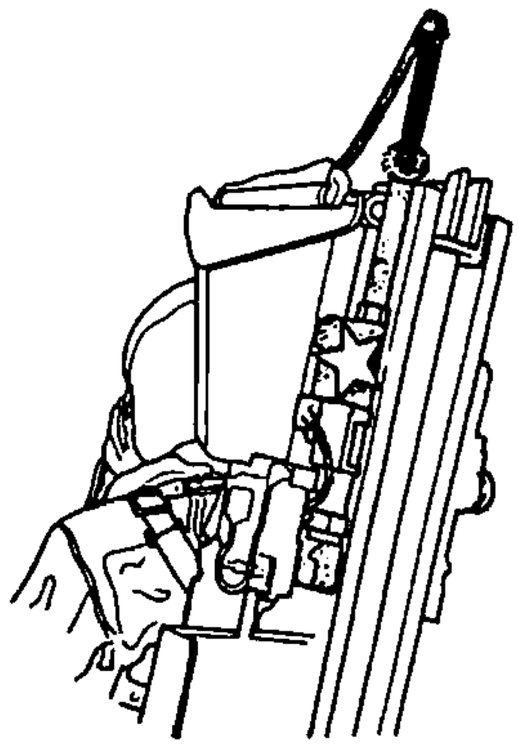
LY

**3**
EJECTION GUN
TUBES SEPARATE.



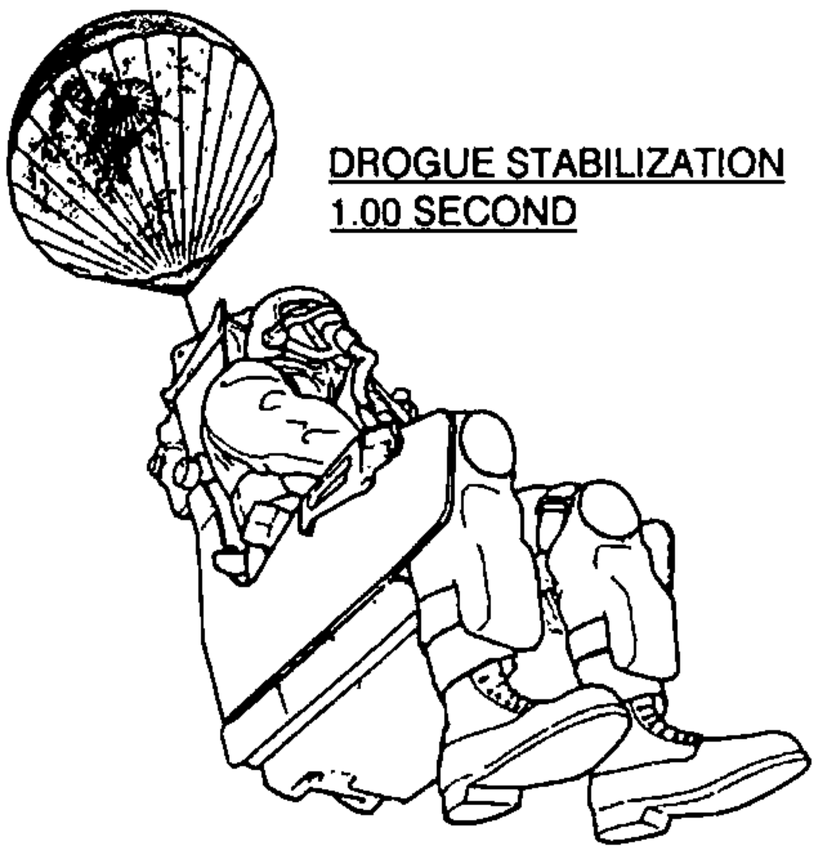
END OF GUN STROKE
0.25 SECONDS

**4**
HALF A SECOND AFTER INITIAL SEAT
MOVEMENT, THE DROGUE GUN FIRES
AND DEPLOYS THE DROGUE.



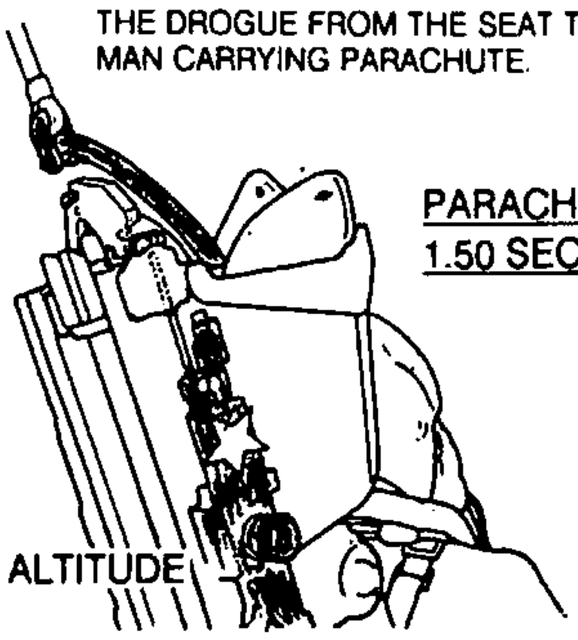
DROGUE GUN FIRING
0.50 SECONDS

**5**
THE DROGUE DEVELOPS TO STABILIZE
AND RETARD THE SEAT



DROGUE STABILIZATION
1.00 SECOND

**6**
ONE AND A HALF SECONDS AFTER INITIAL SEAT
MOVEMENT OR AT BAROSTATIC ALTITUDE, THE
TIME RELEASE MECHANISM OPERATES TO RELEASE
THE PILOT'S HARNESS, AND LEG RESTRAINT
SYSTEM FROM THE SEAT. THE SHACKLE RELEASES
THE DROGUE FROM THE SEAT TO STREAM THE
MAN CARRYING PARACHUTE.



PARACHUTE DEPLOYMENT
1.50 SECONDS

HIGH ALTITUDE

THE TIME RELEASE IS FITTED WITH A BAROSTAT TO AUTOMATICALLY DEPLOY THE PERSONNEL PARACHUTE AT A PRESET HEIGHT. THE BASIC SETTINGS AVAILABLE ARE 10,000 - 13,000 FEET, 11,500 - 14,500 FEET, 5,000 METRES AND 6,000 METRES. SHOULD EJECTION OCCUR ABOVE THE PRESET ALTITUDE, THE PILOT WILL DESCEND IN THE SEAT STABILIZED BY THE DROGUE PARACHUTES AND BREATHING OXYGEN FROM THE SEAT MOUNTED SUPPLY UNTIL THE PARACHUTE IS AUTOMATICALLY OPENED. THE PILOT THEN MAKES A NORMAL PARACHUTE DESCENT.

*Fig FO 1-10*

***Figure FO 1-10***

SECTION 2

# NORMAL PROCEDURES

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| Flight Planning | 2-1 |
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SECTION 2

# NORMAL PROCEDURES

## PREPARATION FOR FLIGHT

### Flight Restrictions

Refer to Section 5 for aircraft operating limitations.

### Flight Planning

Refer to Appendix 1 to determine take-off, cruise control, fuel planning, approach and landing data necessary to complete the sortie.

### Weight and Balance

Refer to Section 5 for weight and balance limitations. For loading information, refer to Appendix 1.

### Checklists

This section includes all procedures that are necessary for the operation of the aircraft, presented in an amplified format. The abbreviated checklists, available in a separate publication, are designed for use by the flight crew in-flight. The separate checklist publication is to be carried by flight crew on every sortie.

## PRE-FLIGHT

### BEFORE EXTERIOR INSPECTION CHECKS

1. Form EE500 - Check and Sign.
2. Seat firing handle - Safety pin in.
3. BAT MASTER - OFF (front and rear cockpits).
4. STARTER - Guarded OFF (front and rear cockpits).
5. IGNITION - Guarded OFF (front and rear cockpits).
6. LG handle - DOWN (front and rear cockpits).
7. PCL - OFF.
8. Command ejection - OFF (rear cockpit).
9. Check for loose articles.
10. Unlock flight controls if conditions permit.

### EXTERIOR INSPECTION CHECKS

Systematically check the aircraft for signs of damage and leaks, and for the security of panels and doors. General items to check include:

1. Damage on all surfaces.
2. Leaks of fuel, oil, hydraulic or other fluids.
3. Security of access doors, panels, covers, filler caps and antennae.
4. Removal of ground safety guards, covers and earthing leads.

Specifically check:

1. Left Wing
   - a. Left aileron - Free movement
   - b. Navigation lights - Serviceable.
   - c. Pitot tube - Cover removed, holes clear.
   - d. AOA probe - Cover removed, vane free.
   - e. Fuel contents filler cap - Secure.
   - f. Maintenance fuel shut-off panel (LB2) - Secure
2. Left Main Landing Gear
   - a. LG oleo extension - Min 85 mm.
   - b. Tyre - No excessive wear, correct inflation, no damage.
   - c. LG wheel brake hose - No damage or leaks.
   - d. Heavy LG indicator - No damage.
3. Nose.
   - a. Oil cooler intake - Blank Removed, clear.
   - b. Exhaust stubs - Covers removed.
   - c. Engine air intake - Blank removed, clear.

**d.** Inertial separator - Flap up, door closed.

**e.** Propeller - Restraint removed, Condition.

**f.** ECS air intake and exhaust - Blanks removed.

**4.** Nose Landing Gear

**a.** Nose leg - Restraint collar removed.

**b.** Tyre - No excessive wear, correct inflation and no damage.

**c.** LG oleo extension - Min 90 mm.

**5.** Right Main Landing Gear

**a.** LG oleo extension - Min 85 mm.

**b.** Tyre - No excessive wear, correct inflation and no damage.

**c.** LG wheel brake hose - No damage or leaks.

**d.** Heavy LG indicator - No damage.

**6.** Right wing

**a.** OAT bulb - No damage.

**b.** Fuel content filler cap - Secure.

**c.** Navigation lights - serviceable.

**d.** Right aileron - Free movement.

**7.** Right Fuselage

**a.** Static Port - Clear.

**8.** Tail Assembly

**a.** Elevator and rudder - Free movement elevator control rod connection secure.

**9.** Left Fuselage

**a.** Baggage compartment - Contents and door secure.

**b.** Static port - Clear.

**c.** Oxygen overpressure indicator - Green.

## REAR COCKPIT SOLO CHECKS

**1.** Harness - Secure. Apron fitted.

**2.** Command ejection - OFF.

**3.** Oxygen regulator - OFF, 100%.

**4.** Oxygen hose - Secure.

**5.** Intercom lead - Secure.

**6.** Anti G valve - Outlet CLOSED.

**7.** Electronics/Avionics - All off.

**8.** All switches - Off/guarded.

**9.** Air distribution lever - Fully up.

**10.** Air louvres - Closed.

**11.** Map cases - Empty and closed.

**12.** Wander light - Secure.

**13.** Canopy breaker knife - Secure.

**14.** C/Bs - SEC ATTD, ADI, HSI C/Bs pulled, all others in.

**15.** Check for loose articles.

## BEFORE ENTERING COCKPIT CHECKS

**WARNING**

Do not stand on the ejection seat firing handle.

**1.** Sear firing handle - Safety pin in.

**2.** Seat pan firing breech - Safety pin removed.

**3.** Leg Restraint Lines - Secure.

**4.** 4 pin Servicing Streamer - Removed and stowed.

**5.** Oxygen - EO handle not pulled, contents in white sector.

**6.** PSP lug - Secure in sticker clip.

**7.** Drogue gun trip rod - Connected, safety pin removed.

**8.** Top latch - Flush.

9. Drogue withdrawal line - Piston secure, drogue line connected.

10. Ejection gun gas initiator - Quick release pin and spring loaded latch correctly engaged.

11. BTRU trip rod - Connected, safety pin removed.

12. Command firing connectors - Secure.

13. Main oxygen quick disconnect - Connected.

14. MOR handle - Safety pin removed.

15. Harness - All attachments secure.

16. Single handed release strap - Correctly routed and connected.

**Rear Cockpit:**

17. Command Ejection - Off.

## BEFORE START CHECKS

1. EPU - ON if available.

2. BAT MASTER - ON (or OFF with external power), check volts.

**WARNING**

Ensure the MOR linkages and QRF are free of obstruction prior to seat height adjustment. Loose objects may jam and bend the MOR linkages possibly causing the Drogue to fire. The QRF may foul the Seat Firing Unit Sear, possibly causing seat ejection.

3. Seat height - Adjust.

4. Rudder pedals - Adjust.

5. Strap in.

6. Oxygen - ON, Check contents, TEST, Set NORMAL.

7. Audio integration panel - Rotary selector position 1. Set VOL.

8. FIREWALL SHUT-OFF handle - Down.

9. BATTERY BUSBAR C/Bs - C/Bs in.

10. Map Case - Secure.

11. LIGHTS and SYSTEM TEST - Check LG lights, CWS and ESDP.

**CAUTION**

A single SICU failure will have no effect on information displayed. A second failure could result in total failure of all indications on front and rear ESDPs.

**NOTE**

If Engine Caution flashes, record failure code on FUEL QTY gauge at end of test and place the aircraft unserviceable.

12. AOA test - HIGH, LOW and ON SPEED.

13. PCL - IDLE. Check boost pumps run. Check movement, adjust friction, set off. Reset BOOST PUMPS.

14. FLAPS - UP.

15. ELS ISOLATE/EMERG FUEL - Guarded OFF

16. ELT - Guarded ARM.

17. PROBES - OFF.

18. TRIM INT RPT - Guarded NORM.

19. LDG LIGHT - OFF.

20. INVERTER - GEN, check INV caption extinguished.

21. AVIONICS -

    a. BAT - OFF.

    b. GEN - OFF.

22. Trim Gauges - Conditions, Trims functioning, set for Take-off.

23. Clock - Set.

24. LG handle - DOWN, 3 greens.

25. EMER LDG GR Handle - IN, safety wired.

26. EMER ALL JTSN - Guarded, safety wired.

27. Standby AI - Uncaged, erect, set pitch bar.

28. Accelerometer - Note readings, reset.

29. Master CAUTION - Press to reset.

30. Flight instruments - Condition.

31. Engine and Secondary instruments - Condition.

32. FUEL - Contents, agrees with analogue gauge, balanced.

33. FUEL USED - Reset if required.

34. N2 HYD pressure - Check.

35. PARKING BRAKE - Set, brakes positive feel.

36. Chocks - Remove.

37. GEN MASTER - OFF.

38. BOOST PUMPS - ARM.

39. EXT FUEL TRANSFER PUMPS - OFF.

40. INSTR LIGHTS - As required.

41. NAV light - OFF.

42. STROBE/BCN lights - OFF.

43. ECS - OFF.

44. GENERATOR BUSBAR C/Bs - C/Bs in.

45. Wander light - Secure.

46. Map case - Secure.

47. Canopy breaker knife - Secure.

48. Standby Compass - Condition.

49. INTAKE ICE handle - In.

50. Air distribution lever - As required.

51. RAM AIR - CLOSED.

52. Controls - Full, Free and correct movement.

**NOTE**

With the IF hood fitted the rudder and elevator movements cannot be seen unless the canopy is open.

# STARTING

## ENGINE STARTING CHECKS

**CAUTION**

The area around the aircraft must be checked clear of personnel and equipment prior to starting the engine. The ground-run danger areas are shown at Figure 2-1.

**1.** Canopy - Closed, locked or latched.

**2.** Propeller area - Clear.

**3.** NAV light - ON.

**4.** STROBE/BCN lights - BCN.

**5.** BOOST PUMPS - ON.

**NOTE**

- Minimum battery voltage for start is 24 VDC.
- If ELU CWS caption illuminates during start, continue with the Start Checks.

**6.** STARTER - ON.

**7.** ENG OIL - Confirm pressure rising.

**8.** IGNITION - ON, check aurally.

At stabilised NG (minimum 12%):

**9.** PCL - IDLE.

**10.** ITT - Monitor, check light up within 10 secs of PCL to IDLE.

**NOTE**

ITT start limits are:

| | | | |
|---|---|---|---|
| **a.** | Normal | - | 800°C max |
| **b.** | Transient for | - | 800°C to 870° max less than 20 secs |
| | | - | 870°C to 1000°C max for less than 5 secs. |
| **c.** | Absolute | - | 1000°C max. |

If ITT exceeds the above limits or rapidly approaches the absolute limit proceed as follows:

**11.** PCL - OFF.

**12.** IGNition - OFF.

**13.** Starter - ON until ITT 800°C, or for a max of 10 secs, then OFF.

**14.** Boost Pumps - ARM.

**15.** Place the aircraft U/S.

**When NG stabilised at or above 56%:**

**16.** IGNITION - OFF.

**17.** STARTER - OFF.

**18.** BOOST PUMPS - ARM.

**19.** ENG OIL pressure and temperature - Green arcs.

**20.** Propeller - Unfeathered.

If engine does not light up within 10 seconds of moving PCL to idle, or if NG stabilises below 45%:

**21.** PCL - OFF.

**22.** IGNITION - OFF.

**23.** STARTER - OFF.

**24.** BOOST PUMPS - ARM.

**25.** Carry out an Engine Dry Run.

# GROUND-RUN DANGER AREAS

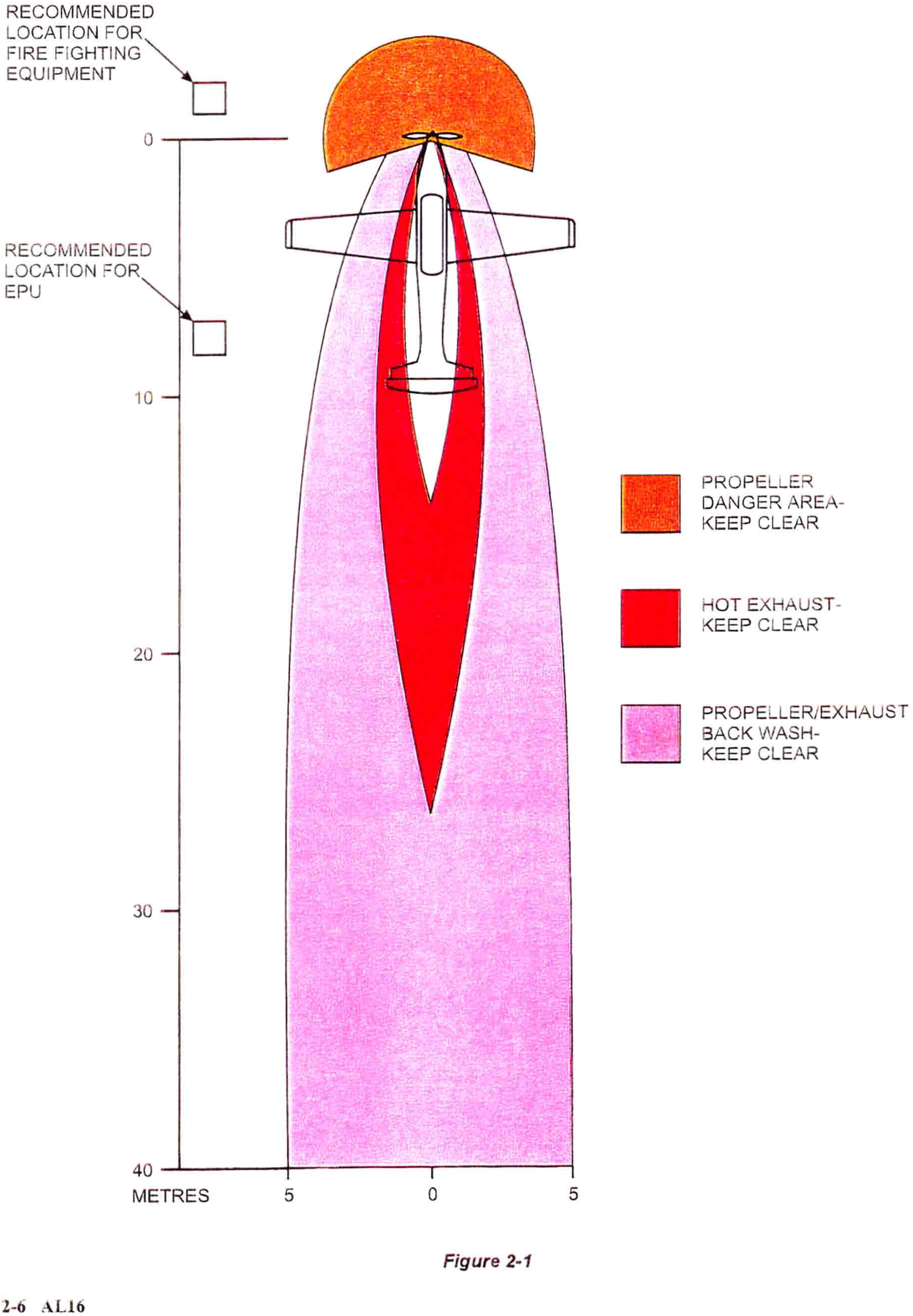



*Figure 2-1*

## ENGINE DRY RUN CHECKS

Allow 30 secs for fuel draining, then:

1. PCL - OFF.
2. IGNITION - OFF.
3. BOOST PUMPS - ON.
4. STARTER - ON for 15 secs then OFF.
5. BOOST PUMPS - ARM.
6. After 1 minute cooling period repeat a normal engine start sequence.

**CAUTION**

After 3 cycles of the starter motor, allow a 30 minute cooling period.

## AFTER START CHECKS

1. EPU - Disconnect.
2. BAT MASTER - ON.
3. CWS - GEN CWS caption only.

**NOTE**

If the ELU CWS caption has illuminated, pull and reset the ELU/TQ C/B. If the C/B will not reset, or the ELU CWS caption remains illuminated, place the aircraft unserviceable.

4. GEN MASTER - ON, GEN caption out, check volts and positive charge.

   If first flight of the day:

   GEN TEST - Overvolt - Reset
   - Overload - Reset.

5. INVERTER - BAT.
6. AVIONICS
   a. BAT - ON.
   b. GEN - ON.
7. ECS - LOW.
8. NG - Minimum 56%.
9. N2 HYD - Green sector.
10. FLAPS - Check function, set UP.
11. AIR BRAKE - Check function, select in.
12. Flight Instruments - Erect, OFF flags away, AI and Compass comparison, EFIS intensity adjusted.

**CAUTION**

Do not taxi until the AHRS is fully erected as indicated by the OFF Flags clearing from the ADI, HSI and RMI.

13. Radio and nav aids - Set as required.
14. Altimeter - set QNH (reading ± 60 ft of actual elevation).
15. CWS - No Red or Amber captions.
16. Seat firing handle - Safety pin remove and stow.

# TAXI

**CAUTION**

- Stabilised operation of the propeller between 1200 and 1700 NP on the ground is prohibited.
- Maintain a minimum distance of 50 metres from preceding aircraft to prevent foreign object damage.

## BRAKING TECHNIQUE WHILE TAXIING

During extended taxiing, the recommended technique is to allow the aircraft to reach a faster than normal taxi speed. Control is exercised by the periodic application of smooth braking to bring the aircraft almost to rest before releasing the brakes completely. In crosswind conditions NWS should be used in preference to differential braking to prevent the aircraft weather-cocking.

**CAUTION**

Continuous brake application may lead to tyre deflation due to activation of the wheel fusible plugs.

## TAXI CHECKS

1. PARKING BRK - OFF.

**NOTE**

Depending upon the ambient temperature, ground slope and nature of the ground surface, the aircraft may commence to move immediately. A small amount of additional torque may be required, however, once moving, idle power is usually sufficient to taxi along level ground.

2. Brakes - Check operation.

3. Nosewheel steering - ON. Check function.

**NOTE**

The aircraft may be taxied with or without the nosewheel steering engaged. When selecting ON or OFF, the status of the nosewheel steering system should be checked by reference to the green nosewheel steering light on the instrument panel.

4. Flight Instruments - Check function.

# PRE-TAKE-OFF

## PRE-TAKE-OFF VITAL ACTIONS

1. Canopy - Closed and locked, light out.
2. Air brake - IN.
3. FLAPS - Select TO, indicating.
4. PROBES - ON, indicating.
5. Trims - Set.
6. Flight Instruments - Functional check carried out. No flags.
7. Engine Instruments - Indications in green arc.
8. Volt/amps - Normal, positive charge.
9. N2 HYD press - Indications in green arc.
10. Fuel - Contents, agrees with analogue gauge, balanced, flow normal (approx 140-170 lbs/hr).
11. CWS - No red or amber captions.
12. Oxygen - Contents, ON, NORMAL, NORMAL, flow and connections.
13. Controls - Full, and free movement.

**CAUTION**

The NWS must be OFF prior to conducting a rudder control check.

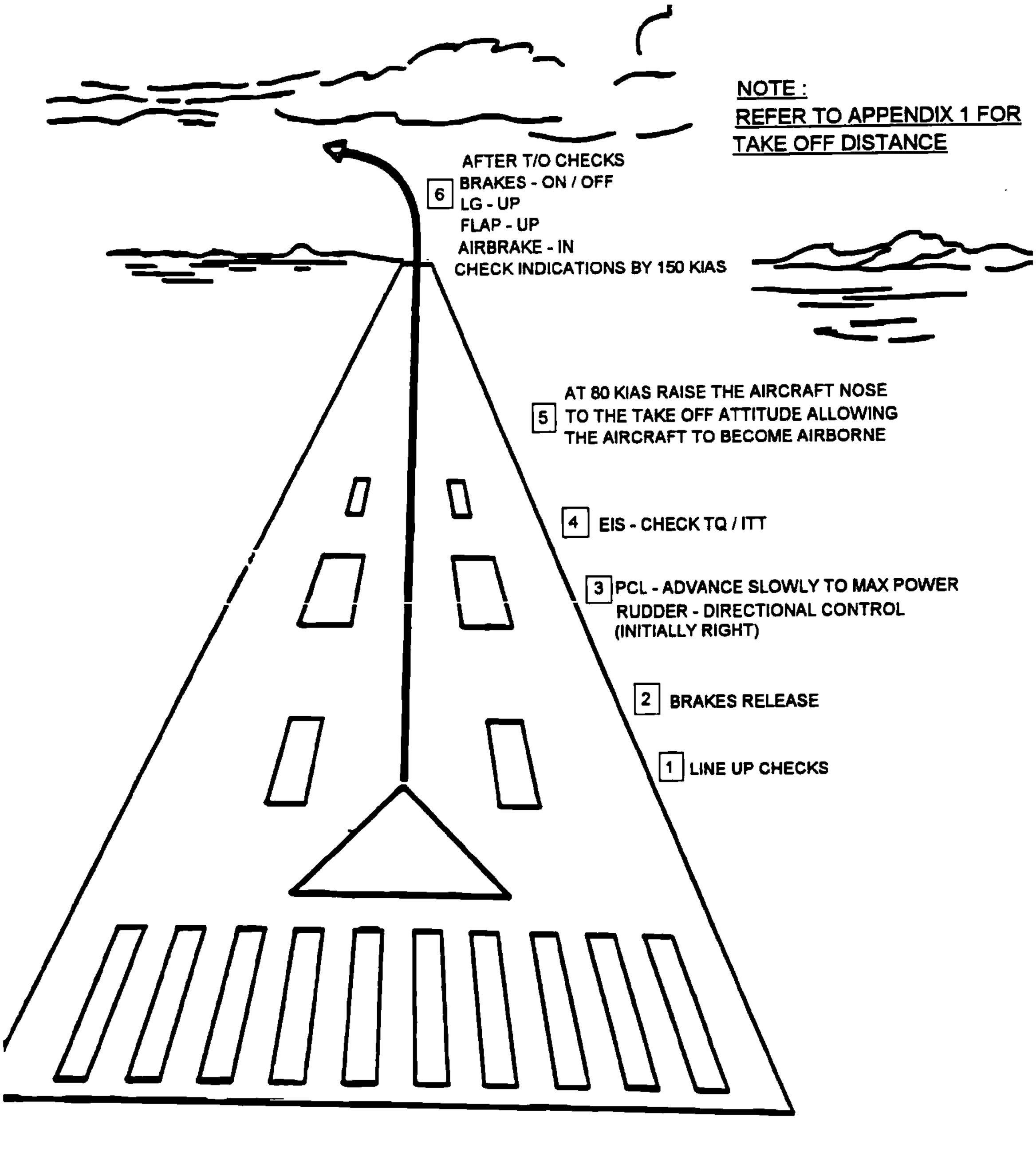
14. Harness - T-handle locked, leg restraints, PSP and G hose connected, chin strap secure, visor(s) down.
15. Seat firing handle - Safety Pin removed.
16. Command ejection - As required.
17. Emergency Briefing - Conducted.

## LINE UP CHECKS

When aligned with the runway heading check:

1. LDG LIGHT - ON.
2. Clock - Start.
3. Transponder (XPDR) - As required.
4. STROBE/BCN lights - As required.
5. AHRS - Fast erect (if required).

# NORMAL TAKE-OFF




NOTE :
REFER TO APPENDIX 1 FOR TAKE OFF DISTANCE

*Figure 2-2*

# TAKE-OFF

## NORMAL TAKE-OFF

The normal take-off profile is shown at Figure 2-2.

1. Brakes - Release.

2. PCL - MAX

3. ESDP - TQ, ITT, Ng limiting.

4. Airspeed - At 80 KIAS rotate.

**NOTE**

- During PCL advance, engine limits may be momentarily exceeded. The ELU should stabilise maximum values within approximately five seconds.

- Large transient overswings of TQ and Np with associated red or amber ESDP captions are likely to occur if the PCL is advanced to MAX while the engine oil temperature is less than 70°C.

Refer to Appendix 1 for take-off distances and speeds.

## OBSTACLE CLEARANCE TAKE-OFF

1. PCL - Maximum available for conditions.

2. ESDP - TQ, ITT, Ng limiting.

3. Brakes - Release.

4. PCL - MAX

5. Airspeed - At 77 KIAS rotate (85 KIAS at 2700 kg).

6. Landing Gear and Flap up, climb at 95 KIAS,until clear of obstacles (100 KIAS at 2700 kg).

## CROSS-WIND TAKE-OFF

During the take-off roll aileron should be used into-wind to assist with the maintenance of wings level.

**CAUTION**

- Maximum TQ versus Crosswind Component limits should be adhered to IAW Section 5.

- During crosswind conditions, the combination of run-up power, propeller wash and crosswind may cause the aircraft to lean nose low to the left, causing the nosewheel to castor uncontrollably. In these circumstances, power should be reduced and the nosewheel straightened before take-off is attempted.

**NOTE**

To prevent side loads on the main landing gear, rotation to the take-off attitude should be accomplished in a positive manner.

## AFTER TAKE-OFF CHECKS

When the aircraft is safely airborne, proceed as follows:

1. LG - UP.

2. FLAPS - UP.

3. AIR BRAKE - In.

4. Check - LG lights out, flap indicating UP and AIR BRAKE indicating IN by 150 KIAS.

At airspeeds above 130 KIAS, the landing gear may not fully retract. Proceed as follows:

5. Reduce speed to below 130 KIAS.

6. Stabilise 1 G flight.

7. Await gear retraction (all green and reds extinguished).

8. Proceed as normal.

# INFLIGHT

## CLIMB/PERIODIC CHECKS

Normal climb speed is 180 KIAS. Refer to Appendix 1 for more detailed climb data. During the climb and at periodic intervals during flight conduct the following checks:

1. Engine - Indications checked.
2. Electrics - greater than 25 VDC and no discharge.
3. N2 HYD press - checked.
4. FUEL - Contents, agrees with analogue gauge, balanced.
5. CWS - No Red or Amber captions.
6. Oxygen - Contents, ON, NORMAL, NORMAL, flow and connections.

## CRUISE

For cruise data refer to Appendix 1.

## FLIGHT CHARACTERISTICS

For information regarding the aircraft flight characteristics refer to Section 6.

## PRE-MANOEUVRE CHECKS

Before manoeuvres such as stalling, spinning or aerobatics, carry out the following checks:

1. Height - Sufficient for recovery.
2. Equipment - No loose articles.
3. Fuel symmetry - Within a quarter tank.
4. Trims - As required.

## DESCENT AND REJOIN

For descent data, refer to Appendix 1. A typical rejoin and landing profile is shown at Figure 2-3.

### Maximum Rate Descent

Entry technique for entering a maximum rate descent will depend upon flight circumstances at the time. Should a maximum rate descent be required, the following procedure is used when at or below cruise speeds and power settings:

1. PCL - Max power (or idle if above cruise power and speed).
2. Attitude - Lower to 22° nose down.

Approaching 250 KIAS:

3. PCL - IDLE.
4. Air Brake - Out.
5. Airspeed - maintain 250 KIAS.

**NOTE**

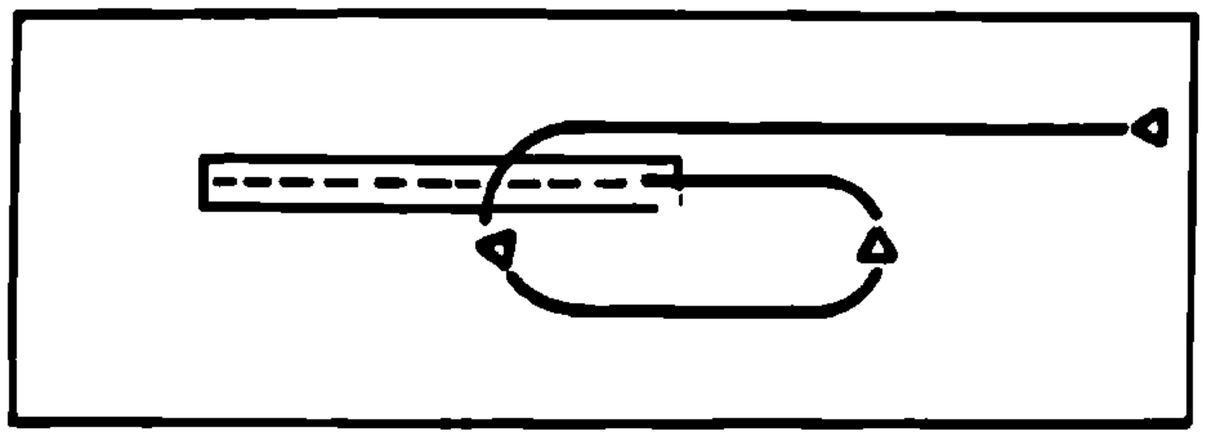
Similar descent performance (10,000 ft/min ROD) can be achieved at 300 KIAS, clean, but with higher control forces.

## REJOIN CHECKS

Before descent or before entering the circuit pattern, carry out the following checks:

1. FUEL - Contents, agrees with analogue gauge, balanced.
2. Instruments - Erect, Off Flags away, Compass comparison.
3. Radio and navigation aids - Tune and identify.
4. Altimeter - Set QNH as required.
5. ECS - LOW or High (as required).
6. Demist - As required.

# REJOIN AND LANDING PROFILE








DOWNWIND POSITION

NOTE:

1. ADJUST BANK TO ROLL OUT DOWNWIND WITH ROUNDEL ON RUNWAY.
2. FOR GUSTY SURFACE WIND INCREASE THRESHOLD SPEED BY 0.5 OF THE GUST FACTOR TO A MAX OF 10 KIAS

*Figure 2-3*

# NORMAL CIRCUIT

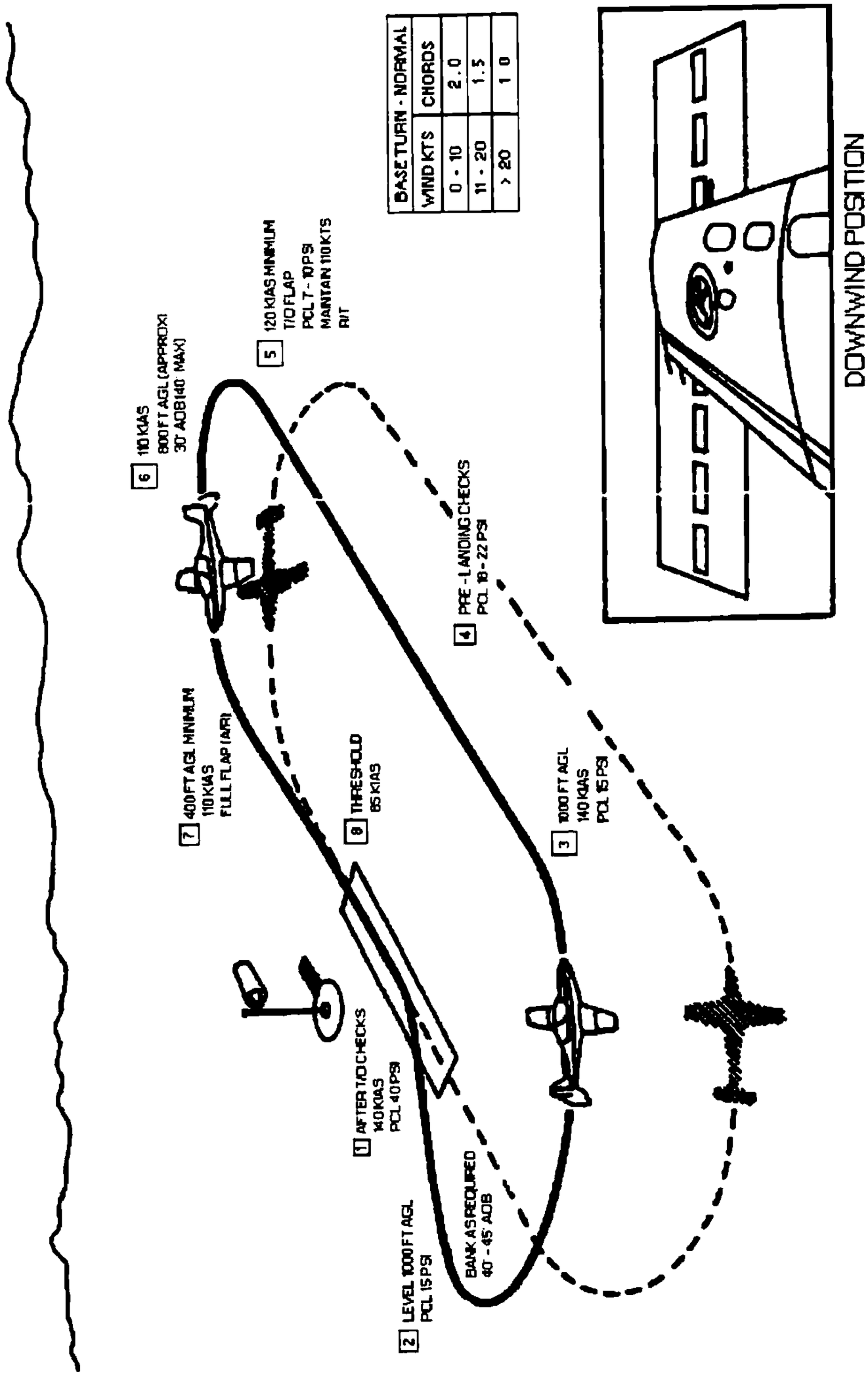
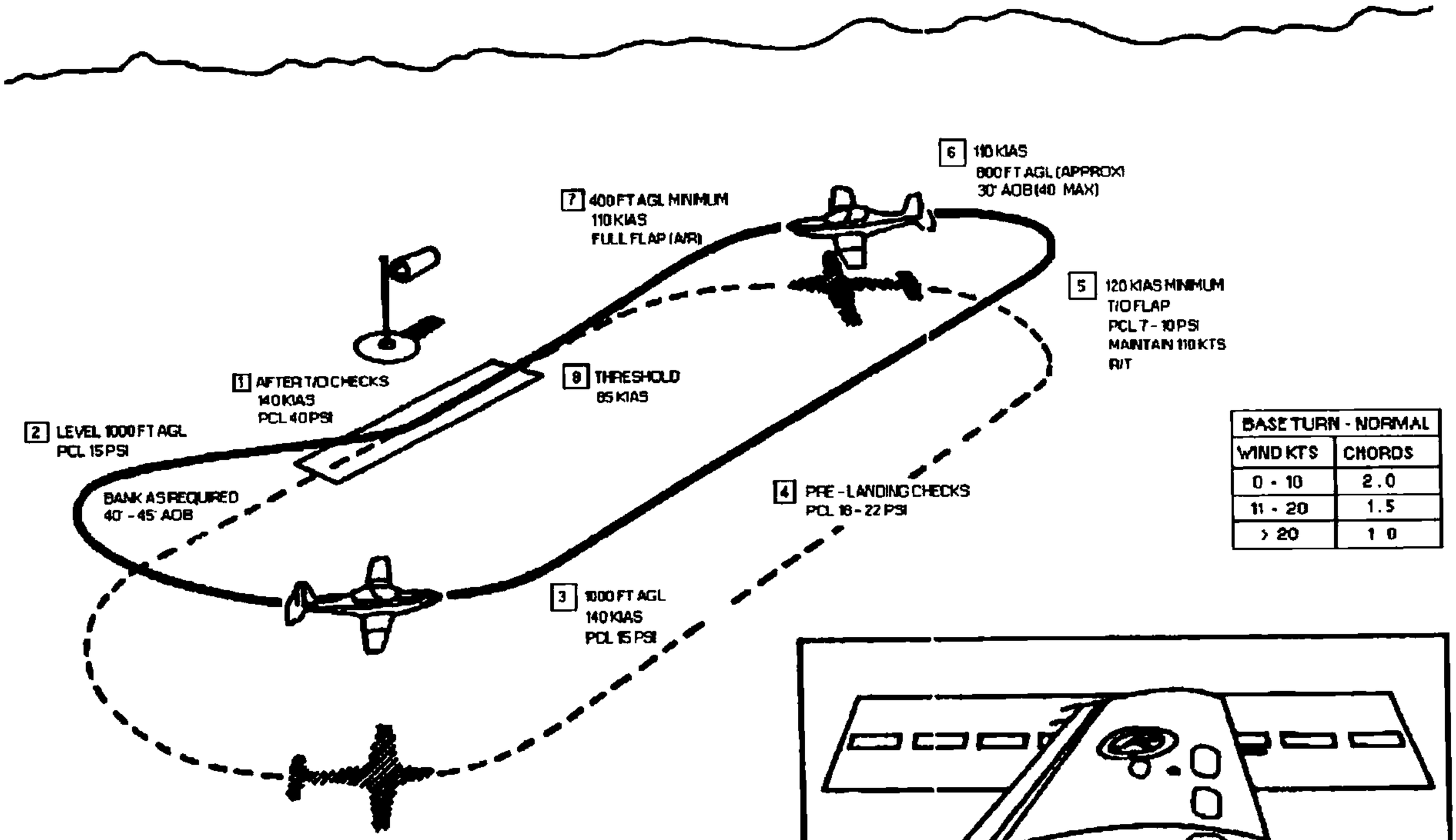



*Figure 2-4*




| BASE TURN - NORMAL | |
|---|---|
| WIND KTS | CHORDS |
| 0 - 10 | 2.0 |
| 11 - 20 | 1.5 |
| > 20 | 1.0 |



DOWNWIND POSITION

# LANDING

## PRE LANDING CHECKS

1. Airspeed - Below 150 KIAS.
2. AIR BRAKE - IN.
3. LG - DOWN, 3 greens.
4. Brakes - Positive feel.
5. FUEL - Contents, agrees with analogue gauge, balanced.
6. Threshold speed - Nominate, bug set.
7. FLAP - As required.

## THRESHOLD SPEEDS

Threshold speeds are as follows:

| Land Flap | Flapless | Glide |
|---|---|---|
| 85 KIAS | 100 KIAS | 100 KIAS |

**NOTE**

When landing in gusty surface wind conditions, increase the calculated threshold speed by half the gust factor up to 10 kts.

## NORMAL PROCEDURE

The normal landing procedures are outlined in Figure 2-4. Acknowledge landing instructions and depress the LG DOWN TONE button on the landing gear selector panel to confirm that the landing gear is down and locked. Lower FLAPS (as required) and maintain a constant approach path. At or near the runway threshold, commence a flare then slowly close the PCL to IDLE. Fly the main wheels onto the ground to achieve a positive touchdown.

## BRAKING TECHNIQUE

Braking in the PC9/A is effective but the application of only moderate pressure can be sufficient to lock the wheels. Correct braking techniques are essential to avoid blowing tyres or overheating the brakes and activating the fusible plugs in the wheels.

**CAUTION**

There is minimal feedback to warn of impending wheel lock, and tyre blow-out can occur almost instantaneously.

Following touchdown, maximise aerodynamic braking, but lower the nosewheel onto the runway before elevator effectiveness is lost. Commence braking below 60 KIAS (as required), and introduce backstick but do not allow the nose oleo to extend. Equal weight on both main wheels can be maintained by use of aileron into wind.

### General Considerations

Wheel braking converts kinetic energy to heat. Moderate to heavy braking should be applied from as low a ground speed as possible rather than light braking from high speed. Braking should be continuous rather than intermittent pulsing of the brakes.

## MAXIMUM EFFORT LANDING

Fly an accurate threshold speed (if required an AOA indexer approach may be flown). Lower the nose wheel immediately after landing and commence moderate braking. Introduce backstick, but do not allow the nose oleo to extend. When below 60 KTS groundspeed increased brake pressure may be smoothly applied. Maintain directional control by use of rudder and differential braking.

## CROSS-WIND LANDING

During final approach either the crabbing or wing down technique may be used to maintain runway centre-line. In strong or gusty crosswind conditions a combination of these techniques may be necessary, and consideration should be given to using less than LAND flap. Rudder is to be used to align the aircraft with runway centre-line during the flare prior to touchdown if necessary, and the into-wind wing lowered to prevent drift. After touchdown the nosewheel should be lowered and gentle braking commenced. During the landing roll, coordinated inputs of aileron and rudder are required to keep wings level and to maintain directional control. If difficulty is experienced keeping the wings level or the aircraft straight during the landing roll raise the flap.

**CAUTION**

The wing down technique should not be used when ferry tanks are installed.

## WET RUNWAY LANDING

When landing on a wet runway, ensure accurate threshold speeds are flown and a positive touchdown is made close to the threshold. Braking may be commenced when below 60 KIAS groundspeed to minimise dynamic aquaplaning. Use smooth and gentle braking to decelerate remaining clear of painted surfaces. When landing in crosswind ensure that the weight is evenly distributed on both main wheels by use of aileron into wind.

## GO-AROUND

When the decision to go-around is made, advance the PCL to MCP, level the wings and select the climb attitude. Carry out the normal after take-off checks when safely climbing away. The normal go-around profile is shown at Figure 2-5.

**CAUTION**

Engine acceleration from idle to MCP may take up to 7 secs. The decision to go-around must be made with sufficient airspeed and height to allow the engine to accelerate.

**NOTE**

Speed acceleration, once MCP is achieved, will be quite rapid and the airspeed may exceed 130 KIAS before the landing gear is fully retracted. Proceed as per the After Take-off checks.

## AOA INDEXER APPROACH

Under favourable wind and runway conditions a lower threshold IAS may be obtained using the AOA indexer. A normal circuit profile is flown up to the base turn point. At the base turn point, power is reduced to 6-8 psi and LAND flap selected. IAS is reduced until the green donut is obtained, and power used to control ROD. With wings level on finals, an AOA validity check is carried out. If IAS is less than 80 KIAS with only the green donut illuminated, convert to a normal circuit. If AOA indications are verified, maintain the green donut to touchdown using power, and use attitude to maintain aimpoint/glidepath. Due to the lower threshold IAS, flare response will be reduced and care should be taken that a high ROD does not develop on finals.

# AFTER LANDING

## AFTER LANDING CHECKS

When clear of the runway, carry out the following checks:

1. Seat Firing Handle - Safety pin in.
2. Command Ejection - Off.
3. AIR BRAKE - In.
4. FLAPS - UP.
5. PROBES - OFF
6. LDG LIGHT - OFF.
7. Transponder (XPDR) - SBY.
8. STROBE/BCN lights - BCN.

## SHUTDOWN CHECKS

Proceed as follows to shutdown the engine.

1. PCL - IDLE.
2. PARKING BRK - ON.
3. Standby AI - Cage.
4. AVIONICS -
   a. BAT - OFF.
   b. GEN - OFF.
5. INVERTER - OFF

**NOTE**

ITT must be stabilised for a minimum of one minute prior to shutdown.

6. PCL - OFF.
7. ECS - OFF.
8. GEN MASTER - OFF.
9. Note time and fuel.

When propeller stopped:

10. Chocks - In.
11. NAV, STROBE/BCN lights - OFF
12. BAT MASTER - OFF.
13. PARKING BRK - As required.
14. OXYGEN - OFF, 100%.
15. Flight controls - Lock

# GO-AROUND PROFILE

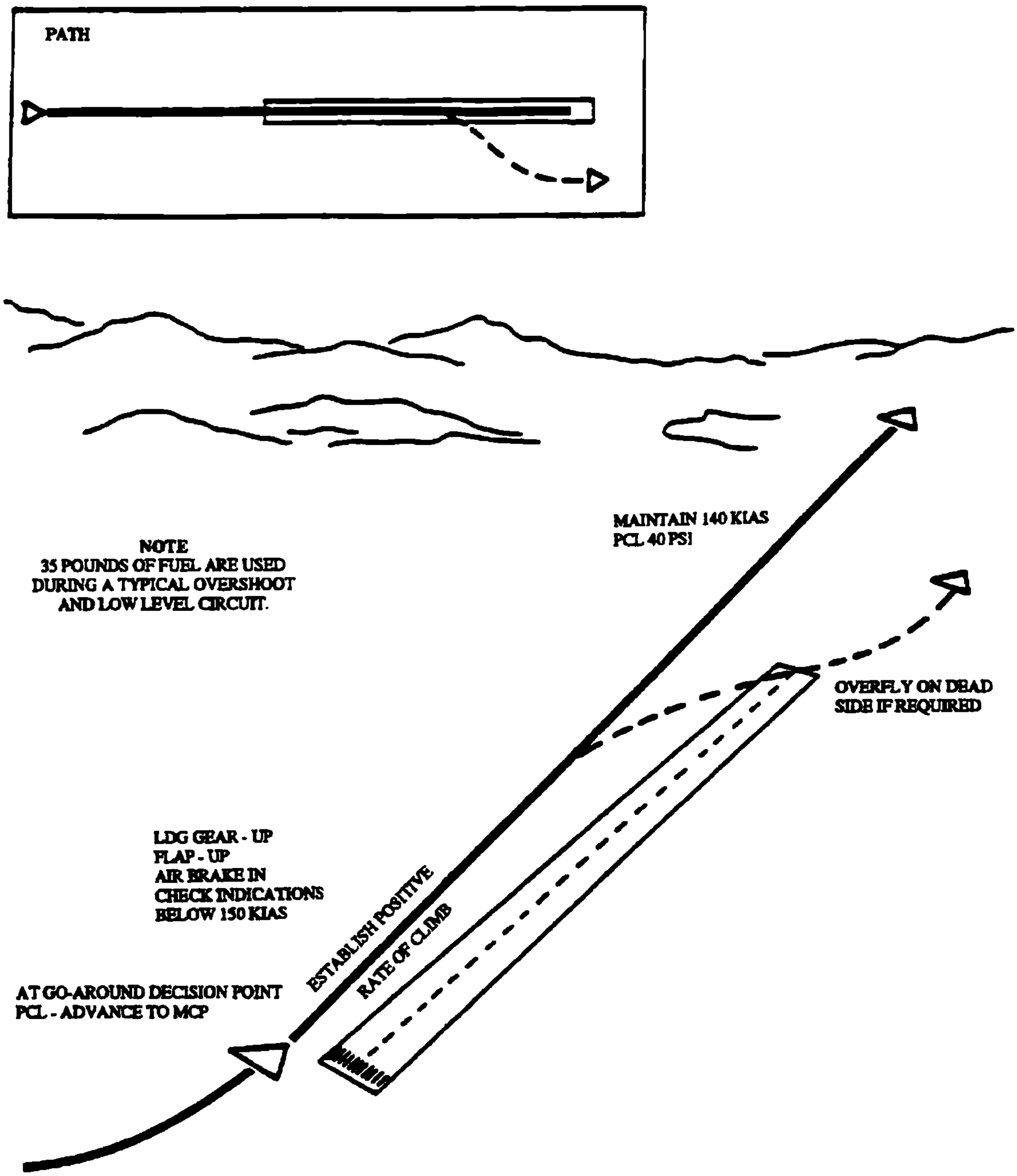



*Figure 2-5*

# AOA INDEXER CIRCUIT

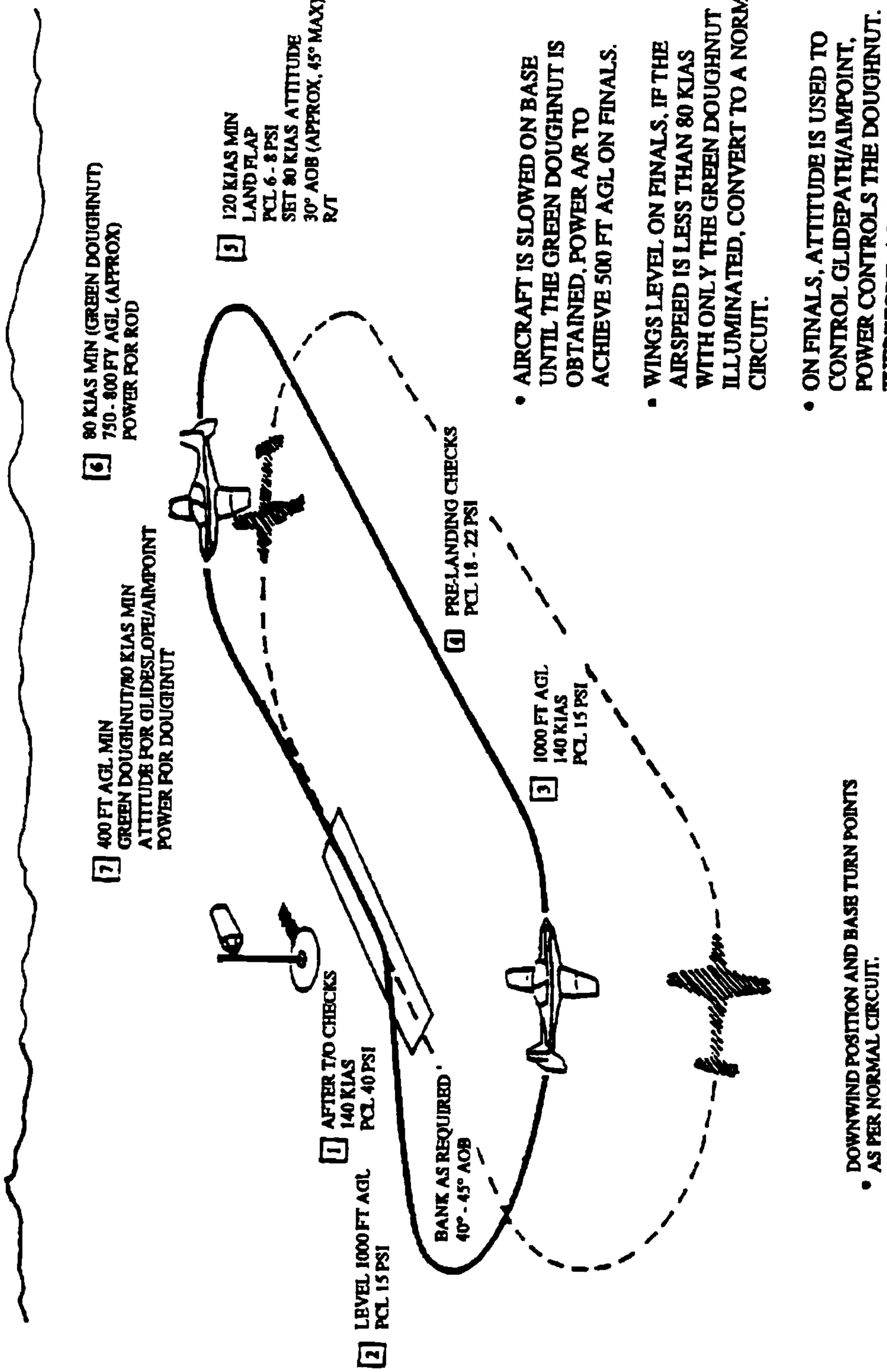
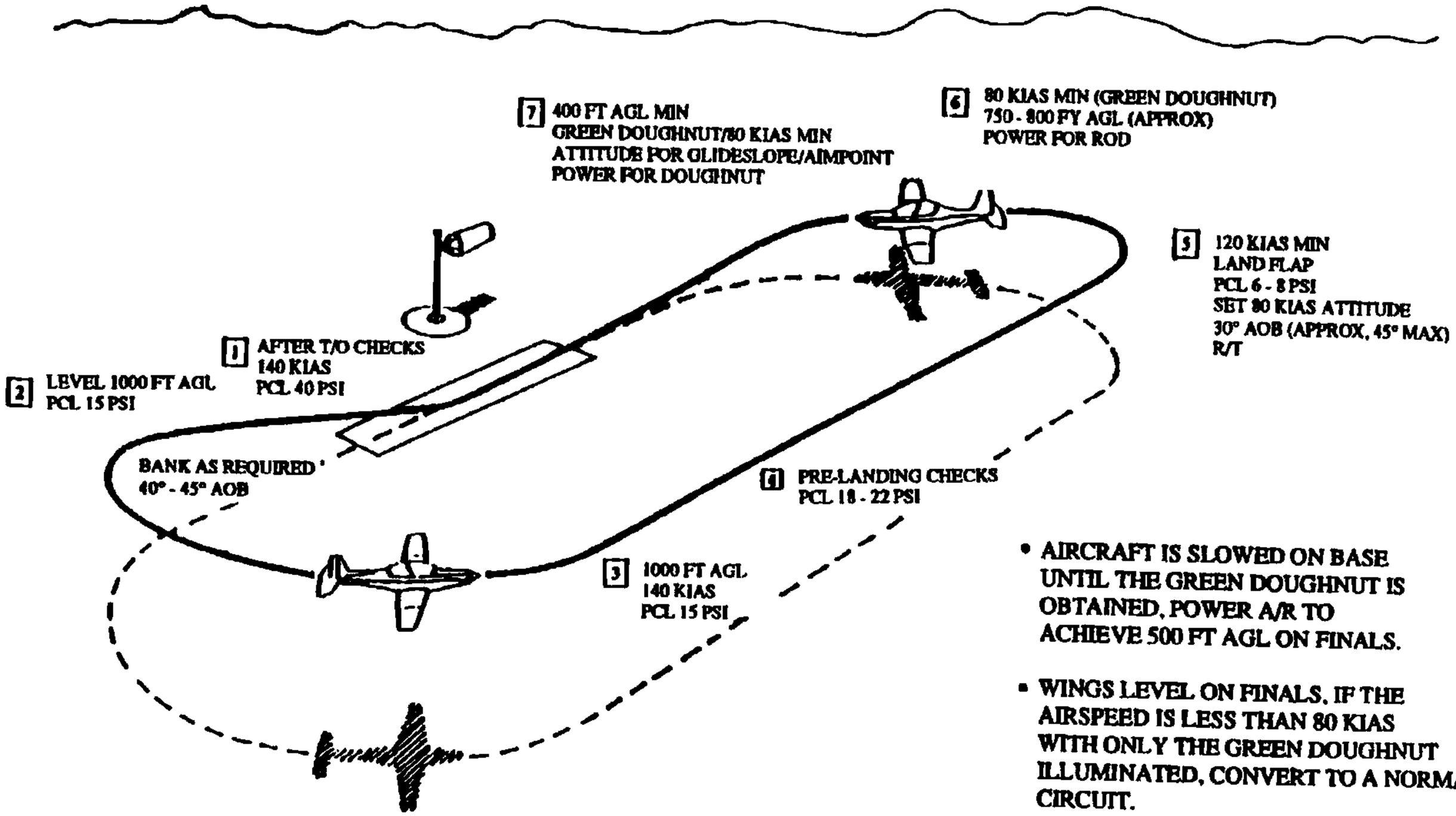



*Figure 2-6*




- DOWNWIND POSITION AND BASE TURN POINTS AS PER NORMAL CIRCUIT.

- AIRCRAFT IS SLOWED ON BASE UNTIL THE GREEN DOUGHNUT IS OBTAINED, POWER A/R TO ACHIEVE 500 FT AGL ON FINALS.

- WINGS LEVEL ON FINALS, IF THE AIRSPEED IS LESS THAN 80 KIAS WITH ONLY THE GREEN DOUGHNUT ILLUMINATED, CONVERT TO A NORMAL CIRCUIT.

- ON FINALS, ATTITUDE IS USED TO CONTROL GLIDEPATH/AIMPOINT, POWER CONTROLS THE DOUGHNUT. THEREPORE, A RED CHEVRON WILL REQUIRE AN INCREASE IN POWER.

# GLIDE CIRCUIT

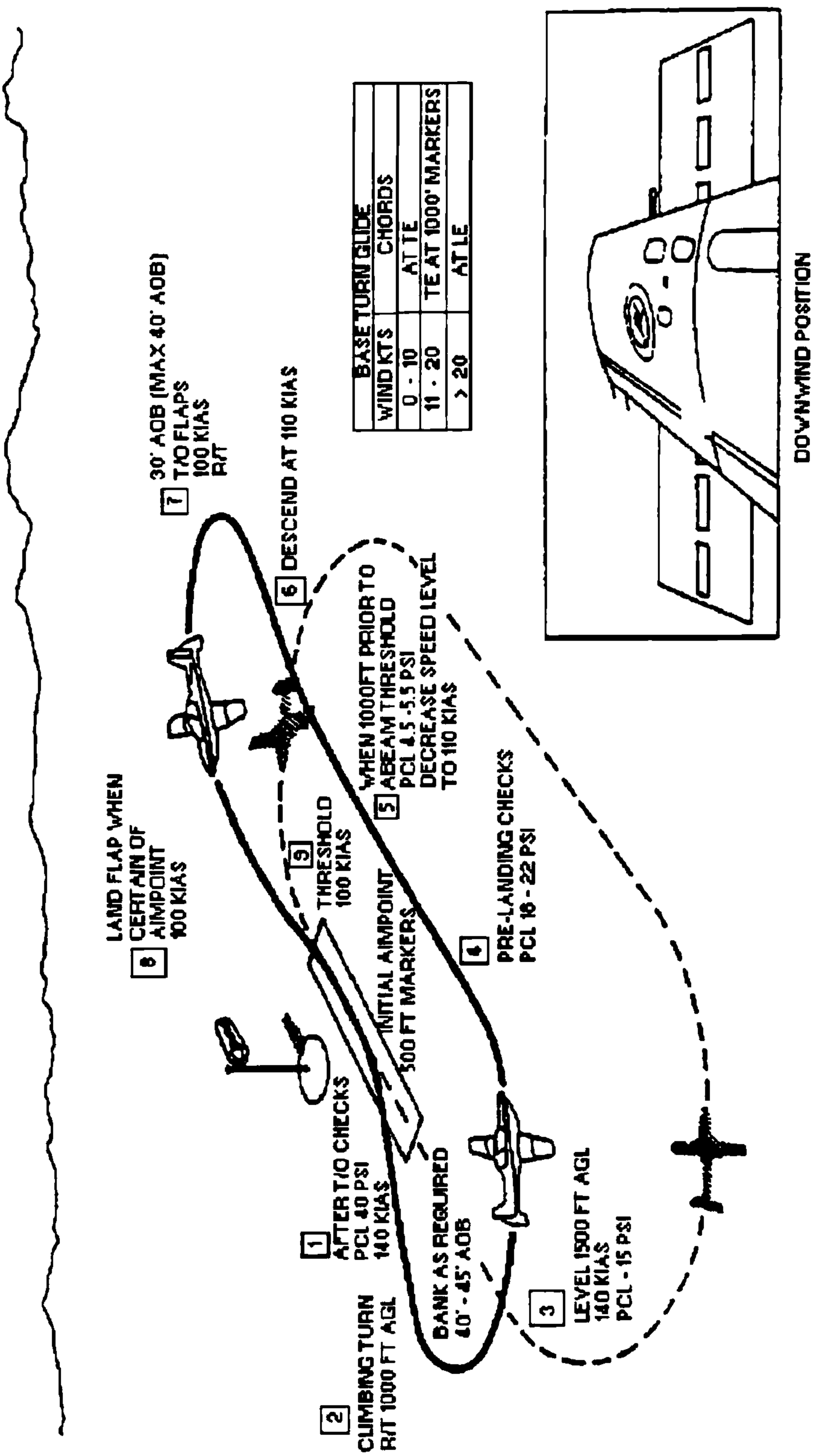
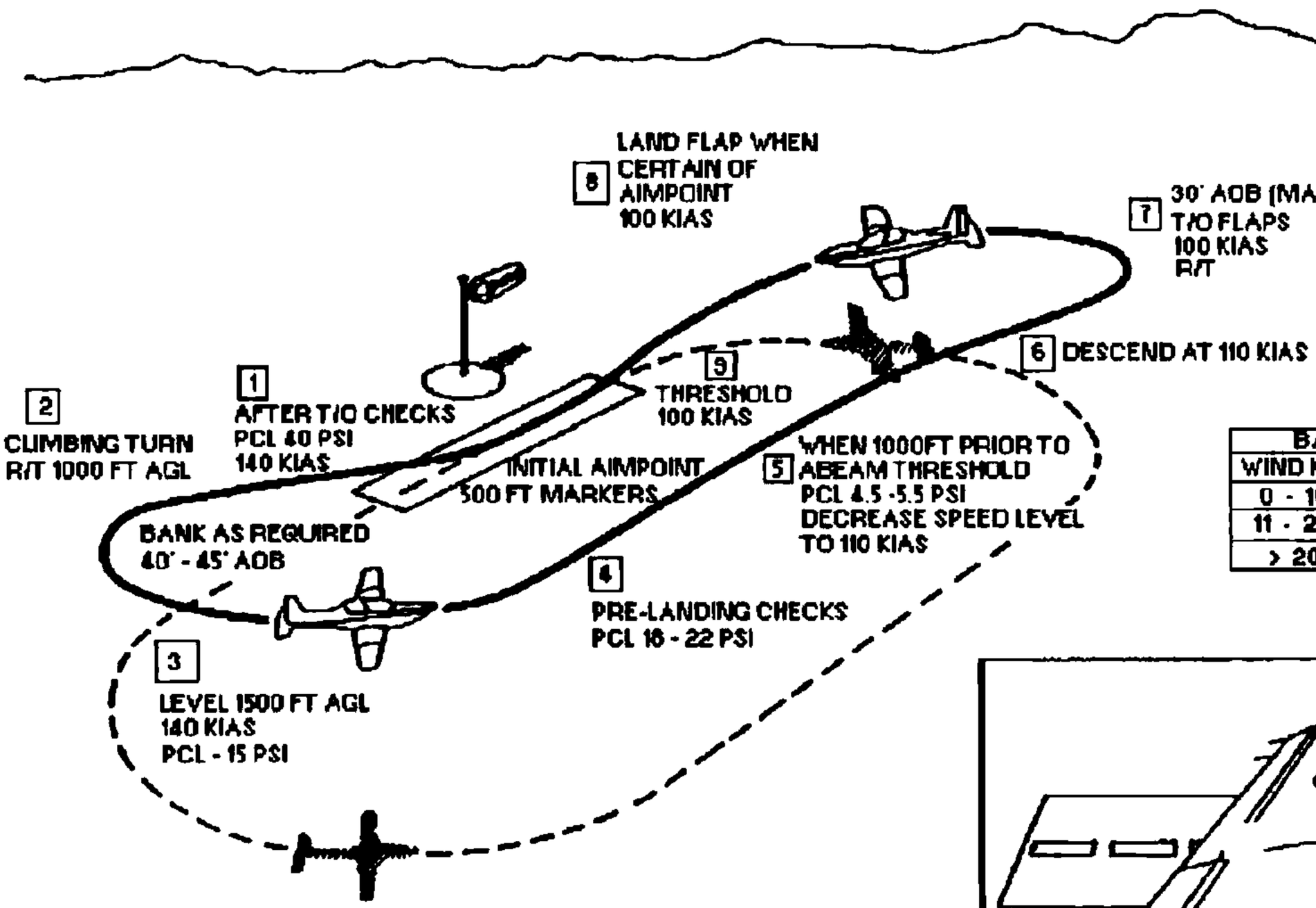
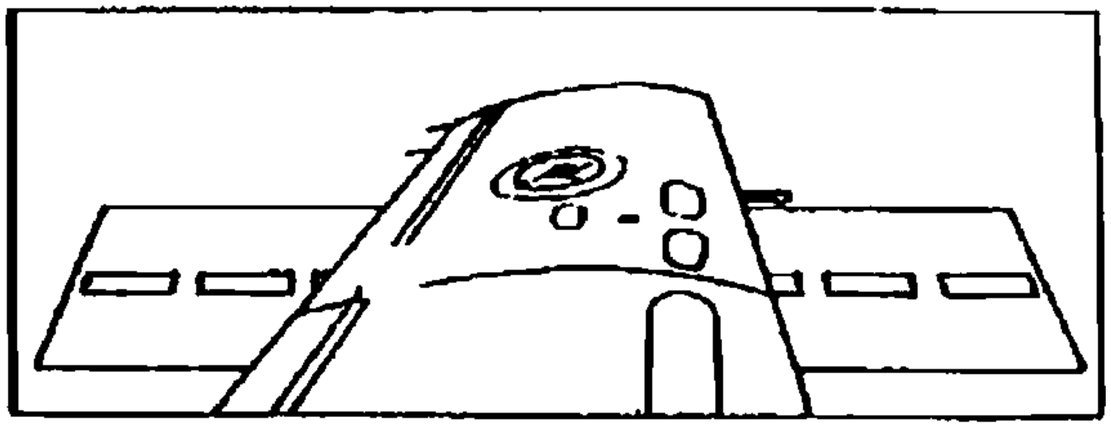



*Figure 2-7*




| BASE TURN GLIDE | |
|---|---|
| WIND KTS | CHORDS |
| 0 - 10 | AT TE |
| 11 - 20 | TE AT 1000' MARKERS |
| > 20 | AT LE |



DOWNWIND POSITION

SECTION 3

# EMERGENCY PROCEDURES

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SECTION 3

# EMERGENCY PROCEDURES

## INTRODUCTION

This section contains procedures to be followed to recover from an emergency situation. Multiple emergencies, adverse weather and other peculiar conditions may dictate modification of these procedures. Critical items are printed in **BOLD FACE TYPE.** These items are considered to be those which must be performed immediately to prevent aggravation of the emergency. The time scale for completion of critical checklist items will not permit direct reference to a checklist and bold face items should therefore be committed to memory. Subsequent checks should normally be performed with reference to the checklist. Each emergency procedure is designed to cover the worst situation for a given emergency. When the nature and severity of an encountered emergency are established common sense and sound judgement are to be used in determining whether all, or part only, of a published procedure is relevant or whether a variation to the procedure is required. When an emergency occurs notify as soon as possible, other crew/flight members and the controlling agency, of the existence of the emergency and the intended course of action. In all emergency situations, three basic rules must be observed to optimise chances of safe recovery:

1. Maintain positive control of the aircraft. If in formation, nominate whether you wish to lead or fly wing and then do so.
2. Analyse the situation and then take the corrective action.
3. Land when appropriate given the nature and severity of the emergency.

The WARNING/CAUTION PRESS TO RESET LIGHTS should be reset as soon as practicable after illumination. This will also reset the audio warning system. If the attention getters are not reset the only indication of any subsequent failure may be the illumination of a CWS annunciator panel caption.

Following any engine malfunction the aircraft should be landed via a precautionary forced landing pattern if weather conditions permit.

## DEFINITIONS

### Land As Soon As Possible:

A landing must be accomplished at the nearest suitable airfield considering the severity of the emergency, weather conditions, field facilities and command guidance.

### Land When Practicable:

The mission should be terminated. Landing is to be accomplished at a suitable alternate or the planned destination at the discretion of the pilot after taking into account the nature and severity of the emergency.

### Absolute Minimum Ejection Seat Design Parameters (AMEP):

AMEP are the minimum parameters (altitude, IAS, and ROD) as stated by the ejection seat manufacturer, from which pilot survival can be expected, assuming correct seat function.

AMEP are Ground Level, at 60 KIAS (no downward vector) with zero pitch and zero roll.

### Minimum Ejection Decision Height (MEDH):

MEDH is the minimum height at which an ejection decision should be made in an 'under control' power-off situation. Ejection following a decision at or above this height should result in pilot survival, assuming correct seat function. If the aircraft captain elects to continue flying the aircraft below this height, he or she should be fully confident of carrying out a successful forced landing and be aware that the chances of a successful ejection after this point may be seriously degraded.

MEDH is 300 ft AO.

### Minimum Safe Ejection Height (MSEH) - Under Control:

MSEH - Under Control, is the minimum height for premeditated ejection straight and level at the recommended IAS. Ejection from this height will provide the pilot with sufficient time to carry out corrective actions in the case of a seat malfunction and to complete the parachute descent checks.

MSEH - Under Control is 1000 ft AO.

### Minimum Safe Ejection Height (MSEH) - Out of Control:

MSEH - Out of Control, is the minimum ejection height at maximum likely ROD out of control (in a spin or following structural failure). Ejection from this height will provide the pilot with sufficient time to carry out corrective actions following a seat malfunction and to complete the parachute descent checks.

MSEH - Out of Control is 3000 ft AO.

# GROUND OPERATION EMERGENCIES

## ENGINE FIRE ON THE GROUND

If the engine fire occurs whilst taxying, bring the aircraft to a stop, preferably clear of aircraft and buildings.

Actions:

1. **PCL - OFF.**
2. **IGNITION - OFF.**
3. **STARTER - OFF.**
4. **FIREWALL SHUT-OFF HANDLE- PULL.**
5. **BAT AND GEN MASTER - OFF.**
6. **PARK BRAKE - AS REQUIRED.**
7. **Execute EMERGENCY EGRESS checklist.**

**NOTE**

If time permits, notify ATC prior to selection of BAT MASTER to OFF.

## EMERGENCY EGRESS

**WARNING**

Do not attempt ejection below 60 knots. Do not pull the MOR handle while the ejection seat remains in the aircraft.

If an emergency exit is necessary whilst taxying, bring the aircraft to a stop. The engine should be shutdown prior to vacating the aircraft. Proceed as follows:

1. **SEAT FIRING HANDLE - SAFETY PIN IN.**
2. **LEG RESTRAINTS - RELEASE.**

**NOTE**

The leg restraints should be released by operating the leg restraint line manual release lever. (refer fig 1-37)

3. **HARNESS - RELEASE.**
4. **OXYGEN AND INTERCOM - DISCONNECT.**
5. **PSP - DISCONNECT.**
6. **CANOPY - OPEN.**
7. Exit cockpit

**NOTE**

If canopy is jammed closed, break perspex with canopy breaker tool. Slide the fastening pin out of the canopy breaking tool located on the right side of the canopy frame and remove the tool. Strike the canopy as hard as possible at a point above the canopy frame on one side. This will require holding the tool in both hands and turning to face the side being struck. Once the canopy has been penetrated, strike ahead of the crack in the direction required to make an escape hole large enough to exit the aircraft.

## BRAKE FAILURE

Brake malfunctions may be either mechanical o hydraulic failures and may result in temporary loss o directional control. Due to the independent hydraulic wheel brake system it is very unlikely that complet failure will occur. Proceed as follows:

1. **NOSE WHEEL STEERING - ENGAGE.**
2. **SERVICEABLE BRAKE - APPLY.**
3. Rudder - as required

**CAUTION**

If the PCL is selected off, nose wheel steering will cease.

**NOTE**

Directional control can be maintained by use of differential braking and nose wheel steering.

# TAKEOFF EMERGENCIES

## ABORT

Take-off should not be commenced when calculated take-off distance required (see Appendix 1) precludes safe stopping using moderate braking in the runway stop-distance available, after abort at the lift-off point. Actions in the event of an abort at this point should be pre-briefed in the Pre-Takeoff checks. Proceed as follows:

1. **PCL - IDLE.**
2. **WHEEL BRAKES - AS REQUIRED.**

If runway overrun and risk of collision likely:

3. **PCL - OFF.**
4. **FIREWALL SHUT-OFF HANDLE - PULL.**
5. **BAT AND GEN MASTER - OFF.**

**WARNING**

Do not attempt ejection below 60 KIAS.

**NOTE**

- If circumstances permit, transmit abort call prior to selection of BAT and GEN MASTER to OFF.
- Most efficient braking is obtained by using one smooth continuous application of brakes (Refer to Braking Technique in Section 2). Should a lock up occur, momentarily cease all braking.
- Whenever moderate to heavy braking has been used, the wheel brakes will be extremely hot. It is preferable not to leave the park brake on as the pads may weld to the discs.

## TYRE FAILURE DURING TAKE-OFF

If sufficient runway remains

1. **Execute ABORT Checklist.**

If insufficient runway remains

2. Continue the takeoff.
3. Do not raise the landing gear.
4. Land when practicable as for LANDING WITH BLOWN TYRE.

**CAUTION**

Differential braking and nose wheel steering may be necessary to maintain directional control. Nose wheel steering activation with a large rudder deflection could lead to rapid departure from the runway. Therefore, nose wheel steering should be used with great caution, and a blip style technique is suggested.

**NOTE**

- Pronounced vibration may occur due to nose wheel shimmy.
- Use minimum braking during a tyre failure abort. If the nose wheel tyre has blown, hold full back elevator to minimise forces on the nose wheel.

## ENGINE FAILURE AFTER TAKEOFF

Actions in the event of engine failure after takeoff should be pre-briefed in the Pre-Takeoff Vital Actions. Careful consideration must be given to various factors such as:

**a.** Cut-off altitudes for safe forced landing, ejection and turn-back.

**b.** Wind direction, strength, gust factor and any possibility of wind shear.

**c.** Terrain in the airfield overrun area and the area beyond for suitability as a forced landing area.

**d.** Weather, including visibility and cloud.

**WARNING**

This procedure is for a time critical response. When height and/or airspeed is sufficient, refer to ENGINE FAILURE/LOSS OF POWER checklist.

**1. LAND AHEAD, EJECT OR TURN BACK (HEIGHT PERMITTING).**

**2. PCL - OFF.**

If landing:

**3. FIRE WALL SHUT-OFF HANDLE -PULL.**

**4. LG HANDLE - DOWN.**

**5. EMER LDG GR HANDLE - PULL.**

**6. FLAPS - AS REQUIRED.**

**7. BAT AND GEN MASTER - OFF.**

**NOTE**

If time permits a Mayday call should be made prior to selection of BAT and GEN MASTER to OFF.

**WARNING**

- The decision to eject should be made as early as possible to increase chances of success. When both cockpits are occupied, the order to eject must be given on intercom and not the radio. For maximum chances of survival in an ejection following an engine failure after take-off, the decision to eject should be made before MEDH and the aircraft should be climbing with wings level at the time of ejection.
- Careful attention to airspeed must be paid after a large and sudden loss of power to avoid inadvertent stalling.

**NOTE**

- Flap selections cannot be retracted after using the emergency method of lowering the landing gear. The pilot must be sure of glide path when making flap selections.
- Lowering the LG handle prior to lowering the landing gear by the emergency method will ensure the gear up/low power/low airspeed warning will not activate, and permit use of the normal LG DOWN TONE.

# INFLIGHT EMERGENCIES

**WARNING**

- During any inflight emergency, when structural damage is known or suspected, a controllability check should be performed in the intended landing configuration at a safe height (above MSEH - Out of Control) if possible. If adequate control response is available and if it is practical, maintain this configuration and land as soon as possible.
- Safety of the crew is the overriding consideration in any emergency. The final decision to eject or attempt a forced landing rests with the pilot. It can be difficult to assess landing area length and surface condition from the air. Structural integrity could be prejudiced if landing on an inadequate surface. If at any stage doubt exists as to the safe completion of a forced landing, the aircraft should be abandoned.

It is impossible to establish a pre-determined set of rules and instructions which would provide a ready made decision applicable to all cases. The following basic conditions are of prime importance in determining whether to eject or attempt a forced landing:

**a.** Condition of the aircraft.

**b.** Pilot experience and ability.

**c.** Weather conditions including wind and cloud base.

**d.** Dimensions and surface conditions of the intended landing area. A hard, smooth, level surface with at least 3000 ft for landing is considered the normal minimum for forced landing, providing the first 500 ft is used as the touch down zone. If at MEDH, the pilot in command determines that the aircraft will not touchdown with 2500 ft of runway remaining, then the aircraft should be abandoned.

**e.** Approaches to the intended landing area.

# ENGINE

## ENGINE FAILURE/LOSS OF POWER

Engine fire or mechanical failure may be indicated by any combination of the following:

**a.** Illumination of the CAUTION PUSH TO RESET and/or WARNING PUSH TO RESET warning lights.

**b.** Visible smoke, flames, oil misting on the canopy.

**c.** Chip light.

**d.** Rapid increase or decrease in NG.

**e.** Rapid increase or decrease in torque.

**f.** Increasing ITT resulting in illumination of ESDP red warning light.

**g.** Airframe vibration.

**h.** NP increasing or decreasing markedly.

Engine flame out will normally be indicated by decreasing:

**a.** Ng < 40%, and

**b.** ITT < 400°C.

FCU failure is indicated by the engine power reducing to IDLE with no response from PCL movement. ELS failure is indicated by erratic and uncommanded changes in engine power output. If the ELU CWS caption illuminates, execute ELU FAILURE checklist.

If power is unreliable or insufficient for flight, proceed as follows:

**1. CONVERT EXCESS SPEED TO HEIGHT.**

**2. CHECK FOR FIRE OR MECHANICAL FAILURE.**

If Fire or Mechanical failure is indicated:

**3. Execute ENGINE FIRE OR MECHANICAL FAILURE checklist.**

If Fire or Mechanical Failure not indicated:

**4. CHECK Ng.**

If Ng > 40%:

**5. ELS ISOLATE/EMERG FUEL SWITCH - ISOL/ARM.**

If engine responds immediately:

**6. PCL - SELECT REQUIRED TORQUE.**

7. Land when practicable.

**CAUTION**

Selecting the ELS ISOLATE/EMERG FUEL switch to ISOL/ARM removes the protection provided by the ELU, and illuminates the CWS ELU and WARNING press to reset captions. If the PCL is at a high power setting, the engine will immediately accelerate and may exceed Torque and/or ITT limits unless manually limited.

If engine does not respond immediately:

8. **TOGGLE SWITCH - SELECT REQUIRED TORQUE.**

**CAUTION**

Acceleration from IDLE to MAX may take up to 10 seconds. Up to 6 seconds may be required to obtain any engine response. Observe engine limits manually.

If engine responds and when a safe altitude has been obtained:

9. PCL - IDLE.

10. Land when practicable using the EFCU.

If Ng decays below 40% or the engine is not responding to EFCU inputs:

11. **PCL - OFF.**

12. Turn towards a suitable landing area and carry out a FORCED LANDING.

**WARNING**

If below MSEH - Under Control, do not prejudice the safety of the crew by persisting in attempts to restore engine power. If sufficient power for flight is not obtained and a suitable landing area cannot be reached, ejection is to be initiated.

**NOTE**

With ITT below 400°C and NP above 1000 RPM, the ESDP warning light illuminates.

## ENGINE FIRE OR MECHANICAL FAILURE

1. **PCL - OFF.**

2. **FIREWALL SHUT-OFF HANDLE - PULL.**

3. **BOOST PUMPS - ARM/RESET.**

4. **GEN MASTER - OFF.**

If fire continues:

5. **EJECT.**

If fire is extinguished:

6. Execute FORCED LANDING checklist.

## FORCED LANDING

1. Glide 125 KIAS.

2. Assess glide potential.

3. Aim for high key at 2500 ft AGL, or low key at 1500 ft AGL.

4. Transponder (XPDR) - Emergency.

5. Transmit - 'MAYDAY'.

If altitude/time permits:

6. Execute NORMAL RELIGHT checklist.

**WARNING**

If the aircraft has been shutdown due to Fire or Mechanical Failure, a relight is not to be attempted.

If relight unsuccessful and no more attempts to be made:

7. FIREWALL SHUT-OFF Handle - Pull.

8. Harness - Locked and tight.

When landing is assured:

9. LG HANDLE - DOWN

10. EMER LDG GR HANDLE - PULL.

11. FLAPS - as required.

12. BAT and GEN MASTER - OFF.

**WARNING**

If at any stage during the forced landing, there is doubt about achieving a safe landing - Eject.

**NOTE**

- Lowering the LG handle prior to lowering the landing gear by the emergency method will ensure the gear up/low power/low airspeed warning will not activate, and permit use of the normal LG DOWN TONE.
- Optimum gliding performance at 2250 kg is achieved at the following speeds.

| Best Speed Range | (KTS) | (Nm/100ft) |
|---|---|---|
| CLEAN | 125 | 2.3 |
| LG DOWN FLAPS UP | 110 | 1.5 |

- Successful lowering of the landing gear by the emergency method will give 3 green and 2 red indications.
- The forced landing pattern is shown at Figure 3-1.

## NORMAL RELIGHT

Recommended conditions: Altitude below 25,000 ft. gliding at 125 KIAS. Relighting may be attempted at all airspeeds. Aircraft attitude needs to be adjusted during the relight or up to 10 KIAS will be lost as the propeller unfeathers.

**CAUTION**

- For relight at or below 120 KIAS the propeller may take up to 2 minutes or 3000 ft of height to achieve governing RPM. Moving the PCL prior to the propeller reaching governing range could result in an overspeed.
- Prior to attempting a relight ensure the ELS is reset by selecting ELS ISOLATE/EMERG FUEL to OFF and pulling and resetting the ELU/TORQUE C/B.

**1.** PCL - OFF.

**2.** Electrical loads - Reduce to below 60 AMP (Turn external lighting and PROBES off).

**3.** BAT MASTER-ON.

**4.** LEFT OR RIGHT BOOST PUMP - ON.

**5.** STARTER - ON.

**6.** IGNition - ON.

When NG at or above 12%:

**7.** PCL - IDLE.

**8.** ITT/NG - MONITOR.

When NG stabilised >56%:

**9.** IGNition - OFF.

**10.** STARTER - OFF.

**11.** BOOST PUMPS - ARM/RESET.

**12.** ENG OIL temp/pressure - Green arcs.

**13.** PCL - Select required torque.

**14.** Electrics - As required.

**15.** AHRS - Reset if required.

If no indication of relight within l0 seconds:

**16.** PCL - OFF.

**17.** IGNition - OFF.

**18.** STARTER - OFF.

**19.** BOOST PUMPS - ARM/RESET.

**20.** Continue with FORCED LANDING checklist.

**WARNING**

Engine normal relight should not normally be attempted below 3000 ft AGL as ejection potential will be seriously degraded.

## ELU FAILURE

ELU failure is indicated by illumination of the ELU CWS caption and WARNING PRESS TO RESET light. Proceed as follows:

**1. PCL - CONTROL TORQUE/ITT.**

When sufficient height is available:

**2. PCL - IDLE THEN ADVANCE TO MID-RANGE**

3. Land When Practicable.

If PCL authority still restricted, proceed as for ENGINE FAILURE/LOSS OF POWER checklist.

**CAUTION**

With the ELU failed, torque and ITT limitations must be observed manually.

**NOTE**

The PCL need only be moved momentarily to IDLE before advancing to MID-RANGE.

## ENGINE OIL PRESSURE

An engine oil pressure malfunction will be indicated by the ENG OIL PSI indicating outside the green sector (90-135 psi) with the following warnings:

**a.** Below 60 psi or above 135 psi (red lines) ESDP WARNING (red) illuminated, ENG OIL PSI ESDP segment blinks quickly.

**b.** 60 - 90 psi (yellow sector) - ESDP CAUTION (yellow) illuminated, ENG OIL PSI ESDP segment blinks slowly.

If oil pressure indication is outside green sector:

**1. NG - CHECK ABOVE 72%.**

If the oil pressure is above or below red lines:

**2. LAND AS SOON AS POSSIBLE USING MINIMUM REQUIRED TORQUE.**

If the oil pressure is in yellow sector:

**3. LAND WHEN PRACTICABLE USING LESS THAN 30 PSI TORQUE.**

**NOTE**

During zero-G manoeuvres, transient oil pressure spikes or drops below 40 psi (and possibly as low as zero) are acceptable provided that the engine manoeuvre time limits detailed in section five are not exceeded. During steady state operations the minimum allowable transient oil pressure is still 40 psi as detailed in Figure 5-5.

## ENGINE OIL TEMPERATURE

An engine oil temperature malfunction will be indicated by the ENG OIL °C indicating outside the green sector (0 to 99°C) with the following warnings:

**a.** Temperature between 0 to -40 °C. ESDP CAUTION (yellow) illuminated, ENG OIL °C ESDP segment blinks slowly.

**b.** Temperature below -40 °C or above +99 °C. ESDP WARNING (red) illuminated, ENG OIL °C ESDP segment blinks quickly.

If the ENG OIL temperature indicates outside the green sector (0 to 99°C), proceed as follows:

**1. PCL - REDUCE TO MINIMUM PRACTICABLE TORQUE.**

If at high altitude and conditions permit:

**2.** Descend to a lower altitude.

**CAUTION**

It is possible during extended flight at low OAT (less than -30 °C) for the engine oil temperature to rise higher than 90 °C. This temperature increase is caused by cold, viscous oil partially blocking the oil cooler.

If temperature reached full scale deflection or does not return to normal operating range:

**3. LAND AS SOON AS POSSIBLE.**

If temperature returns to normal:

**4.** Land When Practicable.

## CHIP DETECTOR WARNING

Presence of metallic elements in the reduction gearbox section of the engine oil system is indicated by illumination of the CHIP CWS caption and CAUTION PRESS TO RESET light. Proceed as follows:

1. **PCL - REDUCE TO MINIMUM PRACTICABLE TORQUE.**

2. **Land as soon as possible.**

3. Monitor NP.

If NP Decreasing:

4. Execute ENGINE FIRE OR MECHANICAL FAILURE checklist.

**NOTE**

Monitor engine performance to provide warning of impending reduction gearbox failure.

## ENGINE ICING

Engine icing may be indicated by reduction in torque and/or increase in ITT, possible resonance, vibration and compressor stall. If Engine icing is suspected, proceed as follows:

1. INTAKE ICE handle - Pull.

2. PCL - Retard.

3. Descend clear of icing area.

4. PCL - Advance slowly to 20 PSI and allow de-icing to take place for five minutes.

5. If ITT increases rapidly when the PCL is subsequently advanced, reselect 20 PSI and allow a further five minutes before further advancing the PCL.

If Engine Flame Out Occurs:

6. Execute **NORMAL RELIGHT** checklist.

If Relight is Successful:

7. Carry out ice melting procedure detailed in steps 4 and 5.

If Relight is Unsuccessful:

8. Conduct a further attempt when clear of icing conditions.

**WARNING**

- If engine operation is not restored sufficiently to sustain level flight by the time Minimum Safe Altitude (MSA) is reached, the aircraft should be abandoned unless in VMC and well positioned for a forced landing.
- There is a very low probability of successful clearance of engine icing conditions if engine icing is encountered in cloud or precipitation at MSA. Monitor ITT closely during PCL movements and relight attempts after suspected engine icing.

# ELECTRICAL

## GENERATOR FAILURE

Generator failure is indicated by the following:

**a.** illumination of the GEN CWS caption and CAUTION PRESS TO RESET light, and

**b.** DC VOLTS indicating 25.0 or less, and DC AMPS discharging.

### NOTE

If discharge is 60 AMPS or more, ESDP (yellow) Caution will be ON and DC AMPS blinking.

Proceed as follows:

1. STARTER - Check OFF.

2. GEN MASTER - ON/RESET.

If Generator does not reset:

3. GEN MASTER - OFF.

4. Electrical loads - Non-essentials OFF.

5. GEN MASTER - ON/RESET.

If Generator does not reset:

6. GEN MASTER - OFF.

7. Land before Battery is exhausted.

### NOTE

- Although multiple attempts to reset the GEN do not present a fire risk it is not considered worthwhile persisting with any more than five attempts as this would indicate a failure of the GEN or regulator.
- Selection of non-essential loads will depend on circumstances at the time of failure, including the following:

  **a.** VFR or IFR flight conditions and expected conditions for recovery to nearest suitable airfield,

  **b.** radio and navigation aids necessary for recovery,

  **c.** distance from nearest suitable airfield, and

  **d.** day or night.

- The Battery should last 20 mins if load is reduced to below 60 AMPS or 30 mins if load is reduced to 50 AMPS. Electrical services will begin to fail when volts drop below 21 V DC.

## INVERTER FAILURE

Inverter failure is indicated by:

**a.** illumination of the INV CWS caption and CAUTION PRESS TO RESET light, and

**b.** loss of avionics systems.

Proceed as follows:

1. **INVERTER - SELECT ALTERNATIVE.**

### NOTE

If circuit breaker of failed inverter is popped, wait two minutes, then attempt to reset once only. Do not reselect this inverter unless it is essential for flight.

## BUS-TIE C/B POPPED

Popping of the BUS-TIE C/B is indicated by the following:

**a.** illumination of the BUS-TIE CWS caption and CAUTION PRESS TO RESET light, and

**b.** DC VOLTS indicating 25.0 or less, and DC AMPS discharging.

Proceed as follows:

1. BUS-TIE C/B - Reset.

If C/B resets:

2. Monitor volts and amps.

3. Land when practicable.

If C/B will not reset:

4. Monitor volts and amps.

5. If IMC, attempt to regain VMC.

6. Land as soon as possible.

### NOTE

- The Battery Busbar will not be powered by the Generator if the BUS-TIE C/B will not reset.

- If the Generator fails with the BUS-TIE C/B popped, execute GENERATOR BUSBAR FAILURE checklist.
- Consideration should be given to reducing Battery Busbar load to extend Battery life.

## GENERATOR BUSBAR FAILURE

**CAUTION**

A Generator Busbar failure while the ELU is limiting engine output will lead to immediate engine overtorque or overtemp.

Generator Busbar failure is indicated by illumination of the GEN BUS CWS caption and the WARNING PRESS TO RESET light. Proceed as follows:

1. **PCL - CONTROL TORQUE/ITT.**
2. INVERTER - BAT.
3. DC VOLTS and AMPS - Check.

If DC VOLTS are normal (28 V DC) and DC AMPS are 0 or charging, indications are that the Generator is still supplying power normally to the Battery Busbar. Proceed as follows:

4. Land using services supplied by Battery Busbar.

If DC VOLTS are 25 or less and DC AMPS have normal discharge, indications are that the Generator is not supplying the Battery Busbar. Proceed as follows:

5. BUS-TIE C/B - reset if popped (BUS-TIE CWS caption should be illuminated).

If DC VOLTS/AMPS do not return to normal:

6. Electrical loads - Non-essentials OFF.
7. Land using services supplied by the Battery Busbar before the Battery is exhausted.

**NOTE**

The following systems are inoperative: AOA, ECS, AIR BRAKE, NWS, Probes, Aural Warning, RH Boost Pump, COMM2 VHF, TACAN, ADF, DU2, Marker and LG position lights.

## BATTERY BUSBAR FAILURE

Battery Busbar failure is indicated by illumination of the BAT BUS CWS caption and the WARNING PRESS TO RESET light. Proceed as follows:

1. **INVERTER - GEN. (EFIS SWAP IF IMC)**
2. DC VOLTS and AMPS - Check.

If DC VOLTS are normal (28 V DC) and DC AMPS are 0 or charging, indications are that the Generator is still supplying power normally to the Generator Busbar, and charging the Battery. Proceed as follows:

3. Land using services supplied by the Generator Busbar.

If DC VOLTS are 25 or less and DC AMPS have discharge, indications are that the Generator is not supplying the Battery. Proceed as follows:

4. BUS-TIE C/B - Reset if popped (BUS-TIE CWS caption should be illuminated).

If DC VOLTS/AMPS do not return to normal:

5. Land using the services supplied by the Generator Busbar.

**NOTE**

The following systems will be lost; normal Landing Gear extension, COMM 1, LH Boost Pump, EFCU, normal audio, STBY AI, trims, fuel flow, DUI, RMI, TACAN and VOR, XPNDR and the Landing Gear down tone. The Landing Gear will have to be lowered using the EMERGENCY LANDING GEAR EXTENSION checklist.

## EFIS DU FAILURE

A DU failure may be indicated by blanking or irregular display indication of the affected screen or smoke/fumes emanating from the vicinity of the DU. It may also be accompanied by a yellow DU1/DU2 screen caption and/or unusual static breakthrough on the aircraft intercom. Proceed as follows:

If smoke or fumes are evident:

1. **Execute SMOKE AND FUMES IN THE COCKPIT/ELECTRICAL FIRE checklist.**

If smoke or fumes are not evident:

2. EFIS SWAP - As required.
3. EADI/EHSI circuit breaker (affected DU) - PULL

**CAUTION**

The DU Brightness/Blanking Control knob only controls screen intensity. Power is not isolated from the DU until the circuit breaker is pulled.

4. Land when practicable.

## SMOKE AND FUMES IN THE COCKPIT/ ELECTRICAL FIRE

**NOTE**

It is common for fuel smells to be present in the cockpit during aerobatic manouvres. If fumes persist execute SMOKE AND FUMES IN THE COCKPIT/ELECTRICAL FIRE checklist.

Smoke or fumes in the cockpit may be obvious in the form of visible smoke or may be subtle and therefore insidious in nature. All odours not identifiable should be considered toxic. If fumes and/or odours in the cockpit are suspected proceed as follows:

1. **OXYGEN - ON, 100% AND EMERGENCY, TOGGLE DOWN.**

2. **RAM AIR - OPEN.**

3. **LAND AS SOON AS POSSIBLE.**

If indications of smoke and fumes continue and electrical fire is suspected:

4. **PCL - RETARD TO MID RANGE.**

5. **BAT AND GEN MASTER - OFF.**

**WARNING**

Isolating all electric power in IMC, even momentarily, is potentially hazardous. The STBY AI can be expected to supply reliable attitude information for up to 9 minutes.

**CAUTION**

- Turning the BAT and GEN Master off while the ELU is limiting engine output will lead to immediate engine overtorque or overtemp.
- If in IMC consideration should be given to attempting to isolate the fire/fumes to a single BUSBAR by initially switching off the BAT MASTER only, then pulling the Bus-Tie C/B. If the fire/fumes continue, switch the BAT MASTER ON and then select the GEN MASTER OFF.

6. **BUS TIE C/B - PULL.**

If electrical and/or other systems are required proceed as follows:

7. BAT MASTER - ON.

All electrical systems not required for the safe termination of the flight should be selected off.

If fire/fumes continue:

8. BAT MASTER - OFF.

9. GEN MASTER - ON.

If fire/fumes continue:

10. Suspect Engine Fire.

**NOTE**

- Only those systems that are required for the safe termination of the flight should be selected on.
- If fault is isolated to a particular Busbar and power from that Busbar is required for the safe termination of the sortie, further isolation can be achieved by using the following procedure. Remove power from the Busbar and then switch off all electrical items supplied by that Busbar. Return power to the Busbar and select services individually in an attempt to isolate the fault.

## BATTERY HOT

Battery hot is indicated by illumination of the BAT HOT CWS caption and CAUTION PRESS TO RESET light. Proceed as follows:

1. **BAT MASTER - OFF.**

2. **LAND AS SOON AS POSSIBLE.**

If the BAT HOT caption goes out, and the Battery is essential for recovery, wait 5 mins then:

3. BAT MASTER - ON.

If the BAT HOT CWS caption re-illuminates:

4. BAT MASTER - OFF.

# OXYGEN SYSTEM

## OXYGEN FAILURE

If the aircraft oxygen system fails above 10,000 ft proceed as follows:

1. **EO HANDLE - PULL.**
2. **MAIN OXYGEN HOSE - DISCONNECT.**
3. **DESCEND BELOW 10,000 FT.**

**WARNING**

Ensure the oxygen hose is disconnected at the lower of the two connection points on the connecting elbow.

If the oxygen system fails below 10,000 ft:

4. Land when practicable.

## SUSPECTED HYPOXIA

If symptoms of hypoxia or hyperventilation are suspected, proceed as follows:

1. **OXYGEN - ON, 100% AND EMERGENCY, TOGGLE DOWN.**
2. Oxygen - Check contents, flow and connections.
3. Mask seal - Check.
4. Descend below 10,000 ft.
5. Land when practicable.

**NOTE**

Control rate of breathing (normal respiration is one breath per 4 to 5 seconds).

If Hypoxia persists:

6. Execute OXYGEN FAILURE checklist.

**WARNING**

Avoid any high G manoeuvres following a suspected hypoxia incident.

## LOW OXYGEN CONTENTS

Low oxygen contents (<300 psi) is indicated by illumination of the OXYGEN CWS caption and CAUTION PRESS TO RESET light. Proceed as follows:

1. Monitor contents.

If pressure falls below 100 psi:

2. Descend below 10,000 ft.
3. Land when practicable.

**NOTE**

- Once below 10,000 ft, consideration should be given to disconnecting from the system to conserve oxygen in case of smoke and/or fumes.
- If pressure falls below 50 psi, the aircraft must be placed unserviceable.

# HYDRAULIC SYSTEM

## HYDRAULIC FAILURE

Loss of hydraulics can result from an electrical or hydraulic problem. Loss of hydraulic pressure is indicated by illumination of the HYDR PX CWS caption and the CAUTION PRESS TO RESET light. Proceed as follows:

1. HYDR SYS C/B - Check in, reset if popped.
2. HYDR PX caption - Check.

If C/B will not reset, or the HYDR PX CWS caption remains illuminated:

3. MONITOR N2 HYD.
4. Land when practicable.

**NOTE**

All normal hydraulic services are inoperative. The Landing Gear will need to be lowered by EMERGENCY LANDING GEAR EXTENSION checklist.

## EMERGENCY HYDRAULIC FAILURE

Loss of emergency hydraulic pressure is indicated by illumination of the HYDR E CWS caption and the CAUTION PRESS TO RESET light. Proceed as follows:

**1. N2 HYD PRESSURE - CHECK.**

If the pressure has decreased (below 2900 psi):

**2. Execute EMERGENCY LANDING GEAR EXTENSION checklist.**

**NOTE**

Emergency landing gear lowering is not available if N2 HYD pressure falls below 2400 psi. With low N2 HYD pressure, reducing airspeed may improve the chances of landing gear lowering.

# FUEL SYSTEM

## LOW FUEL PRESSURE

Low fuel pressure is indicated by illumination of the FUEL PX CWS caption and CAUTION PRESS TO RESET light. The L and R FUEL P lights will also illuminate, indicating BOOST PUMPS have come on automatically. The FUEL PX caption may extinguish before it can be identified. Proceed as follows:

1. PMP RESET - PRESS.

If the Boost Pumps will not reset and the FUEL PX, L FUEL P and R FUEL P captions remain on:

2. Land when practicable.

## BOOST PUMP FAILURE

During operations with the boost pumps ON, Boost Pump failure will be indicated by extinguishing of the respective L or R FUEL P CWS caption. Proceed as follows:

1. AFFECTED BOOST PUMP - ARM.
2. AFFECTED BOOST PUMP LH/RH C/B - Reset.
3. BOOST PUMP - ON.

**NOTE**

If the Boost Pump subsequently fails, do not reset the circuit breaker.

## SUSPECTED FUEL LEAK

A fuel leak may be indicated by a large fuel imbalance combined with a discrepancy between digital and analogue fuel indications and/or visible fuel. Proceed as follows:

1. BOOST PUMP (leaking side) - ON.
2. BOOST PUMP (good side) - ARM/RESET.
3. Fuel state - Monitor.
4. Land when practicable.

**NOTE**

- Under certain temperature and flight conditions fuel may vent from the wing tanks.
- Two segments on the analogue gauge are approximately equal to 35 lbs.

## FUEL ASYMMETRY

Fuel asymmetry is indicated by imbalance between left and right tanks. Proceed as follows:

1. BOOST PUMP (on fuller side) - ON.
2. Fuel state - Monitor.

When fuel balanced:

3. BOOST PUMP - ARM.

If fuel imbalance cannot be corrected:

4. Suspect fuel leak.

**NOTE**

- Maximum fuel asymmetry is 1/4 tank.
- The balancing process may take several minutes and the pilot should institute some reminder system to select the Boost Pump back to the ARM position. Forgetting to complete the procedure will result in major fuel imbalance on the opposite side.

# AIRCRAFT ABANDONMENT

## CONSIDERATIONS FOR EJECTION

Every emergency in which ejection is considered will have its particular set of circumstances involving such factors as speed, attitude, control and altitude. The ejection seat AMEP allow ejection at runway level provided the aircraft speed in level flight is between 60 and 400 KIAS. However, under controlled level flight conditions, eject not below MSEH - Under Control (1 000 ft AO), whenever possible. In an uncontrolled situation, eject not below MSEH - Out of Control (3000 ft AO), whenever possible.

**WARNING**

Do not delay ejection below MSEH - Under Control or MSEH - Out of Control in futile attempts to start the engine, regain power or regain control (whichever is applicable), which may commit you to marginal conditions for safe ejection.

**NOTE**

If an instrument flying hood is fitted, normal ejection is made through the hood.

## OPTIMISING EJECTION

An ejection at low altitude is optimised by commencing a climb and converting excess speed to height. This manoeuvre imparts an upward vector to the ejection seat, providing greater ground clearance during the ejection sequence than would be the case if ejection is performed in level flight. Provided ejection takes place while a positive rate of climb is maintained, the increase in ground clearance will increase the chances of a successful ejection. As with all aircraft ejection systems, safe ejection is enhanced by establishing the best conditions possible prior to ejection. The ideal ejection conditions are:

**a.** Level flight above 2,000 ft AO, and

**b.** 200 to 250 KIAS.

## EJECTION PROCEDURE

The ideal ejection procedure to be used in a premeditated ejection is as follows:

**1.** Aircraft - Position over unpopulated area.

**2.** Height - Ideally above 2,000 ft AO.

**3.** Airspeed - 200 to 250 KIAS.

**4.** Attitude - Level or climbing, no bank.

**5.** Harness - Locked and tight. PSP connected.

**6.** Helmet visor(s) - Down.

**7.** Seat firing handle safety pin - Confirm removed.

**8.** Command ejection - On.

**9.** Trim - Third right rudder.

**10.** Warn other aircraft occupant (if applicable).

**11.** Adopt the EJECTION POSTURE.

**12.** PCL - OFF.

**13.** SEAT FIRING HANDLE - PULL.

**WARNING**

- The canopy must be closed and locked for all ejections.
- The firing handle may be pulled with either the left or right hand with a vertical pull. The other hand should grip the wrist of the pulling hand to prevent flailing at high speeds.
- A conscious effort must be made to adopt the correct ejection posture.
- Simultaneous ejection of both pilots is inadvisable as seat entanglement could occur.

## EJECTION POSTURE

The correct ejection posture is:

**a.** Buttocks well back in seat, straps tight.

**b.** Legs forward with feet touching the rudder pedals to ensure optimum thigh support.

**c.** Back pressed firmly against the seat back.

**d.** Head erect, eyes facing straight forward.

## MANUAL OVERRIDE

If after ejection it is suspected that the barostat unit has failed, proceed as follows:

1. **SEAT FIRING HANDLE - PULL.**

2. **MOR - OPERATE.**

**WARNING**

If the front-seat pilot is ejected by the command ejection system, the MOR handle cannot be operated until the front seat Firing Handle has been pulled.

## POST EJECTION PROCEDURES

### Descent Onto Land

1. Disconnect the oxygen mask at the supply tube bayonet fitting.

2. If the mask is not required for facial protection it may be discarded.

3. If landing in open terrain, release one side (preferably the right hand side) of the single handed release strap connectors.

4. After a landing operate the QRF to release the parachute.

### Descent Into Water

1. Release one side (preferably the right hand side) of the single handed release strap connectors. The personal survival pack will fall free (still connected to the Secumar jacket by the lanyard) and the dinghy will inflate.

2. Remove mask (discard) and remove gloves (place in a pocket).

3. When entering water (and not before) operate the QRF to release the parachute.

4. Pull in dinghy, enter over the narrow end.

5. Pull the survival kit attached to the lanyard aboard.

# OTHER INFLIGHT EMERGENCIES

## LOSS OF CONTROL

The loss of control procedure will recover the aircraft at the incipient stage. If a full spin develops, but cannot be positively identified, the loss of control procedure will recover the aircraft although height loss during recovery will be increased. If control of the aircraft is lost, or the pilot is unsure of a manoeuvre or response of the aircraft, proceed as follows:

1. **PCL - IDLE.**

2. **CONTROLS - VISUALLY CENTRALISED.**

When control regained:

3. Recover from the dive.

If a spin is positively identified, carry out the spin recovery actions as follows:

4. **TURN NEEDLE - IDENTIFY DIRECTION.**

5. **RUDDER - FULL OPPOSITE (TO TURN NEEDLE).**

6. **CONTROL COLUMN - VISUALLY CONFIRM CENTRALISED.**

When the spin stops:

7. **RUDDER - CENTRALISED.**

8. Recover from the dive.

## STUCK PCL

A stuck PCL may involve in-flight shut-down of the engine using the Firewall Shut-off Handle in order to land the aircraft. The engine will take up to 7 seconds to shutdown depending on power setting (experience indicates 3-4 seconds at 45 psi TQ).

**CAUTION**

The propeller will not feather when the engine is shutdown using the Firewall Shut-off Handle. Aerodynamic drag from a windmilling propeller will be significant, and will result in a high rate of descent (approximately 4000 ft/min) during a 125 KIAS glide.

An alternative resolution of a stuck PCL situation involves a straight in approach with Undercarriage, Flaps and Airbrake extended, ECS high and Intake Ice Handle pulled to control airspeed, and then shutting the engine down using the Firewall Shut-off Handle over the threshold. When the engine is shutdown a large increase in drag and decrease in IAS will occur. Experience indicates that with the PCL stuck at MCP, this configuration will result in a minor overspeed of Undercarriage and Flaps.

In deciding whether to use an unfeathered propeller forced landing or a high-drag, powered straight-in approach to resolve a stuck PCL situation, consideration should be given to the power setting at which the PCL is stuck, and consequently the delay in engine shut-down using the Firewall Shut-off Handle, wind conditions, runway length available and pilot experience. Particular attention should be given to maintaining the aircraft within safe ejection parameters.

**NOTE**

An accurate descent profile in the unfeathered propeller configuration will not be available until ARDU flight trials are completed. Consequently the performance figures provided are approximate and are offered for guidance only.

## TRIM RUNAWAY

Trim runaway will be indicated by a progressively increasing force in a single flight control direction and by the relevant trim indicator driving in the appropriate direction. Proceed as follows:

1. **CONTROLS AND TRIM SWITCH - OPPOSE THE RUNAWAY.**

If trim runaway continues:

2. **TRIM INTERRUPT SWITCH - INT RPT.**

3. Circuit breaker of failed trim - Pull.

4. TRIM interrupt switch - NORM.

**CAUTION**

If the elevator trim is permitted to run to full travel at speeds in excess of 240 KIAS, control forces may be greater than the pilot can oppose.

**NOTE**

The aileron and elevator trims are powered from the same circuit breaker on the Battery Busbar. Reduce speed to relieve control forces.

## DAMAGED OR UNSAFE CANOPY

**WARNING**

The chances of a successful ejection through an unlocked canopy are unknown.

An unsafe canopy will be indicated by illumination of the CANOPY CWS caption and CAUTION PRESS TO RESET light. If the canopy is damaged or unsafe, proceed as follows:

1. **AIRSPEED - REDUCE TO 120 KIAS.**
2. **ENSURE NO YAWING FORCES.**
3. Lower ejection seat.

**WARNING**

- Ensure the MOR linkages on the right of the seat pan are free of obstruction prior to seat height adjustment. Loose objects may jam and bend the linkages, causing the drogue to fire.

4. Make a straight-in approach using minimum torque.

## COMMUNICATIONS FAILURE

If unable to transmit and/or receive on COMM 1 or COMM 2 check the following items:

1. Radio control panel ON/OFF knob - ON.
2. Radio control panel volume - Fully ON (clockwise).
3. Audio integration panel volume - Fully ON (clockwise).
4. Rotary selector - Correct COMM selection.
5. Correct frequency.
6. Mic/Tel lead and helmet connections.
7. Squelch - OFF (COMM 2 only - PUSH TST).
8. C/Bs (on respective busses) - IN.

If unsuccessful:

9. Try reselecting the frequency.
10. Try previous/other frequency.
11. Try other COMM unit.

## INTERCOM FAILURE

If intercommunication between the crew is not possible, proceed as follows

1. Oxygen mask microphone switch - ON
2. Audio integration panel volume - Fully ON
3. Rotary selector - Correct COMM position.
4. Mic/Tel lead and Helmet connections - Check
5. Audio C/B (Battery Busbar) - Check IN.

If intercommunication is still not possible:

6. Connect Mic/Tel lead to emergency audio connector (banded yellow/attached to oxygen hose) in BOTH cockpits.
7. Rotary selector BOTH cockpits - Select IC.

**NOTE**

- Gen Avionics BUS BAR must be powered for emergency intercom operation.
- Both occupants need to connect to the emergency audio connectors with the rotary selector-in the IC position to use aircraft emergency intercom system. To transmit on a COMM unit, select the rotary selector to the transmitter desired and return to IC when finished. The audio receiver selectors must be selected IN on the COMM unit/s desired for monitoring R/T.

# LANDING EMERGENCIES

## FLAP

### FLAP FAILURE

Flap failure will be indicated by the flap position indicator not moving or indicating unsafe after flap selection. Proceed as follows:

1. FLAPS CONT 1 (BAT Busbar) and FLAPS CONT 2 (GEN Busbar) C/Bs - Check in, reset if popped.

If normal flap function is not restored and the flaps fail DOWN, no action is possible. If the flaps fail UP and are required for landing, proceed as follows:

2. Airspeed - Below 150 KIAS.

3. LG handle - DOWN.

4. EMER LDG GR handle - Pull.

5. FLAPS - Select as required.

If flaps do not function, no further actions are possible.

**CAUTION**

Landing distance without flaps is approximately 40% longer than with FLAPS.

**NOTE**

- Flaps cannot be retracted if the EMER LDG GR handle has been pulled.
- The flapless circuit procedure is shown at Fig 3-2.

# LANDING GEAR

**CAUTION**

Do not recycle Landing Gear until checklist procedures have been read.

## EMERGENCY LANDING GEAR EXTENSION

**1. AIRSPEED - BELOW 150 KIAS.**

**2. LG HANDLE - DOWN.**

**3. EMER LDG GR HANDLE - PULL.**

**NOTE**

- If range is critical, consideration should be given to leaving the landing gear and flaps up until visual with the airfield environment.
- The landing gear lights will indicate three greens and two reds indicating the main landing gear doors are still open.

If the 3 green lights are not obtained:

4. Obtain external inspection of aircraft.

5. According to Landing Gear position roll and/or sideslip and/or apply positive and negative G in an attempt to lock the landing gear down.

If this is not successful and the 3 greens are still not obtained:

6. Execute LANDING WITH GEAR UP OR PARTIALLY DOWN checklist.

## LANDING GEAR UP SELECTION FAILURE

Landing Gear up selection failure will be indicated by all wheels remain locked down (3 greens) after the landing gear handle is selected UP. Proceed as follows:

**1. Airspeed - Maintain below 150 KIAS.**

2. LG Handle - Down.

3. Land When Practicable.

If Landing Gear retraction is required for operational reasons, the following procedure may provide retraction:

4. Rear LG Handle - UP (if dual).

5. LDG GR HANDLE C/B - Check in, reset if popped.

**NOTE**

If the front and rear LG handles are not mechanically interconnected, ensure both front and rear landing gear handles are selected down in the before landing checks.

## PARTIAL LANDING GEAR RETRACTION

If full retraction is not achieved after slowing the aircraft to below 130 KIAS and achieving stabilised 1 G flight, proceed as follows:

**1. Airspeed - Maintain below 150 KIAS.**

2. LDG GR HANDLE C/B- Check in, reset if popped.

3. Land when practicable

If Landing Gear fails to retract:

4. LG Handle - Reselect DOWN.

If three greens not obtained:

5. Execute EMERGENCY LANDING GEAR EXTENSION checklist.

## PARTIAL LANDING GEAR EXTENSION

Partial extension of the Landing Gear, after selection of the LG handle to DOWN, will be indicated by the landing gear unlocking, but one or more wheels do not lock down (less than 3 greens or a red on the nose wheel). Proceed as follows:

1. LDG GR HANDLE C/B - Check in, reset if popped.

2. Land when practicable.

If 3 greens not obtained:

3. Execute EMERGENCY LANDING GEAR EXTENSION checklist.

## LANDING GEAR DOWN SELECTION FAILURE

Landing gear down selection failure will be indicated by the landing gear remaining locked UP (all lights out), after the LG handle is selected DOWN. Proceed as follows:

1. Rear LG HANDLE - DOWN (if dual)

2. LDG GR HANDLE C/B - Check in, reset if popped.

If landing gear remains locked UP:

3. HYDR PX CWS Caption - Check out.

If HYDR PX CWS caption is illuminated:

4. Execute HYDRAULIC FAILURE checklist.

If caption is not illuminated:

5. LG Handle - Re-cycle UP then DOWN.

If the landing gear remains UP:

6. Execute EMERGENCY LANDING GEAR EXTENSION checklist.

**NOTE**

If the front and rear LG handles are not mechanically interconnected, the captain should land when practicable.

## LANDING WITH GEAR UP OR PARTIALLY DOWN

**WARNING**

Serious damage to the aircraft and injury to the crew is likely when landing with one main wheel up.

1. FUEL - REDUCE TO MINIMUM SAFE FOR LANDING.

2. FLAPS - LAND.

When landing is assured:

3. FIREWALL SHUT-OFF HANDLE - PULL.

4. PCL - OFF.

5. BAT and GEN MASTER - OFF.

6. Use aileron to keep wings level. Lower nose gently before elevator effectiveness is lost.

**NOTE**

- Pulling the Firewall Shut-off Handle will not cause instantaneous shutdown, and will allow PCL and BAT and GEN MASTER OFF selections without compromising aircraft control.
- The landing lights will be inoperative. The AOA indexer will only operate if the nose gear is down and locked.

## LANDING WITH A BLOWN MAIN OR NOSE WHEEL TYRE

If landing with a blown main or nose wheel tyre proceed as follows:

1. Reduce aircraft weight as much as possible.

2. Fly a normal approach.

3. Land on the side of the runway opposite the blown tyre.

4. Touch down on the good tyre and hold the blown tyre off the runway for as long as possible.

5. Proceed as for ABORT procedure

# FORCED LANDING

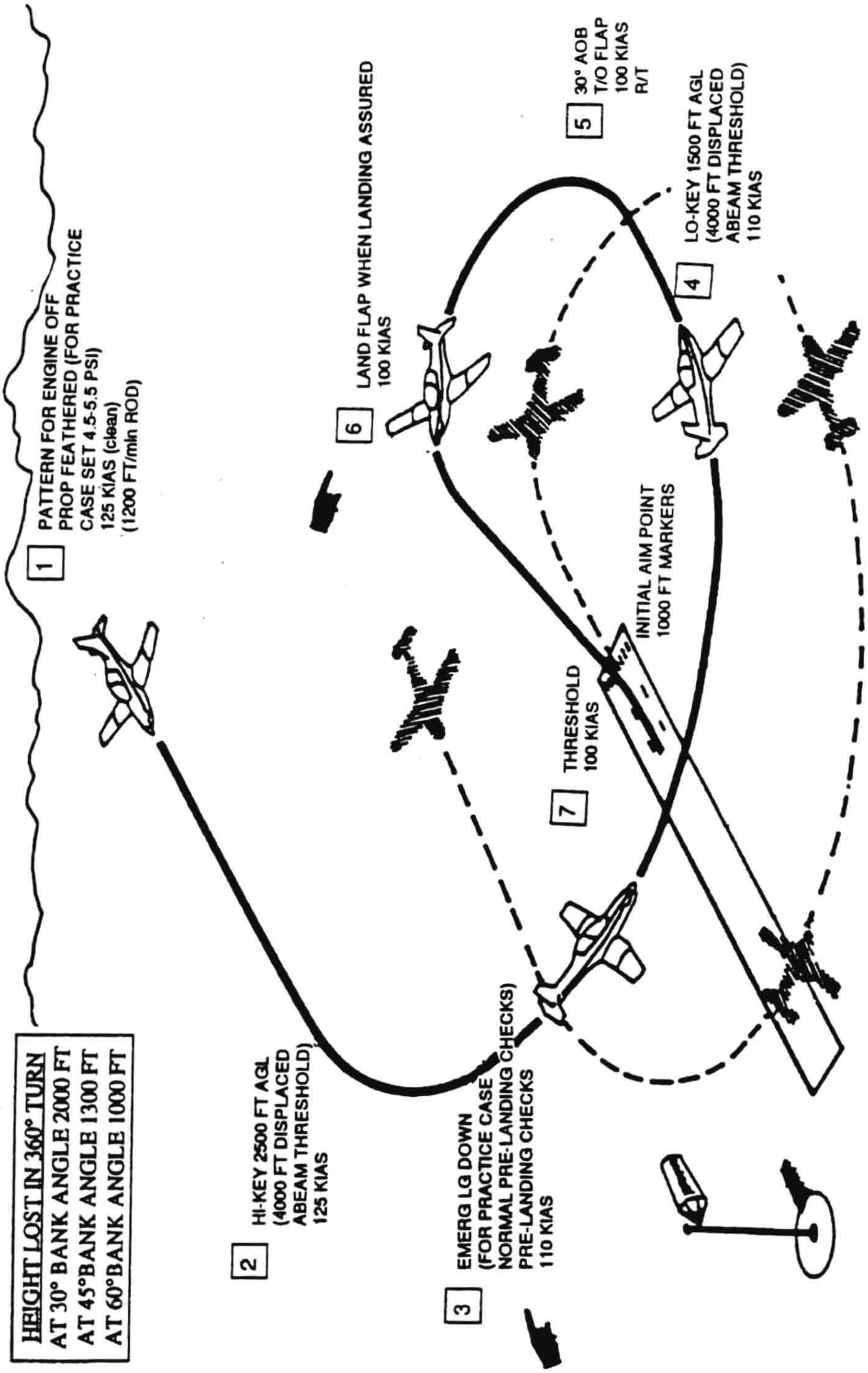



*Figure 3-1*

# FORCED LANDING

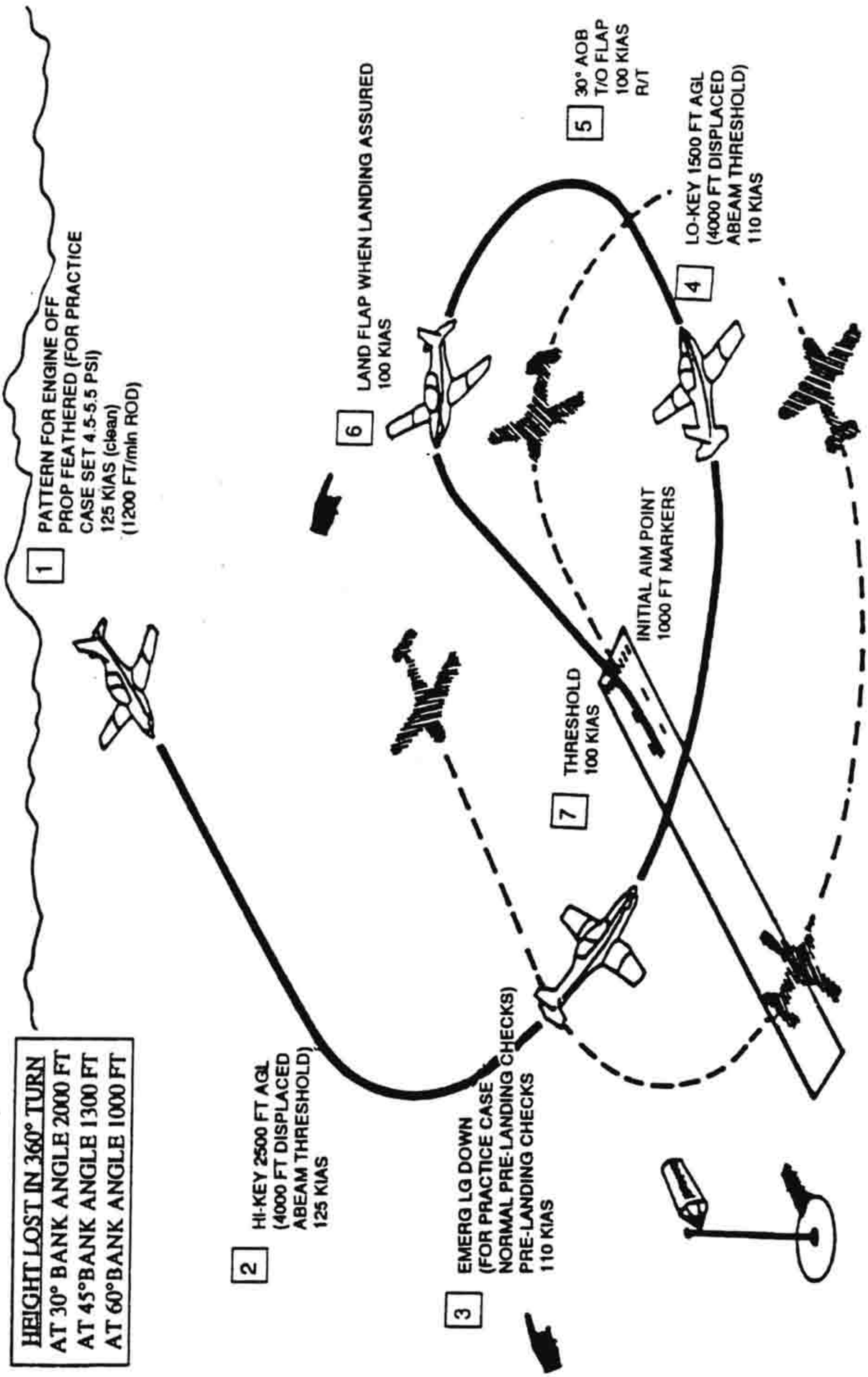



*Figure 3-1*

# FLAPLESS CIRCUIT

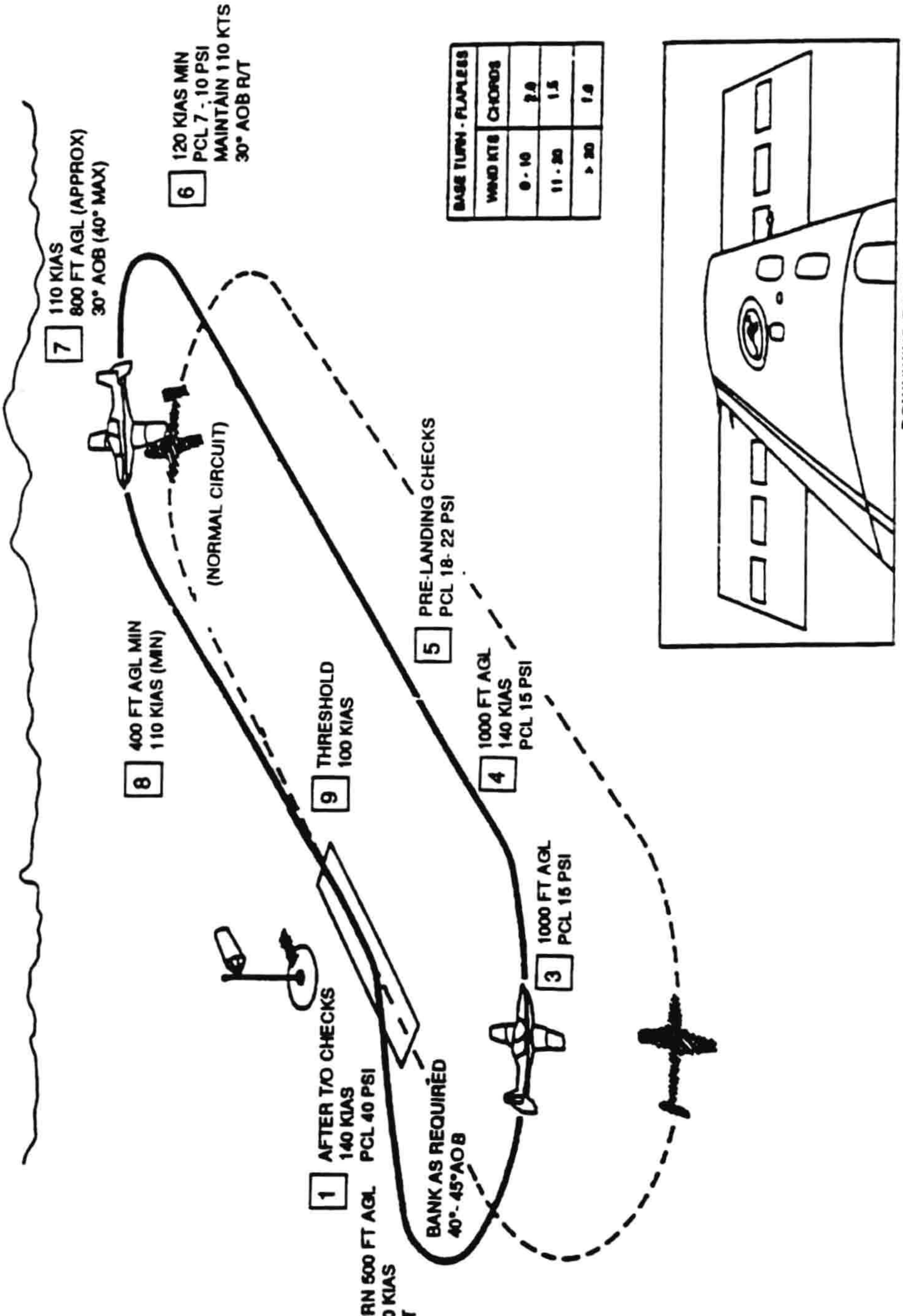



*Figure 3-2*

# FLAPLESS CIRCUIT

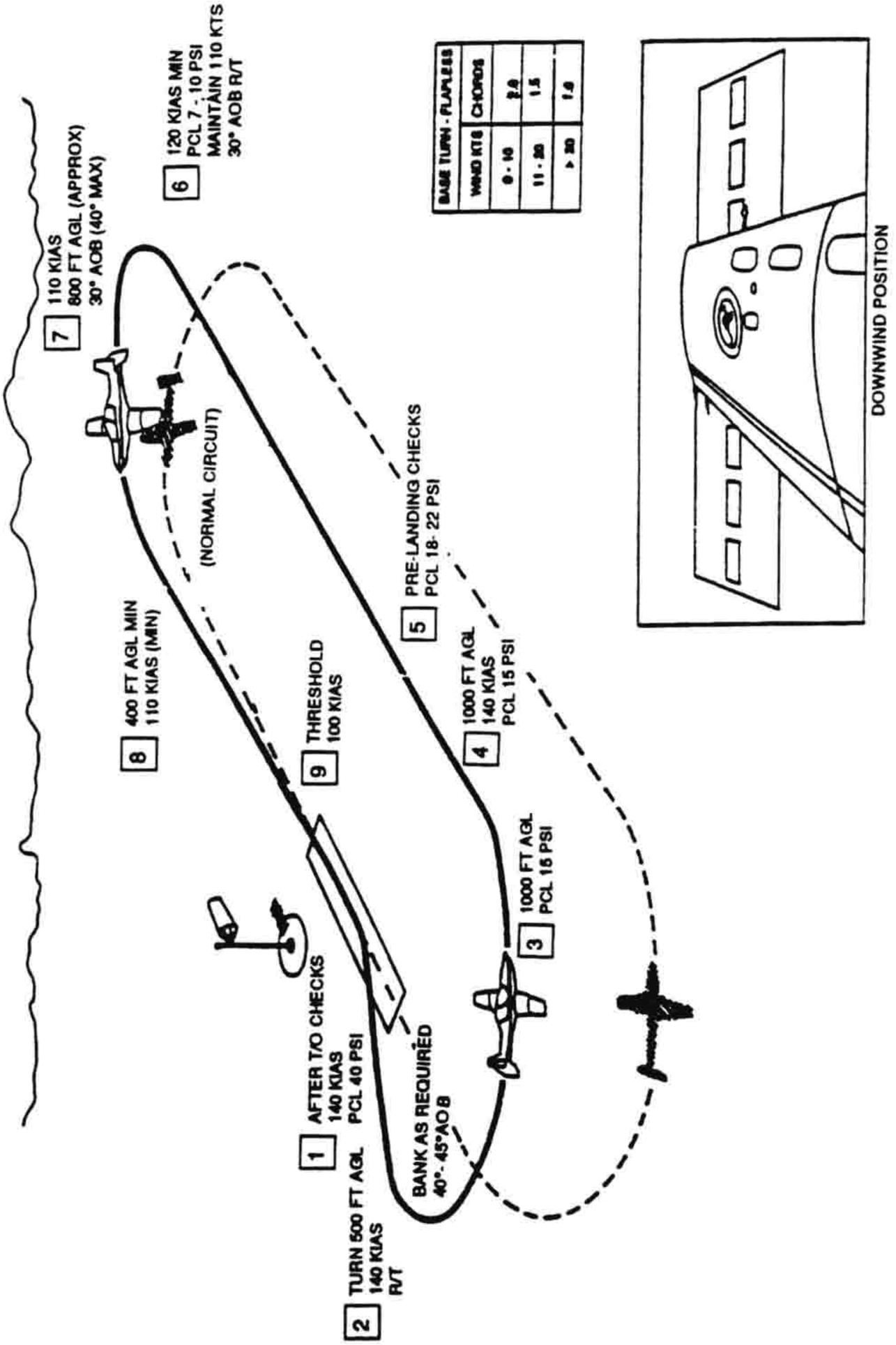



*Figure 3-2*

SECTION 4

# CREW DUTIES

This section is not applicable to the PC9/A aircraft.

SECTION 5

# OPERATING LIMITATIONS

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SECTION 5

# OPERATING LIMITATIONS

## INTRODUCTION

Special attention should be given to the instrument marking illustration, Fig 5-1, since these limitations are not necessarily repeated under their respective sections.

## MINIMUM CREW REQUIREMENTS

The minimum crew for normal flight is one pilot. The aircraft may only be flown solo from the front seat.

## INSTRUMENT MARKINGS

| INSTRUMENT | MARKING | VALUES | MEANING |
|---|---|---|---|
| TORQUE PRESSURE GAUGE | GREEN SECTOR | 0 - 63.8 psi | Normal operating range. |
| | YELLOW SECTOR | 63.8 - 67.4 psi | Time limited to 10% of total engine operating time. |
| | RED LINE | 67.4 psi | Max continuous torque. |
| | RED DIAMOND | 74.3 psi | Max transient (20 sec). |
| | | 80 psi | Max overswing (10 sec). |
| ITT GAUGE | GREEN SECTOR | 400 - 775°C | Normal operating range. |
| | YELLOW SECTOR | 775 - 800°C | Time limited to 10% of total engine operating time. |
| | RED LINE | 800°C | Max continuous ITT. |
| | RED DIAMOND | 1000°C | Max ITT during start (5 sec only > 870°C). |
| GAS GENERATOR TACHOMETER | GREEN SECTOR | 56 - 104% | Normal operating range. |
| | RED LINE | 104% | Maximum. (NG) |
| OIL TEMPERATURE GAUGE | GREEN SECTOR | 0 - 99°C | Normal operating range. |
| | RED LINE | 99°C | Never exceed temp. |
| OIL PRESSURE GAUGE | RED DIAMOND | 40 psi | Min transient. |
| | RED LINE | 60 psi | Min steady pressure. |
| | YELLOW SECTOR | 60 - 90 psi | Pressure low. |
| | GREEN SECTOR | 90 - 135 psi | Normal operating range. |
| | RED LINE | 135 psi | Max steady pressure. |
| | RED DIAMOND | 200 psi | Max on start and transient. |
| N2 HYD PRESSURE GAUGE | RED LINE | 2900 psi | Min operating pressure. |
| | GREEN SECTOR | 2900 - 3100 psi | Normal operating range. |
| | (last segment illuminated indicates that a pressure of 3600 psi has been reached. This could occur under certain conditions and requires no pilot action). | | |

*Figure 5-1*

## INSTRUMENT MARKINGS

Refer to Fig 5-2 for ESDP instrument markings, Fig 5-3 for Mach/Airspeed indicator and Fig 5-4 for accelerometer markings.

**Note**

Limits are based on fuel grade of F34 Aviation Turbine Kerosene (AVTUR with FSII (S748) with SG of 0.785).

# ENGINE AND SECONDARY INSTRUMENTS DISPLAY PANEL

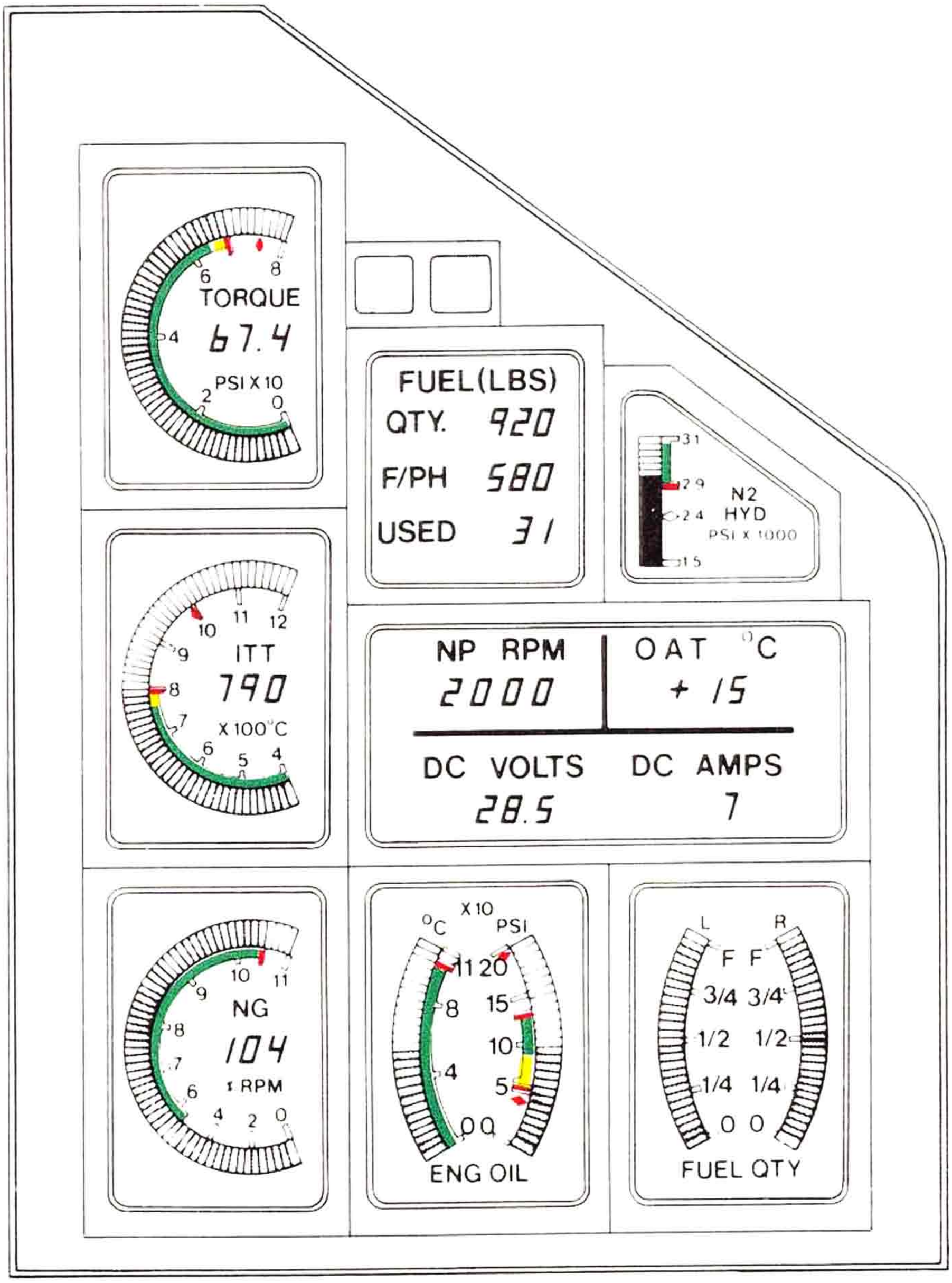



*Figure 5-2*

# MACH/ASI MARKINGS




**Figure 5-3**

| | |
|---|---|
| Red Radials | 320 KIAS / 0.68 Mach. Maximum permissable SL to 35,000 ft. |
| Green Arc | 79 to 320 KIAS. Normal operating range. Lower limit is max weight stall speed with flaps and landing gear retracted. Upper limit is max structural speed SL to 35,000 ft. |
| White Arc | 70 to 150 KIAS. LAND flap operating range. Lower limit is max weight stall speed and upper limit is max speed permissible with flaps extended. |
| Yellow Diamond | Pilot selectable airspeed bug. |

# ACCELEROMETER MARKINGS

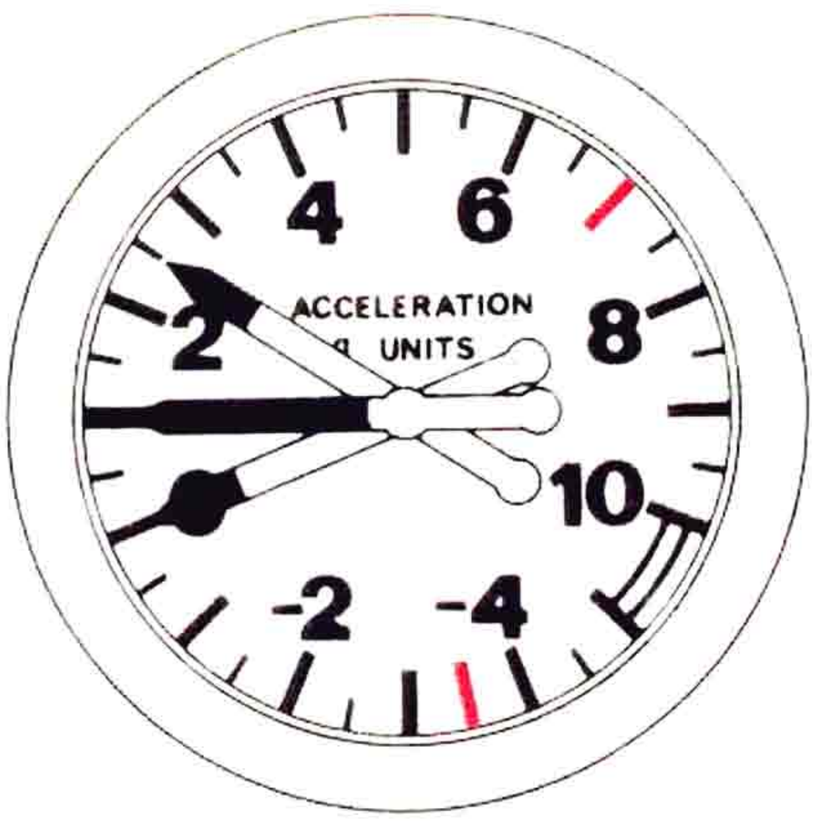



**Figure 5-4**

| | |
|---|---|
| Red Bands | At 7 G maximum positive. At −3.5 G maximum negative. |

# ENGINE LIMITATIONS

| POWER SETTING | TORQUE PSI | ITT °C | NG % | NP RPM | Oil Pres PSI | Oil Temp °C | Time |
|---|---|---|---|---|---|---|---|
| MAXIMUM – | 67.4 | 800 | 104 | 2000 | 90–135 | 0–99 | 10% oper. |
| TRANSIENT – | 74.3 | 870 | 104 | 2205 | 40–200 | 0–99 | 20 sec |
| OVERSWING – | 80 | 870 | 104 | 2205 | 40–200 | 0–99 | 10 sec |
| MAX CRUISE | 63.8 | 775 | 104 | 2000 | 90–135 | 0–99 | |
| IDLE – | | 800 | 56 Min | | 60 min | – 40–99 | |
| STARTING – | | 870 | | | 0–200 | –40 Min | 20 sec |
| | | 1000 | | | 0–200 | – 40 Min | 5 sec |

**Figure 5–5**

## ENGINE LIMITATIONS

All normal engine limitations are shown in Fig 5-5. EIS warning and caution parameters are shown in Fig 5-6. Deliberate inflight engine shutdowns are not permitted.

**NOTE**

- Oil pressure of 90-135 psi is normal with Ng above 72% and oil temperature 60-70° C. Pressure between 60 and 90 psi is undesirable and the flight should be terminated as soon as possible using a maximum of 30 psi torque. Oil pressures below 60 psi are unsafe and the engine must be shut down as soon as possible.

- During zero-G manoeuvres, transient oil pressure spikes or drops below 40 psi (and possibly as low as zero) are acceptable provided that the engine manoeuvre time limits detailed in this section are not exceeded. During steady state operations the minimum allowable transient oil pressure is still 40 psi as detailed in Figure 5–5.

## EIS CAUTIONS AND WARNINGS

The EIS gives the following automatic warning indications:

a. **CAUTION.** The Engine Caution lamp, located above the EIS panel, illuminates steady yellow as long as the abnormal condition exists. The associated indication flashes 40 times a minute.

b. **WARNING.** The Engine Warning lamp, located above the EIS panel, illuminates steady red as long as the abnormal condition exists. The associated indication flashes 80 times a minute. Illumination of the Warning automatically extinguishes the Caution.

A summary of EIS cautions and warnings and their associated causes is given at Fig 5-6.

## Engine Out Of Limits

Land as soon as possible whenever the engine exceeds:

a. 80 PSI torque,

b. 74.3 to 80 PSI for more than 10 seconds,

c. 67.4 to 74.3 PSI torque for more than 20 seconds,

d. 870°C ITT,

e. 800°C to 870°C ITT for more than 20 seconds,

f. 104% NG, or

g. 2205 RPM NP.

**CAUTION**

Engine overspeeds have a cumulative effect and each case must be recorded in the EE500.

## Inter Turbine Temperature

The engine must be shut down wherever ITT exceeds:

a. 1000°C, or

b. 870°C to 1000°C for more than five seconds during engine start.

## Engine Manoeuvre Limitations

The engine manoeuvre limitations are as follows:

a. Inverted flight - 60 seconds maximum.

b. Vertical flight nose up - 15 seconds maximum.

c. Vertical flight nose down - 3 seconds maximum-20 seconds maximum at idle power.

d. Level flight wings vertical - 5 seconds maximum.

# EIS CAUTIONS / WARNINGS

| Parameter | Caution Range | Warning Range |
|---|---|---|
| Torque | 68.3 - 74.8 psi (15 sec delayed) | > 68.3 psi (20 sec delayed)<br>> 74.8 psi (instant) |
| ITT - Norm | 813 - 870°C (15 sec delayed)<br>870 - 883°C (instant) | > 813°C (20 sec delayed)<br>> 883°C (instant)<br>, 400°C and NP > 1000 RPM |
| ITT _ Start | > 870°C (instant) | > 883°C (5 sec dealyed)<br>> 1033°C (instant) |
| NG | None | > 104% |
| Oil Temp | -40°C to 0°C (inhib on start) | < -40°C (inhib on start)<br>> +99°C |
| Oil Press | 60 to 90 psi (inhib on prestart, start and shutdown) | < 60 psi (inhib on prestart, start and shutdown)<br>> 135 psi (inhib on start) |
| NP | < 1960 RPM and NG > 90% (3 sec delayed) (inhib on prestart, start and shutdown)<br>2040 to 2205 RPM (3 sec delayed) (inhib on prestart, start and shutdown) | > 2205 RPM (instant) (inhib on prestart, start and shutdown)<br>1200 to 1700 RPM and aircraft on the ground (10 sec delayed), (inhib on prestart, start and shutdown) |
| DC Volts | 18 to 22VDC (inhib on prestart and start)<br>29.5 to 32.2VDC (inhib on prestart and start) | < 18VDC (inhib on prestart and start)<br>> 32.2VDC (inhib on prestart and start) |
| DC Amps | Discharge > -60A | None |
| OAT | < +4°C and Probes switch off | None |

*Figure 5-6*

### Engine Operation In Icing Conditions

Flight into known icing conditions is not permitted.

**CAUTION**

Where icing conditions are not forecast but the temperature range is between +4°C and -30°C with visible moisture present, engine icing may occur.

**NOTE**

Refer to Section 7 for details regarding flight in icing conditions.

## PROPELLER LIMITATIONS

Stabilised operation on the ground between 1200 and 1700 NP is prohibited.

## EJECTION SEAT LIMITATIONS

### Ejection Speeds

The minimum speed for ejection at ground level is 60 KIAS and the maximum is 400 KIAS. For ejection at higher altitudes the speed is unlimited.

### Ejection Seat Operating Minima

a. Absolute Minimum Ejection Seat Design Parameters (AMEP) - Ground Level at 60 KIAS (no downward vector) with zero pitch and zero roll.

b. Minimum Ejection Decision Height (MEDH) -: 300 ft AO.

c. Minimum Safe Ejection Height -Under Control (MSEH - Under Control) - 1000 ft AO.

d. Minimum Safe Ejection Height - Out of Control (MSEH - Out of Control) - 3000 ft AO.

**WARNING**

Successful ejection at low altitudes is dependant on the resultant vertical speed vectors of the ejection seat and the descent rate. Ejection at low altitudes with a high rate of descent may not be successful. Every attempt should be made to eliminate or reduce the aircraft rate of descent before initiating the ejection sequence.

## AIRFRAME LIMITATIONS

### AIRSPEED LIMITATIONS

### Maximum Operating Speed

The maximum operating speed (VMO/MMO) is:

a. 320 KIAS, or

b. 0.68 Mach (whichever is lower).

### Landing Gear Operation Speeds

The maximum speed for landing gear operation or for flight with the landing gear extended is 150 KIAS.

### Wing Flap Operation Speeds

The maximum speed for wing flap operation or for flight with the flaps extended to beyond the FLAPS UP position is 150 KIAS.

### Air Brake Operation Speeds

The air brake may be operated at any airspeed to the limit of the the aircraft never-exceed speed (VMO). Exceeding a speed of 250 KIAS will induce larger elevator forces (greater than 10 kgs).

**CAUTION**

Operation of the air brake at high airspeeds will induce significant nose up pitch forces (greater than 10 kg). Operation of the air brake at high airspeeds whilst manoeuvring may result in excessive aircraft load factor.

### Manoeuvring Speed

The maximum manoeuvring speed (VA) for the aircraft is:

a. 210 KIAS at weights below 2250 kg.

b. 200 KIAS at weights above 2250 kg.

Full or abrupt control deflections above the applicable speed is prohibited.

### CANOPY LIMITATIONS

The canopy must not be fully open during taxiing. At normal taxiing speeds, the canopy may be latched in the partially open position. If the canopy is left open at high taxiing speeds or with high engine power settings, damage to the canopy and associated locking mechanisms may result. Care should be taken when opening the canopy in a cross-wind.

**WARNING**

Opening the canopy in-flight will result in fatal or serious injury to the crew and structural damage to the aircraft.

**NOTE**

The canopy has been certified by Pilatus to provide a resistance to a 0.45 kg (1 lb) bird at a horizontal speed of 270 knots, and a 1.82 kg (4 lb) bird at 140 knots.

## PROHIBITED MANOEUVRES

The following manoeuvres are prohibited:

**a.** Intentional inverted spinning,

**b.** Flick manoeuvres, and

**c.** Spinning with power above IDLE and/or Air Brake extended.

**d.** Intentional tail-slides.

### Flight Under Negative Load Factor

Flight under negative load factor is limited to:

a. Inverted flight - 60 seconds maximum.

## FUEL ASYMMETRY LIMITATION

Maximum fuel asymmetry is 1/4 tank. Steps should be taken to correct the imbalance prior to reaching the limit. Aerobatic, spinning or stalling manouvres are prohibited while the imbalance is greater than 1/4 tank.

## ACCELERATION LIMITATIONS

### Maximum G In Clean Configuration

Service and structural limits (fatigue) are:

a. +7 maximum positive G and -3.5 maximum negative G for symmetrical manoeuvres at weights up to 2250 kg.

b. +4.5 maximum positive G and -2.25 maximum negative G for symmetrical manoeuvres at weights above 2250 kg.

### Non-Symmetrical (Rolling) `G' Limits

a. The maximum `G' for full aileron manoeuvres at weights up to 2250 kg is +4.6 positive and -2.3 negative.

**WARNING**

The G limitations stated are only an average for the stated conditions and exclude factors such as turbulence. For example, during a rolling pull-out a combination of high AUW, high speed and severe turbulence at or below the stated limits could result in structural failure.

**NOTE**

With two crew and full internal fuel the take off weight is approximately 2290 kg.

### Maximum G with Flaps or Landing Gear Extended

The maximum load factor for flight with flaps or landing gear extended is:

a. +2 maximum positive G.

b. 0 maximum negative G.

## WEIGHT LIMITATIONS

### Maximum Ramp Weight

The maximum ramp weight of the aircraft is 2710 kg (5962 lbs).

### Maximum Take-Off Weight

The maximum take-off weight of the aircraft is 2700 kg (5940 lbs). For normal training operations with two crew and 860 lb of fuel, or one crew and full fuel, the take-off weight is approximately 2250 kg (4960 lb).

### Maximum Landing Weight

The maximum landing weight of the aircraft is 2565 kg (5655 lb).

### Maximum Zero Fuel Weight

The maximum zero fuel weight of the aircraft is 1900 kg (4190 lb).

### Baggage Compartment

The maximum weight in the baggage compartment is 25 kg (55 lbs).

## CENTRE OF GRAVITY LIMITATIONS

At all weights up to 2250 kg:

**a.** Allowable forward CG limit - 4235.5 mm aft of datum, 22% MAC.

**b.** Allowable aft CG limit - 4367.5 mm aft of datum, 30% MAC.

At 2700 kg:

**a.** Allowable forward CG limit - 4244.0 mm aft of datum, 22.5% MAC.

**b.** Allowable aft CG limit - 4351.0 mm aft of datum, 29% MAC.

**NOTE**

- There is a straight line variation between 2250 kg and 2700 kg.
- The reference datum is located 3000 mm forward of the front face of the engine firewall.

## ENVIRONMENTAL LIMITATIONS

### Maximum Operating Altitude

The maximum operating altitude is 25,000 ft with a service ceiling of 38,000 ft.

### Operating Temperature Limits

The following temperature limitations apply for aircraft operations.

**a.** -55°C minimum, and

**b.** +50°C maximum.

During flight at outside temperature below +4°C the PROBES switch must be ON.

### Intake Icing

During flight at outside temperature below +4° with visible moisture present consideration should be given to pulling the INTK ICE handle.

## CROSSWIND LIMITS

### Maximum Take-Off Crosswind Component

The maximum crosswind component for take-off is 35 KTS. When performing a cross-wind take-off, power should be limited to the values in the following table:

| Crosswind Component | Maximum Recommended Torque before Brakes Release |
|---|---|
| 5 kts | 67.4 psi |
| 15 kts | 30 psi |
| 20 kts | 20 psi |
| >20 kts | Idle |

### Maximum Landing Crosswind Component

The maximum crosswind component for landing on a dry runway is 30 KTS.

## RUNWAY SURFACE LIMITATIONS

The minimum runway strength for operation on even, hard grass fields and hard sand is 0.38 Mpa (ICAO).

### TRAMPLING OF BAK-12 AAS CABLE (ref ARDU-TASK-0044)

Trampling of the BAK-12 AAS (with flat-type pendant cable support blocks fitted) by the PC9-A is recommended for all landing configurations in accordance with the flight manual limitations with the following additional restrictions:

**a.** Aircraft trampling weight is not to exceed 2310 kg.

**b.** Speed brake is to be retracted.

**c.** Power is to be constant or decreasing when trampling the cable.

**d.** Aircraft is to be in a stabilised ground roll prior to trampling.

**e.** Brakes are not to be applied while trampling the cable.

**f.** A minimum separation of five seconds is to be maintained between aircraft trampling the cable.

**g.** Only flat-type pendant cable support blocks are to be installed on the BAK-12 AAS for PC9-A trampling.

**CAUTION**

Aircraft fitted with external stores are NOT cleared to trample the BAK-12 AAS.

# RAAF SUPPLEMENT NO 7

# ENGINE OIL SYSTEM

## Introduction

**1.** Several incidents of high engine oil temperature and low engine oil pressure have occurred during high altitude operations of the PC9/A. Pilatus have advised that extended flight at an outside air temperature (OAT) less than -30°C may result in engine oil temperature rising higher than 90°C due to oil congealing in the oil cooling system. A copy of this supplement is to be inserted in two places; facing page 1-11 and page 5-8.

## Variation

**2.** High altitude operations are to be limited to a minimum indicated OAT of -20°C, unless operationally necessary.

**Reference:** DOAT 10/502/1/Air Pt5

**Date of Issue:** 6 June 1996

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SECTION 6

# FLIGHT CHARACTERISTICS

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SECTION 6

# FLIGHT CHARACTERISTICS

## General

The stability and control characteristics of the aircraft are satisfactory throughout the flight envelope. Trim changes resulting from changes in engine power settings are significant, but can be readily corrected with the flight controls.

# STALLING

## General

Aircraft behaviour up to the stall is characterised by the activation of the aural stall warning at an AOA equivalent to stall plus 5 to 10 knots, and the onset of light pre-stall buffet 5 knots above the stall itself. Pitch attitude may be very nose high in power-on approaches to the stall. The aerodynamic stall is characterised by a very small nose down pitch change, a plus one to two knot jump in airspeed, and a wing drop. At the stall, any forward movement of the stick will produce immediate recovery, with the aircraft pitching nose down to just below the horizon. The approach to the stall with flap and landing gear down is similar to that experienced in the clean configuration but at reduced airspeed and with a slightly more nose down attitude. Full flap produces light to moderate buffet which tends to mask the onset of pre-stall aerodynamic buffet. At the stall the stick position is further forward than with the clean configuration.

## Stall Symptoms

The symptoms approaching a 1G stall are summarised as follows:

a. Below 100 kts the aircraft becomes less responsive to control inputs although positive control is available until the point of stall.

b. High nose attitude.

c. Airframe vibration and elevator buffet commencing 5 kts above the stall speed and intensifying as the stall develops.

d. The stall speeds at idle power are:

(1) clean - 75 kts.

(2) take-off flap - 70 kts, and

(3) land flap - 66 kts.

PC9/A unaccelerated stall indications are summarised as:

a. slight nose drop with wing drop.

b. airframe buffet, and

c. high rate of descent.

## Stall Recovery

The recovery procedure is:

Simultaneously:

a. apply full power,

b. control column forward until the buffet stops (sufficient to unstall the wings).

c. rudder to prevent further yaw (sideslip).

When the wings are unstalled,

d. level the wings with aileron,

e. recover from the dive to the climb attitude, and

f. Carry out the after take-off checks when visually climbing away or when the altimeter and VSI start increasing.

Recovery from the dive should be flown so as to achieve minimum height loss. This will be achieved by raising the nose to the pre-stall attitude while maintaining the buffet. The pitch up forces, resulting from the application of power, must be controlled or the aircraft may restall. As a guide during early stall recoveries, 85 to 90 kts should be used as the stage to commence raising the nose in the clean configuration. Recovery for the wings level stall may take up to 300 ft of altitude loss for an aircraft mass of 2250 kg and 500 ft for aircraft mass of 2700 kg.

### High Speed Stalls

Pre-stall audio warning is activated approaching maximum AOA. This is followed by light buffet and mild wing rock which can cause lateral overcontrolling.

**WARNING**

Continued aft stick movement after reaching maximum AOA results in rapid autorotative departure from controlled flight.

## SPINNING

### GENERAL

The aircraft is extremely reluctant to enter an unintentional spin. However, intentional erect spinning may be conducted at all aircraft gross weights. Intentional spinning should be conducted with the trims set to neutral and with a maximum FUEL QTY asymmetry of ¼ tank.

### Erect Spinning

At 80 KIAS, with the PCL at idle, apply full rudder in the intended spin direction and simultaneously move the control column centrally to the aft stop. As the control applications are made, the nose will rise to approximately 20° above the horizon and the aircraft will begin rolling and yawing in the direction of applied rudder. As the aircraft continues to roll, the nose will drop to almost the vertical before returning to the horizon at the completion of the first turn. The aircraft will normally complete a second rolling turn, however by the third turn the nose will settle at an attitude oscillating between 40 and 60° nose down. The exact attitude is affected by AUW and CG position. During the spin moderate oscillations in roll rate may occur, with the roll rate reducing to zero at times. This may appear as a `spin hesitation'. Each turn of the spin takes approximately 2.5 seconds, with a height loss of 350-400 feet. The airspeed indicator will normally show 110 and 115 KIAS for left and right spins respectively, however these indications are subject to pressure errors and the aircraft actual speed approximates 70 knots. At high altitude, spins will tend to be flatter with a higher rate of rotation. Rudder loads during extended spins are moderately high (55 daN) and some `tramping' in the rudder circuit may be experienced, especially during hesitation in the roll or yaw oscillations. The aural stall warning will sound throughout the spin. Spins which are inadvertently entered with power applied will have different characteristics to those stated. If the aircraft has a lateral fuel asymmetry, spinning in the opposite direction to the heavier tank will produce a smoother spin, with reduced roll oscillation. Spinning in the same direction as the heavier tank will produce a more oscillatory roll motion.

### Incipient Recovery

The first two turns of the spin can be considered the incipient phase, and recovery can be accomplished at any time during this phase by smoothly centralising all controls. The neutral position can be defined as when the control column handgrip is midway between the CRS and HDG set knobs on the EHSI control panel, at the distance at which the gustlock would engage. Entry to the incipient phase may be prolonged during mishandled departures, as typically there will be a period of balance between control forces, aircraft inertia, and aircraft stability. The aircraft will normally stabilise in the fully-developed spin within two turns. If airframe buffet persists or recurs after incipient recovery action has been taken, confirm that the elevator control is at the neutral position. Following incipient spin recovery action, the spin will normally cease within one further turn. If the spin continues, the standard spin recovery action should be taken. Approximately 1500 feet of altitude will be lost between centralising the controls and recovery to wings-level, climbing flight if recovery action is initiated at the completion of the incipient phase.

### Erect Spin Recovery

The aircraft normally recovers within 1.5-2.5 turns after recovery controls are applied. When rotation has ceased, smoothly centralise the controls. Approximately 4500 feet is required to conduct a six turn spin and recovery. If an incorrect recovery technique is used, or the recovery controls are held after spin cessation, additional height will be lost and the aircraft may commence spinning in the opposite direction.

**CAUTION**

If the landing gear, flaps or air brake were extended prior to spin entry, retract the air brake only and if necessary accept a moderate overspeed of the landing gear and flap during the recovery dive.

### Spiral Manoeuvres

When full pro-spin controls are not maintained during spin entry, a descending spiral manoeuvre may result. This will be characterised by rapidly rising IAS and `g' loading. Spiral recovery is accompanied by centralising all flight controls until the turning stops, and then recovering from the dive.

# RAAF SUPPLEMENT NO 3

# MANOEUVRING FLIGHT

**Instruction**

1. This supplement is to be inserted facing page 6–3.

**Action**

2. The following is to be read in association with 'Manoeuvring Flight'.

**NOTE**

During manoeuvring flight using MCP or MP, the engine is susceptible to torque increases, especially after the aircraft has been rolled through the wings vertical position. This is a known characteristic of the electronic limiting system and is acceptable provided the Section 5 torque/ITT limits are not exceeded.

**NOTE**

During manoeuvring flight, especially when the aircraft has been rolled through the wings vertical position at less than 1 'g', torque indications displayed on the ESDP may be unreliable due to a decrease in the engine oil pressure at the torquemeter. This is a known deficiency with the torque indicating system and is acceptable provided the PCL is not increased until normal torque indications are regained.

**Date:** OCT 91

### Inverted spinning

The aircraft is not cleared for intentional inverted spinning. Due to the effect of high engine torque at low airspeeds, positive identification of an inverted spin may be difficult. As soon as control is lost all flight controls should be centralised. If the aircraft does not recover to controlled flight, the standard spin recovery action should be applied.

**WARNING**

Aircraft motion in an inverted spin is sometimes so erratic that interpretation of the turn needle may not be possible until all flight controls are centralised.

## FLIGHT CONTROLS

In 1 G flight the elevators are light and give precise pitch control throughout the aircraft's speed range. Stick forces increase with increasing airspeed and G. The ailerons are light and produce a good response throughout the flight envelope. Full aileron can easily be applied single-handed up to the maximum manoeuvre speed (VA) of 210 knots. A light spring in the aileron circuit improves stick centering for flight in turbulent conditions. The maximum roll rate of 130° per second is achieved at 210 knots. In inverted flight some lightening of aileron loads may be experienced below -2 G. A similar phenomenon may also be noticed if ice is allowed to build up on the aileron surface. The rudder circuit has a light to moderate centering spring. Rudder forces increase rapidly with increasing airspeed. The rudder is highly geared with only a short pedal movement and is very powerful throughout the flight envelope. The aircraft may be turned using the rudders alone, but response to control is slow.

**WARNING**

Large and rapid rudder deflections cause rapid pitch nose down due to blanking of the tailplane.

### Trims

The electrically activated elevator trim rate is moderately fast and during level acceleration or deceleration should be progressively trimmed to follow up the changing stick forces. During acceleration from very low speed at high power settings the trim rate may be too slow initially to maintain the aircraft in trim, and some stick force may have to be held manually. The electronic rudder trim rate is moderate, but too slow to respond to rapid power changes. At low speed and with high power, about one third right rudder or half rudder pedal defection and 40 kg foot load is required. During aerobatics and during manoeuvring flight the rudder trim should be set neutral and the aircraft balanced with the rudder until a stabilised flight condition is subsequently reached. The lateral and directional trim of the aircraft can only be accurately set during wings level stabilised flight. The electric aileron trim is fast acting but normally is set neutral and only used to balance the lateral effects of changed directional trimming.

### Air Brake

The air brake may be used at all speeds. At low speeds the air brake has little effect but at higher speeds the effect progressively increases. Above 250 KIAS operation of the air brake produces a moderate to high nose up trim change.

### Flaps

Operation of the wing flaps produces negligible pitch trim change.

## ENGINE POWER CHANGES

Directional trim of the aircraft is particularly sensitive to power changes. The selection of high power without correcting rudder will result in marked yaw left and sideslip, but no roll. If uncorrected, the resulting yaw will produce roll in the same direction. Movement of the PCL from idle is followed by a two to three second delay before the resulting engine power change. Above approximately 35 psi torque, engine acceleration is rapid and closely follows movement of the PCL. To maintain balanced flight throughout the range of engine accelerations, anticipation of the rate of engine power change is necessary. At low airspeeds, typically in the range used during the approach to land, operation of the engine below 3.5 psi torque produces a net propeller drag force rather than thrust. Overcontrolling of PCL during the approach to land may produce a noticeable thrust, drag and yaw oscillation with subsequent speed changes. The oscillation may be controlled by setting the PCL at one torque value until a stabilised speed is achieved. Optimum engine response will be produced by using small, smooth inputs with appropriate anticipation of power response.

### Climbing

Optimum climb performance will be attained by using the recommended climb speeds. Higher climb speeds may be used to achieve improved forward (over-the-nose) visibility and higher aerodrome departure speeds.

### Level Flight Characteristics with Various Airspeeds

At a constant power setting the directional trim of the aircraft is such that above approximately 120 KIAS the aircraft requires progressive small changes of rudder trim. Below 120 KIAS however, directional trim changes are greater for a given speed change such that more right rudder is required with reducing airspeed to maintain balanced flight.

## MANOEUVRING FLIGHT

### AEROBATICS

The aircraft is cleared for aerobatics excepting the manoeuvres listed at Section 5. Ample power is available to conduct aerobatics up to 25,000 feet.

**WARNING**

If control is lost whilst manoeuvring the PCL should be set to idle and the controls held firmly centered until stabilised flight is recovered.

The following aerobatic manoeuvres are permitted singly or in a combination:

a. Inverted Flight,

b. Lazy Eight,

c. Steep Turn,

d. Derry Turn,

e. Wing Over,

f. Slow Roll,

g. Aileron Roll,

h. Barrel Roll,

i. Climbing Half Roll,

j. Hesitation Roll,

k. Vertical Roll,

l. Stall Turn,

m. Loop Positive,

n. Cuban Eight,

o. Roll off the Top,

p. Rolling Turn,

q. Erect Spin,

r. Knife Edge.

## DIVES

As speed is increased towards the limit of 320 knot/MO.68 the aircraft's controls become progressively heavier. The rudder and rudder trim remain very powerful and should be used with care. Selecting the air brake out above 250 knots produces a moderately strong nose up pitching movement. If the IAS/Mach limit is inadvertently exceeded, the aircraft may exhibit a marked increase in longitudinal stick force per G. Longitudinal control remains effective. Dive recovery will be assisted by use of the elevator trim and selecting the air brake out, with anticipation for the subsequent nose-up pitching movement.

## FLIGHT IN TURBULENCE

The aircraft is very sensitive to turbulence; gust loads imposed on the aircraft increasing directly with the speed. At high speeds in turbulent air, gust loads may be very high. If the gusts occur during high-speed manoeuvring flight, the limit load factor may be exceeded. Additionally, considerable lateral movement of the aircraft may be experienced. The recommended turbulence penetration speed is 150 KIAS.

## SECTION 7

# ALL WEATHER OPERATION

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# SECTION 7

# ALL WEATHER OPERATIONS

## INTRODUCTION

In general, this section consists of procedures and information which differ from, or are supplementary to, the normal operating procedures in Section 2. In some cases, however, repetition has been necessary for emphasis, clarity, or continuity of thought.

## INSTRUMENT FLIGHT PROCEDURES

### Before Take-off

1. Aircraft - Line up visually with centreline of runway.
2. Heading references - Checked and heading bug set on runway heading.

### Instrument Take-off

After normal pre take-off checks:

1. Brakes - Release.
2. Direction - Maintain by external and/or compass reference.
3. Aircraft - Rotate at 80 KIAS.
4. When a positive rate of climb is indicated, carry out the after take-off checks.

### Instrument Climb

Normal climb techniques are suitable for instrument climbs.

### Instrument Cruising Flight

Normal cruise techniques are suitable for instrument flight. Maximum bank angle of 30° is normally used.

### Instrument Descent

Typical instrument descent configuration is:

1. PCL - 10 psi.
2. Air brake - As required.
3. IAS - 200 KIAS.

**WARNING**

The use of the aircraft airbrake may have a large and negative effect on the static pressure error. With the airbrake extended, actual altitude may be significantly less than indicated and terrain clearance reduced. Operation of the airbrake should be avoided during instrument final approach in Instrument Meteorological Conditions.

### Holding

Holding should be accomplished at 120 KIAS.The recommended angle of bank at 150 KIAS or less is 20°.

### NDB/VOR Approach

Refer to Fig 7-1 for NDB/VOR pattern.

### PRA/SRA

Refer to Fig 7-2 for PRA/SRA pattern. Descend to PRA pick-up altitude and transition to landing configuration as directed by PRA. Final approach configuration is:

a. PCL - As required (approx 13 psi).
b. Landing gear - Down.
c. Flap - Take-off.
d. IAS - 110 KIAS.

**Note**

A missed approach followed by a second PRA or TACAN requires a further 80 - 90 lb of fuel.

### ILS/LLZ Approach

Refer to Fig 7-3 for ILS/LLZ pattern.

### TAC Approach

Refer to Fig 7-4 for TAC pattern.

### Missed Approach Procedure

Simultaneously select:

a. PCL - MAX CRUISE Power.
b. Attitude - 10° NU. When the altimeter and VSI indicate a positive rate of climb:

1. Perform the After Take-Off Checks.
2. Reduce PCL to 40 PSI and adopt a 140 KIAS climb, maintaining a minimum rate of climb of 500 ft/min until missed approach complete.

## REDUCED VISIBILITY CIRCUIT

Low cloud base and poor visibility may preclude a circling approach at normal circuit heights. In this case follow the low level circuit procedure shown in Fig 7-5.

## ICING

The aircraft is susceptible to engine and airframe icing during flight in cloud or precipitation where ambient temperature is below 4°C. Symptoms of engine icing are loss of torque and increase in ITT. Resonance and vibration may occur, followed by a compressor stall or possibly a flameout.

**WARNING**

- Flight in visible moisture below +4°C indicated OAT is likely to cause engine icing and the INTK ICE handle should be pulled.
- Deliberate flight in known icing conditions is prohibited.

**Note**

Engine icing may occur without any unusual noises.

# NDB/VOR APPROACH

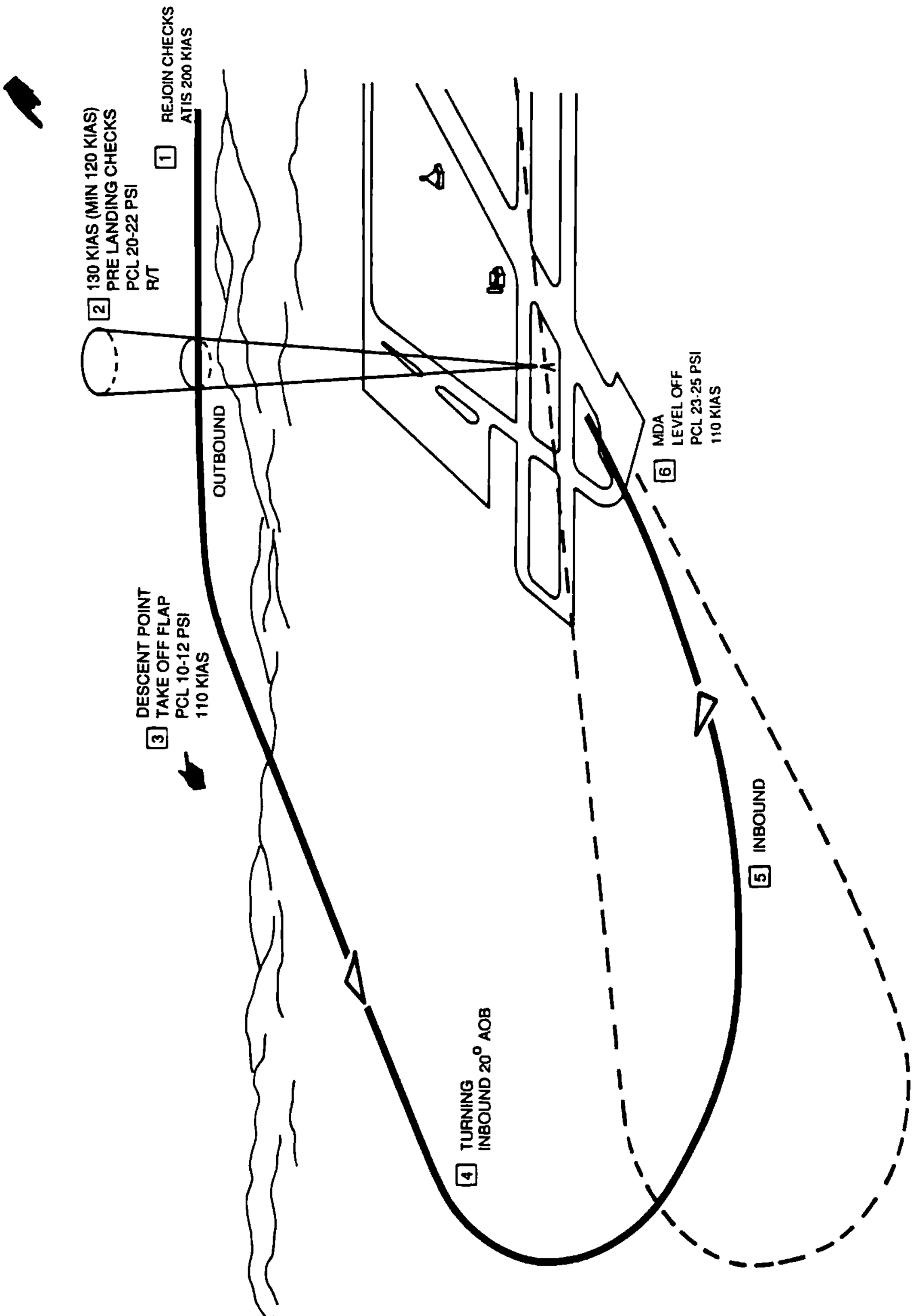



*Figure 7-1*

2 130 KIAS (MIN 120 KIAS)
PRE LANDING CHECKS
PCL 20-22 PSI
R/T

1 REJOIN CHECKS
ATIS 200 KIAS

3 DESCENT POINT
TAKE OFF FLAP
PCL 10-12 PSI
110 KIAS

OUTBOUND

4 TURNING
INBOUND 20° AOB

5 INBOUND

6 MDA
LEVEL OFF
PCL 23-25 PSI
110 KIAS

# PRA/SRA

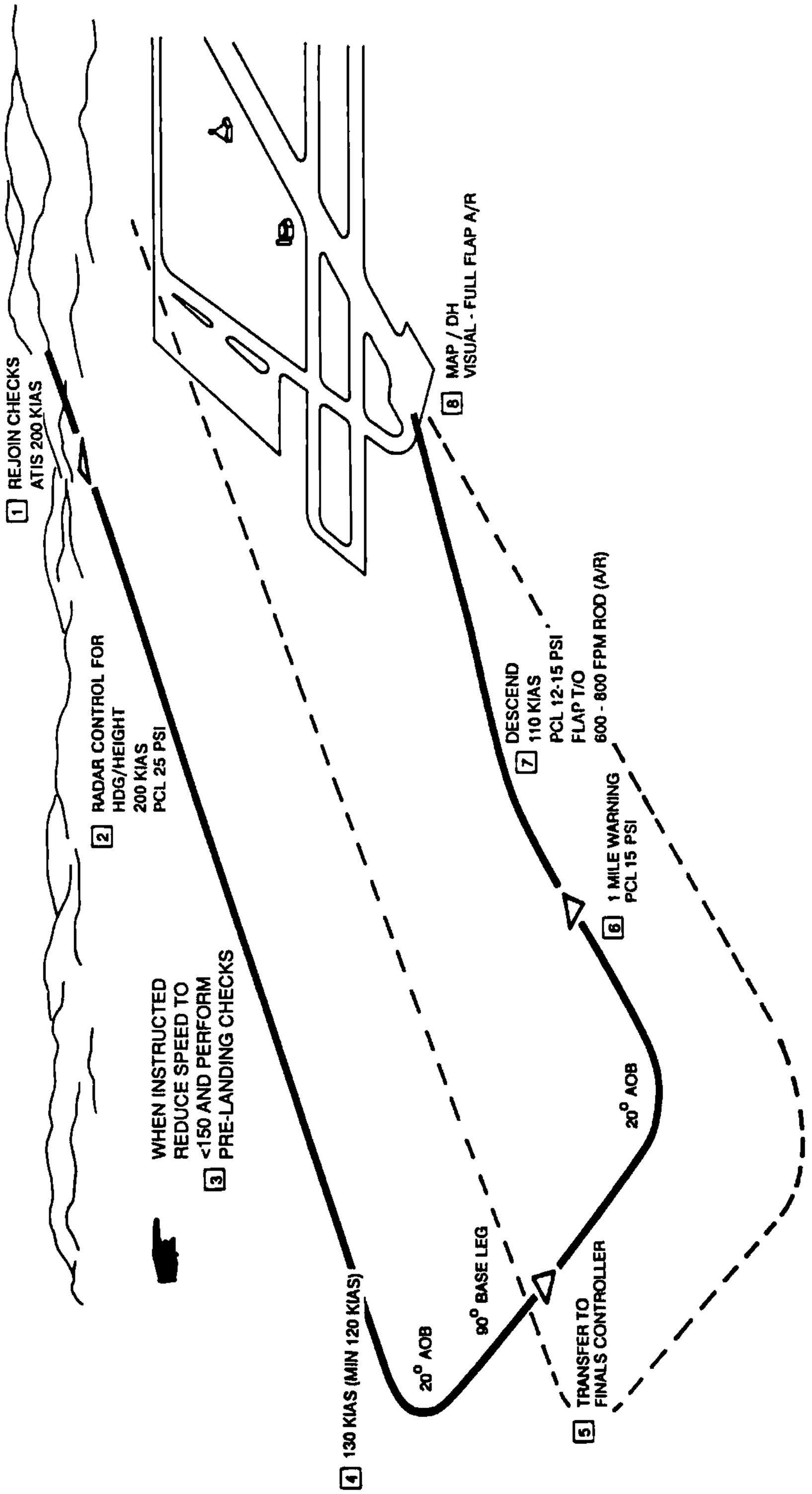



*Figure 7-2*

1 REJOIN CHECKS
ATIS 200 KIAS
2 RADAR CONTROL FOR
HDG/HEIGHT
200 KIAS
PCL 25 PSI
WHEN INSTRUCTED
REDUCE SPEED TO
3 <150 AND PERFORM
PRE-LANDING CHECKS
4 130 KIAS (MIN 120 KIAS)
20° AOB
90° BASE LEG
5 TRANSFER TO
FINALS CONTROLLER
20° AOB
6 1 MILE WARNING
PCL 15 PSI
7 DESCEND
110 KIAS
PCL 12-15 PSI
FLAP T/O
600 - 800 FPM ROD (A/R)
8 MAP / DH
VISUAL - FULL FLAP A/R

# ILS/LLZ APPROACH

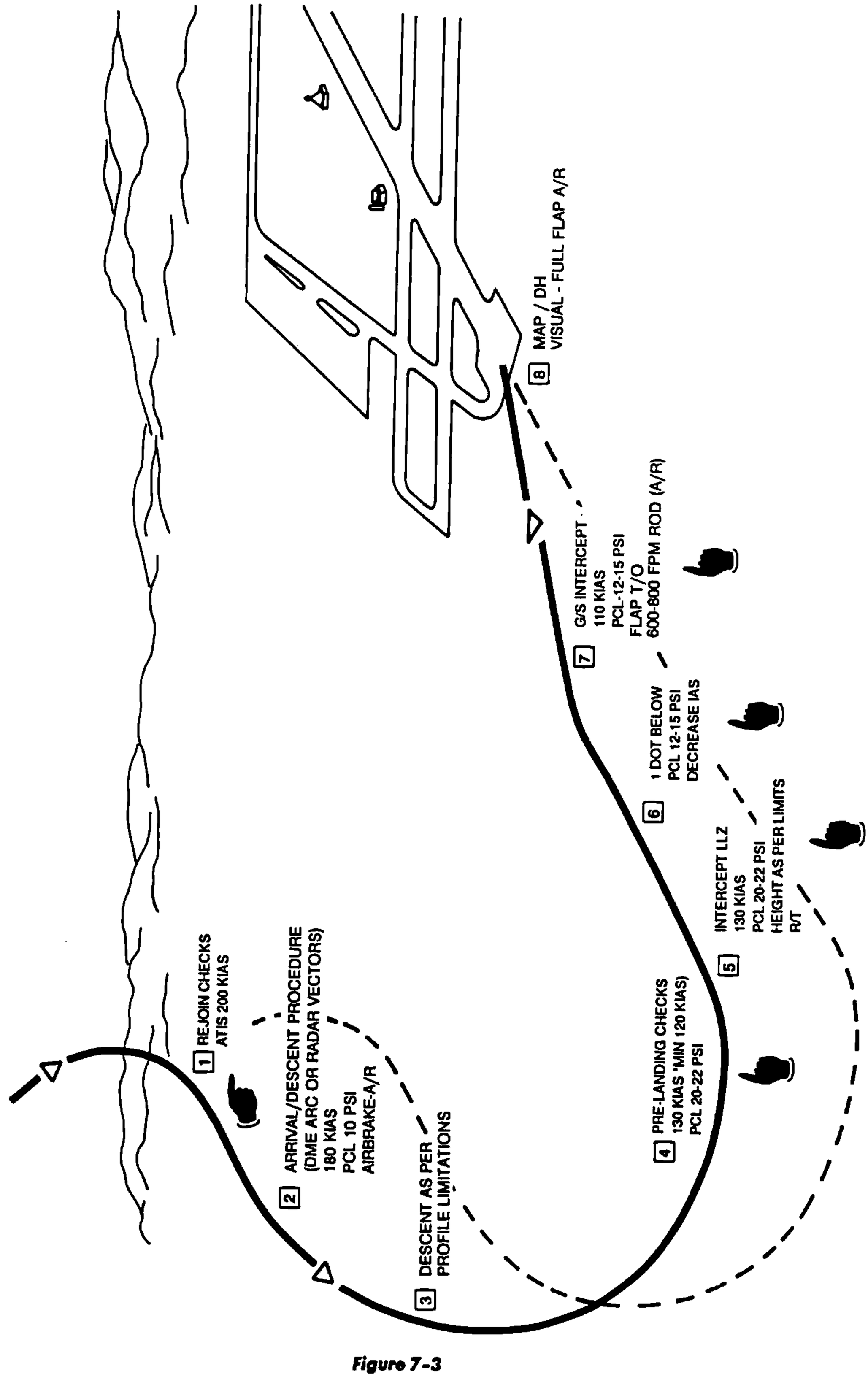



*Figure 7-3*

1 REJOIN CHECKS
ATIS 200 KIAS

2 ARRIVAL/DESCENT PROCEDURE
(DME ARC OR RADAR VECTORS)
180 KIAS
PCL 10 PSI
AIRBRAKE-A/R

3 DESCENT AS PER
PROFILE LIMITATIONS

4 PRE-LANDING CHECKS
130 KIAS *MIN 120 KIAS)
PCL 20-22 PSI

5 INTERCEPT LLZ
130 KIAS
PCL 20-22 PSI
HEIGHT AS PER LIMITS
R/T

6 1 DOT BELOW
PCL 12-15 PSI
DECREASE IAS

7 G/S INTERCEPT
110 KIAS
PCL-12-15 PSI
FLAP T/O
600-800 FPM ROD (A/R)

8 MAP / DH
VISUAL - FULL FLAP A/R

# TAC APPROACH

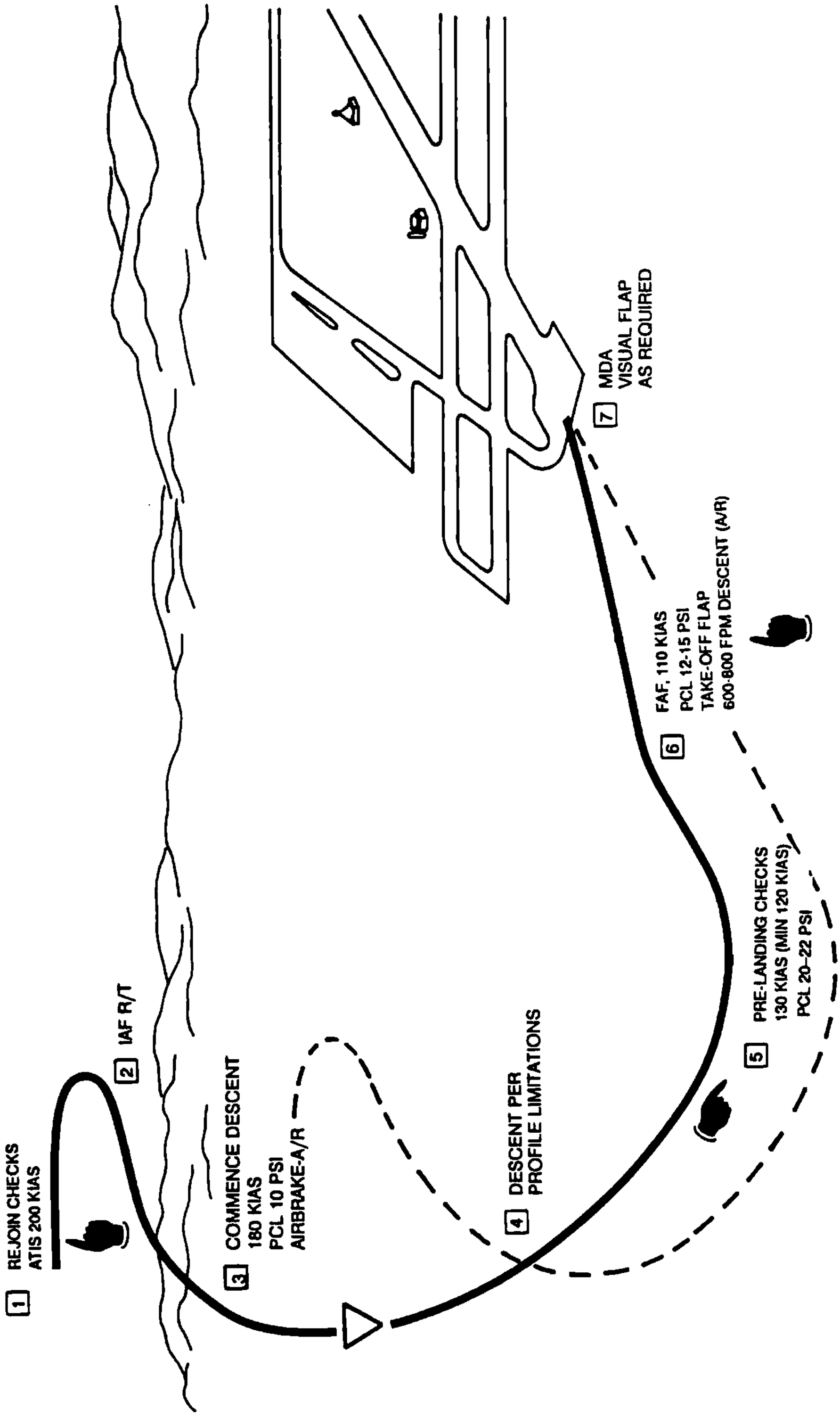
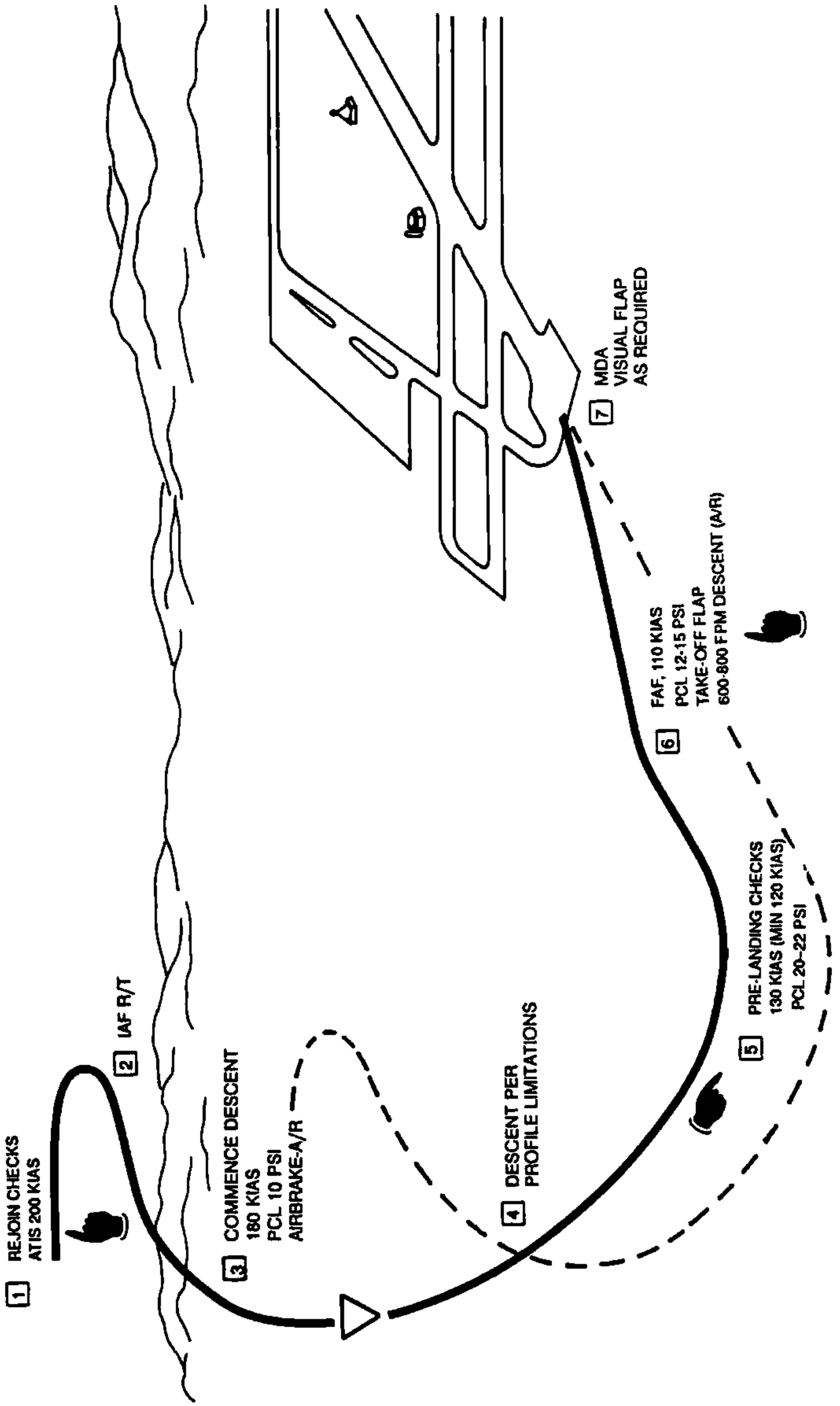


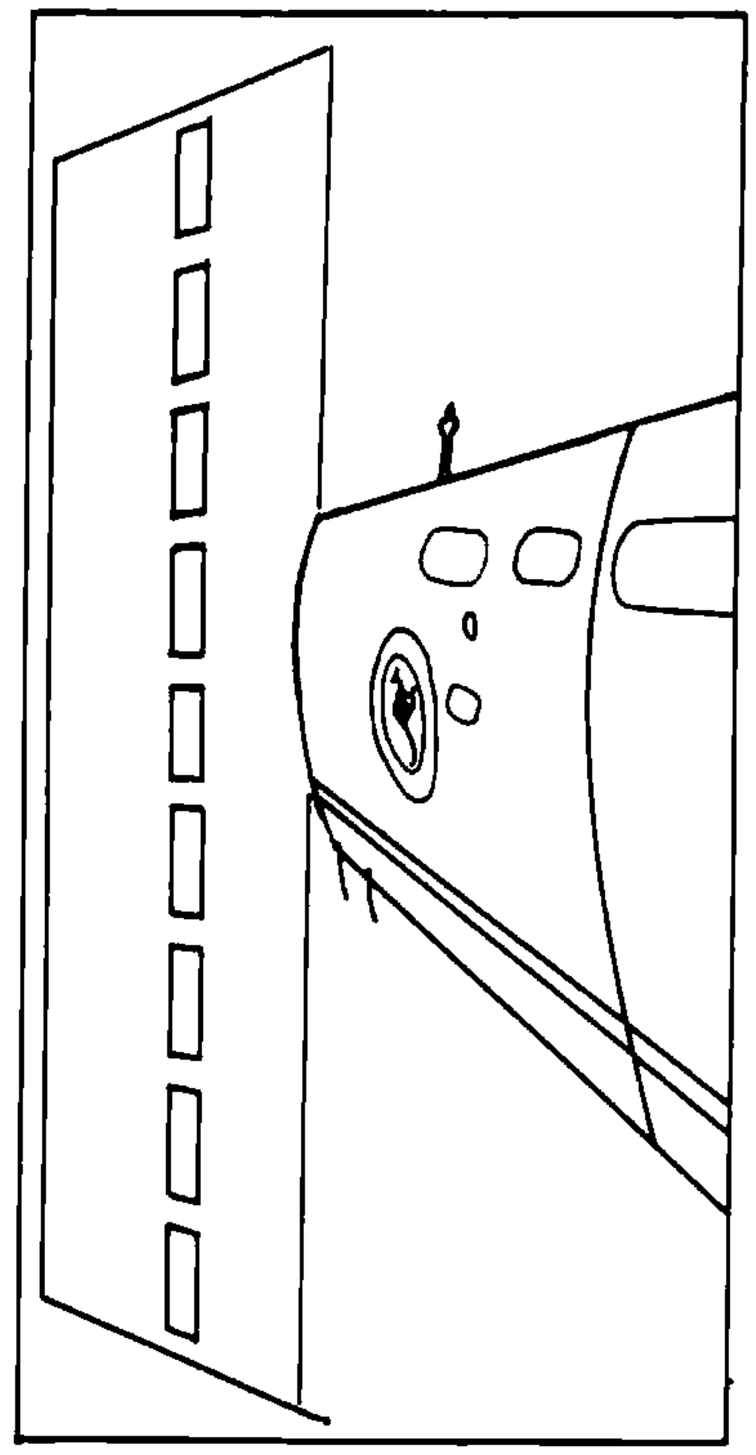

*Figure 7-4*

# TAC APPROACH



# LOW LEVEL APPROACH

| BASE TURN | |
|---|---|
| WIND KTS | CHORDS |
| 0 - 10 | 2.0 |
| 11 - 20 | 1.5 |
| > 20 | 1.0 |



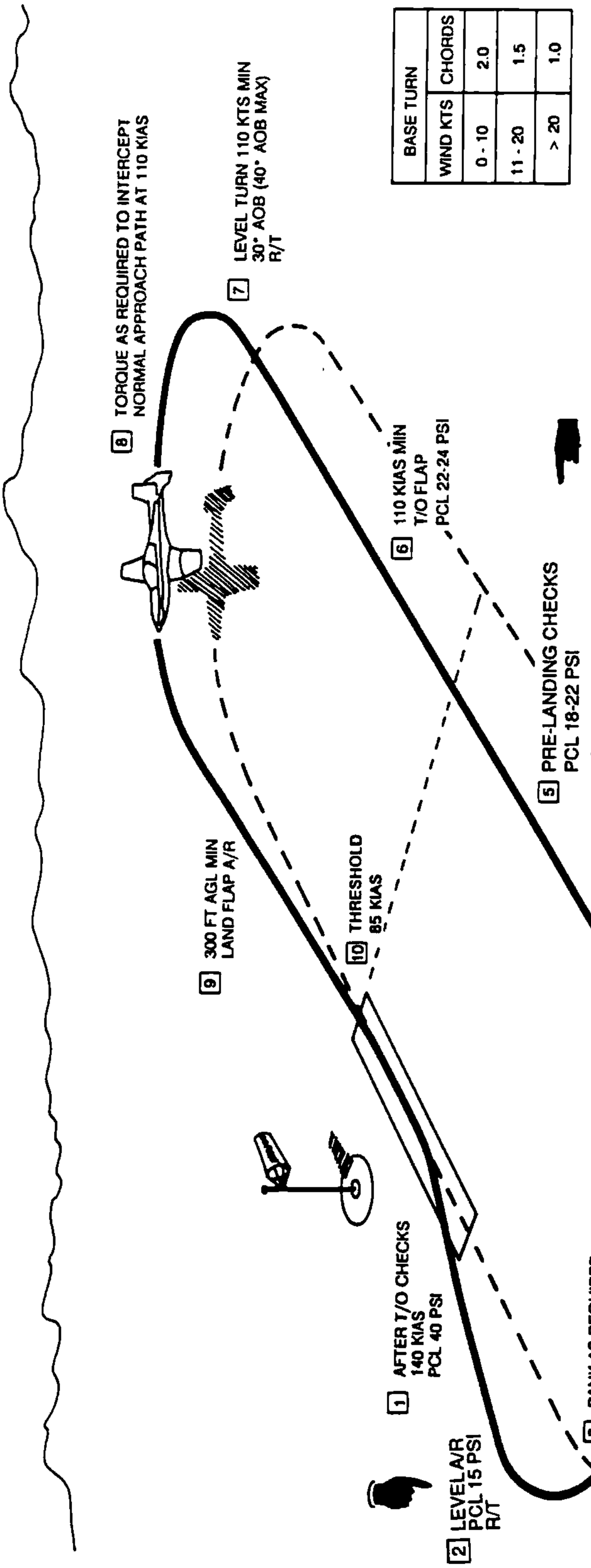
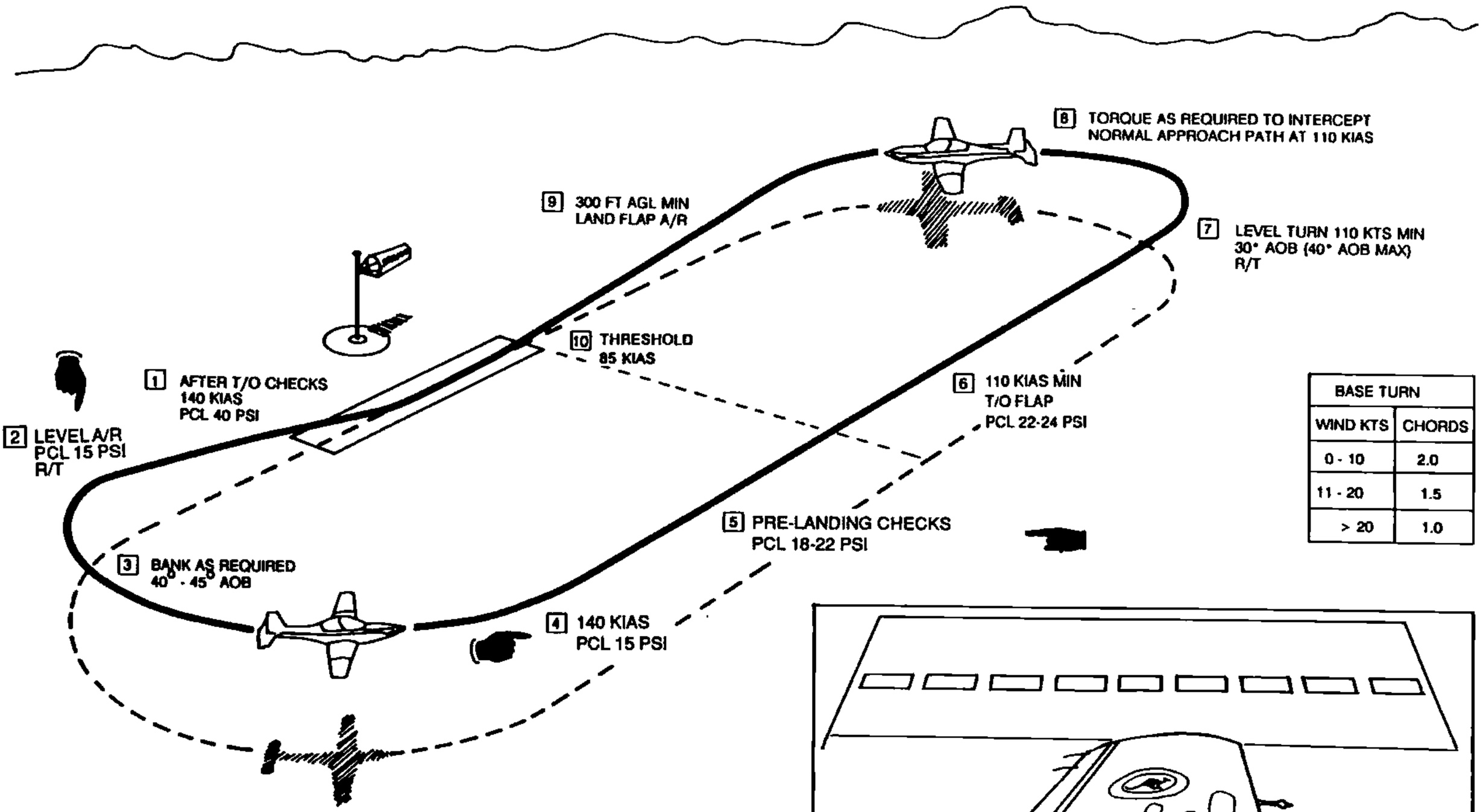




*Figure 7-5*

# LOW LEVEL APPROACH




| BASE TURN | |
|---|---|
| WIND KTS | CHORDS |
| 0 - 10 | 2.0 |
| 11 - 20 | 1.5 |
| > 20 | 1.0 |



DOWNWIND POSITION

## OPERATIONS ON WET RUNWAYS

Care must be exercised when operating on a wet runway. Aquaplaning can be expected above 67 kts (G/S) on take-off and when above 57 kts (G/S) on landing (based on a tyre pressure of 55 psi). Braking and harsh rudder movements should not be attempted when the aircraft is on a wet runway and above aquaplaning speeds.

## TURBULENCE AND THUNDERSTORMS

An airspeed of 150 KIAS is recommended for thunderstorm penetration and for operation in areas of known severe turbulence. To reduce vision deterioration from lightning, the cockpit lighting should be adjusted to full brightness before penetration.

**WARNING**

Flight through thunderstorm activity or known severe turbulence is not recommended and should be avoided where possible. If flight in the storm is unavoidable, attempt to maintain attitude and accept altitude and airspeed fluctuations.

## NIGHT FLYING

Techniques used in night flying do not differ appreciably from those used in daylight operation.

## COLD WEATHER OPERATIONS

### Before Entering Aircraft

**WARNING**

All accumulated ice and snow must be removed from the aircraft before flight is attempted. Ensure that water does not accumulate on control surfaces or other critical areas where refreezing may cause damage or binding.

**CAUTION**

To avoid damage to aircraft surfaces, do not permit ice to be chipped or scraped away. Remove all protective covers and duct bungs. Check to see that all surfaces, ducts, struts, drains and vents are free from snow, ice and frost. Ice and encrusted snow may be removed by using de-icing fluid or by direct air flow from a portable ground heater. Inspect the aircraft carefully for fuel and hydraulic leaks caused by the contraction of fittings or by shrinkage of packings. Inspect areas behind the aircraft to ensure that water or snow will not be blown onto personnel and equipment during engine start.

### Starting Engine

Use normal procedures to start the engine. Normal power settings may be used within the engine limitations (refer to Section 5).

**CAUTION**

Engine starting is not permitted with oil temperature below -40°C. Direct air flow from a portable ground heater may be used to raise the temperature prior to start.

### Taxiing

**WARNING**

Nose wheel steering and braking may not be completely effective when taxiing on ice or hard packed snow. Exercise care and taxi at reduced speed while operating on these surfaces. Increase the normal interval between aircraft to ensure safe stopping distance and to prevent icing of aircraft surfaces by melted snow and ice in the jet blast or propeller wash of preceding aircraft.

**CAUTION**

Painted areas on runways, taxiways and hard-stands are significantly more slippery than unpainted areas, particularly when wet. In addition, painted areas sometimes serve as condensation surfaces and it is possible to have wet, frosty or even icy conditions on those areas when the overall weather condition is dry.

### Take-off

Ensure that the take-off planning allows for reduced braking efficiency due to ice and snow on a runway in the event of an abort. Make a normal take-off.

### After Take-off

After take-off from a wet snow covered or slush covered field, operate the landing gear through several complete cycles to prevent gear freezing in the retracted position.

### Before Leaving the Aircraft

Leave the canopy partly open (weather conditions permitting); this will allow circulation within the cockpit to reduce canopy frosting.

## HOT WEATHER AND DESERT OPERATIONS

Hot weather and desert operations require additional precautions to be taken to prevent damage from dust, sand and high temperatures. During conditions of blowing sand and dust, the canopy should be closed and sealed and all protective covers installed when the aircraft is not in use.

**CAUTION**

- Do not attempt take-off or engine operation in a sand storm or dust storm if avoidable.

- During flight in dust or flying sand the INTAKE ICE handle should be pulled.

### Exterior Inspection

Inspect the exposed areas of the shock strut and actuator pistons on the landing gear and have them cleaned as required. Check all intakes for sand accumulation. Check tyres for signs of blistering, and check for over-inflation of tyres and struts due to high ambient temperatures. Check for fuel or hydraulic leakage due to thermal expansion of sealing materials. Inspect the area aft of the aircraft to make sure that propeller wash will not cause sand or dust to be blown onto personnel or equipment when the engine is started.

### Before Taxiing

Ground testing should be completed as rapidly as is practicable.

### Taxiing

Brakes should be used as little as possible to prevent overheating. The ECS may be selected to HIGH if full cold selection in LOW is insufficient to maintain temperature.

### Engine Acceleration

**WARNING**

**Failure to make allowance for the increased engine acceleration time when operating in high ambient air temperatures could lead to a hazardous situation, particularly in the landing pattern.**

### Cruise

At low altitude ensure that engine temperature limits are not exceeded.

### Approach and Landing

Maintain normal approach and landing speeds. Brake overheating may occur.

**CAUTION**

**High surface temperature conditions may often result in unexpected gusts and wind shears near the ground.**

### Before Leaving the Aircraft

**CAUTION**

**Leave the canopy partially open (weather conditions permitting), if the aircraft is to be parked in locations with high humidity or excessive Diurnal temperature variation.**

### Taxiing

Brakes should be used as little as possible to prevent overheating. The ECS may be selected to HIGH if full cold selection in LOW is insufficient to maintain temperature.

### Engine Acceleration

**WARNING**

Failure to make allowance for the increased engine acceleration time when operating in high ambient air temperatures could lead to a hazardous situation, particularly in the landing pattern.

### Cruise

At low altitude ensure that engine temperature limits are not exceeded.

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### Before Leaving the Aircraft

**CAUTION**

Leave the canopy partially open (weather conditions permitting), if the aircraft is to be parked in locations with high humidity or excessive diurnal temperature variation.

APPENDIX 1

# PERFORMANCE DATA

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APPENDIX 1

# PERFORMANCE DATA

## PART 1 - INTRODUCTION

### GENERAL

Information contained in the Performance Data section is based on, and consistent with, the recommended operating procedures and techniques set forth elsewhere in this manual. The performance data are pessimistic because they are based on a hypothetical worst condition engine. This will mean that engines in good conditions will experience lower fuel flows for any given torque settings. When operating on automatic temperature limiting at higher OAT or altitude the engine will show a higher torque and a higher fuel flow than the data shown in the following tables. It is therefore necessary, especially for maximum cruise performance, to realise that the range may be affected if flown on automatic temperature limiting. If range is critical a reduction in power is recommended to achieve the fuel flows indicated in the table. It is also recommended for maximum range performance to set the torque given in the appropriate table to produce the given speed. A slightly lower fuel flow may result.

### ATMOSPHERE

ICAO standard atmosphere is assumed throughout unless otherwise specified.

### FUEL GRADE

The performance data contained herein is based on engine operations using JET A1 with a specific gravity (SG) of 0.806 kg/L). At very low fuel states, to transfer the maximum possible quantity of fuel from the tanks to the engine, the aircraft must be flown in an erect, level attitude with no acceleration or sideslip. Steady pitch attitudes of up to 30 degrees nose up or 10 degrees nose down give an increase in unusable fuel of not more than 2 litres. An attitude 30 degrees nose down pitch increases unusable fuel by 10 litres.

### AIRSPEED POSITION ERROR CORRECTION

The speeds quoted in the performance tables are KCAS and KIAS. For KIAS zero instrument error has been assumed. The airspeed position error correction table is shown at Fig A1-1. To obtain calibrated airspeed (CAS) from rectified airspeed (RAS), enter the graph at the desired RAS and follow the speed line vertically until it intersects the desired configuration line. From the intersection read horizontally to the Position error correction. Add the position error to the initial RAS to obtain CAS. The following example highlights this procedure:

Example: Flaps up, LG up, speed 90 KTS RAS.

CAS = 90 + 3.4 = 93.4 kts CAS.

### IDENTIFICATION OF CHARTS

All performance data is based on the Pratt and Whitney Canada engine PT6A-62, flat rated to 950 SHP, as fitted to the Pilatus PC9/A aircraft. The propeller fitted is the Hartzell HC-D4N-2A/D9512A, with 4 blades, variable pitch, feathering type.

### STANDARD UNIT CONVERSION TABLE

The standard unit conversion table, shown at Fig A1-2, provides a means for direct conversion of temperature from degrees Celsius to degrees Fahrenheit, and of distance and speed from feet, nautical miles, feet per second, and feet per minute to the metric equivalents of metre, kilometres, metres per second, and metres per minute, respectively.

### SYMBOLS AND DEFINITIONS

**Reserved.**

### WEIGHT AND BALANCE

The centre of gravity datum for moment measurement is an imaginary point 3 metres forward of the firewall, along the extended aircraft longitudinal axis. Changes to equipment or structure which affect the aircraft weight and balance are to be recorded and a new basic aircraft empty weight and moment calculated. The basic empty weight of the aircraft includes all operating equipment that has a fixed weight, the airframe, powerplant, required equipment, fitted optional or special equipment, hydraulic fluid, oil, residual and unusable fuel. The permissible centre of gravity (CG) envelope for aircraft weights to 2250 kg is shown at Fig A1-3 and for aircraft weights to 2700 kg is shown at Fig A1-4.

### CALCULATION OF CG

To calculate if the aircraft weight is within the permitted limits, complete a CG calculation. Enter the moment (mkg) obtained from the CG calculation in the relevant CG envelope. The total aircraft weight must also be within the limits specified in Section 5. The fuel arms, for internal tanks only, are as follows:

a. Full Fuel (540 L/423.9 kg/933 lb) = 4.178 m,

b. 3/4 Fuel (405 L/317.9 kg/700 lb) = 4.154 m,

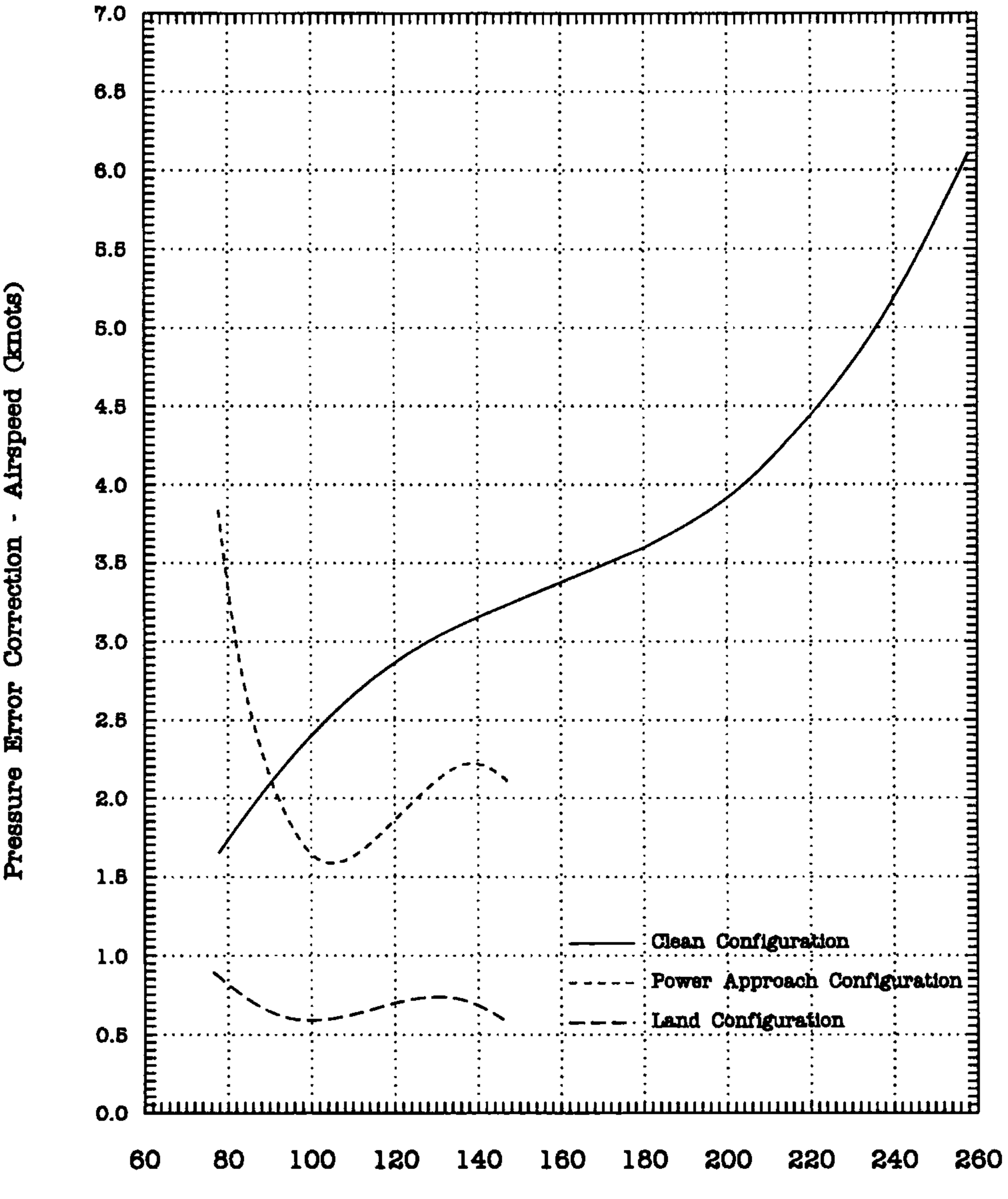
c. 1/2 Fuel (270 L/211.9 kg/467 lb) = 4.129 m,

d. 1/4 Fuel (135 L/106.0 kg/233 lb) = 4.065 m, and

e. 1/8 Fuel (70 L/55.0 kg/121 lb) = 3.700 m.

The arms for external stores are listed in Fig A1-5:

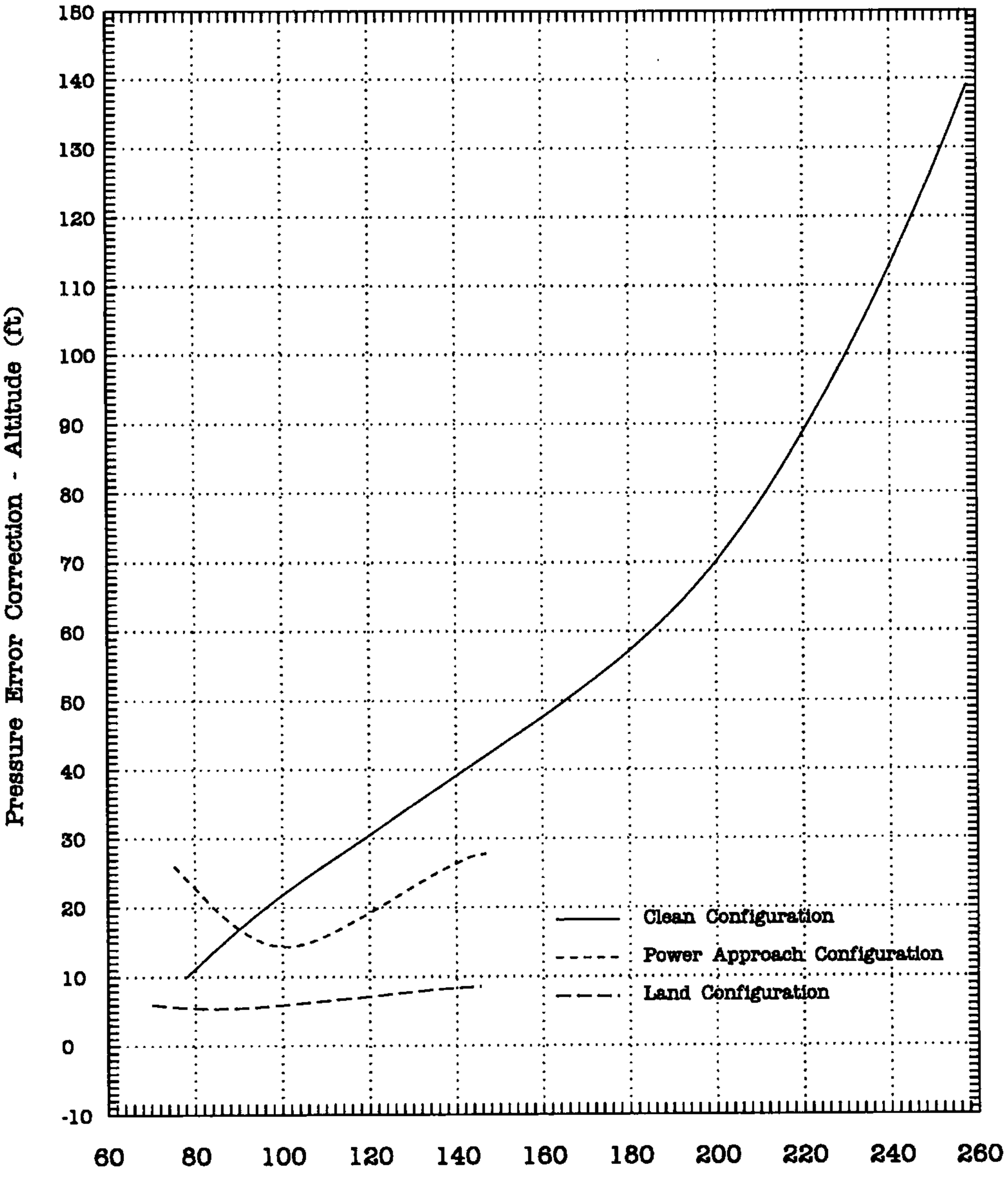
# PRESSURE ERROR CORRECTION - AIRSPEED




**NOTE:** Equivalent Airspeed = Rectified Airspeed + PEC.

**Figure A1-1**

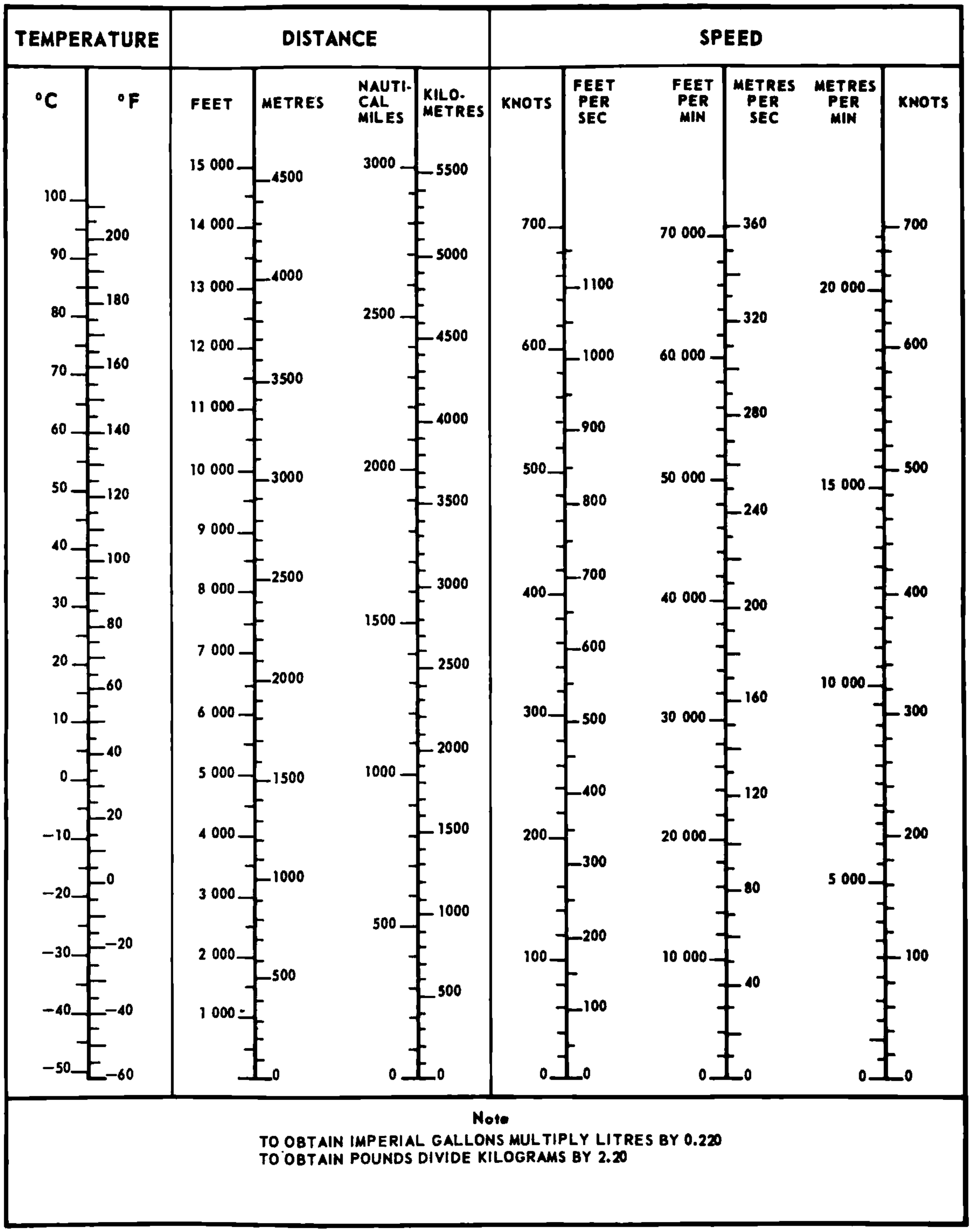
# PRESSURE ERROR CORRECTION - ALTITUDE




**NOTE:** True Altitude = Indicated Altitude + PEC.

**Figure A1-1A**

# STANDARD UNITS CONVERSION




**Note**

TO OBTAIN IMPERIAL GALLONS MULTIPLY LITRES BY 0.220
TO OBTAIN POUNDS DIVIDE KILOGRAMS BY 2.20

*Figure A1-2*

# CG ENVELOPE (2250 KG)

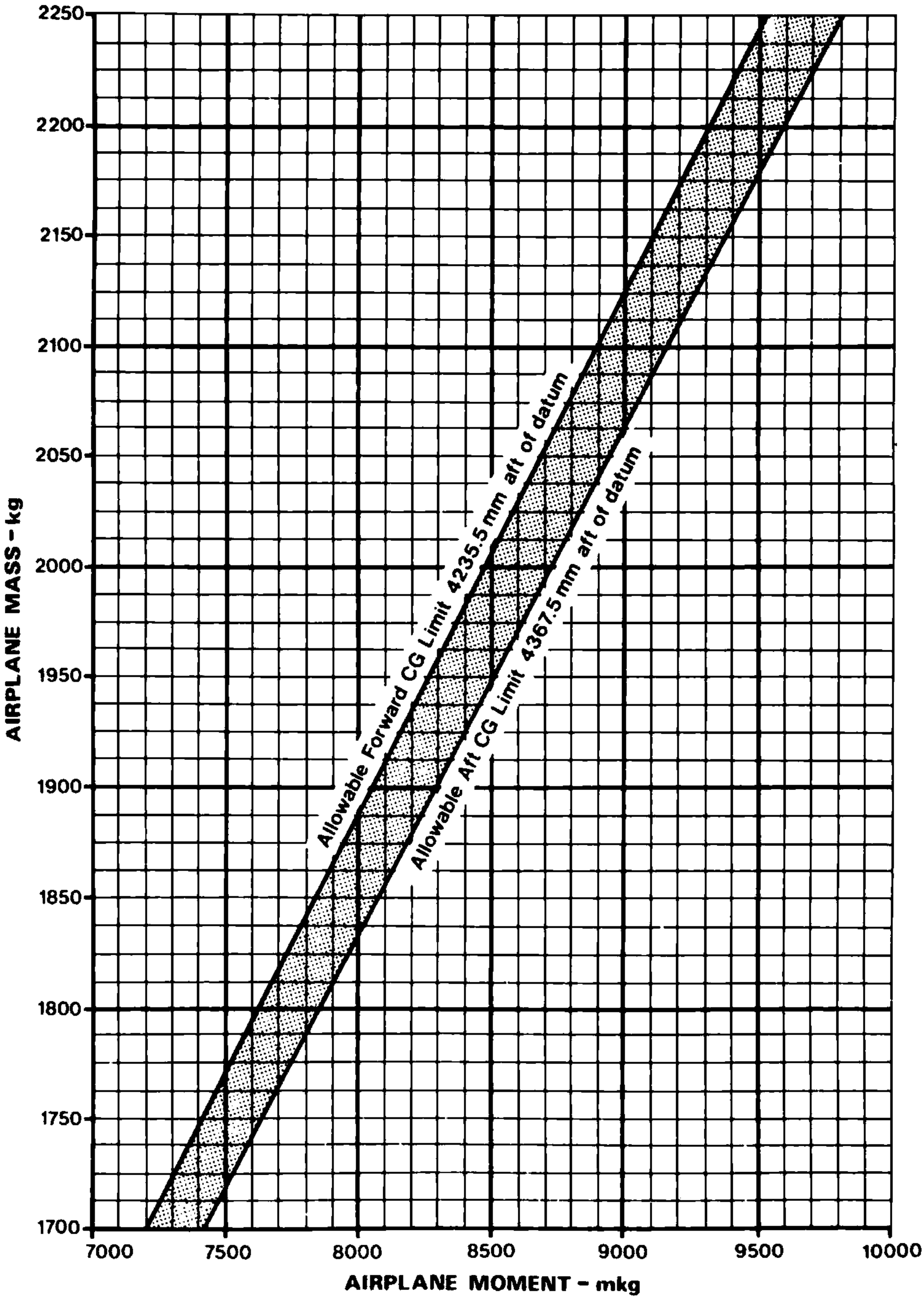



*Figure A1-3*

# CG ENVELOPE (2700 KG)

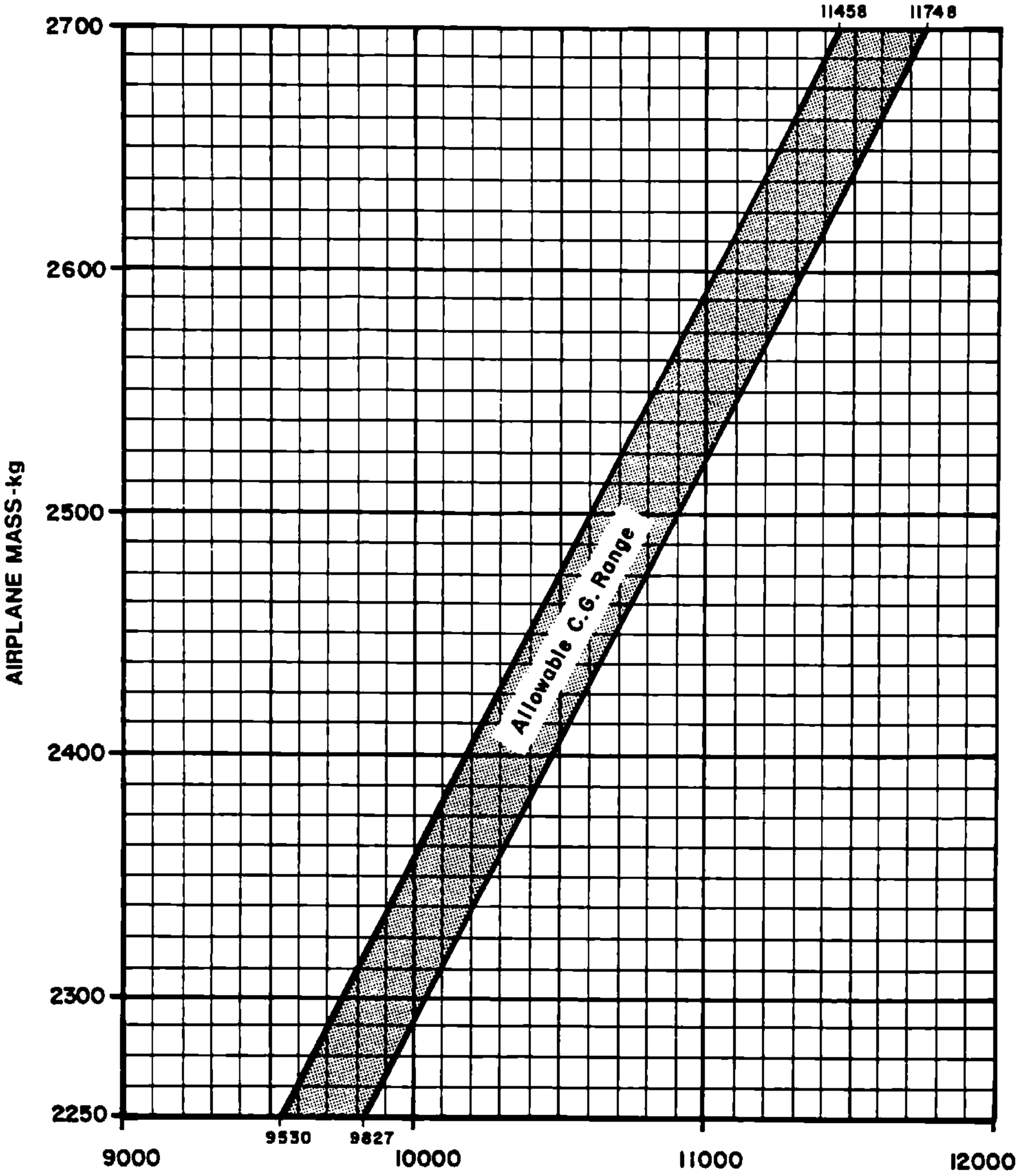



*Figure A1-4*

# EXTERNAL STORES

| | | MASS kg | ARM m | MOMENT kg/m |
|---|---|---|---|---|
| INNER STORE STATION | 1. PYLON (RACK) 6110 (including fairings) (ea) | 8.2 | 4.36 | 35.752 |
| | 2. FUEL TANK 254 L - EMPTY | 31.5 | 4.47 | 140.805 |
| | 3. FUEL TANK 254 L - FULL (439 lb) (inc tanks) (ea) | 236.2 | 4.26 | 1006.212 |
| MID STORE STATION | 4. PYLON (RACK) 6110 (including fairings) (ea) | 8.2 | 4.37 | 35.834 |
| | 5. FUEL TANK 254 L - EMPTY | 31.5 | 4.48 | 141.120 |
| | 6. FUEL TANK 254 L - FULL (439 lb) (inc tanks) (ea) | 236.2 | 4.27 | 1008.574 |
| OUTER STORE STATION | 7. PYLON (RACK) 6110 (including fairings) (ea) | 8.2 | 4.45 | 36.490 |
| | 8. EXT CONTAINER - EMPTY (ea) | 18.9 | 4.40 | 83.160 |
| | 9. EXT CONTAINER - MAX LOADED WITH TOOLS (ea) | 101.8 | 4.40 | 447.920 |

***Figure A1-5***

The most recent basic aircraft weight and moment, including all operating equipment that has a fixed location and is actually installed, can be found in the aircraft maintenance log book. The weight includes the weight of the airframe, powerplant, required equipment, fitted optional or special equipment, hydraulic fluid, oil, residual and unusable fuel. A sample CG calculation, for a fuel SG of 0.785 kg/L, is shown at Fig A1-6.

**TAKE-OFF AND LANDING WIND COMPONENT**

**Reserved.**

**STALLING SPEED**

Stalling speeds are provided for two configurations. Fig A1-7 gives stalling speeds for an aircraft weight of 2250 kg and idle power.

Fig A1-8 gives stalling speeds for an aircraft weight of 2700 kg and idle power.

# SAMPLE CG CALCULATION

| | KG | M | MKG |
|---|---|---|---|
| 1. BASIC EMPTY WEIGHT | 1701.6 | 4.267 | 7260.852 |
| 2. INTERNAL FUEL | 423.9 | 4.178 | 1771.054 |
| 3. PYLON (MIDDLE) (x2) | 16.4 | 4.360 | 71.504 |
| 4. EXT FUEL (INC TANK) | 472.4 | 4.270 | 2017.148 |
| 5. FRONT PILOT | 80.0 | 4.061 | 334.000 |
| 6. REAR PILOT (solo) | 0.0 | 5.486 | 0.000 |
| 7. BAGGAGE (max 25 kg) | 0.0 | 6.898 | 0.000 |
| **RAMP WEIGHT** | 2694.3 | 4.251 | 11454.558 |
| 8. LESS START/TAXY FUEL | −3.2 | 5.113 | −16.670 |
| **TAKE-OFF WEIGHT** | 2691.1 | 4.250 | 11437.888 |
| *** WITHIN WEIGHT LIMIT** | yes | *WITHIN CG LIMIT* | yes |

*** RESTRICTIONS:** Addition of a second pilot would require defuel of 71 kg (156 lb). Maximum 4.5 G and -2.25 G until fuel reduced by 444.3 kg (978.5 lb), to achieve authorised aerobatic weight of 2250 kg (4960 lb). Fuel weight is calculated as follows:

| | |
|---|---|
| Ramp fuel | 833.3 kg (1835 lb) |
| Fuel burn | 444.3 kg (978 lb) |
| **REMAINING** | **389.0 kg (857 lb)** |

***Figure A1-6***

# STALLING SPEEDS (2250 KG)

| Configuration | BANK ANGLE | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| | 0° | | 20° | | 40° | | 60° | |
| | KIAS | KCAS | KIAS | KCAS | KIAS | KCAS | KIAS | KCAS |
| LANDING | 65 | 70 | 67 | 72 | 77 | 80 | 97 | 99 |
| TAKE-OFF | 68 | 73 | 71 | 75 | 80 | 83 | 101 | 103 |
| CLEAN | 74 | 79 | 76 | 81 | 87 | 90 | 110 | 112 |

***Figure A1-7***

# STALLING SPEEDS (2700 KG)

| Configuration | BANK ANGLE | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| | 0° | | 20° | | 40° | | 60° | |
| | KIAS | KCAS | KIAS | KCAS | KIAS | KCAS | KIAS | KCAS |
| LANDING | 72 | 76 | 76 | 79 | 84 | 87 | 107 | 108 |
| TAKE-OFF | 77 | 80 | 80 | 83 | 90 | 92 | 112 | 113 |
| CLEAN | 82 | 86 | 85 | 89 | 95 | 98 | 120 | 122 |

***Figure A1-8***

# PART 2 - TAKE-OFF

## TAKE-OFF GROUND ROLL

Take-off ground roll is the distance from brake release to lift-off with the following assumptions:

a. Flaps - Take-off,

b. PCL - MAX power,

c. ECS - Low,

d. Hard surface runway, and

e. Rotate speed - 77 KIAS (81 KCAS) for aircraft weight of 2250 kg and 85 KIAS (88 KCAS) for aircraft weight of 2700 kg.

Take-off ground roll for operation on a soft surface runway increases by 12%. Take-off ground roll for an aircraft weight of 2250 kg is shown at Fig A1-9.

Take-off ground roll for an aircraft weight of 2700 kg is shown at Fig A1-10.

## *TAKE-OFF GROUND ROLL (2250 KG)*

| 2250 kg MASS | ALTITUDE FT | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|---|
| TAKE-OFF GROUND ROLL (m) | S. L. | 195 | 205 | 220 | 230 | 270 | 340 | 440 | 510 |
| Distance from brake | 2000 | 210 | 225 | 240 | 255 | 310 | 395 | 525 | 615 |
| release. Wing flap | 4000 | 235 | 250 | 265 | 295 | 360 | 465 | 625 | 745 |
| in TO position. | 6000 | 255 | 275 | 295 | 345 | 420 | 540 | 755 | 905 |
| Max. engine power. | 8000 | 285 | 305 | 340 | 400 | 485 | 630 | 915 | 1120 |
| ECS LOW. | 10000 | 315 | 345 | 395 | 460 | 565 | 740 | 1110 | 1350 * |

***Figure A1-9***

## *TAKE-OFF GROUND ROLL (2700 KG)*

| 2700 kg MASS | ALTITUDE FT | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|---|
| TAKE-OFF GROUND ROLL (m) | S. L. | 330 | 350 | 380 | 400 | 450 | 540 | 680 | 780 |
| Distance from brake | 2000 | 360 | 390 | 410 | 450 | 520 | 630 | 800 | 930 |
| release. Wing flap in | 4000 | 400 | 430 | 460 | 510 | 600 | 720 | 960 | 1120 |
| TO position; max. en- | 6000 | 450 | 480 | 520 | 580 | 670 | 830 | 1140 | 1340 |
| gine power; ECS LOW | 8000 | 500 | 540 | 600 | 680 | 790 | 970 | 1350 | 1580 |

***Figure A1-10***

## TAKE-OFF DISTANCE

Take-off distance is the distance from brakes release to clear a 15 m obstacle with the following assumptions:

a. Flaps - Take-off,

b. PCL - MAX power,

c. ECS - Low,

d. Hard surface runway,

e. Rotate speed - 77 KIAS (81 KCAS) for aircraft weight of 2250 kg and 85 KIAS (88 KCAS) for aircraft weight of 2700 kg, and

f. Climb speed - 93 KIAS (95 KCAS) for aircraft weight of 2250 kg and 97 KIAS (100 KCAS) for aircraft weight of 2700 kg.

Take-off distance for operation on a soft surface runway increases by 12%. Take-off distance for an aircraft weight of 2250 kg is shown at Fig A1-11.

Take-off distance for an aircraft weight of 2700 kg is shown at Fig A1-12.

## ACCELERATION CHECK TIME

**Reserved.**

## REFUSAL SPEED

**Reserved.**

# TAKE-OFF DISTANCE (2250 KG)

| 2250 kg MASS | ALTITUDE FT | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|---|
| TAKE-OFF DISTANCE (m) | | | | | | | | | |
| Distance from brake | S. L. | 315 | 335 | 355 | 375 | 435 | 565 | 740 | 875 |
| release to clear | 2000 | 345 | 365 | 390 | 415 | 505 | 655 | 885 | 1060 |
| 15 m obstacle. Wing | 4000 | 375 | 405 | 430 | 480 | 585 | 770 | 1075 | 1315 |
| flap in TO position. | 6000 | 415 | 445 | 475 | 560 | 685 | 905 | 1320 | 1645 |
| Max. engine power. | 8000 | 455 | 490 | 545 | 650 | 795 | 1060 | 1635 | 2110 |
| ECS LOW. | 10000 | 505 | 550 | 635 | 750 | 930 | 1255 | 2045 | 2665 * |

***Figure A1-11***

# TAKE-OFF DISTANCE (2700 KG)

| 2700 kg MASS | ALTITUDE FT | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|---|
| TAKE-OFF DISTANCE (m) | | | | | | | | | |
| Distance from brake | S. L. | 520 | 550 | 600 | 630 | 710 | 870 | 1090 | 1250 |
| release to clear 15 m | 2000 | 570 | 620 | 650 | 710 | 820 | 1000 | 1280 | 1490 |
| obstacle. Wing flap | 4000 | 630 | 680 | 730 | 800 | 950 | 1160 | 1540 | 1800 |
| in TO position; max. | 6000 | 710 | 760 | 820 | 910 | 1060 | 1330 | 1830 | 2150 |
| engine power; ECS LOW | 8000 | 790 | 850 | 950 | 1070 | 1240 | 1560 | 2160 | 2530 |

***Figure A1-12***

## PART 3 - CLIMB

**BEST RATE OF CLIMB**

The best rate of climb performance data is based on the following assumptions:

a. Flaps - Up,

b. LG - Up,

c. PCL - MAX CRUISE power, and

d. ECS - Low.

Best rate of climb performance data for an aircraft weight of 2250 kg is shown at Fig A1-13.

# *BEST RATE OF CLIMB (2250 KG)*

Conditions: Mass at take-off 2250 kg, max. cruise power, LG and flaps UP, ECS LOW.

| ALTITUDE FT | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|
| | fpm | fpm | fpm | fpm | fpm | fpm | fpm | fpm |
| S. L. | 4200 | 4175 | 4150 | 4100 | 3450 | 2600 | 1950 | 1600 |
| 2000 | 4175 | 4150 | 4100 | 4050 | 3275 | 2475 | 1800 | 1475 |
| 4000 | 4150 | 4100 | 4075 | 3800 | 3100 | 2350 | 1700 | 1325 |
| 6000 | 4125 | 4075 | 4050 | 3550 | 2900 | 2200 | 1550 | 1200 |
| 8000 | 4100 | 4050 | 3900 | 3325 | 2700 | 2050 | 1425 | 1075 |
| 10000 | 4075 | 4025 | 3650 | 3100 | 2500 | 1875 | 1300 | 950 |
| Airspeed | | | | | | | | |
| KIAS | 140 | 140 | 140 | 140 | 138 | 128 | 118 | 108 |
| KCAS | 142 | 142 | 142 | 142 | 140 | 130 | 120 | 110 |

***Figure A1-13***

Best rate of climb performance data for an aircraft weight of 2700 kg is shown at Fig A1-14.

**BEST ANGLE OF CLIMB**

The best angle of climb performance data is based on the following assumptions:

a. Flaps - Up,

b. LG - Up,

c. PCL - MAX CRUISE power, and

d. ECS - Low.

Best angle of climb performance data for an aircraft weight of 2250 kg is shown at Fig A1-15.

## *BEST RATE OF CLIMB (2700 KG)*

Conditions: Mass at take-off 2700 kg, max. cruise power, LG and flaps UP, ECS LOW

| ALTITUDE FT | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|
| | fpm | fpm | fpm | fpm | fpm | fpm | fpm | fpm |
| S.L. | 3280 | 3270 | 3260 | 3250 | 2650 | 2050 | 1400 | 1050 |
| 2000 | 3270 | 3260 | 3250 | 3150 | 2550 | 1950 | 1300 | 950 |
| 4000 | 3260 | 3250 | 3240 | 3050 | 2400 | 1800 | 1150 | 800 |
| 6000 | 3250 | 3240 | 3230 | 2900 | 2250 | 1650 | 1000 | 650 |
| 8000 | 3240 | 3230 | 3050 | 2700 | 2050 | 1450 | 800 | 450 |
| Airspeed | | | | | | | | |
| KIAS | 142 | 142 | 142 | 142 | 140 | 130 | 121 | 118 |
| KCAS | 144 | 144 | 144 | 144 | 142 | 132 | 123 | 120 |

***Figure A1-14***

## *BEST ANGLE OF CLIMB (2250 KG)*

Conditions: Mass at take-off 2250 kg, max. cruise power, LG and flaps UP, ECS LOW.

| | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|
| Airspeed KIAS | 88 | 88 | 88 | 88 | 88 | 88 | 88 | 88 |
| KCAS | 91 | 91 | 91 | 91 | 91 | 91 | 91 | 91 |

***Figure A1-15***

Best angle of climb performance data for an aircraft weight of 2700 kg is shown at Fig A1-16.

**GO-AROUND CLIMB PERFORMANCE**

Go-around rate of climb performance data is based on the following assumptions:

a. Flaps - Land,

b. LG - Down,

c. PCL - MAX power, and

d. ECS - Low.

Go-around rate of climb performance data for an aircraft weight of 2250 kg is shown at Fig A1-17.

## *BEST ANGLE OF CLIMB (2700 KG)*

Conditions: Mass at take-off 2700 kg, max. cruise power, LG and flaps UP, ECS LOW

| | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|
| AIRSPEED | | | | | | | | |
| KIAS | 92 | 92 | 97 | 97 | 97 | 97 | 97 | 97 |
| KCAS | 95 | 95 | 100 | 100 | 100 | 100 | 100 | 100 |

***Figure A1-16***

## *GO-AROUND CLIMB PERFORMANCE (2250 KG)*

Conditions: Aircraft mass 2250 kg, maximum power, LG and flaps in landing configuration, ECS LOW.

| ALTITUDE FT | | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|---|
| | | fpm | fpm | fpm | fpm | fpm | fpm | fpm | fpm |
| S. L. | | 2750 | 2725 | 2700 | 2625 | 2300 | 1800 | 1250 | 950 |
| 2000 | | 2700 | 2675 | 2650 | 2550 | 2150 | 1600 | 1075 | 775 |
| 4000 | | 2625 | 2600 | 2575 | 2400 | 1950 | 1400 | 900 | 625 |
| 6000 | | 2575 | 2550 | 2450 | 2175 | 1750 | 1250 | 700 | 450 |
| 8000 | | 2500 | 2450 | 2250 | 1975 | 1550 | 1050 | 500 | 300 |
| Airspeed for best rate | KIAS | 95 | 95 | 95 | 95 | 95 | 92 | 85 | 82 |
| | KCAS | 97 | 97 | 97 | 97 | 97 | 94 | 88 | 85 |
| Airspeed for best angle | KIAS | 78 | 78 | 78 | 78 | 78 | 78 | 78 | 78 |
| | KCAS | 81 | 81 | 81 | 81 | 81 | 81 | 81 | 81 |

***Figure A1-17***

Go-around rate of climb performance data for an aircraft weight of 2700 kg is shown at Fig A1-18.

**CLIMB PERFORMANCE DATA**

The climb performance data is based on the following assumptions:

a. Flaps - Up,

b. LG - Up,

c. PCL - MAX CRUISE power, and

d. ECS - Low.

## GO-AROUND CLIMB PERFORMANCE (2700 KG)

**Conditions: Aircraft mass 2700 kg, maximum power, LG and flaps in landing configuration, ECS LOW**

| ALTITUDE FT | | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 |
|---|---|---|---|---|---|---|---|
| | | fpm | fpm | fpm | fpm | fpm | fpm |
| | S.L. | 2150 | 2100 | 2050 | 2000 | 1670 | 1170 |
| | 2000 | 2100 | 2050 | 2000 | 1900 | 1500 | 1000 |
| | 4000 | 2050 | 2000 | 1950 | 1760 | 1320 | 810 |
| | 6000 | 2000 | 1925 | 1825 | 1600 | 1120 | 610 |
| | 8000 | 1950 | 1850 | 1650 | 1400 | 920 | 400 |
| Airspeed for best rate | KIAS | 98 | 98 | 98 | 98 | 98 | 98 |
| | KCAS | 100 | 100 | 100 | 100 | 100 | 100 |
| Airspeed for best angle | KIAS | 88 | 88 | 88 | 88 | 88 | 88 |
| | KCAS | 90 | 90 | 90 | 90 | 90 | 90 |

***Figure A1-18***

Climb performance data for an aircraft weight of 2250 kg is shown at Fig A1-19.

Climb performance data for an aircraft weight of 2700 kg is shown at Fig A1-20.

# CLIMB PERFORMANCE DATA (2250 KG)

Conditions: Mass at take-off 2250 kg, max. cruise power, LG and flaps UP, ISA, ECS LOW.

| ALTITUDE | SPEED FOR BEST RATE | | TIME* (FROM S.L.) | DISTANCE (FROM S.L.) | FUEL USED (FROM S.L.) |
|---|---|---|---|---|---|
| FT | KIAS | KCAS | MIN/SEC | NM | LBS |
| S. L. | 140 | 142 | - | - | 0 |
| 5000 | 140 | 142 | 1/20 | 4 | 14 |
| 10000 | 134 | 136 | 2/50 | 8 | 29 |
| 15000 | 127 | 129 | 4/30 | 13 | 45 |
| 20000 | 120 | 122 | 6/30 | 19 | 65 |
| 25000 | 114 | 116 | 9/30 | 26 | 86 |

* After lift-off

***Figure A1-19***

# CLIMB PERFORMANCE DATA (2700 KG)

Conditions: Mass at take-off 2700 kg, max. cruise power, LG and flaps UP, ISA, ECS LOW

| ALTITUDE | SPEED FOR BEST RATE | | TIME* (FROM S.L.) | DISTANCE (FROM S.L.) | FUEL USED (FROM S.L.) |
|---|---|---|---|---|---|
| FT | KIAS | KCAS | MIN/SEC | N.M. | LBS |
| 0 | 143 | 145 | - | - | - |
| 5000 | 143 | 145 | 2/00 | 4 | 17 |
| 10000 | 138 | 140 | 3/30 | 9 | 37 |
| 15000 | 133 | 135 | 5/30 | 16 | 56 |
| 20000 | 128 | 130 | 8/30 | 24 | 80 |
| 25000 | 118 | 120 | 12/30 | 35 | 110 |

* after lift off

***Figure A1-20***

## PART 4 - CRUISE

### MAX SPEED PERFORMANCE

The maximum speed performance data is based on the following assumptions:

a. Flaps - Up,

b. LG - Up,

c. Nil wind,

d. ISA environment, and

e. ECS - Low.

f. Initial fuel - 951 lb,

The fuel type used for the calculations was JET A-1, with an SG of 0.806 kg/L. The climb schedule used was that described in Climb Performance Data (this section). The descent profile used was 178 KIAS (180 KCAS) for aircraft weight of 2250 kg and 118 KIAS (120 KCAS) for aircraft weight of 2700 kg, 10 psi torque and air brake IN. Fuel reserves were 5% of total fuel plus 20 minutes holding in the endurance configuration. Maximum speed performance data for an initial aircraft mass at take-off of 2250 kg is shown at Fig A1-21.

Maximum speed performance data for an initial aircraft mass at take-off of 2700 kg is shown at Fig A1-22.

### MAXIMUM SUSTAINED G PERFORMANCE

**Reserved.**

## MAX SPEED PERFORMANCE (2250 KG)

| ALTITUDE | TORQUE | FUEL FLOW | AIRPSEED | | | RANGE | | ENDURANCE |
|---|---|---|---|---|---|---|---|---|
| FT | PSI | LBS/HR | KTAS | KIAS | KCAS | NM | KM | HR/MIN |
| S.L. | 63.8 | 590 | 269 | 267 | 269 | 352 | 652 | 1/19 |
| 5000 | 63.8 | 550 | 284 | 262 | 264 | 409 | 757 | 1/29 |
| 10000 | 58.7 | 500 | 291 | 250 | 252 | 470 | 870 | 1/40 |
| 15000 | 52.4 | 440 | 295 | 235 | 237 | 545 | 1009 | 1/56 |
| 20000 | 45.1 | 380 | 298 | 220 | 222 | 631 | 1169 | 2/14 |
| 25000 | 39.1 | 320 | 296 | 200 | 202 | 727 | 1346 | 3/35 |

***Figure A1-21***

## MAX SPEED PERFORMANCE (2700 KG)

| ALTITUDE | TORQUE | FUEL FLOW | AIRPSEED | | | RANGE | | ENDURANCE |
|---|---|---|---|---|---|---|---|---|
| FT | PSI | LBS/HR | KTAS | KIAS | KCAS | NM | KM | HR/MIN |
| S.L. | 63.8 | 590 | 269 | 267 | 269 | 358 | 663 | 1/19 |
| 5000 | 63.8 | 550 | 283 | 262 | 264 | 420 | 778 | 1/35 |
| 10000 | 58.7 | 500 | 288 | 247 | 249 | 500 | 926 | 2/1 |
| 15000 | 52.4 | 440 | 291 | 231 | 233 | 535 | 991 | 2/23 |
| 20000 | 45.1 | 380 | 290 | 212 | 214 | 622 | 1152 | 2/48 |
| 25000 | 39.1 | 320 | 289 | 194 | 196 | 793 | 1468 | 3/19 |

***Figure A1-22***

# PART 5 - RANGE

## MAX RANGE PERFORMANCE

The maximum range performance data is based on the following assumptions:

a. Flaps - Up,

b. LG - Up,

c. Nil wind,

d. ISA environment, and

e. ECS - Low.

f. Initial fuel - 951 lb,

The fuel type used for the calculations was JET A-1, with an SG of 0.806 kg/L. The climb schedule used was that described in Climb Performance Data. The descent profile used was 178 KIAS (180 KCAS) for aircraft weight of 2250 kg and 118 KIAS (120 KCAS) for aircraft weight of 2700 kg, 10 psi torque and air brake IN. Fuel reserves were 5% of total fuel plus 20 minutes holding in the endurance configuration. Maximum range performance data for an initial aircraft mass at take-off of 2250 kg is shown at Fig A1-23.

Maximum range performance data for an initial aircraft mass at take-off of 2700 kg is shown at Fig A1-24.

## *MAX RANGE PERFORMANCE (2250 KG)*

| ALTITUDE FT | TORQUE PSI | FUEL FLOW LBS/HR | AIRSPEED KTAS | AIRSPEED KIAS | AIRSPEED KCAS | RANGE NM | RANGE KM | ENDURANCE HR/MIN |
|---|---|---|---|---|---|---|---|---|
| S. L. | 29.5 | 390 | 202 | 200 | 202 | 401 | 743 | 1/59 |
| 5000 | 25.0 | 340 | 202 | 186 | 188 | 467 | 865 | 2/20 |
| 10000 | 21.5 | 285 | 202 | 172 | 174 | 566 | 1048 | 2/49 |
| 15000 | 20.0 | 245 | 204 | 161 | 163 | 665 | 1232 | 3/16 |
| 20000 | 19.0 | 215 | 205 | 148 | 150 | 752 | 1393 | 3/40 |
| 25000 | 18.2 | 190 | 208 | 139 | 141 | 847 | 1569 | 4/06 |

***Figure A1-23***

## *MAX RANGE PERFORMANCE (2700 KG)*

| ALTITUDE FT | TORQUE PSI | FUEL FLOW LBS/HR | AIRSPEED KTAS | AIRSPEED KIAS | AIRSPEED KCAS | RANGE NM | RANGE KM | ENDURANCE HR/MIN |
|---|---|---|---|---|---|---|---|---|
| S. L. | 30 | 395 | 204 | 202 | 204 | 400 | 741 | 1/57 |
| 5000 | 28 | 350 | 208 | 191 | 193 | 483 | 895 | 2/22 |
| 10000 | 25 | 305 | 208 | 177 | 179 | 578 | 1070 | 2/57 |
| 15000 | 22 | 260 | 208 | 164 | 166 | 686 | 1270 | 3/31 |
| 20000 | 21 | 225 | 208 | 151 | 153 | 794 | 1470 | 4/5 |
| 25000 | 21 | 205 | 216 | 144 | 146 | 893 | 1654 | 4/31 |

***Figure A1-24***

# PART 6 - ENDURANCE

## ENDURANCE

Endurance performance data is unavailable. The stated endurances in the maximum range performance section are airborne times while in the maximum range configuration and should not be confused with the endurance configuration.

# PART 7 - DESCENT

## CRUISE DESCENT

Cruise descent performance data is unavailable. Current procedures assume the following profile:

a. PCL - 35 PSI Torque,

b. Flaps - Up,

c. LG - UP,

d. Air brake - In, and

e. Airspeed - 250 KIAS.

## MAX RATE DESCENT

Maximum rate descent performance data is unavailable. Current procedures assume the following profile:

a. PCL - IDLE,

b. Flaps - Up,

c. LG - UP,

d. Air brake - Out, and

e. Airspeed - 230 KIAS.

This profile gives an approximate performance of 10,000 ft/min rate of descent.

# PART 8 - APPROACH AND LANDING

## LANDING DISTANCE

The landing distance is the distance an aircraft requires to land from a 15 m height to a full stop. The assumptions made in the performance data are:

a. Flaps - Land,

b. Moderate braking,

c. PCL - IDLE,

d. Green donut on AOA for approach, and

e. Hard runway.

The landing distances are increased by 13% for landing on a soft runway. The landing distances for an aircraft weight of 2250 kg are shown at Fig A1-25.

The landing distances for an aircraft weight of 2565 kg are shown at Fig A1-26.

## *LANDING DISTANCE (2250 KG)*

| 2250 kg MASS | ALTITUDE FT | ISA −30 | ISA −20 | ISA −10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|---|
| LANDING DISTANCE (m) | | | | | | | | | |
| Distance from 15 m | S.L. | 445 | 460 | 475 | 485 | 500 | 510 | 525 | 530 |
| obstacle to stop. | 2000 | 465 | 480 | 495 | 510 | 520 | 535 | 550 | 555 |
| Wing flap in LD | 4000 | 485 | 500 | 515 | 530 | 545 | 560 | 575 | 585 |
| position | 6000 | 510 | 525 | 540 | 560 | 575 | 590 | 605 | 615 |
| | 8000 | 535 | 550 | 570 | 585 | 605 | 620 | 640 | 650 |
| | 10000 | 560 | 580 | 600 | 620 | 635 | 655 | 680 | 690 |

***Figure A1-25***

## *LANDING DISTANCE (2565 KG)*

| 2565 kg MASS | ALTITUDE FT | ISA −30 | ISA −20 | ISA −10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|---|
| LANDING DISTANCE (m) | | | | | | | | | |
| Distance from 15 m | S.L. | 475 | 495 | 510 | 530 | 545 | 560 | 580 | 585 |
| obstacle to full | 2000 | 500 | 520 | 540 | 560 | 575 | 595 | 615 | 620 |
| stop. Wing flap in | 4000 | 530 | 550 | 570 | 590 | 615 | 635 | 655 | 665 |
| LD position | 6000 | 560 | 585 | 605 | 630 | 650 | 675 | 700 | 705 |
| | 8000 | 595 | 620 | 640 | 665 | 690 | 715 | 740 | 750 |

***Figure A1-26***

**LANDING GROUND ROLL**

The landing ground roll is the distance an aircraft requires to stop from touchdown to a full stop. The assumptions made in the performance data are:

a. Flaps - Land,

b. Moderate braking,

c. PCL - IDLE,

d. Green donut on AOA for approach, and

e. Hard runway.

The ground roll distances are increased by 13% for landing on a soft runway. Ground roll distances for an aircraft weight of 2250 kg are shown at Fig A1-27.

Ground roll distances for an aircraft weight of 2565 kg are shown at Fig A1-28.

## LANDING GROUND ROLL (2250 KG)

| 2250 kg MASS | ALTITUDE FT | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|---|
| LANDING GROUND ROLL (m) | | | | | | | | | |
| Distance from touch | S. L. | 325 | 340 | 350 | 365 | 375 | 385 | 400 | 405 |
| down to stop. | 2000 | 345 | 360 | 370 | 385 | 395 | 410 | 420 | 430 |
| Wing flap in LD | 4000 | 370 | 380 | 395 | 405 | 420 | 435 | 450 | 455 |
| position. | 6000 | 385 | 400 | 415 | 430 | 445 | 460 | 475 | 485 |
| | 8000 | 410 | 425 | 440 | 460 | 475 | 490 | 510 | 515 |
| | 10000 | 435 | 450 | 470 | 485 | 505 | 520 | 540 | 550 |

*Figure A1-27*

## LANDING GROUND ROLL (2565 KG)

| 2565 kg MASS | ALTITUDE FT | ISA -30 | ISA -20 | ISA -10 | ISA | ISA +10 | ISA +20 | ISA +30 | ISA +35 |
|---|---|---|---|---|---|---|---|---|---|
| LANDING GROUND ROLL (m) | | | | | | | | | |
| Distance to full stop | S L | 365 | 380 | 395 | 405 | 420 | 435 | 445 | 450 |
| Wing flap in LD posi- | 2000 | 385 | 400 | 510 | 430 | 445 | 460 | 470 | 480 |
| tion | 4000 | 410 | 425 | 440 | 455 | 470 | 490 | 505 | 510 |
| | 6000 | 430 | 450 | 465 | 485 | 500 | 515 | 535 | 540 |
| | 8000 | 460 | 475 | 495 | 515 | 530 | 550 | 570 | 575 |

*Figure A1-28*

# PART 9 - MISCELLANEOUS CHARTS

**Reserved.**