FLIGHT MANUAL
MIG-29

CHANGE NOTICE

LATEST CHANGED PAGES SUPERSEDE THE SAME PAGES OF PREVIOUS DATE.
INSERT CHANGED PAGES INTO BASIC PUBLICATION.
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This Flight Manual is incomplete without GAF T.O. 1F-MIG29-1CL-1.
See INDEX GAF T.O. 0-1-1A for current status of Flight Manual, Safety and Operational Supplements, Flight Crew Checklist.

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LIST OF EFFECTIVE PAGES

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With Change 4, the following supplements are considered to be incorporated into the manual, the status of the changed pages has been included in the List of Effective Pages, and the title pages of those supplements are to be destroyed:
GAF T.O. 1F-MIG29-1S-30; GAF T.O. 1F-MIG29-1S-35, GAF T.O. 1F-MIG29-1S-36, GAF T.O. 1F-MIG29-1S-37

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The information in this manual is reflected in the checklist GAF T.O. 1F-MIG29-1CL-1
*) The asterisk indicates pages changed, renewed, added or deleted by the current change.

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INFORMATION SUMMARY

- List of Safety and Operational Supplements -

1. List of incorporated supplements

With Change 4 the following supplements are considered incorporated into the manual:

Type and No. of Supplement	Date	Short Title	Affected Page(s)
GAF T.O. 1F-MIG29-1S-35	6 Mar 01	Changes to Limitations	1-62, 1-119B, 1-120, 3-19, 3-20, 5-5, 5-6, 5-9, 5-10, 5-12, 5-13, 6-5, 6-6
GAF T.O. 1F-MIG29-1S-36	3 May 01	Minimum Equipment List	5-14
GAF T.O. 1F-MIG29-1S-37	19 Jul 01	Taxi Checks	2-12, 2-13

2. List of supplements not incorporated

The following table lists supplements which atill have to be observed.
This table is to be updated by the holder.

Type and No. of Supplement	Date	Short Title	Affected Page(s)
GAF T.O. 1F-MIG29-1S-27	12 Apr 99	GPS / TSPI-POD	FSIntOS
GAF T.O. 1F-MIG29-1S-30	1 Sep 99	ICAO II and GPS	

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FOREWORD

Шек сикс фор МЙГс

SCOPE

This manual contains the necessary information for safe and efficient operation of the MiG-29 aircraft. These instructions provide you with a general knowledge of the aircraft and its characteristics and specific normal and emergency operating procedures. Your experience is recognized; therefore, basic flight principles are avoided. Instructions in this manual are for a crew inexperienced in the operation of this aircraft. This manual provides the best possible operating instructions under most circumstances. Multiple emergencies, adverse weather, terrain etc. may require modification of the procedures.

PERMISSIBLE OPERATION

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, which is not specifically permitted in this manual, is attempted.

HOW TO BE ASSURED OF HAVING LATEST DATA

Refer to GAF T.O. 0-1-1A for a listing of all current flight manuals, safety supplements, operational supplements, and checklists. Also, check the flight manual cover page, the title block of each safety and operational supplement, and all status pages contained in the flight manual or attached to formal safety and operational supplements. Clear up all discrepancies before flight. For the latest data refer to the INDEX GAF T.O. 0-1-1A, which is issued every three month and the status page of the latest supplement. If you have any questions about the date of issue, check with your supply personel.

SAFETY SUPPLEMENTS

Information involving safety will be promptly forwarded to you in a safety supplement. Urgent information is published in interim safety supplements. The supplement title block and status page should be checked to determine the supplement's effect on the manual and other outstanding supplements.

OPERATIONAL SUPPLEMENTS

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

HOW TO HANDLE THE SUPPLEMENTS

The supplements have to be inserted in the following order:
- Operational supplements on top of the flight manual and
- Safety supplements on top of the operational supplements.

Pen and ink changes in the manual and checklist are not authorized unless otherwise stated.
Write the number of the supplement alongside the effected portions of the flight manual.

CHECKLIST

The flight manual contains itemized procedures with necessary amplifications. The checklist contains itemized procedures without the amplification. Primary line items in the flight manual and checklist are identical. If a formal safety or operational supplement affects your checklist, the affected checklist page will be replaced by an interim change.

CHANGE SYMBOLS

The change symbol is a black line in the outer margin of the affected paragraph. It indicates text and tabular illustrations changes made to the current issue. Changes to illustrations (except tabular and plotted illustrations) are indicated by a pointing hand. Changes to the list of effective pages are indicated by an asterisk.

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WARNINGS, CAUTIONS AND NOTES

WARNING	Operating procedures, techniques, etc., which could result in personal injury or loss of life if not carefully followed.
CAUTION	Operating procedures, techniques, etc., which could result in damage to equipment if not carefully followed.
NOTE	Operating procedure, techniques, etc., which are considered essential to emphasize.

"SHALL", "WILL", "SHOULD" AND "MAY"

The words "shall" or "will" shall be used to express a mandatory requirement. The word "should" shall be used to express non-mandatory provisions. The word "may" shall be used to express permissiveness.

YOUR RESPONSIBILITY - TO LET US KNOW

Every effort is made to keep this manual up-to-date. However, we cannot correct an error unless we know of its existence. In this regard, it is essential that you do your part. For any questions and information use the following address:

LwMatKdo III A
Postfach 90 61 10 / 503
51127 Köln

DEFICIENCY REPORT AND PROPOSALS FOR CORRECTION OR IMPROVEMENT

Discrepancies and proposals for correction or improvement concerning this manual shall be reported to LwMatKdo I C 1 using AFTO Form 22, Publication Deficiency Report in three copies.

NOTE

Discrepancies in publications which endanger personnel or jeopardize Flight Safety have to be reported immediately by telex to LwMatKdo I C and to LwMatKdo III A.

LIST OF ABBREVIATIONS

A

A/A	Air to Air
A/C	Aircraft
A/D	Aerodrome
AB	Afterburner
AC	Alternating Current, Aircraft Commander
ACCRY	Accessory
ACFT	Aircraft
ACN	Aircraft Classification Number
ACS	Armament Control Switch
ADC	Air Data Computer
ADF	Automatic Direction Finding
ADI	Attitude Director Indicator
AFCS	Automatic Flight Control System
AGL	Above Ground Level
Ah	Ampere-Hours
AIL	Aileron
AIS	Aircraft Instrumentation Subsystem
ALT	Altitude
AJ	Active Jammer
Aj1, Aj2	Nozzle Area
AM	Amplitude Modulation
ANT	Antenna
AOA	Angel of Attack
AOB	Angle of Bank
AP	Autopilot
APS	Auxiliary Power System
APU	Auxiliary Power Unit
ASAP	As Soon As Possible
ATC	Air Trafic Control
ATT	Attitude
AUTO	Automatic

B

BAT	Battery
BIT	Built-In Test
BITE	Built-In Test Equipment
BRG	Bearing
BS	Boresight

C

C	Celsius
CAJ	Compensation Active Jammer
CAS	Calibrated Airspeed
CC	Close Combat
CCW	Counter Clockwise
CDP	Compressor Discharge Pressure
CG	Center of Gravity
CHAN	Channel
CIT	Compressed Intake Temperature
CL	Center Line
CMBT	Combat
CMPTR	Computer
COC	AOA Limiter System
COMP	Compass
CPI	Combined Pressure Indicator
CRIT	Critical
CRT	Cathode Ray Tube
CW	Clockwise

D

DASS	Defensive Aids Subsystem
DC	Direct Current
DI	Drag Index
DIM	Dimmer
DISCON	Disconnect
DLU	Acceleration Sensor
DME	Distance Measurement Equipment
DUSU	Angular Rate Sensor

E

E	East
EAS	Equivalent Airspeed
ECM	Electronic Counter Measures
ECP	Engine Control Pump
ECU	Engine Control Unit
EGT	Exhaust Gas Temperature
EMER	Emergency
EMERG	Emergency
ENG	Engine
ENG GBX	Engine Gearbox
EPM	Electronic Protection Measures
EXT	External, Extinguisher
ext	Extension

F

F/C	Front Cockpit
FAF	Final Approach Fix
FCS	Fire Control System
FDR	Flight Data Recorder
FHS	Front Hemisphere
FM	Frequency Modulation
FT, ft	Feet
FO	Foldout
FOD	Foreign Object Damage
FSP	Failure Simulation Panel

G

G	German
g	(Unit of) Gravity
GAF	German Air Force
GCA	Ground Controlled Approach
GEN	Generator
GENER	Generator
GBX	Accessory Gearbox
GND	Ground
GT	German Trainer

H

HDD	Head Down Display
HDG	Heading
HF	High Frequency
HMS	Helmet-Mounted Sight
HP	High-Pressure
HSI	Horizontal Situation Indicator
HUD	Head Up Display
HYD	Hydraulic
Hz	Hertz (Cycles per second)

I

I/C	Intercom
I/P	Identification of Position
IAS	Indicated Airspeed
ICAO	International Civil Aviation Organization
IFF	Identification Friend or Foe
IFR	Instrument Flight Rules
IGV	Inlet Guide Vane
ILS	Instrument Landing System
ILLUM	Illumination
IMC	Instrument Meteorological Conditions
IN	Inertial Navigation
in	Inch
INBD	Inboard
IP	Instructor Pilot
IR	Infrared
IRSTS	Infrared Search and Track System

J

JETT	Jettison

K

KG, kg	Kilogram
kHz	Kilohertz
KIAS	Knots Indicated Airspeed
KM, Km	Kilometer
KTAS	Knots True Airspeed
KTS, kts	Knots
kPa	Kilopascal
kp	Kilopond
kVA	Kilovolt-Ampere
kW	Kilowatt

L

L	Litre
LCN	Load Classification Number
LDR	Light Dependent Resistor
LG	Landing Gear
LDG, ldg	Landing
LED	Light Emitting Diode
LEF	Leading Edge Flaps
LH	Left Hand
LP	Low-Pressure
LPM	Limited Power Mode
LRF	Laser Range Finder

M

M	Mach
m, mtr	Meter
MAC	Mean Aerodynamic Cord
Mag	Magnetic
MAN	Manuell
MAX, max	Maximum
MDA	Minimum Descent Altitude
MHz	Megahertz
MIC	Microphone
MID	Middle
MIN, min	Minimum
min	Minutes
MIL	Military
MLG	Main Landing Gear
MPa	Megapascal
MRK	Marker
MSL	Mean Sea Level, Missile

N

N	North
NA	Not applicable
NAV	Navigation
NAVIG	Navigation
NDB	Non Directional Beacon
NE	Not established
NH	High-Pressure Compressor Speed
NL	Low-Pressure Compressor Speed
NLG	Nose Landing Gear
NM	Nautical Miles
NORM	Normal
NPM	Normal Power Mode
NWS	Nose Wheel Steering

O

OAT	Outside Air Temperature
OPT	Optical, Optimum
OUTBD	Outboard

P

Pa	Pascal
PAR	Precisson Approach Radar
PCN	Pavement Classification Number
PEC	Personal Equipment Connector
PH	Phase
PIO	Pilot Induced Oscillation
Pos	Position
PTO	DC/AC Converter
PTT	Press to Transmit
PWR, pwr	Power

Q

QFE	Barometric Pressure at Airfield Level
QNH	Barometric Pressure at Sea Level

R

R/C	Rear Cockpit
RCVR	Receiver
RDR	Radar
REC	Receive
RH	Right Hand
RHAW	Radar Homing and Warning
RHS	Rear Hemisphere
RNG	Range
RPM	Revolutions per Minute
RUD	Rudder
RWY, rwy	Runway

S

S	South
SAS	Stability Augmentation System
sec	Seconds
SIF	Selective Identification Feature
SL	Sea Level
SP	Simulation Panel
SPO	RHAW Receiver
SQLCH	Squelch
STBY	Standby
sw	Switch
SWL	Single Wheel Load
SYS	System

T

T/O	Takeoff
T/R	Transmit/Receive
TAC	Tactical
TACAN, TCN	Tactical Air Navigation
TAS	True Airspeed
Temp	Temperature
TGT	Target
TLP	Telelight Panel
TO/LD	Takeoff/Landing
TR	Transmit
TFU	Trim Feel Unit
TURB	Turbine
TWF	Track-While-Scan Feature

U

UHF	Ultra High Frequency

V

V	Volt
VENT	Ventilation
VFR	Visual Flight Rules
VHF	Very High Frequency
VIBR	Vibration
VIWAS	Voice Information and Warning System
VMC	Visual Meteorological Condition
VOL	Volume
VSI	Vertical Speed Indicator
VVI	Vertical Velocity Indicator

W

W	West
WCS	Weapon Control System
WDT	Wing Drop Tank
WP	Way Point

X, Y, Z

SECTION 1

DESCRIPTION AND OPERATION

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THE AIRCRAFT

The MiG-29 flown in the single-seat and double-seat (tandem) trainer version is a light-weight, high-performance, all-weather fighter interceptor, designed by the Mikojan company, with look-down / shoot-down and ground-attack capability.

Mission capability includes air defense with radar and infrared guided missiles and a 30 mm gun.

The aircraft is equipped with two Tumansky RD-33 dual-shaft, axial-flow turbofan engines with variable air intake ducts and variable exhaust nozzle sections.

Two air intakes are installed in nacelles below the wing roots. For foreign object damage (FOD) prevention, the air intakes are closed after landing and generally on the ground. With the intake ramps closed, engine air is taken in through a series of louvers in the upper surface of the wing root.

Each engine drives an associated engine gearbox (ENG GBX). Both ENG GBX are interconnected to the aircraft accessory gearbox (GBX). For engine start, an auxiliary power unit (APU) provides torque to the GBX which drives all accessories.

Normally, the thrust-to-weight ratio is greater than 1 (depending on the aircraft load and configuration). It enables high velocities, high rates of acceleration and high turn rates.

The aircraft shape is characterized by an integrated fuselage-to-wing design which forms an overall airfoil.

The almost flat bottom of the fuselage is an integrated part of the lower surface of the airfoil.

The aircraft structure comprises cantilever low-wing monoplane wings with leading edge flaps (LEF), trailing edge slotted flaps and ailerons. The tail of the cantilever structure includes two vertical stabilizers with small inset rudders and two tailerons. Dual irreversible hydraulic actuators position the control surfaces.

Electrical power is provided by an AC and a DC generator driven by a gearbox. Two batteries supply emergency power.

The fuel supply system incorporates internal fuselage and wing tanks, single point refueling and a fuel tank vent system. An external centerline tank (CL tank) and two wing drop tanks (WDT) can be installed.

The hydraulic power supply system provides pressure to the hydraulic actuators. Two separate and independent systems supply hydraulic pressure to the main and to the boost system. An emergency pump supplies pressure in the event of a main pump malfunction.

The pneumatic pressure supply consists of a main and an emergency system to control and pressurize aircraft systems.

The landing gear is hydraulically operated. It includes pneumatically powered brakes, anti-skid for all wheels and nose wheel steering.

A drag chute contained in the aft section of the fuselage significantly reduces landing roll distance.

A rocket-assisted ejection seat is designed to provide safe escape under minimum speed / zero altitude conditions. It is fully automatic throughout the ejection sequence.

The oxygen system is divided into a main and an emergency system. The main system supplies oxygen to the pilot during normal flight conditions and supports APU start and engine relight.

An emergency oxygen bottle is installed in the ejection seat to provide the pilot with emergency oxygen.

The pitot system includes a main and an emergency pitot boom. To prevent icing, both booms are electrically heated.

An angle of attack (AOA) limiter system and an automatic flight control system (AFCS) with automatic pitch control is incorporated.

The weapon delivery system comprises fire control, missile launchers and 30 mm gun.

The fire control system includes the pulse Doppler radar, an infrared search and track system (IRSTS) and a weapon computer.

Two digital computers process armament control data to provide displays of navigation, steering information and weapon aiming data to the head up display (HUD) and head down display (HDD).

A helmet mounted sight (HMS) system can be used to designate visually acquired targets to the radar, the infrared search and track system and to the infrared (IR) seekers of the missiles.

Target range information is provided by an integrated laser range finder (LRF) in conjunction with IRSTS operation.

Navigation equipment such as tactical air navigation (TACAN) is incorporated.

The aircraft has electronic protection measures (EPM) capabilities and a radar homing and warning receiver (RHAW).

A flare dispenser system is installed for protection against IR missiles.

Information and warning equipment is installed to attract the pilot's attention to failures in aircraft systems by audio and visual means i.e. telelight panel (TLP), voice information and warning system (VIWAS) and AEKRAN.

Flight data are continuously recorded for further processing after the mission.

A HUD camera is installed to record display and visual target information.

MIG-29 GT Trainer Version

The tandem-seat trainer version has a continuous framed canopy.

The GT has no radar but is capable of employing IR missiles and the 30 mm gun.

A radar simulation control panel is installed in the rear cockpit to display simulated targets into the HUD and the HDD. The emergency simulation control panels are deactivated.

For safe ground operation, a periscope system enables the rear occupant to have visual contact with the area in front of the aircraft.

AIRCRAFT GROSS WEIGHT

The approx. average gross weights are as follows:

	G	GT
Operating weight	11 001 kg	10 856 kg
Operating weight plus full internal fuel load	14 454 kg	14 409 kg
Operating weight plus full internal fuel load and full external centerline tank	15 775 kg	15 730 kg

AFTER MODIFICATION WITH WING DROP TANKS

	G	GT
Operating weight plus full internal fuel load, full centerline tank and two full wing drop tanks	17 906 kg	

NOTE

The operating weight includes the crew member (for GT two crew members) unusable fuel, oil and the gun without ammunition.

For detailed information, refer to GAF T.O. 1F-MIG29-5.

AIRCRAFT DIMENSIONS MIG-29 G

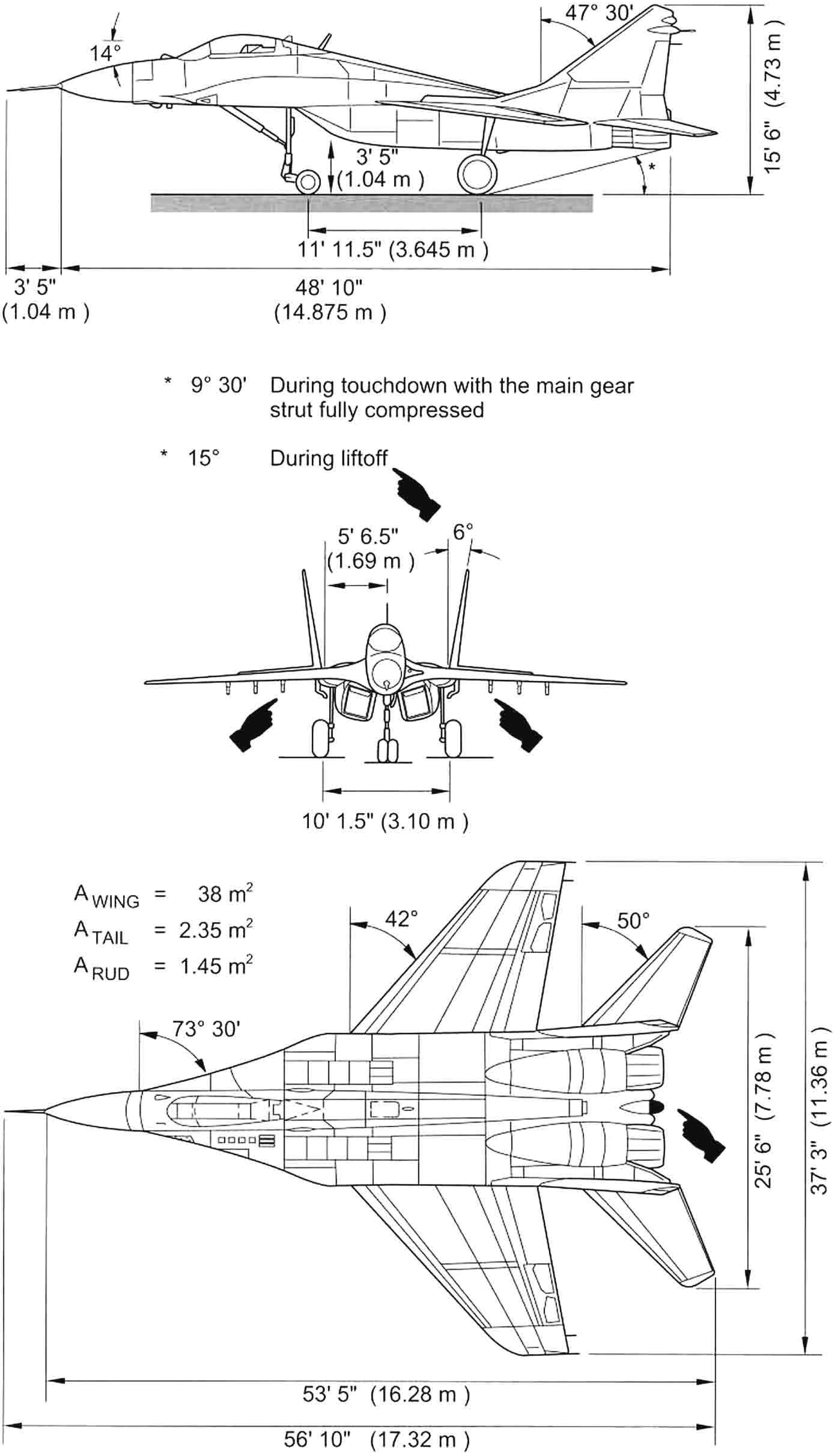


Figure 1-0

AIRCRAFT DIMENSIONS MIG-29 GT

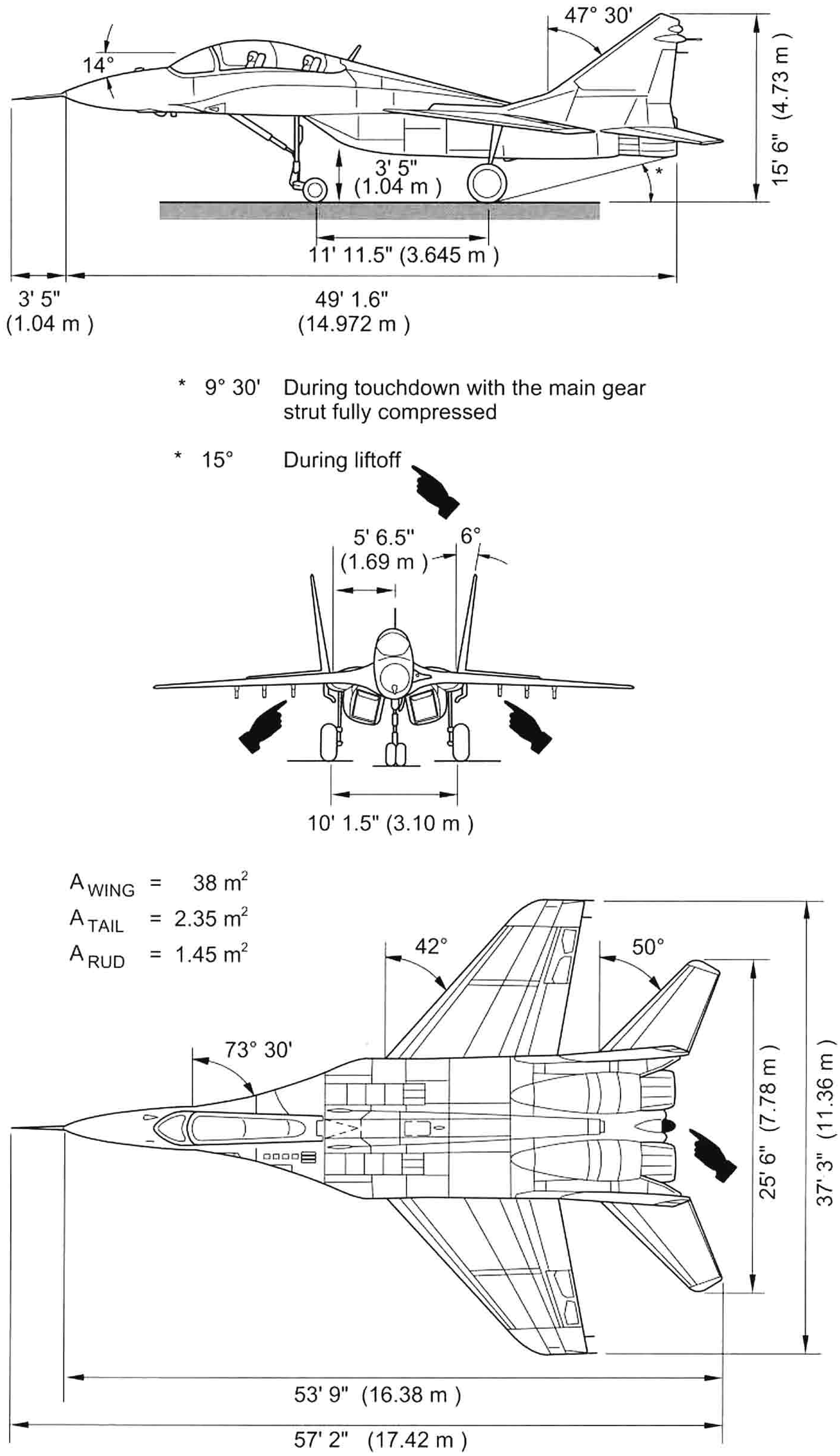


Figure 1-0A

ENGINES

The aircraft is powered by two Tumansky RD-33 thirteen-compressor-stage, dual-shaft, axial-flow turbofan engines. They are equipped with ring combustion chambers and afterburners, variable air intake systems and variable exhaust nozzles. Refer to figure FO 1-5.

For standard day / sea level conditions, the approximate static thrust ratings are as follows:

- NPM = Normal Power Mode
- LPM = Limited Power Mode

G / GT	NPM	LPM
Maximum AB thrust	8 300 kp	7 520 kp
Minimum AB thrust	5 600 kp	5 280 kp
MILITARY thrust	5 040 kp	4 680 kp
IDLE thrust	180 kp	180 kp

The two engines are mounted side by side in the aft section of the fuselage.

An APU gas turbine engine started by an electric motor is used to crank the engines for starting. Either the batteries or an external electrical power source can be used to provide electrical power during engine start.

The engines are supplied with separate intakes located below the wing roots. On the ground, air is provided through louvers in the upper surface of the wing roots since the variable ramps of the intakes are closed to prevent FOD.

In flight, engine air flow is controlled by variable ramps and variable stator vanes of the first two stages of the high-pressure (HP) compressor. It allows optimum engine performance over a wide range of aircraft operating conditions.

Engine air is routed through the four stages of the low-pressure (LP) compressor. After the LP compressor the airflow is divided into two streams, a hot main stream and a cold bypass flow.

The bypass air flows through an annular duct surrounding the HP compressor, the combustion chamber and the turbine section to rejoin the main flow in the air mixer of the afterburner (AB) section. The main stream flows through the nine stages of the HP compressor to the annular combustion chamber, where a controlled quantity of fuel is injected and ignited during start by ignitor plugs.

The hot high-pressure gas from the combustion chamber expands through the turbine section and mixes with the cold bypass stream for further expansion.

The turbine section of the engine consists of two single-stage turbines driving the HP compressor and the LP compressor. The two rotor shafts are mechanically independent of each other.

Engine speeds are indicated by a tachometer showing the HP compressor speed of both engines as a percentage of nominal maximum RPM.

During AB operation, additional fuel is injected into the hot gas stream by AB spray bars located in the AB chamber behind the turbine section, producing a substantial gain in thrust.

The exhaust nozzle area is fully variable and automatically controlled to obtain the desired thrust within engine operating limits.

The engine control unit (ECU) controls the hydro-mechanical equipment of the engine control system and supplies discrete fail signals to the warning equipment.

The engine system is described in the following paragraphs:

- Bleed air system
- Engine oil system
- Engine fuel system
- Engine control system
- Engine anti-surge system
- AB fuel system
- Exhaust nozzle system
- Engine air intake system
- Variable stator system
- Engine ignition system
- Engine starting system
- Engine AB system
- Throttles
- Engine controls and indicators
- Engine fire detection system
- Engine fire extinguisher system
- Engine operation

BLEED AIR SYSTEM

Engine compressor bleed air taken from the LP and HP compressors (refer to figure FO-5) at three locations is utilized for the following functions:

LP compressor:

- Fuel accumulator tank pressurization.
- External tank pressurization and fuel transfer.
- Internal tank pressurization.

HP compressor 5th stage:

- HP and LP turbine rotor and stator cooling.

HP compressor 7th stage:

- Air-conditioning and pressurization system.
- Anti-ice system of the LP compressor intake.

ENGINE OIL SYSTEM

Each engine is equipped with a self-contained, dry-sump full pressure oil system to provide circulation of oil for lubrication and cooling of the engine main bearings and of the ENG GBX. Venting of the bearings and of the oil tank is provided to prevent excessive pressure build-up. Refer to figure FO-4.

Oil is drawn from the oil supply tank by a main lube pump and delivered through a pressure filter to the accumulator and the oil pressure sensor. Tappings are provided to feed the three main engine bearings and the ENG GBX.

Oil from the engine main bearings is returned by scavenge pumps to the oil tank via two separate fuel-cooled oil coolers (engine and AB fuel systems) to cool the engine oil. Filters are fitted in the oil return lines in front of the scavenge pumps.

A magnetic chip detector to provide an indication of engine wear and warning of engine components breakdown and an oil temperature sensor are provided downstream in the return line. Two suction pumps draw return oil from the ENG GBX and feed it to the output side of the main lube pump.

When the engine is shut down, oil from the forward main bearing is drained to a separate return tank which is connected to a scavenge pump.

The three engine bearing chambers, the oil tank and the oil-air separator are vented to the ENG GBX which in turn is vented overboard via a centrifugal breather.

For negative g flights a pendulum-like suction pipe inside the oil supply tank ensures oil supply to the main lube pump.

INDICATIONS AND WARNINGS

The equivalent information will be recorded by the flight data recorder.

Engine Oil Temperature

Engine oil temperature is sensed by a temperature probe in the oil return line. The engine fault detection unit will illuminate a red warning caption on the telelight panel (TLP).

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	OIL PRESS LEFT	Oil temperature above 195° C.
AEKRAN	OIL TEMP LEFT	
VIWAS	"SCHMIERSTOFFTEMPERATUR IM LINKEN TRIEBWERK ZU HOCH" "DREHZAHL VERRINGERN"	

Engine Oil Pressure Low

A pressure sensor in the pressure line illuminates a red warning caption via the engine fault detection unit.

<table>
<tr><th></th><th>INDICATION</th><th>FAULT/EFFECT</th></tr>
<tr><td>MASTER CAUTION</td><td>LIGHT FLASHING</td><td></td></tr>
<tr><td>TLP</td><td>OIL PRESS LEFT</td><td rowspan="2">Oil pressure LH engine low for more than 20 sec:
If the actual pressure falls below 1.8 ±0.18 kp/cm² at 50 % to 89 % RPM or below 2.7 ±0.27 kp/cm² at RPM > 89 % for more than 20 sec.</td></tr>
<tr><td>AEKRAN</td><td>OIL PRESS LEFT</td></tr>
<tr><td>VIWAS</td><td colspan="2">"SCHMIERSTOFFDRUCK IM LINKEN TRIEBWERK ZU GERING"
"DREHZAHL VERRINGERN"</td></tr>
</table>

Engine Chip

Metal chips are sensed by a chip detector in the oil return line. The engine fault detection unit will illuminate a red warning caption on the TLP.

<table>
<tr><th></th><th>INDICATION</th><th>FAULT/EFFECT</th></tr>
<tr><td>MASTER CAUTION</td><td>LIGHT FLASHING</td><td></td></tr>
<tr><td>TLP</td><td>OIL PRESS LEFT</td><td rowspan="2">The system remains operational for 17 sec when oil pressure is low. It is assumed that abrasion will start after 20 sec.</td></tr>
<tr><td>AEKRAN</td><td>CHIP LEFT</td></tr>
<tr><td>VIWAS</td><td colspan="2">"SPÄNE IM SCHMIERSTOFF DES LINKEN TRIEBWERKS"
"DREHZAHL VERRINGERN"</td></tr>
</table>

ENGINE FUEL SYSTEM

The engine fuel system pressurizes, meters, atomizes and injects fuel into the HP compressor discharge airstream, refer to figure FO-5.

The system is controlled by the engine fuel control as a function of various internal operating signals.

The engine fuel system consists of a low pressure and a high pressure system. Fuel is supplied to the low pressure fuel pump of the low pressure system by two fuel booster pumps located inside the engine supply tank. The pressurized fuel passes through a filter and is distributed to the engine control pump (ECP), the AB fuel pump and the nozzle HP pump of the HP system.

The ECP meters the fuel according to throttle position and various engine parameters.

The engine fuel is routed via the drain and cut-off valve, the fuel-cooled oil coolers and the engine fuel flow divider valve to the nozzles of the first and second manifold of the engine combustion chamber, where injection into the airstream occurs.

The ECP supplies fuel to position the actuators of:

- Variable stator vanes of the HP compressor inlet
- AB ignition control
- Automatic engine and AB control equipment.

ENGINE CONTROL SYSTEM

The engine control system is a hydro-mechanical system manually controlled by throttle inputs and operated by an electronic engine control unit (ECU), refer to figure 1-1A.

Main system components of the engine control system include:

- ECU
- ECP
- AB and nozzle control unit
- Engine starter unit.

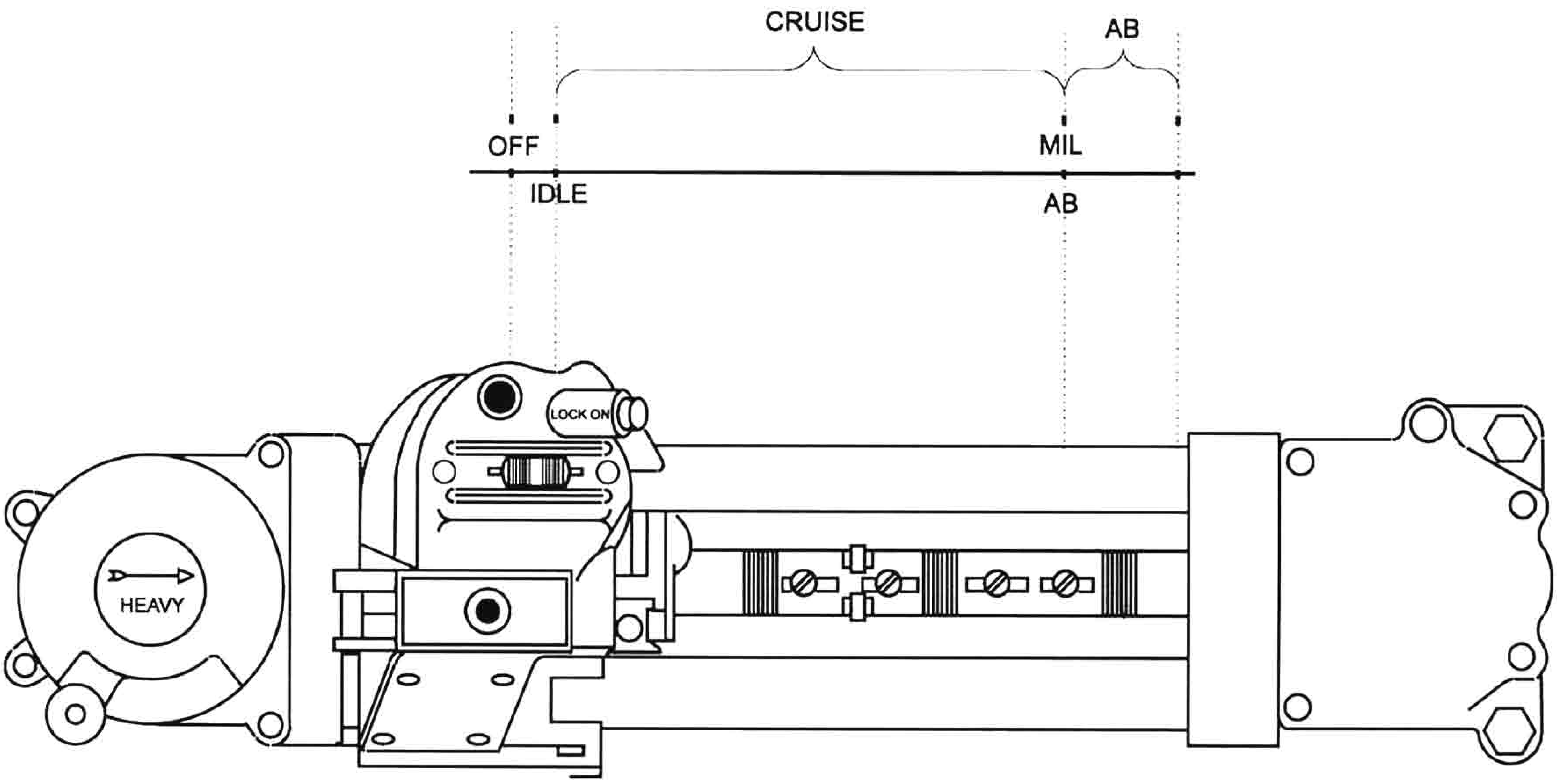
The throttle produces an engine speed demand signal which is routed mechanically to the ECP and to the AB and nozzle control unit. The ECU supplies electrical control signals to solenoids in the ECP, the AB and nozzle control unit and to the engine starter unit in order to modify engine performance within safe operation limits.

The entire performance range of the engine may be divided into four distinct operation regions:

- IDLE
- Cruise
- MIL
- AB

Which system(s) or system components are active to control the engine depend on in which region the throttle is positioned, refer to figure 1-1.

In the column throttle position, the four regions of the engine performance range are listed. In the same row as the throttle position, to the right, the control systems or units are listed which are active for that particular throttle position, while in the columns underneath, the parameters being adjusted or modulated to control the engine.



Throttle position	ECU	ECP	AB and Nozzle Control Unit
IDLE	-	NH	NL
Cruise	NL Correction depending on air mass flow	NH	NL
MIL	NH, NL, T4	-	Nozzle area
AB	T4	-	-

Figure 1-1

In addition to the normal ECS the following functions are provided:

- Automatic engine start sequence on the ground and in flight.
- Control of the variable air intake guide vanes (IGV) of the HP compressor.
- Providing a fuel pressure signal (servo fuel) to control the AB ignition system.
- Supply of servo fuel to the control valve of the engine anti-ice system.
- Control of air intake flow during weapon deployment to prevent stall.

ENGINE CONTROL SYSTEM

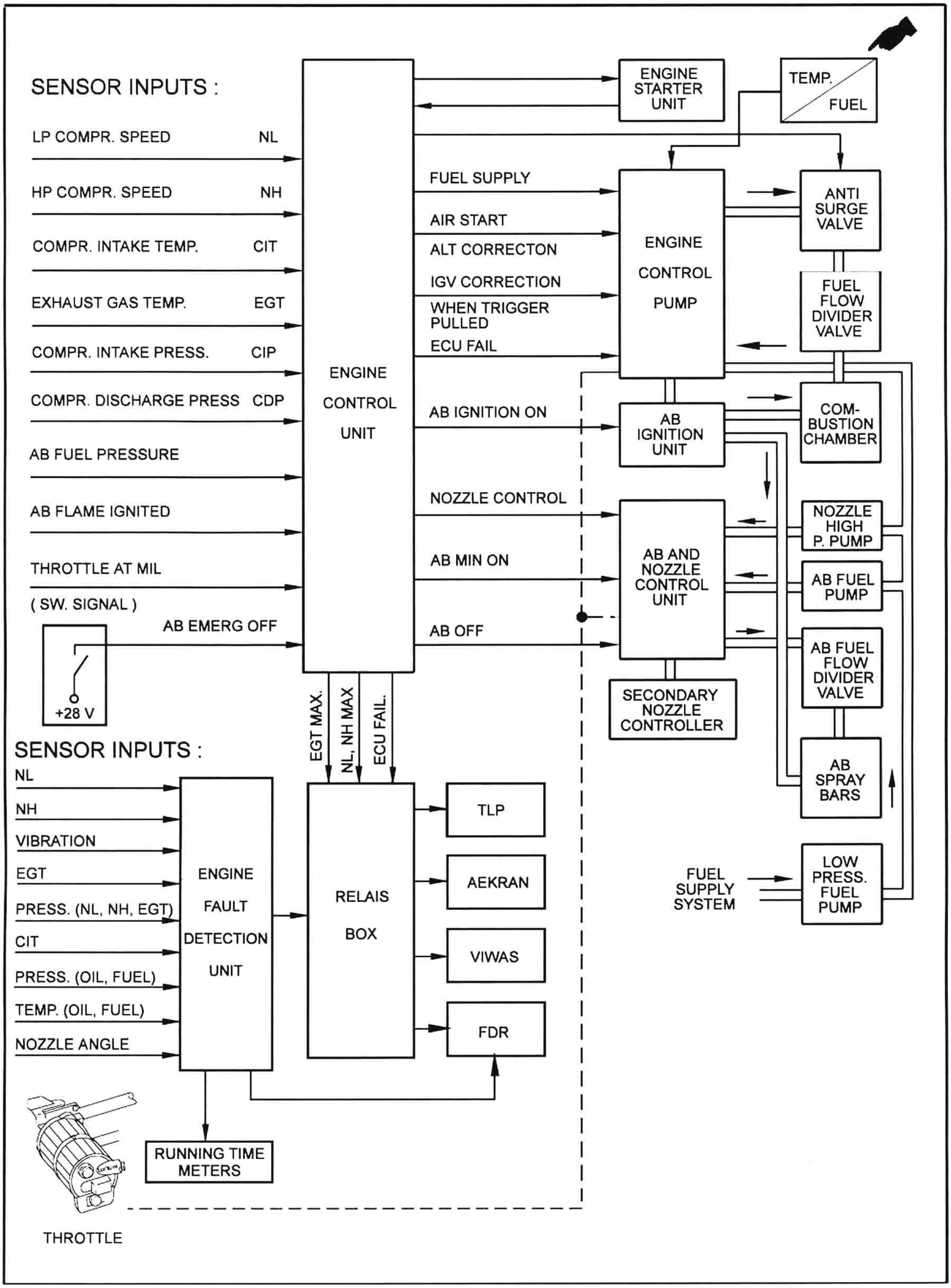


Figure 1-1A

ENGINE CONTROL UNIT

The ECU optimizes the fuel flow for thrust demand, which is controlled by the throttle, by consideration of environmental and engine-specific parameters.

The parallel electronic function lanes of the ECU continuously receive speed signals from the shafts of the HP and LP compressor, LP compressor inlet temperature (CIT), LP turbine outlet and exhaust gas temperature (EGT). They also receive signals from CIT and from the exhaust nozzle outlet by sensors in the engine.

The speed of the high-pressure compressor is abbreviated NH and the speed of the low-pressure compressor as NL. NH is indicated on the engine RPM indicator.

The ECU compares these parameters against preset guiding schedules and limit values and responds with electrical control signals to solenoids in control units of the engine fuel, AB, IGV, exhaust nozzle and air intake.

The ECU, together with the hydro-mechanical control devices provides at

Engine start:

- Control of start sequence as a function of throttle setting, engine fuel pressure, and of air pressure ratio between compressor discharge and ambient air.

Engine run-up to IDLE:

- Limiting of EGT as a function of CIT

Cruising operation:

- Limiting of NL as functions of CIT and throttle setting.

- Maintaining the relation between NH and NL by modulation of the primary exhaust nozzle area.

MIL power and AB operation:

- Limiting of NH and EGT as a function of CIT by modulation of engine fuel flow.
- Scheduling of NL as a function of CIT by modulation of the primary exhaust nozzle area.
- Limiting of NL as a function of CIT by modulation of engine fuel pressure.

AB selection:

- Control of the AB ignition logic.

Engine operation boundaries:

- Protection against compressor surge.

FUNCTION OF THE ENGINE CONTROL UNIT (ECU)
NORMAL POWER MODE

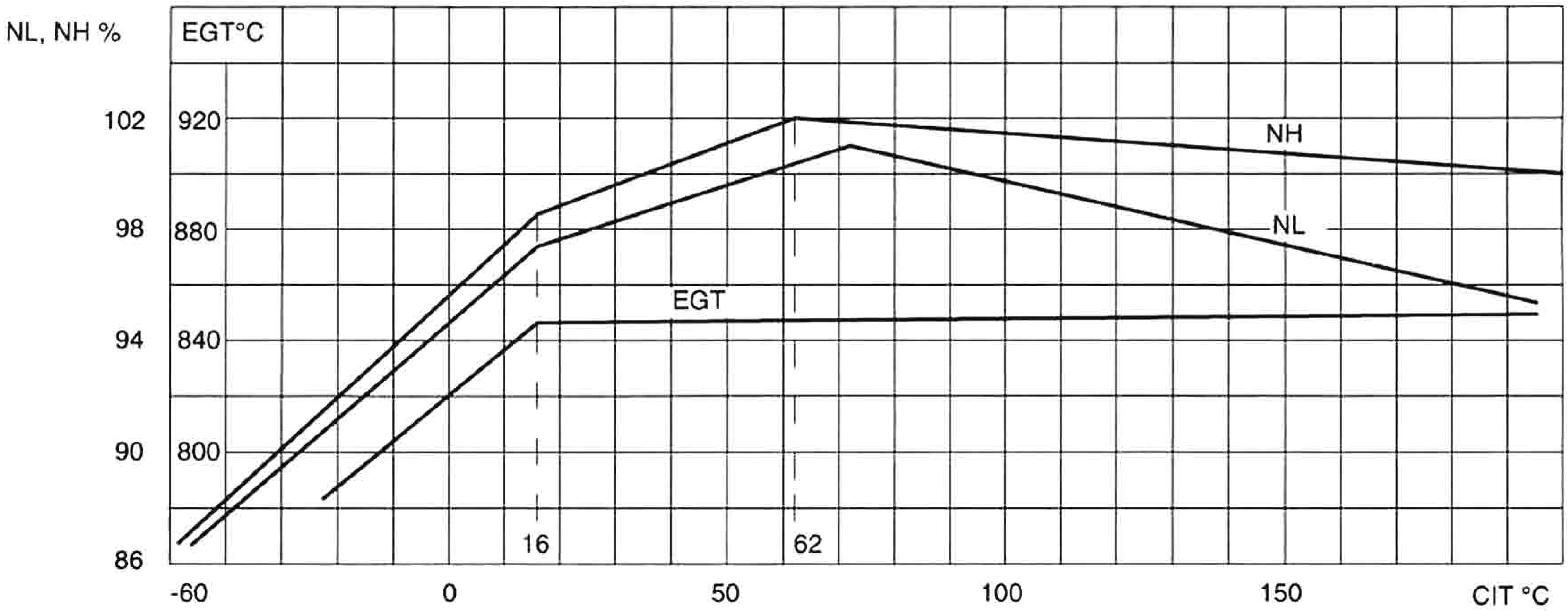


Figure 1-2

FUNCTION OF THE ENGINE CONTROL UNIT (ECU)
(LIMITED POWER MODE)

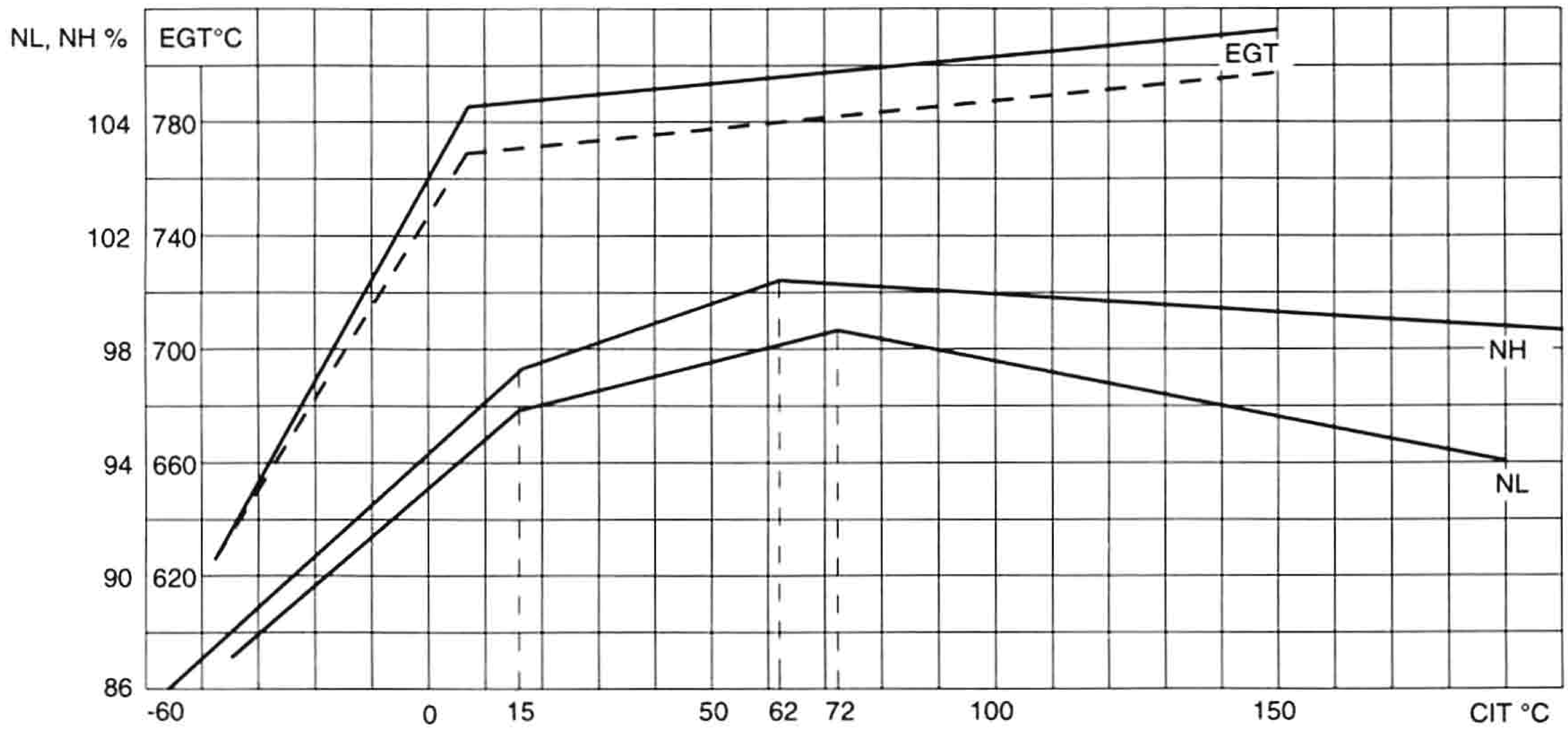


Figure 1-3

INDICATIONS AND WARNINGS

Built-In Test Equipment

The Built-In Test Equipment (BITE) of the ECU provides self-test of limitation lanes and selects a reversionary lane in case of failure. This reversionary lane controls the NH speed controller to reduce the scheduled NH by 6 % to 7 % and issues information to the warning equipment which responds as follows:

	INDICATION	FAULT/EFFECT
AEKRAN	LEFT ENG STBY SYS	Automatic reduction of the scheduled NH by 6 % to 7 %.
VIWAS	"LINKES TRIEBWERK IM RESERVEREGIME" "BEACHTE TEMPERATUR UND DREHZAHL"	

NH SPEED CONTROLLER

The NH speed controller, as part of the engine control pump, adjusts the HP compressor RPM as a function of the throttle position and CIT.

The controller consists of a centrifugal governor which is driven at a speed which is a function of NH and controlled by throttle angle via a mechanical linkage. The set position is continuously modified by a hydraulic actuator which receives a fuel pressure signal representing CIT.

The output actuator of the NH speed controller represents corrected RPM demand and operates the fuel metering valve to set up the required engine fuel flow. At a selected throttle setting, the engine thrust remains constant, regardless of aircraft speed or altitude, except when overridden by any limiters.

The engine maximum speed is controlled by the scheduled limits of the engine control unit which receives actual NH from a pulse probe. The electrical signal is converted to a positioning signal for an electrical pressure control solenoid which modifies the fuel flow.

To avoid conflicts between the corrected RPM demand signal and the limit signal during maximum engine speed, the ECU generates an offset speed signal to set the corrected RPM signal about 3 % above the limit signal.

FUNCTION OF THE NH SPEED CONTROLLER
(NORMAL POWER MODE)

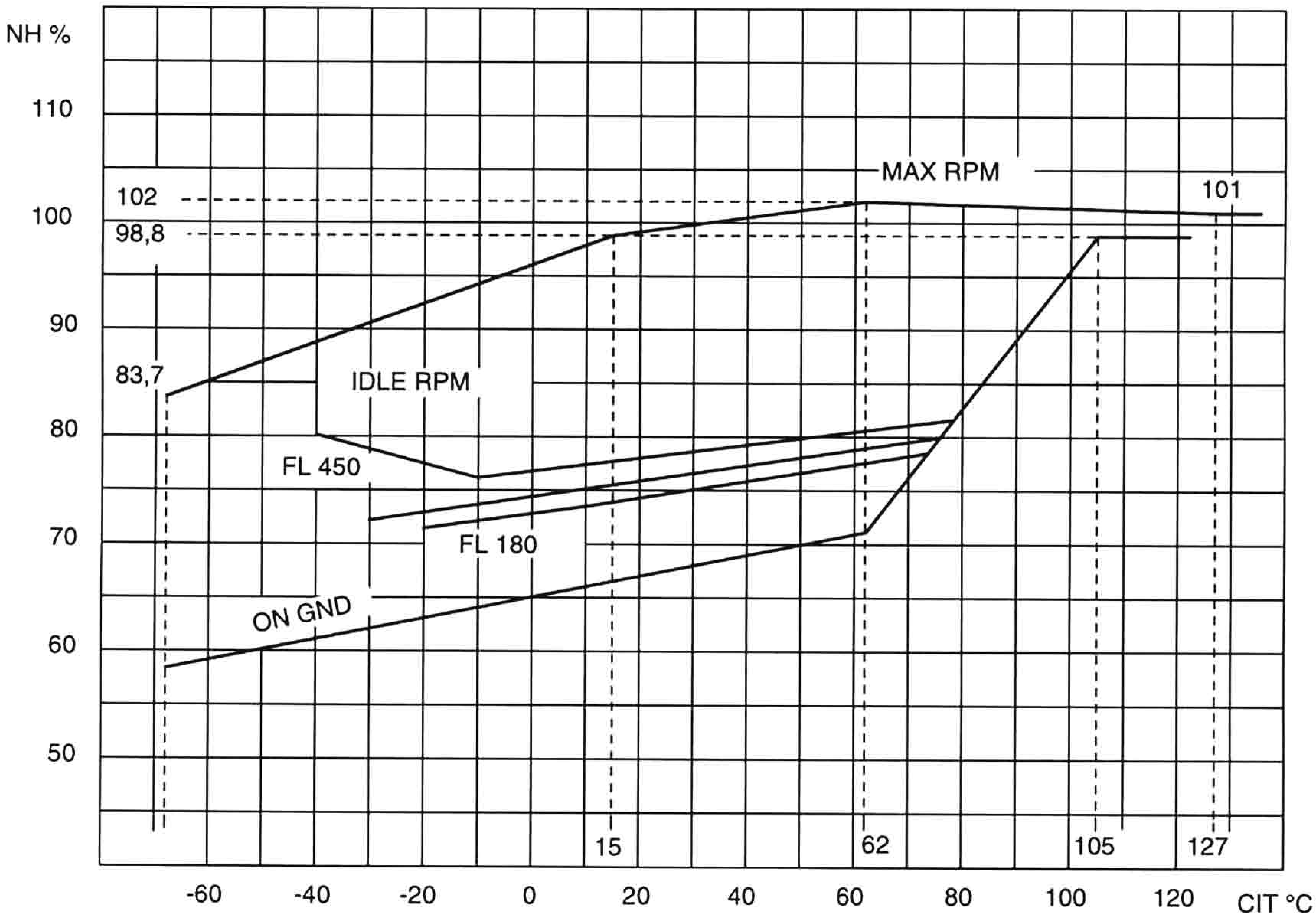


Figure 1-4

FUNCTION OF THE NH SPEED CONTROLLER
(LIMITED POWER MODE)

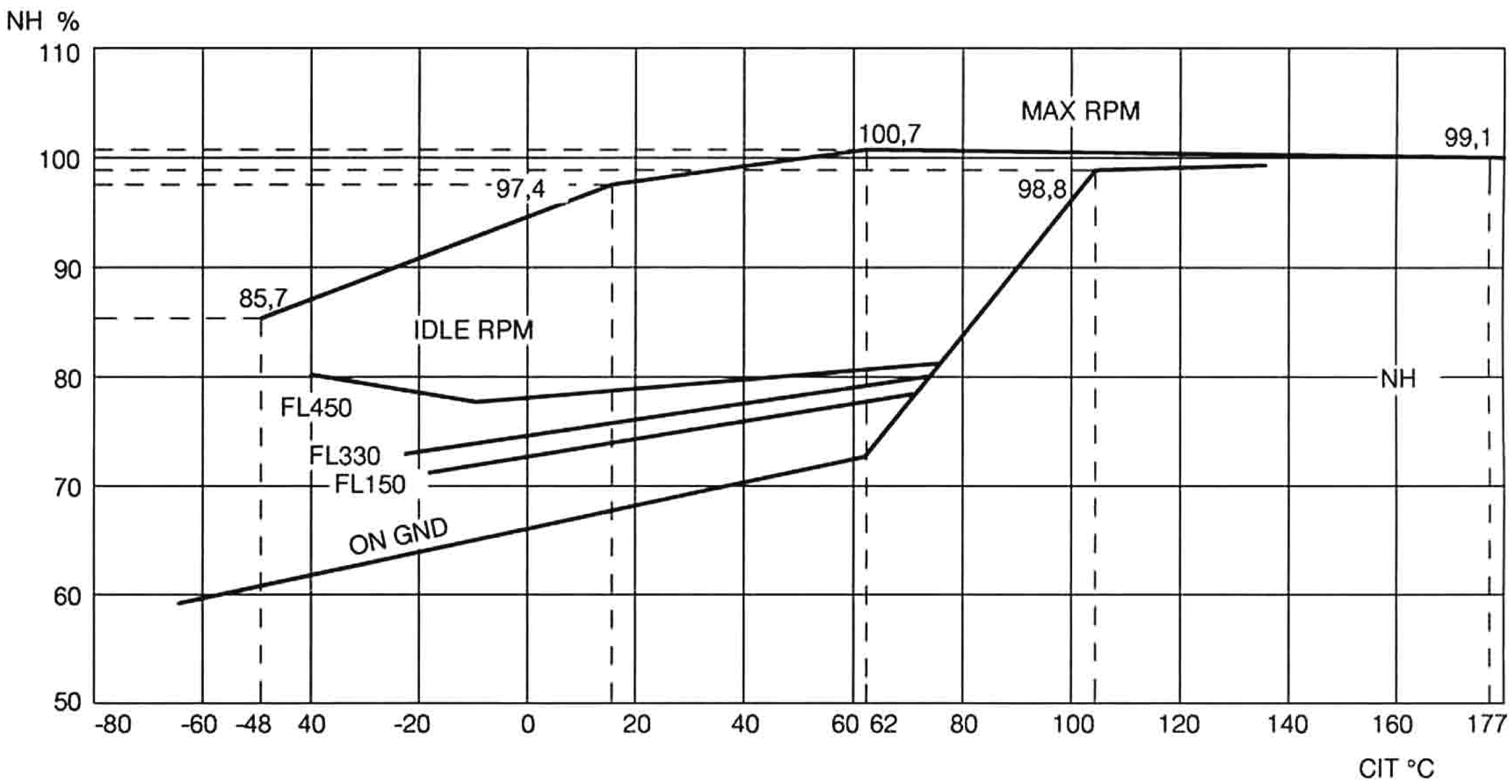


Figure 1-5

INDICATIONS AND WARNINGS

NH Overspeed

Should the ECU detect an NH overspeed by the input from the NH pulse probe, then it assumes a failure of the NH speed controller.

When the throttle setting is reduced, the primary exhaust nozzle controller reduces the primary nozzle area. As a result, the compressor discharge pressure increases and NL, is reduced as well.

The equivalent information will be recorded by the flight data recorder.

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	REDUCE RPM LH ENG	When NH increases by more than 2 % above the scheduled value the ECU issues signals to the warning system after a time delay of 2 sec. The same warnings are initiated by the engine fault detection unit if NH accelerates to 103.5 % RPM within 2 to 3 sec.
AEKRAN	LEFT OVER SPEED	
VIWAS	"DREHZAHL LINKES TRIEBWERK ZU HOCH" "DREHZAHL VERRINGERN"	

STEADY STATE CONTROL

At IDLE and cruise steady state operation, NH is controlled only by the corrected RPM function of the NH speed controller and by primary nozzle modulation for NL guidance. RPM fluctuations of compressor speed are kept within ± 1.5 % RPM at IDLE and ± 1.2 % RPM at cruise speeds.

At MIL and AB steady state operation, maximum engine speed is controlled by the schedules of the ECU, the corrected RPM function of the NH speed controller and the AB nozzle control unit for thrust and NL guidance. RPM fluctuations of compressor speed are kept within ± 0.6 %.

TRANSIENTS CONTROL

Engine acceleration and deceleration between IDLE and MIL power is accomplished by repositioning of the NH speed controller by throttle setting.

The ECU receives values of compressor intake pressure and compressor discharge pressure and controls air flow through the compressor by regulating the necessary fuel. The correct fuel-to-air ratio is monitored by the ECU to ensure combustion.

The air flow is also controlled by air inlet ramps and two-stage inlet guide vanes of the HP compressor on demand of the ECU.

NL SPEED CONTROL

The LP compressor provides an air flow to the HP compressor and a secondary airstream to cool the hot section of the engine.
Further, the LP compressor provides a high increase of thrust due to increase of air flow by mixing the cold secondary air with the hot gases from the turbine section.

The gas pressure in the air mixer must be kept lower than the turbine exhaust pressure to ensure laminar gas flow through the two turbine sections to prevent turbine stall.

When the pressure in the gas mixer decreases and the difference of pressure across the turbines increases, the turbines can extract more energy from the mass flow and NH and NL will increase. The ECU will control the increase of NH by reducing the engine fuel flow, whereas the increase of NL is controlled by increasing the mixer pressure, thus decreasing the pressure difference across the turbine section.

The mixer pressure is controlled by modulation of the primary exhaust nozzle area. The relation of NL to NH is scheduled and limited by the engine control unit by affecting the nozzle jet area at cruising, MIL and AB power. Both parameters NL and NH are corrected for CIT.

INDICATIONS AND WARNINGS

NL Overspeed

To prevent NL overspeed when the primary exhaust nozzle controller fails, a NL maximum speed limiter lane within the ECU is incorporated. This lane prevents an increase of NL of more than 2 % RPM above the scheduled value by reduction of the engine fuel flow, i.e. affecting the NH speed control setting.

The equivalent information will be recorded by the flight data recorder.

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	REDUCE RPM LH ENG	When NL overspeeds by 2 % RPM, the ECU issues signals to the warning system after a time delay of 2 sec. The same warnings are triggered by the engine fault detection unit if the NL accelerates to 103.5 % RPM within 2 to 3 sec.
AEKRAN	LEFT OVER SPEED	
VIWAS	"DREHZAHL LINKES TRIEBWERK ZU HOCH" "DREHZAHL VERRINGERN"	

ENGINE FAULT DETECTION UNIT

The engine fault detection unit compares actual engine parameters to preset schedules and generates discrete signals for the warning system if parameters exceed the limits.

In addition, the unit issues a signal to engage the AC electrical power generator and activates the nitrogen system to pressurize the fuel system when NH exceeds 55 % RPM.

The engine fault detection unit is connected to the ENG SYS switch on the system power control panel at the RH console.

INDICATIONS AND WARNINGS

Engine Overtemperature

When the throttle is retarded, NH and compressor discharge pressure are decreased to normal values. The limiter of the ECP is activated to reduce the engine fuel flow. NH and EGT are stabilized.

The equivalent information will be recorded by the flight data recorder.

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	REDUCE RPM LH ENG	When the sensed EGT exceeds the scheduled value by 40° to 60° C for 0.2 sec.
AEKRAN	OVERHEAT LEFT	
VIWAS	"ÜBERHITZUNG LINKES TRIEBWERK" "DREHZAHL VERRINGERN"	

Engine Vibration

In the immediate vicinity of vital engine components, e.g. bearings and gears, vibration sensors are installed which convert mechanical oscillations and vibrations into an electrical signal. The amplitude of this signal is proportional to the magnitude of the vibration.

As soon as the amplitude of the signal exceeds a scheduled value, the engine fault detection unit issues a warning signal to the various warning systems.

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	REDUCE RPM LH ENG	Excessive vibration in left engine. The vibration level exceeds a present limit for NH greater than 35 % RPM.
AEKRAN	VIBR LEFT	
VIWAS	"VIBRATION IM LINKEN TRIEBWERK" "DREHZAHL VERRINGERN"	

Engine Fuel Pressure

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	REDUCE RPM LH ENG	Engine fuel pressure exceeds 7.35 ±0.49 MPa for 2 to 3 sec.
AEKRAN	FUEL PRESSURE LEFT	
VIWAS	"KRAFTSTOFFDRUCK LINKES TRIEBWERK ZU HOCH" "DREHZAHL VERRINGERN"	

ENGINE ANTI-SURGE SYSTEM

The engine anti-surge system is selected with the ANTI SURGE switch on the system power panel. The activated system automatically detects and counteracts engine surge. A sensor detects fluctuations of HP compressor discharge pressure.

When fluctuations occur, the ECU issues control signals. After fluctuations have ceased, the signals will persist for additional 0.5 sec, however they will not exceed a total of 2.4 sec. The control signals are:

- Signal to close the inlet guide vanes (IGV) by 25 degrees.
- Close signal to the drain and shut-off valve to interrupt the engine fuel flow to the combustion chamber for up to 3 sec.
- Close signal to the ECP to close the IGV of the HP compressor.
- Open signal to the NL control lane of the ECU to shift the program schedule for 5 % NL. As a result, the primary nozzle area will open for a corrected value.
- Signal to close the intake ramp for an additional 10 % maximum, limited by the applicable ramp travel schedule. Refer to variable duct ramp system in this section.
- Start signal to the engine starter unit to actuate a preventive engine start cycle for 8 sec. This activates the green caption LH / RH ENG START on the TLP.

If the discharge pressure sensor of the HP compressor fails, the control signals from the ECU are automatically switched off after 2.5 sec.

When the anti-surge lane is activated and an additional fault signal from the engine high temperature lane is present, anti-surge actions are operative for the time of the high temperature condition plus 0.5 sec.

Total time of extended system operation is limited to a maximum of 8 sec. If the high temperature condition persists for more than 2.4 sec, engine fuel flow will be alternately interrupted by the drain and shut-off valve for 2.4 sec and permitted for 1.2 sec.

The engine anti-surge system is deactivated at altitudes below 9 000 ft MSL with airspeeds below M 1.15.

Exception:

If the 30 mm gun or a missile is fired, or if an overheat condition exists, the system is activated and a preventive engine relight cycle is initiated.

INDICATIONS AND WARNINGS

The equivalent information will be recorded by the flight data recorder.

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	**REDUCE RPM LH ENG**	Engine in surge condition and overheat left / right.
AEKRAN	**OVERHEAT LEFT**	
VIWAS	"ÜBERHITZUNG LINKES TRIEBWERK" "DREHZAHL VERRINGERN"	

AFTERBURNER FUEL SYSTEM

The AB fuel system supplies and regulates fuel flow into the engine tailpipe for AB combustion. The ignited fuel-air mixture increases engine thrust. Refer to figure FO-5.

The AB system consists of an annular casing which contains the air mixing chamber, and a diffusor type of AB combustion chamber. The diffusor ensures reduction of air flow velocity which results in a gain of time for hot and cold air mixing, AB fuel injection and ignition. The injection system consists of flame holders and three spraybars with injection nozzles for radial fuel injection.

The required fuel flow for AB ignition will be controlled by the ECU. It also sets the exhaust nozzle to a position for minimum AB power. Simultaneously the AB ignition unit is activated.

The system ensures ignition in the AB combustion chamber, stable burning at minimum AB thrust, modulation of AB fuel flow at different power settings, AB thrust transients control and AB ignition timing.

AB fuel is controlled by throttle and CIT inputs to the AB and nozzle control unit and is limited to safe operating ranges by the ECU.

AFTERBURNER IGNITION

To initiate AB operation, the throttle must be advanced into AB range. AB ignition is possible at engine speeds of at least 72 % to 76 % RPM.

The fuel from the LP fuel pump of the LP system is supplied to the AB fuel pump.

When the throttle is advanced into AB range, the AB fuel pump increases the fuel pressure and supplies it to the AB and nozzle control unit. The pressurized fuel is applied via the AB fuel-cooled oil cooler to the AB pressurizing valve and to the nozzles of the first of three spraybars. A fuel pressure sensor detects the system pressure and issues an appropriate signal to the ECU.

The required fuel flow for AB ignition will be controlled by the ECU. It also sets the exhaust nozzle to a position for minimum AB power. Simultaneously the AB ignition unit is activated.

The ignition unit controls the internal pressure and fuel flow to the ejector nozzle and spin nozzle of the AB torch ignitor.

Starter jet fuel is supplied to the ejector nozzle inside the combustion chamber and to the spin nozzle. Starter jet fuel from the ejector nozzle ignites inside the combustion chamber. This torch will ignite the starter jet fuel supplied to the spin nozzle. This will extend the torch to the AB combustion area to ignite the limited, minimum AB fuel from the first spraybar.

Two flame sensors are installed behind the AB flame holder to detect the ignited AB fuel and to initialize the shut-off of the AB ignition by the ECU. Simultaneously, the ECU establishes the normal fuel flow for minimum AB.

INDICATIONS AND WARNINGS

When the AB flame sensors detect a flame, the ECU issues a status signal to the warning equipment which responds as follows:

The equivalent information will be recorded by the flight data recorder.

	INDICATION	FAULT / EFFECT
TLP	**LH ENG AB**	Left engine AB on.

AFTERBURNER CONTROL SYSTEM

The AB control system is a hydro-mechanical system using fuel as actuating fluid. It is manually controlled by throttle inputs and automatically operated by the ECU.

The main system components are the ECU and the AB and nozzle control unit.

The AB and nozzle control unit regulates the fuel to spraybars according to throttle setting.

Since throttle position in AB is beyond MIL, the basic ECP senses a demand for 100 % RPM for all AB throttle settings.

As soon as AB is selected, the ECU, the AB and nozzle control unit, and the AB ignition unit control will regulate:

- AB ignition sequence as a function of throttle setting and of compressor discharge pressure corrected for CIT.
- Time of torch ignition as a function of fuel pressure.
- Primary exhaust nozzle setting to minimum cross section.

During AB operation:

- AB operation as a function of throttle setting and of compressor discharge pressure corrected for CIT.
- Primary exhaust nozzle cross section according to throttle settings.

The AB fuel pressurizing valve delivers the fuel to the first spraybar for AB start, to the second and third spraybar as a function of fuel pressure.

AB acceleration according to a throttle burst is limited only by constructional drag inside the control units.

During throttle burst from IDLE to AB MAX, the AB will not ignite below 72 % to 76 % RPM.

When AB blow-out occurs during AB operation, the ECU will limit the AB fuel flow to the quantity required for minimum AB operation after 0.5 sec.

For an emergency AB shut-down, the AB EMERG OFF switch on the engine emergency panel has to be set to OFF. The ECU together with the AB and nozzle control unit and AB fuel divider valve will reduce the fuel pressure so that the spraybars close in sequence.

EXHAUST NOZZLE SYSTEM

Since thrust is directly proportional to gas velocity at the exhaust, pressure and temperature in the AB area have to be as high as possible to obtain a high nozzle pressure ratio at the Laval nozzle. Refer to figure FO-5.

Two sets of cylindrical nozzles, operating together, make up the variable exhaust nozzle system. The primary nozzle (inner nozzle) controls the convergent portion of the nozzle, while the secondary nozzle (outer nozzle) controls the divergent portion of the nozzle.

Both nozzles are mechanically linked for common operation by the synchronization unit, using fuel as actuating fluid. The exhaust gas leaves the primary nozzle at subsonic velocity and is accelerated to supersonic velocity by controlled expansion of the gas. The pressure of the gas, before leaving the secondary nozzle, equals ambient air pressure.

NOZZLE AREA CONTROL

Pressurized fuel as a control medium is supplied from the HP nozzle pump to the nozzle control section of the AB and nozzle control unit. Depending on the inputs, the AB nozzle control unit modifies fuel pressure to the synchronization unit for nozzle area adjustment.

Throttle setting and corrected NL as a function of corrected NH are utilized to schedule the correct primary nozzle area (Aj1). During engine operation below MIL, the nozzle is scheduled to open fully at IDLE and the area is decreased as the throttle is advanced toward the MIL position.

For AB ignition, the area is closed to minimum AB operation.
During engine operation in the MIL and AB range the nozzle control system modulates the primary nozzle area (Aj1) to maintain NL according to the preset schedule of the ECU. The secondary nozzle area (Aj2) is adjusted synchronously to achieve a complete pressure drop within the Laval nozzle to ambient atmosphere pressure.

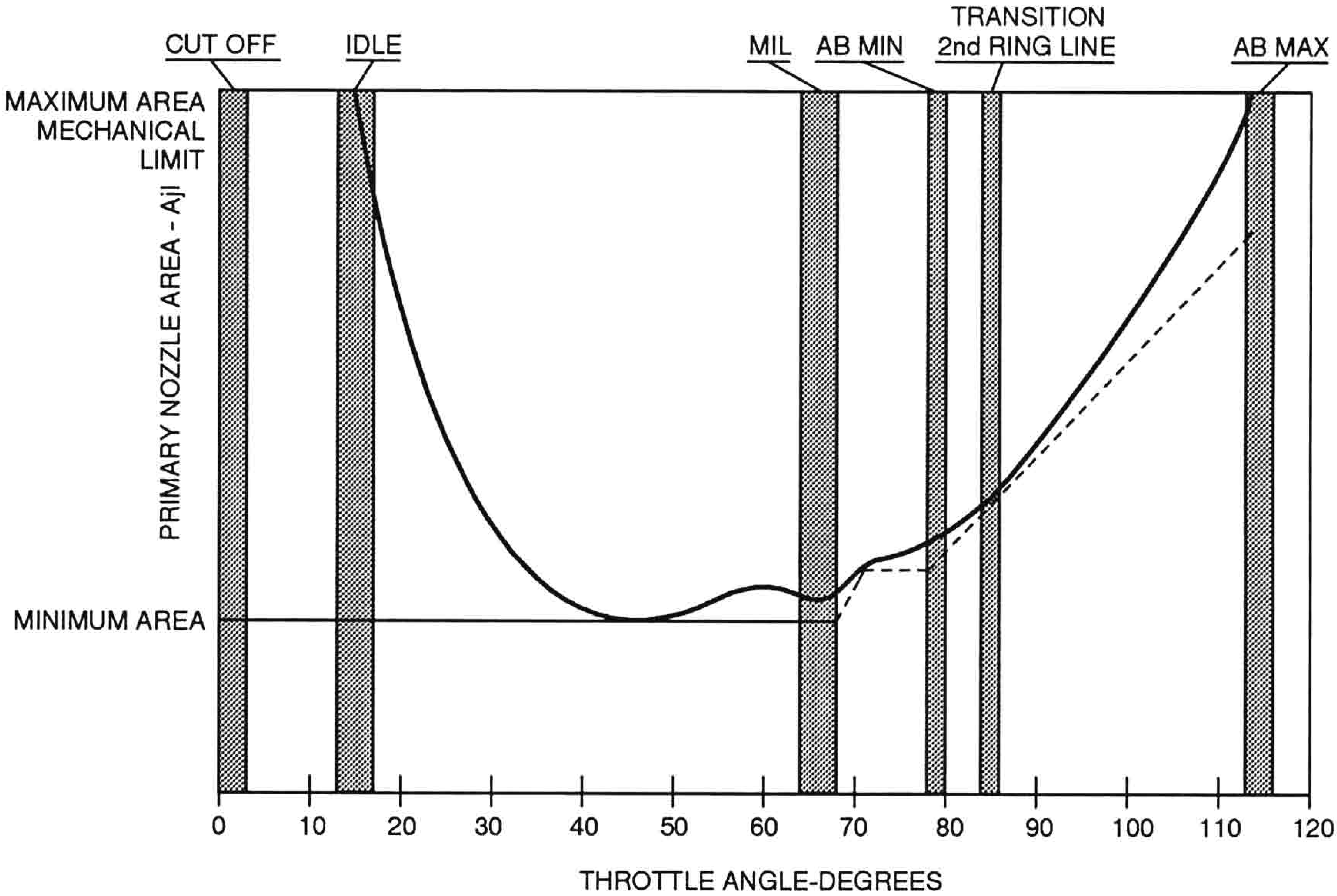
When the nozzle control lane of the ECU fails, the primary nozzle area is modulated by the AB and nozzle control unit as a function of throttle setting and CIT.

INDICATIONS AND WARNINGS

The equivalent information will be recorded by the flight data recorder.

	INDICATION	FAULT / EFFECT
AEKRAN	**LEFT ENG STBY SYS**	Nozzle control lane of LH engine failed.
VIWAS	"LINKES TRIEBWERK IM RESERVEREGIME" "BEACHTE TEMPERATUR UND DREHZAHL "	

VARIATION OF THE NOZZLE AREA WITH THROTTLE POSITION



PRIMARY NOZZLE SCHEDULE GENERATED BY THE ECU AS FUNCTION OF CORRECTED RPM OF HP COMPRESSOR TO CONTROL THE LP COMPRESSOR SPEED.

MINIMUM AREA AS LIMITED BY THE AB AND NOZZLE CONTROL UNIT.

MECHANICAL SCHEDULE AS FUNCTION OF THROTTLE; AB AND NOZZLE CONTROL UNIT WHEN ECU FAILS.

Figure 1-6

ENGINE AIR INTAKE SYSTEM

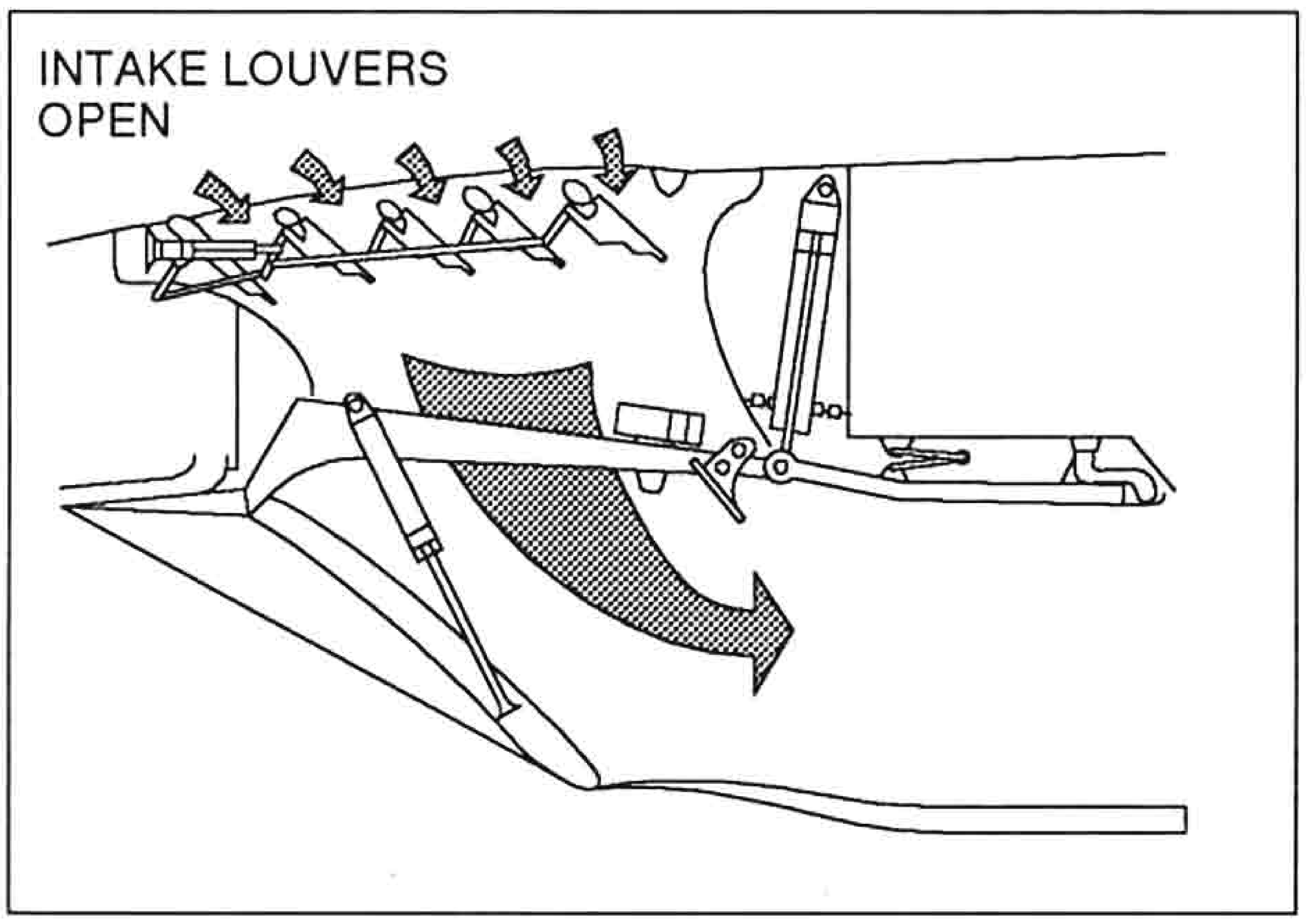
There are two independent air intakes, one for each engine. Each inlet duct is located below the wing root, 2.5 inches apart from the lower surface. This distance allows the boundary layer from the wing root and the inlet duct to pass outside the bellmouth. The components are a variable duct ramp system and an air inlet louver system at the upper wing root surface.
The inlet louver system is interconnected to the forward ramp system by an internal upper air intake duct, refer to figure 1-7.
To prevent FOD during takeoff and landing, the duct ramps are closed and the air intake louvers are opened.

VARIABLE DUCT RAMP SYSTEM

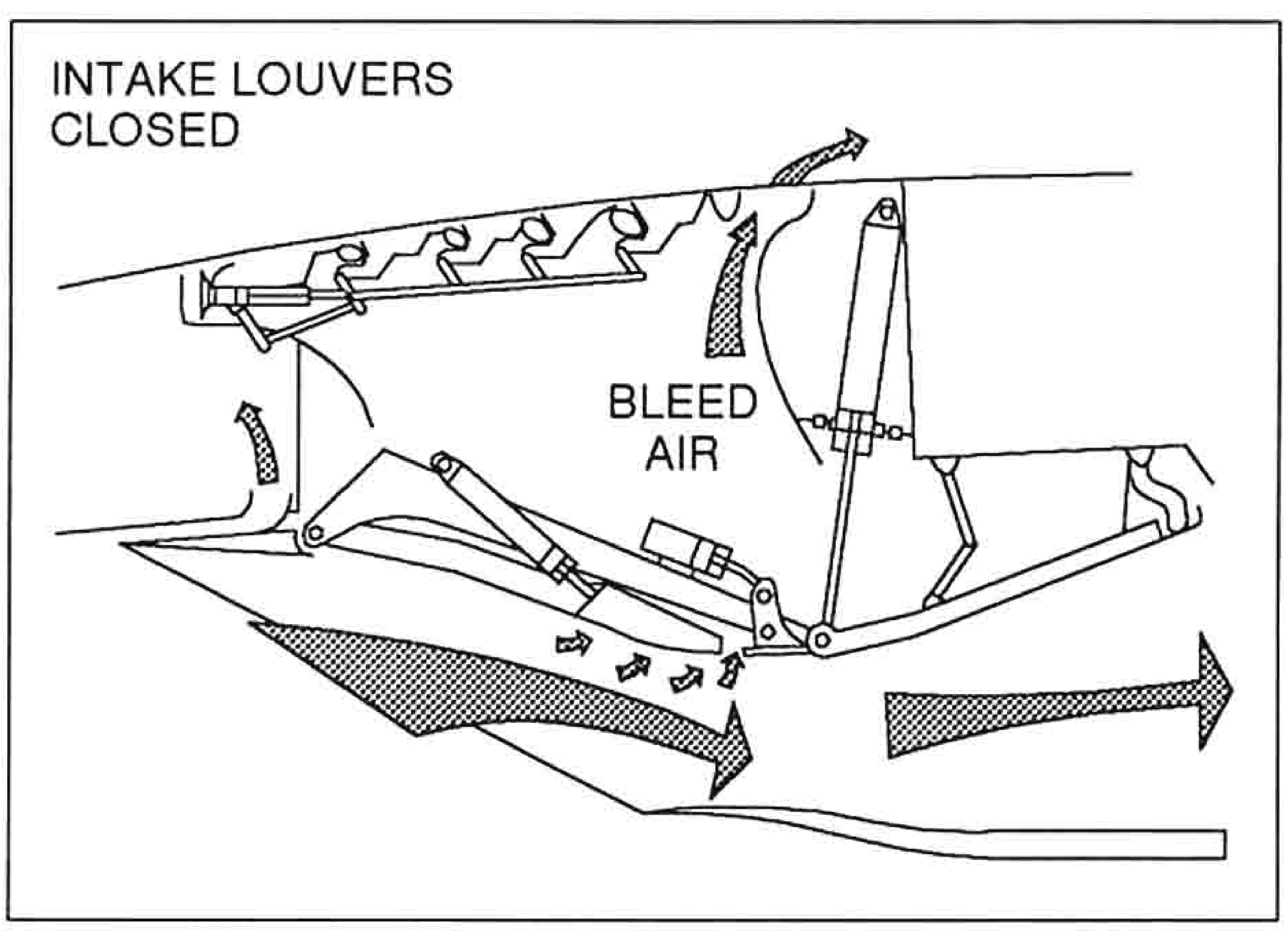
The variable forward ramp system provides engine air at optimum subsonic airflow to the low-pressure compressor face throughout a wide range of speeds. The ramp assembly consists of a variable forward ramp, a variable aft ramp, a bleed-off valve and a ramp control unit. The bleed-off valve and perforated sections of the ramps allow boundary layer air from the forward ramp to be bled-off and exhausted overboard.

In flight the forward and the aft ramp are variable to modify the intake air stream to the engine, refer to figure 1-7.

VARIABLE DUCT RAMP SYSTEM MODES



TAKEOFF/LANDING MODE



IN-FLIGHT MODES

Figure 1-7

AIR INTAKE LOUVERS

The supplementary air intake duct at each upper wing root surface is equipped with five narrow hinged shutters, air intake louvers, allowing the duct to be open or closed. The louvers are spring loaded closed. During takeoff and landing roll, the louvers are opened by vacuum generated by the engine compressor, since the forward ramp is closed. When the forward ramp is retracted at about 108 KIAS, the louvers close. During flight, however, depending on engine RPM, they may open intermittently at Mach numbers ≤ 0.3 in IDLE and ≤ 0.6 in MIL. During engine shut-down, the louvers are locked in the close position when the main hydraulic system pressure subsides.

AIR INTAKE CONTROL

During supersonic flight, the intake airstream has to be decelerated to subsonic speed. Deceleration is attained by four slanting and one straight shock wave. The number of slanting shock waves, their slope angles and the position of the final straight shock wave depend on airspeed, AOA and inclination of the ramps.

The forward and aft ramp are controlled separately by a control unit and hydraulically positioned. The three control schedules are functions of corrected NL, altitude and Mach number. Refer to figure FO-6.

VARIABLE RAMPS SYSTEM OPERATION

On the ground and with engines shut down, no hydraulic pressure is available, and the forward and aft ramp moves to maximum duct opening to provide free access for interior inspection. Refer to figure 1-8.

During engine start, the forward ramp is moved to maximum extended position closing the intake duct.

NOTE

Depending on the setting of the ramp control unit, either both ramps close as soon as hydraulic pressure is available, or the ramp of the starting engine closes when 35 % RPM are reached.

During takeoff, at about 108 KIAS, the forward ramp is retracted to fully open.
In flight, gear retracted, the third travel schedule controls the wedge angle between 0 % and 35 % extension. Refer to figure 1-8.

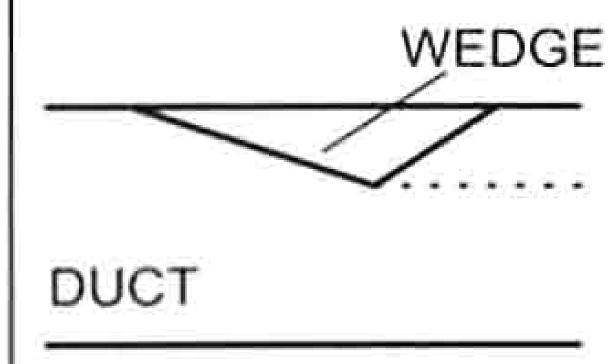
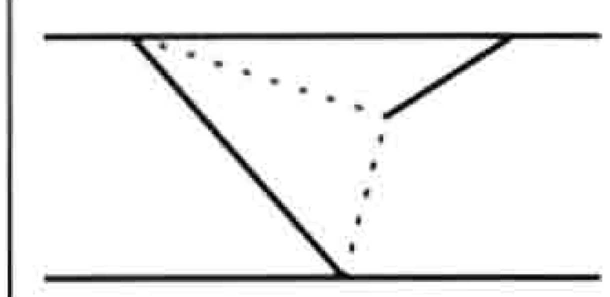
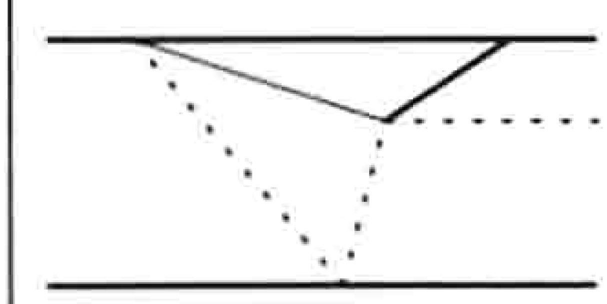
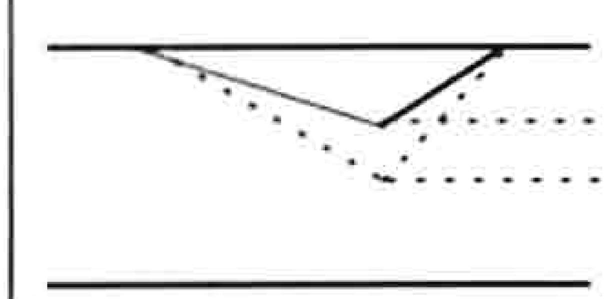
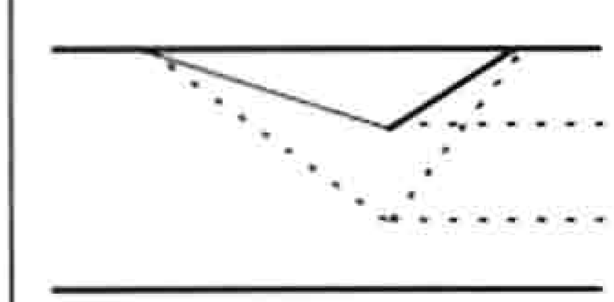
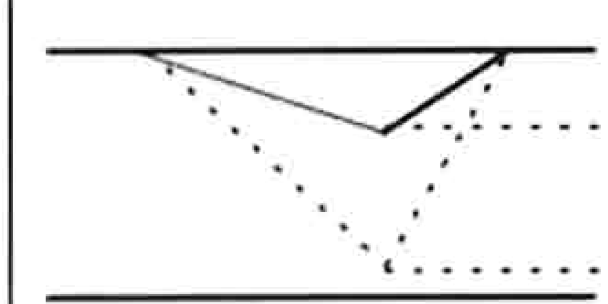
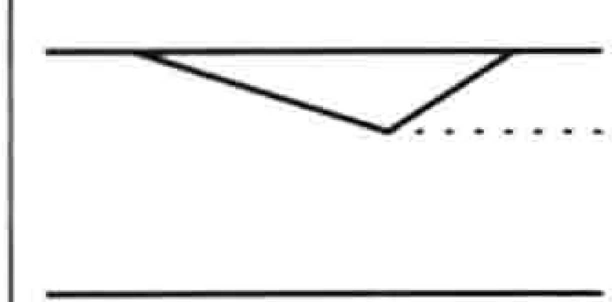
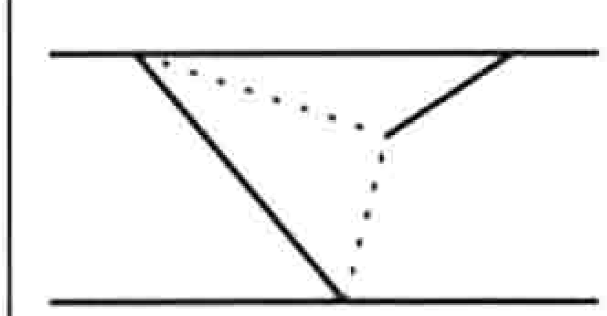
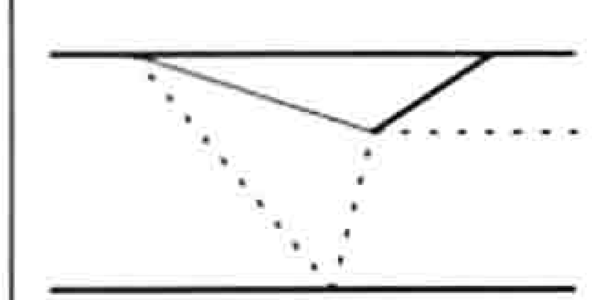
The second travel schedule controls the wedge angle between 0 % and 60 % extension.

The first travel schedule controls the wedge angle between 0 % and 100 % extension.

During landing, with gear extended, the wedge is completely retracted for full duct opening.

After landing, when the speed is reduced below 108 KIAS, the forward ramp is extended to close the air intake.

ENGINE AIR INTAKE SYSTEM

NO.	CONDITION	DESCRIPTION	FWD RAMP POSITION	WEDGE EXTENSION	PRO-GRAM	MECHANICAL CONDITION	INDICA-TION
1	ON GND	ENG SHUT-DOWN NO HYDRAULIC PRESSURE AVAILABLE	FWD PART OF WEDGE	MINIMUM 0 %	-		0 %
2	ON GND	ENG RUNNING HYDRAULIC PRESSURE AVAILABLE	SEPARATES FROM WEDGE AND CLOSES DUCT	AFT RAMP MINIMUM	-		ВП
3	TAKEOFF	SPEED ABOVE 108 KIAS	FWD PART OF WEDGE	MINIMUM 0 %	-		0 %
4	IN FLIGHT	GEAR RETRACTED PROGRAM RANGE: BELOW 10 000 ft ABOVE M 1.15 OR ABOVE 10 000 ft BELOW M 1.15	FWD PART OF WEDGE	0 TO 35 %	3		0 % 35 %
5	IN FLIGHT	ABOVE 10 000 ft M 1.15 TO M 1.5	FWD PART OF WEDGE	0 TO 60 %	2		0 % 60 %
6	IN FLIGHT	ABOVE M 1.5	FWD PART OF WEDGE	0 TO 100 %	1		0 % 100 %
7	LANDING	GEAR EXTENDED	FWD PART OF WEDGE	MINIMUM 0 %	-		0 %
8	AFTER LANDING	SPEED BELOW 108 KIAS	SEPARATES FROM WEDGE AND CLOSES DUCT	AFT RAMP MINIMUM	-		ВП
9	ON GND	ENG SHUT-DOWN NO HYDRAULIC PRESSURE AVAILABLE	FWD PART OF WEDGE	MINIMUM 0 %	-		0 %

NOTE: The wedge extension within a program is a function of corrected NL. A high corrected NL corresponds to minimum extension. Refer to figure FO-6.

Figure 1-8

Built-In Test Equipment

After start of both engines, the two ramp control units are tested automatically when engine RPM is increased to 80 % to 90 % RPM.

Check for illumination of two green captions LH INLET CHECK or RH INLET CHECK on the control and test panel at the aft section of the RH console.

No warning should be issued by the warning equipment.

The ramp control unit consists of two control lanes. In case of a malfunction of one lane, the BITE selects the second lane automatically.

RAMP CONTROL UNIT BITE INDICATION

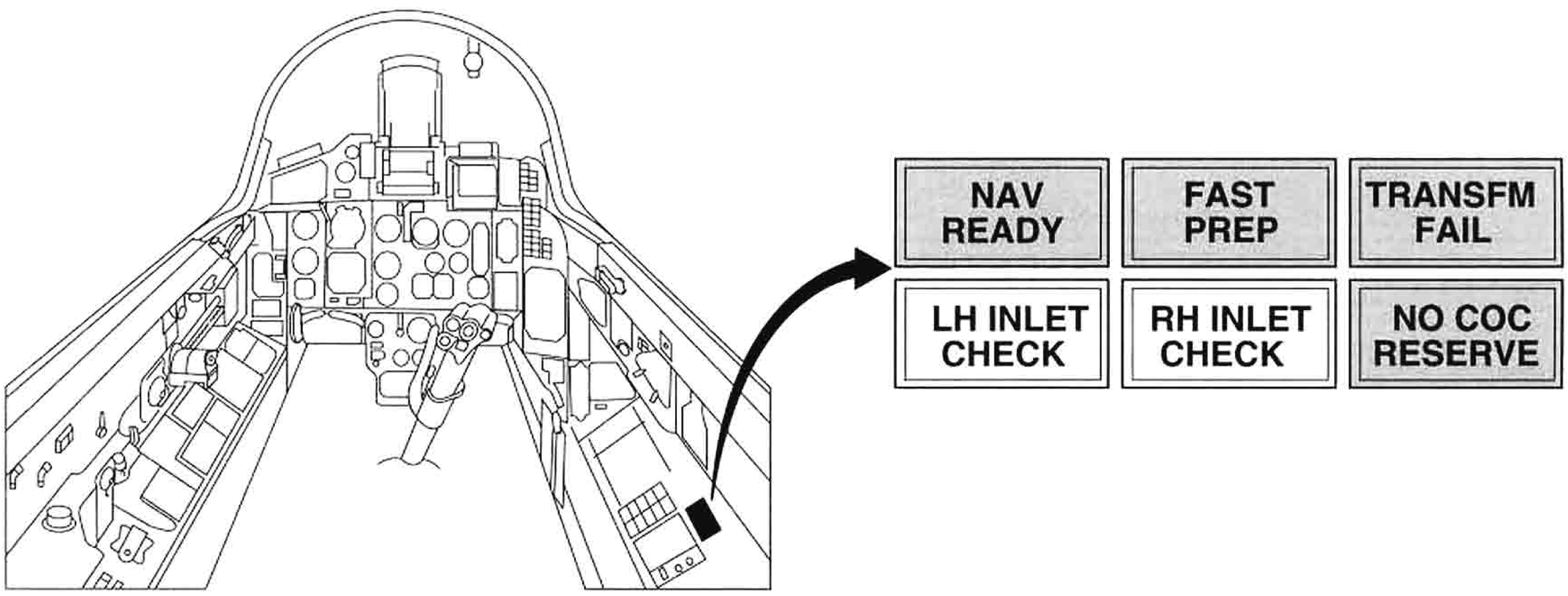


Figure 1-9

PREVENTIVE RAMP EXTENSION

When the ECU receives a signal from the anti-surge system, i.e. surge detection lane of the ECU, an additional 10 % extension demand signal is issued by the ramp control unit. This additional extension is limited by the maximum extension value of the actual variable duct travel schedule.

INDICATIONS AND WARNINGS

- If the gear is retracted, and the forward ramp is not retracted to full duct opening, i.e. the air intake louvers are still open.
- If after takeoff the ramp control is inoperative with main hydraulic system pressure available, the wedge may extend only to 55 % which results in a significant power loss.
- If main hydraulic system pressure is not available and the ramp control system fails, the wedge remains in its last position.

In these cases the warning equipment will issue the warnings listed below:

	INDICATION	FAULT / EFFECT
AEKRAN	UPPER INLET	Air intake remains closed.
VIWAS	"OBERER LUFTEINLAUF GEÖFFNET" "M-ZAHL KLEINER 0,8"	

	INDICATION	FAULT / EFFECT
AEKRAN	LEFT AIR INTK	Ramp control failure.
VIWAS	"AUSFALL DER AUTOMATIK LINKER LUFTEINLAUF" "KEILE ÖFFNEN"	

The equivalent information will be recorded by the flight data recorder.

Manual Wedge Retraction

In case of an intake ramp controller malfunction or a main hydraulic system failure, the wedges are locked in the position at time of failure. To retract the wedges during flight or before landing, thus fully opening the air intake duct, the springloaded and guarded RAMP EMERG RETRACTION LH or RH switch at the engine emergency panel has to be held to RETRACTION for wedge retraction.

If the malfunction is a faulty intake ramps controller, an emergency hydraulic unit powered by the main hydraulic system will retract the wedge upon switch operation. If the main hydraulic system fails as well, both wedges will be retracted by engine intake ram air upon actuation of one or both switches.

After emergency wedge retraction the wedge can extend to the 8 % position when the switch is released.

Ramp Position Indicator

The indicator shows wedge / ramp positions of both air intake systems as a percentage of nominal maximum extension. The 0 % mark represents minimum extension, which corresponds to fully-open air intake duct. For takeoff and landing, when the forward ramp closes the intake duct, the indication will be at the takeoff / landing position ВП.

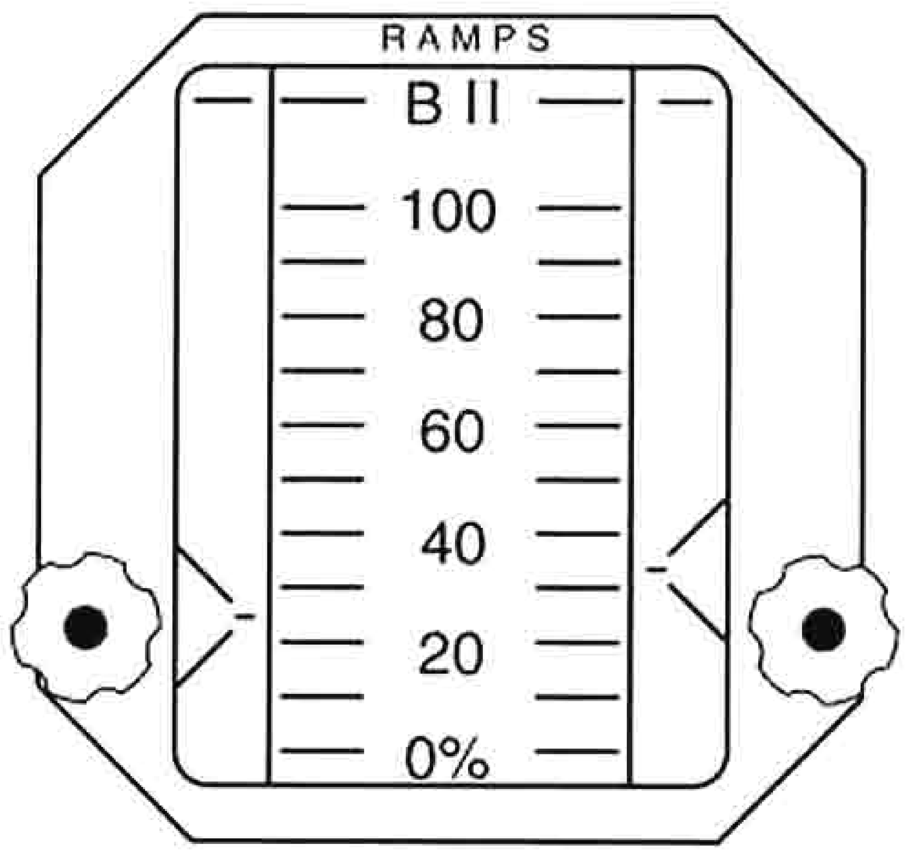


Figure 1-10

Wedge Travel Programs

This diagram shows the three wedge travel programs as a function of corrected NL. The extra travel of the forward ramp and it's function is shown also. Refer to figure FO-6. The wedge travel programs, as function of altitude and Mach number, are shown in figure FO-6 as well.

VARIABLE STATOR SYSTEM

The variable stator system consists of intake guide vanes and stator vanes of the first three stages of the HP compressor. Its function is to prevent compressor surge by limiting the intake air flow angle.

Squeezing either gun or missile trigger will activate the engine anti-surge cycle, thus closing the vanes by 25 degrees. This minimizes adverse effects on the engine caused by gun or missile exhaust gas ingestion. Upon signals from the ECU, the system uses fuel from the ECP as the actuating medium.

ENGINE IGNITION SYSTEM

The engine ignition system initiates ignition of the fuel in the combustion chamber during the starting cycle, and provides an automatic engine ignition source when the weapon release trigger is pressed.

The engine starter unit provides power to each ignitor plug and controls the ignition sequence. The ignition sequence is initiated by pressing the GND START button. When 35 % RPM are reached, the ignition system is switched off.

For air relight, the ignition system is activated for 20 sec for automatic and semi-automatic relight. The ignition system can also be activated manually by operating the AIR RELIGHT LH / RH switch. In this case the ignition system will be active as long as the switch is held in the ON position, but should not exceed 100 sec to prevent damage to the ignition coils.

When either trigger is squeezed, the preventive air relight mode activates the ignition system for 8 sec.

For air relight, electrical power is supplied to the ignitor plugs. At the same time oxygen is guided into the ignition area.

INDICATIONS AND WARNINGS

For the time of ignition the engine fault detection unit issues a signal to the warning system as shown as follows:

	INDICATION	FAULT / EFFECT
TLP	**LH ENG START**	Left engine ignition system active.

ENGINE STARTING SYSTEM

Ground starting of both engines is achieved by the APU via an ENG GBX for each engine. Electrical start-up power is supplied by an external power unit or by two internal batteries.

The HP compressor shafts of the left and right engines are connected to two ENG GBX which are connected to a common GBX by two angular drives. Refer to figure FO-7.

An APU is flanged to the GBX and drives it through a gas-coupled turbine. The two ENG GBX can be interconnected by friction clutches within the GBX, allowing each ENG GBX to be driven by the APU. For engine start, a friction clutch is installed in the GBX. Selecting the appropriate engine with the APU running will cause the clutch controller to pressurize the appropriate clutch section with hydraulic oil and the selected engine to rotate.

At the same time engine ignition will be initiated. As soon as 35 % RPM is reached, ignition will be switched off. When engine speed reaches 50 % RPM the APU and the starting system automatically shut-down under control of the engine fault detection unit. Now the engine winds up to IDLE by itself.

The second engine is started accordingly. If the second engine does not light up, or if an RPM hang-up occurs, this engine can be cold cranked by the APU.

If 50 % RPM are not reached within 50 sec, or if the throttle is retarded to STOP within this period, the engine starter unit will automatically shut-down.

STARTING / RELIGHT MODES

The engine starting system allows two ground start modes and four relight modes:

- manual start in any order,
- automatic start,
- automatic relight,
- semi-automatic rilight,
- manual relight and preventive relight.

AUTOMATIC START OF BOTH ENGINES

An automatic start procedure for both engines is provided when the start-up mode switch is set to START BOTH engines. Check for APU switch in the guarded position START NORM. Set both throttles to IDLE and depress the GND START button.

The APU will drive the RH engine first. When APU shut-off speed is reached, the APU will be shut-off and restart for LH engine start after a break of 10 sec. This break provides speed synchronization between APU, GBX and mechanical drives.

MANUAL START

During the manual starting procedure, the left engine should be started first.

Check that the APU switch on the engine start panel is in the guarded position START NORM. To select the LH engine set start-up mode switch to LH and the LH throttle to IDLE.

Depressing the GND START button activates the APU system, engine ignition system, and the friction clutch control unit.

Accordingly the RH engine is started by selecting start-up mode switch to RH and the RH throttle to IDLE. The GND START button activates the APU ignition and the clutch systems to start the RH engine.

NOTE

Before the start sequence of the RH engine is initiated by pressing the GND START button, the LH engine has to run in IDLE for at least 40 sec.

CAUTION

Activation of the GND START button during engine run-up and engine operation is prohibited.

RELIGHT MODES

In the event of an impending failure, counteractive measures are taken automatically by the engine starting system. However, depending on circumstances, use of the semi-automatic relight or manual relight may be required. During all relight modes, pure oxygen is injected into the combustion chamber of the engine.

The relight controller of the ECP receives an altitude correction signal when above 18 000 ft MSL to lean the mixture. The controller is shut-off when 85 % RPM or more is sensed.

NOTE

- At altitudes below 40 000 ft MSL, airspeed required for engine relight is 220 to 540 KIAS with a minimum windmilling RPM of 12 %.
- At altitudes between 40 000 ft MSL and 56 000 ft MSL, a successful relight may be expected between 300 KIAS and M 1.8 if relight is initiated during engine wind-down, however, a minimum windmilling RPM of 50 % may be required.
- At altitudes above 56 000 ft MSL, a normal relight and acceleration should not be expected.
- To assure a reliable airstart, the upper limits of the speed ranges should be preferred.
- If the engine is not controllable over the entire RPM range, retard the throttle to off and repeat engine relight at higher speed.

AUTOMATIC RELIGHT

In the event that RPM drops below 50 % and throttle setting is between IDLE and MIL, the engine starter unit will activate the ignition system while the oxygen system will inject oxygen into the combustion chamber. Oxygen is supplied until 75 % RPM are reached, however, the duration of injection is limited to 20 sec. The warning system illuminates the green captions LH or RH ENG START on the TLP.

SEMI-AUTOMATIC RELIGHT

If the automatic relight fails or an engine RPM hang-up above 50 % occurs, a semi-automatic relight has to be initiated.

This is done by retarding the throttle of the failed engine to OFF for 2 to 3 sec and then advancing the throttle to IDLE. As soon as the throttle is advanced out of the OFF position, ignition is provided and oxygen is injected into the combustion chamber until an RPM of 75 % is reached, however, the injection duration is limited to 20 sec. The warning system illuminates the appropriate ENG START caption on the TLP.

MANUAL RELIGHT

When the engine does not start with the semi-automatic relight procedure, a manual air relight must be attempted.

To perform a manual relight, retard the throttle of the failed engine to OFF. Position the safety wired LH AIR RELIGHT or RH AIR RELIGHT switch as required. The green LH / RH ENG START caption illuminates. Advancing the throttle between IDLE and MIL will initiate the engine ignition system. Run-up time of the engine may last up to 70 sec.

Oxygen is injected into the combustion chamber as long as the AIR RELIGHT switch is actuated.

When the relighted engine reaches 50 % RPM, position the LH / RH AIR RELIGHT switch to OFF. To prevent damage to the ignition coils, the LH / RH AIR RELIGHT switch must be switched OFF after 100 sec. The oxygen supply allows approximately five air relight cycles.

PREVENTIVE RELIGHT

The preventive relight mode is activated automatically for 8 sec when:

- either trigger is squeezed,
- an overheat condition is sensed, or
- a surge is encountered.

As during the other relight modes, oxygen is supplied to the engines and the ignition system is activated.

THROTTLES

Two throttles (refer to figure 1-11) are located at the LH side wall. A throttle lever controls each engine from OFF to MAX AB passing through IDLE and MIL power settings. In the OFF position, the throttles are mechanically locked. To position the throttles from OFF to IDLE or IDLE to OFF the locks must be disengaged by squeezing latches which are integrated at the back side of the throttle grips.

When advancing the throttles from IDLE into the AB range or vice versa, locks engage at position MIL. These locks can be disengaged by squeezing latches which are integrated at the front side of the throttle grips. They will lock again in the MAX AB position. To move the throttles out of MAX AB or from MIN AB to MIL, the front latches must be disengaged.

Positioning a throttle from OFF to MAX AB actuates two different micro switches in the ECU. The first one is actuated in IDLE to enable the starter unit. The second switch is actuated at MIL to select the appropriate ECU schedules for MIL and AB.

The throttles are not interconnected.

A friction adjustment lever is mounted aft of the throttles to permit adjustment of throttle friction to suit individual requirements.

GT:

Two interconnected throttle control levers are located in each cockpit. Full throttle control from OFF through MAX AB is selectable in either cockpit by a throttle stroke lever on the left cockpit side wall behind the throttles. Refer to figures FO-2 / FO-3. It allows to switch full throttle control from the front cockpit to the rear cockpit and vice versa, but is available in one cockpit at a time only. With the lever in the front cockpit position the squeeze latches on the rear cockpit throttles are disabled.

THROTTLE ASSEMBLY

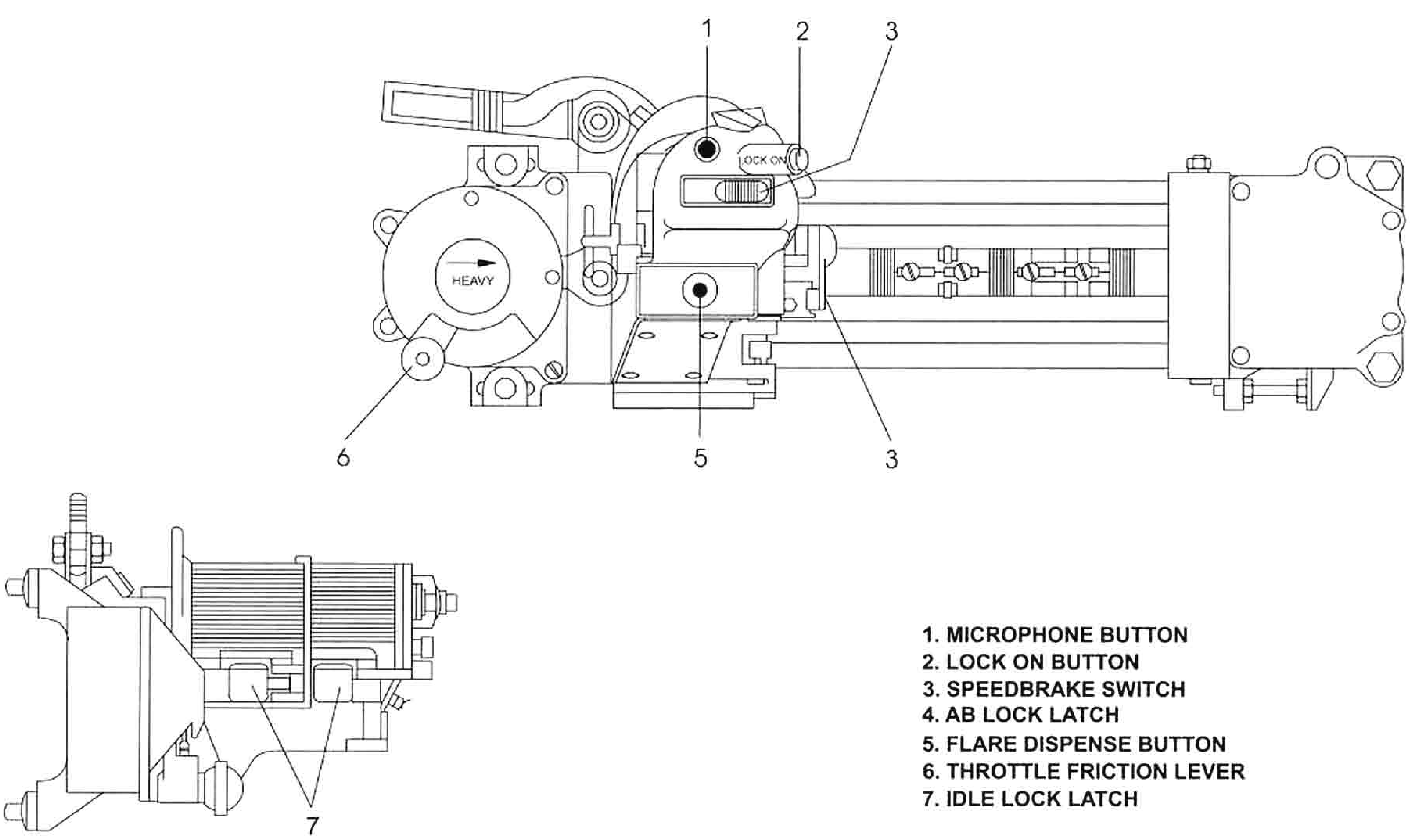


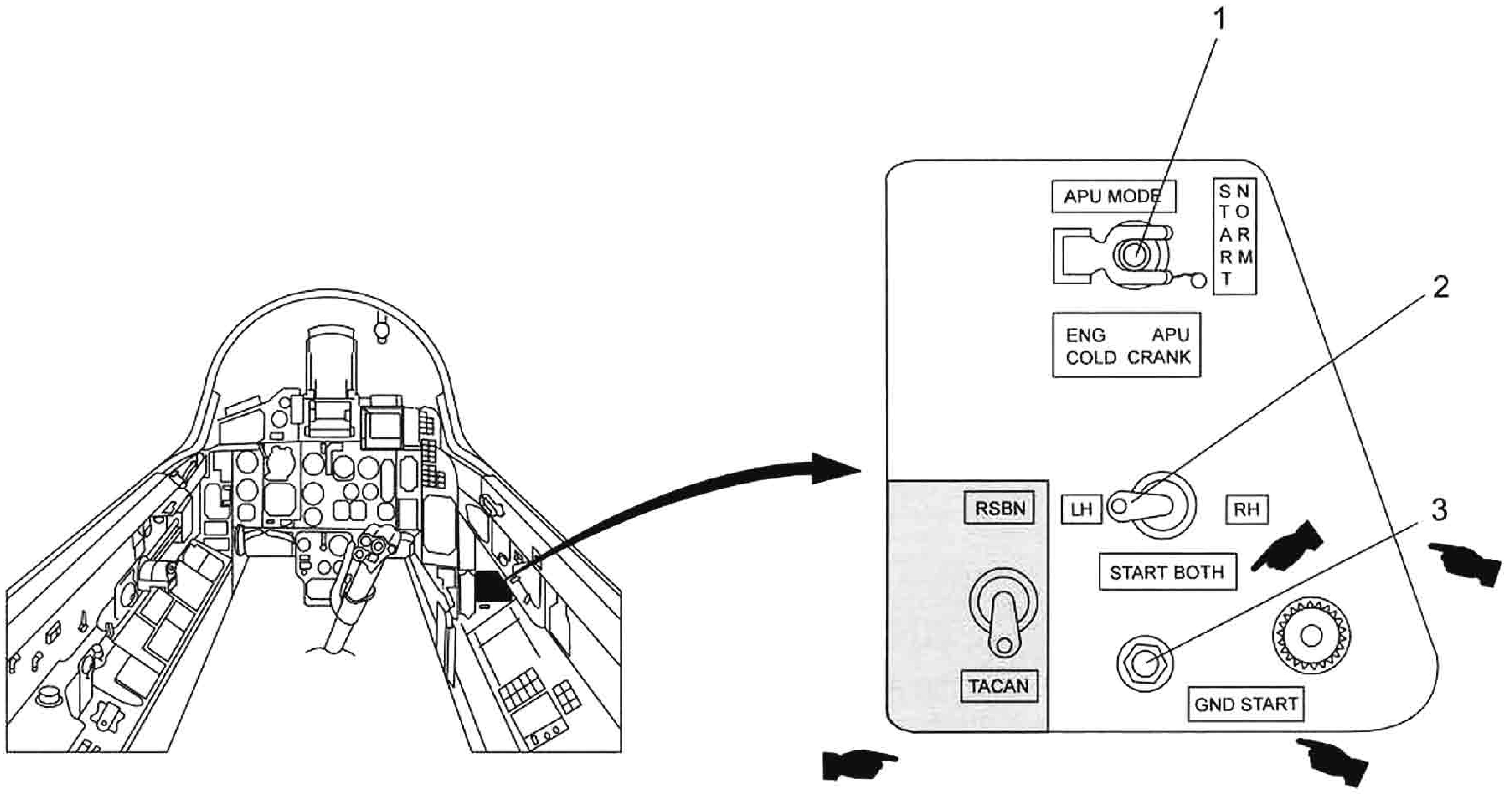
Figure 1-11

ENGINE CONTROLS AND INDICATORS

Engine System Switch

Engaging the ENG SYS switch located on the electric power panel supplies power to the ECU and the APU. Refer to figure 1-22.

ENGINE START PANEL



1. APU SWITCH
2. START-UP MODE SWITCH
3. GND START BUTTON

Figure 1-12

APU Switch

The APU switch located at the engine start panel (refer to figure 1-12) controls the GBX clutch controller and the engine starter unit.

APU MODE	GBX and associated equipment connected.
START NORM	Normal engine start.
ENG COLD CRANK	Engines connected; ignition deactivated. Fuel and oxygen are not supplied.
APU COLD CRANK	APU cranked by the electro starter. Fuel and oxygen are not supplied. Ignition deactivated.

In APU MODE, the various clutches will be engaged and disengaged in such a way that the engines are not connected with the GBX whereas the APU and all the accessories are.

NOTE

Since this mode is used for maintenance purposes only, it should not be selected by the pilot.

The START NORM mode is guarded and safety wired.

Start-Up Mode Switch

The start-up mode switch is located on the engine start panel. It allows selection of individual engine start or start in automatic sequence.

LH	LH engine start
START BOTH	Start both engines sequence RH / LH
RH	RH engine start

Normal position is START BOTH.

Ground Start Button

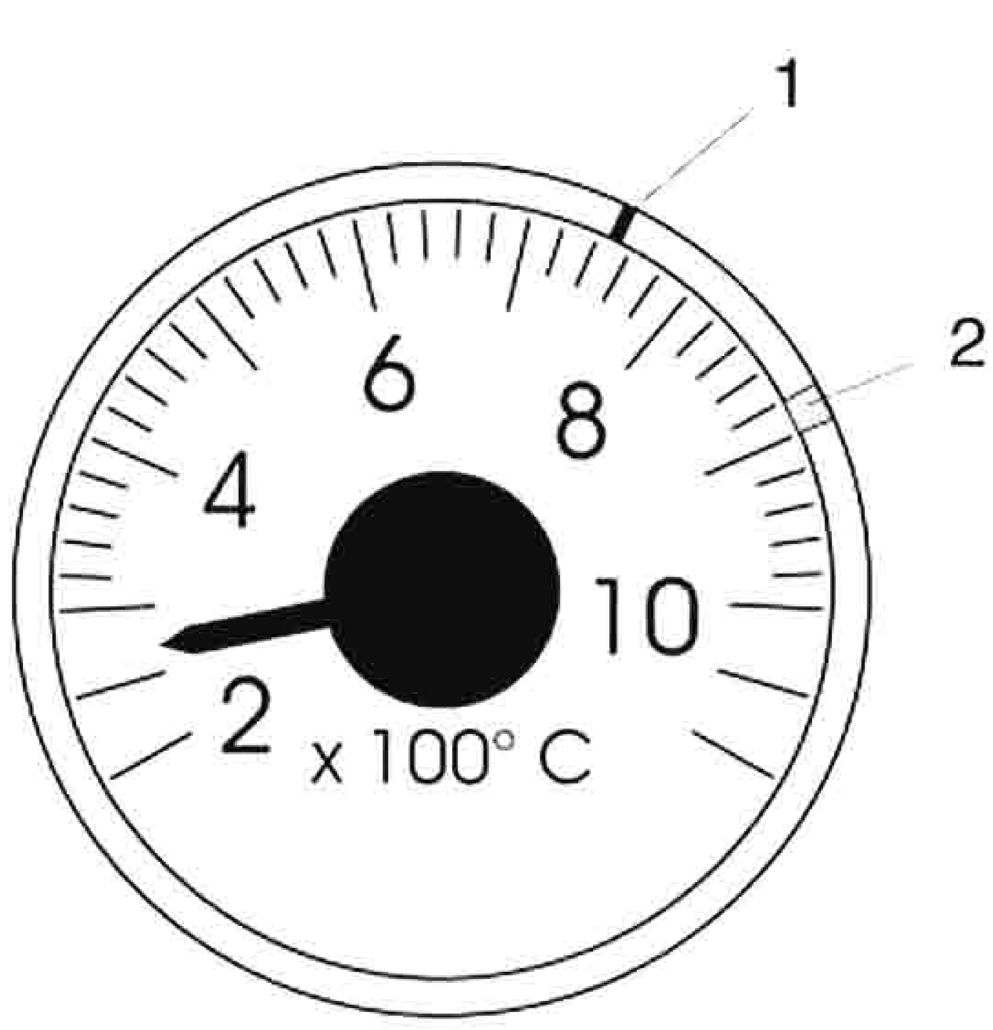
The GND START button is located at the engine start panel. Pressing the button initiates the start and ignition cycle of either the APU and the engines or the APU only depending on APU switch and start-up mode switch setting.

Anti Surge Switch

The ANTI SURGE switch located on the electric power panel (refer to figure 1-22) triggers the activation of the anti-surge system if necessary.

EXHAUST GAS TEMPERATURE INDICATOR

Two indicators provide a rotary pointer display of the EGT from 200° C to 1 100° C in increments of 20° C from 300° C to 1 000° C and 50° C below and above these values. The EGT sensors driving the instruments are located behind the LP turbine outlets. Refer to figure 1-13. The yellow marker and the red sector are adjusted for each individual engine.



1. YELLOW MARKER
2. RED SECTOR

Figure 1-13

ENGINE RPM INDICATOR

A dual pointer RPM indicator displays the RPM of both engines. Refer to figure 1-14.

The scale of the instrument runs from 0 to 110 % at increments of 1 %. Markings on the pointers make reference to the corresponding engine. Metering accuracy is ±1 % below 60 % RPM and above 100 % RPM, and ±0.5 % between 60 % RPM and 100 % RPM.

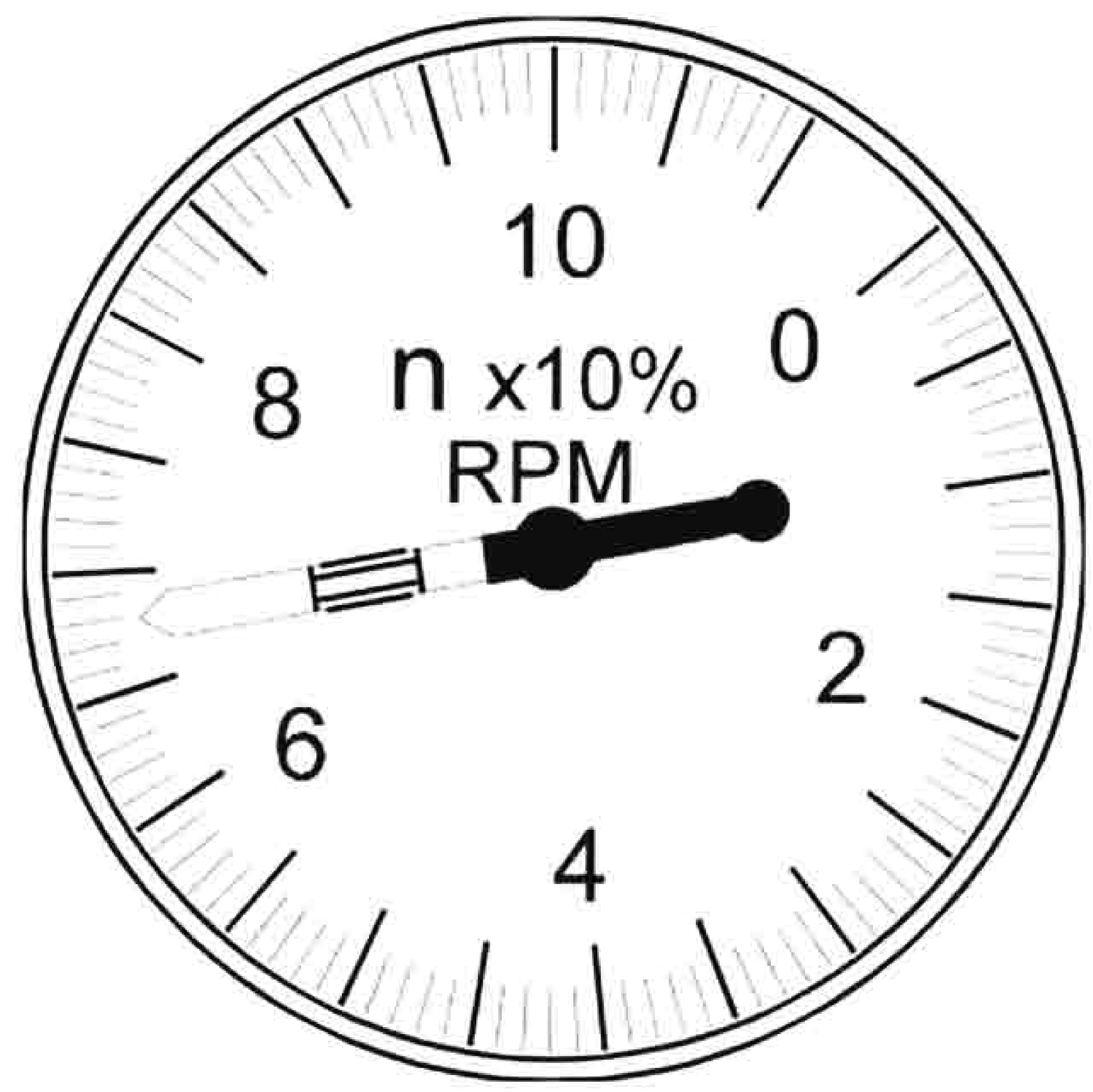


Figure 1-14

Figure 1-15, deleted

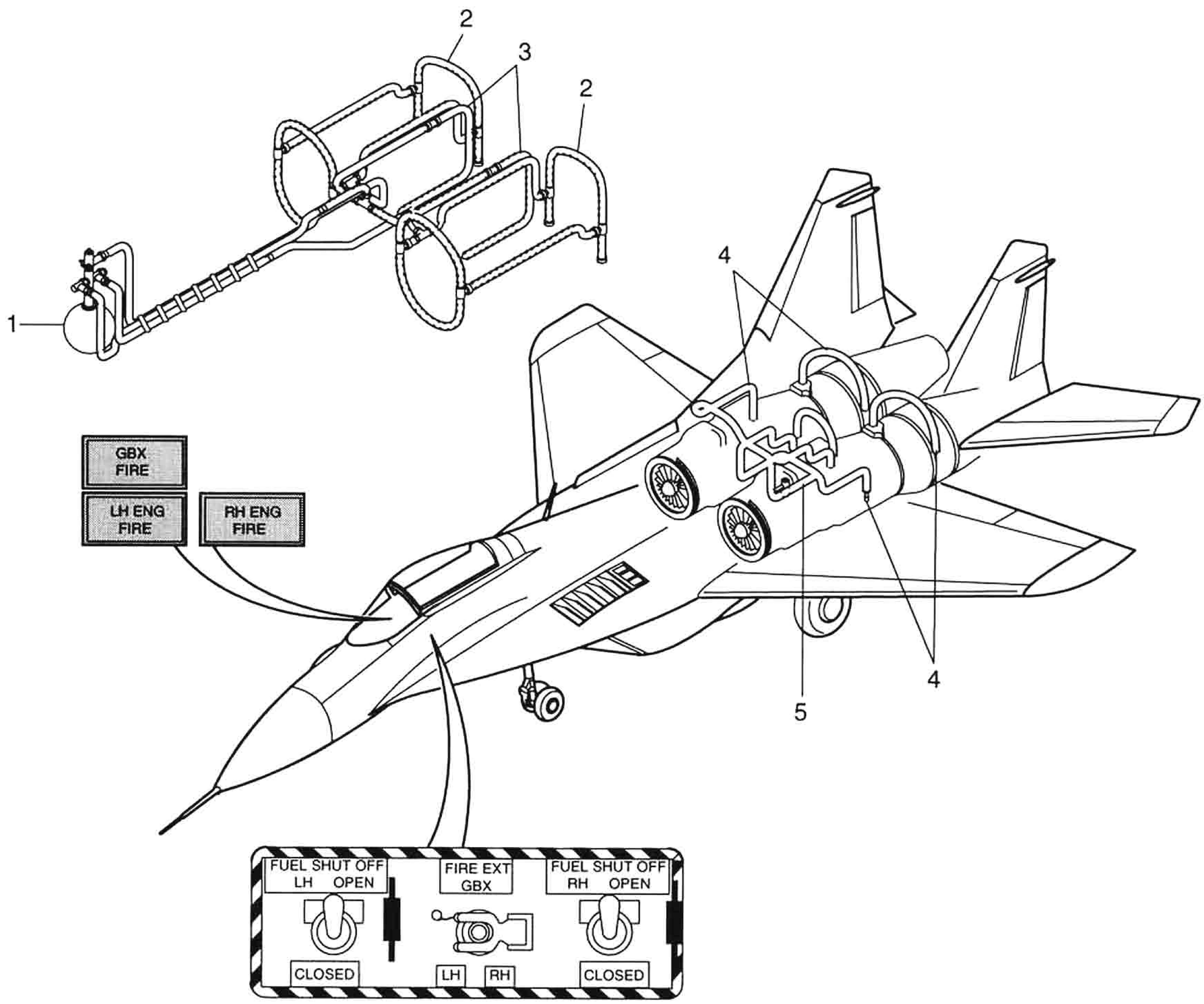
ENGINE FIRE DETECTION SYSTEM

A fire detection system is installed near the GBX and the LH and RH engine. It's function is based on the electrical conductivity of flames. A heat resistant conductor loop is routed through these areas, the distance to the airframe is fixed by high voltage stand-off isolators. In the event of a fire, the high voltage on this conductor loop discharges to the airframe which is electronically detected and initiates a warning signal.

The fire detection system is completely insensitive to high temperatures and will not be triggered by insulation breakdowns as a result of humidity or other electrical short circuits.

The fire detection system will be activated within 3 sec after the appearance of flames.

The extinguisher system consists of a spherical pressure bottle containing an extinguishant foam, three pyro cartridge operated valves and the extinguishing manifolds towards the GBX and the engines, refer to figure 1-16 and 1-17.



1. FIRE EXTINGUISHANT PRESSURE BOTTLE
2. FIRE EXTINGUISHANT SPRAY MANIFOLDS OF ENG COMPARTMENTS
3. FIRE EXTINGUISHANT SPRAY MANIFOLDS OF GBX COMPARTMENTS
4. FIRE WARNING SENSORS IN ENG COMPARTMENTS
5. FIRE WARNING SENSORS IN COMPARTMENTS

Figure 1-16

ENGINE EMERGENCY PANEL

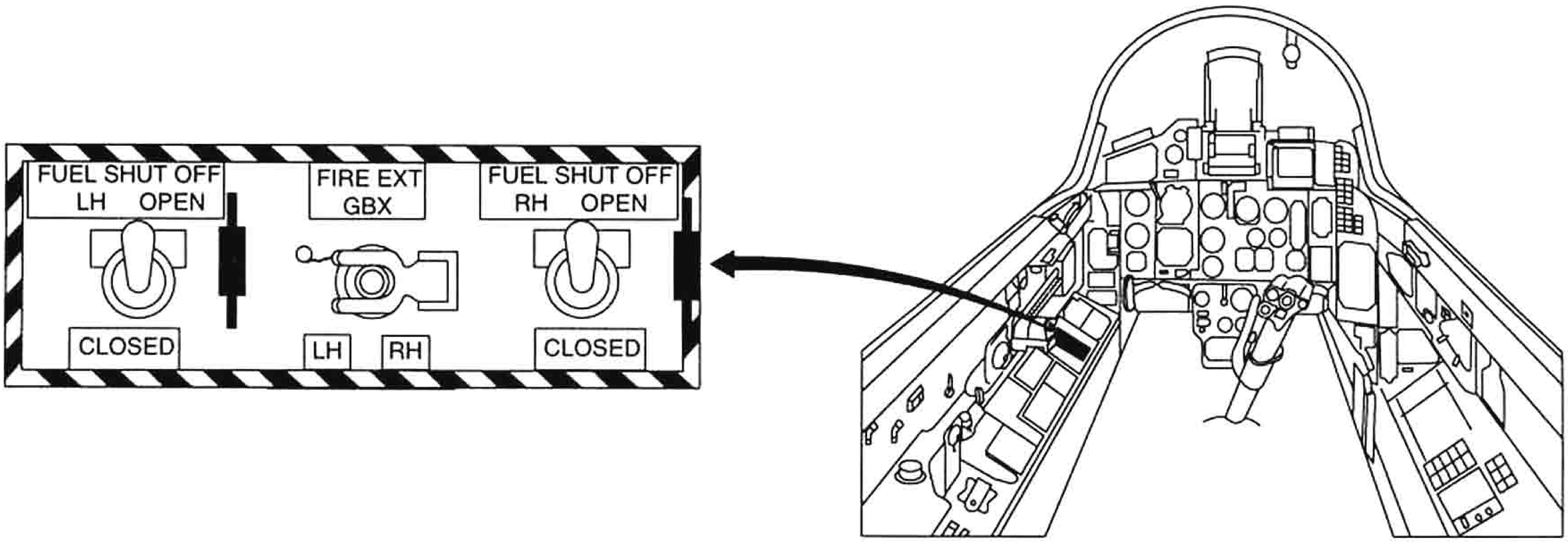


Figure 1-17

FUEL SHUT-OFF LH/RH Switches

Individual FUEL SHUT OFF switches are provided for opening and closing the fuel shut-off valves in the engine boost fuel lines in case of fire, refer to figure 1-17. The fuel shut-off valves are electrically controlled and pneumatically actuated by the pneumatic power supply system.

LH/RH	Fuel shut-off valve open
CLOSED	Fuel shut-off valve closed

Fire Extinguisher Selection Switch

The fire extinguisher selection switch activates the required spray manifolds of the extinguisher system to fight the fire.

GBX	Selects the spray manifolds within the GBX compartment
LH/RH	Selects the spray manifolds within the LH/RH engine compartments

INDICATIONS AND WARNINGS

The red warning captions on the TLP will extinguish if fire fighting has been successful.

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	LH ENG FIRE	Fire in LH engine
VIWAS	"BORDNUMMER, FEUER IM LINKEN TRIEBWERK" (message will be paged twice)	

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	GBX FIRE	GBX fire.
VIWAS	"BORDNUMMER, FEUER IM KSA" (message will be paged twice)	

WARNING

The extinguisher system can be activated only once and all foam will be consumed entirely. The pilot must exercise extreme caution selecting the correct position of the guarded fire extinguisher selection switch.

ENGINE OPERATION

The engine is controlled hydro-mechanically. Under various flight conditions the engine performance is governed electrically by the ECU.

The engine can be operated in three different modes:

- Normal power mode
- Combat mode
- Limited power mode

These modes can only be selected prior to flight by the ground crew.

Control of the engines by throttle bursts within 1 to 2 sec between IDLE and MAX AB is permissible. Time for ignition of AB is 2 to 3 sec. During flight, time for engine acceleration from IDLE to MAX AB is 3 to 7 sec.

NOTE

RPM hang-up is possible momentarily at altitudes above 30 000 ft and airspeeds below 300 KIAS during acceleration from IDLE to MIL.

G-forces can cause RPM changes up to 7 %. However, RPM must not exceed 103 %. A change in EGT is possible as well, but must not exceed the maximum value for MIL operation.

Selecting AB from the MIL power position is assured in the entire range of attitude and velocity. However, during throttle bursts from IDLE to AB at altitudes above 43 000 ft and close to minimum speed, AB ignition may not be available.

NOTE

If AB does not light, the throttle must be retarded to the MIL position. After the engine has stabilized, AB may be reselected.

AB ignition can be verified by observing the green caption LH / RH ENG AB.

NOTE

If the AB does not shut down when deselected, it can be shut down by use of the safety wired AB EMERG OFF switch located on the engine emergency panel. This switch shuts down the AB of both engines.

NORMAL POWER MODE

The performance data throughout the manual, refer to a normal tuned engine, i.e. engine in normal power mode (NPM), except mentioned otherwise.

COMBAT MODE

The combat mode provides additional thrust to MAX AB and is operational under the following prerequisites:

- Combat selector switch МАКС РПТ (MAKS-RPT = normal - combat) located in the LH gear bay, has to be selected prior to flight to the position RPT to achieve the additional thrust. This causes a maximum EGT value of approximately 25° C above the value of normal operation.
- Throttle setting MAX AB.
- Airspeed M ≥ 1.5.

NOTE

Engine operation in combat mode must not exceed 2 % of service life time.

LIMITED POWER MODE

In order to enhance the service life time, the engine is tuned to limited power mode (LPM). Refer to TA 9002.

Tuned to LPM, the ECU limits the maximum EGT about 20° C below the value of normal operation. As a result, NH and NL is reduced also.

RUNNING TIME METERS

Six running time meters located in the LH gear bay record engine operation times in combat and AB modes and total running time.

AUXILIARY POWER SYSTEM

The auxiliary power system (APS) provides facilities for starting the engines on ground and in the air, and transmits mechanical power to drive various accessories. Refer to figure FO-7.

The system consists of an APU, a GBX, and two ENG GBX. The GBX is mounted between the two ENG GBX.

The APU drives the GBX which itself drives the two ENG GBX through two angle drives.

The GBX drives the DC and the AC generator, two hydraulic pumps of the hydraulic system and an active fuel pump of the fuel system. Each ENG GBX drives four associated engine system fuel pumps, three oil pumps and two speed sensors to support its associated engine.

The GBX can be driven by either engine via its ENG GBX through the angle drives or it drives each ENG GBX and the associated engine for engine start.

AUXILIARY POWER UNIT

The APU is a gas turbine using aircraft fuel and oxygen injection to get started. It drives the GBX through an exhaust gas coupled turbine.

The APU is used for engine start and for cold cranking to checkout engine systems. It also provides internal power to drive the accessory equipment for aircraft systems checkout.

APU STARTING SYSTEM

To start the APU, a 28 VDC starter motor is used. The electrical power is indicated on the voltmeter located on the pedestal panel.

Pressing the GND START button activates the engine starter unit to supply oxygen and energizes the engine ignition unit. After 1 sec the starter motor is switched on to wind up the APU compressor shaft. Simultaneously engine fuel is injected into the oxygen atmosphere within the combustion chamber. At 35 % RPM, the starter motor, the ignition and the oxygen supply are switched off. At 100 % RPM, operating speed is controlled by a fuel flow regulating governor. Power output is 77.5 kW.

INDICATIONS AND WARNINGS

CAUTION

If the exhaust gas temperature of the APU exceeds a preset value, the engine fault detection unit will activate the information and warning equipment. In this case the APU has to be shut off immediately.

	INDICATION	FAULT / EFFECT
AEKRAN	**START TURB CRIT CONDITNS**	The exhaust gas temperature of the APU exceeds a preset value.

APU / ENGINE COLD-CRANKING

For ENG or APU system checkout, the engines and the APU can be cold-cranked. To cold-crank an engine the APU switch is set to ENG COLD CRANK, the throttle to OFF and the start-up mode switch to LH or RH. Upon depressing the GND START button, the HP shaft of the selected engine will be driven by the APU. Fuel will not be injected into the engine combustion chamber and the engine ignition system will not be activated.

For cold-cranking the APU, the APU switch has to be set to APU COLD CRANK and the GND START button has to be depressed. In this mode the electrical starter motor will drive the APU compressor shaft. Fuel, oxygen and ignition will not be supplied to the APU.

INTERNAL POWER SUPPLY MODE

On the ground, the APU can be used to generate electrical and hydraulic power. In this mode the two ENG GBX are disconnected from the GBX to prevent engine rotation.

For internal power supply, the ENG SYS switch on the system power panel is set to ON and the APU switch on the engine start panel is set to APU MODE. With this setup, only the APU is started when the GND START button is pressed. The APU will drive the GBX and the associated equipment at a speed equivalent to 70 % engine RPM. The APU is shut down with the ENG SYS switch located on the electric power panel.

NOTE

This function shall not be used.

ACCESSORY GEARBOX

The GBX drives aircraft accessories such as pumps, generators etc. Refer to figure FO-7.

The GBX also transfers torque generated by the APU to the selected engine and controls drive from an engine to the accessory units.

Torque is transferred between the GBX and the two ENG GBX by angle drives.

The lube oil unit and two hydraulic pumps provide cooling and lubrication for the GBX and APU.

Oil quantity is 4.5 l. Sensors for vibration and oil pressure are installed.

A cross drive shaft and three friction clutches are engaged by oil pressure from the hydraulic pumps. The clutches are controlled by an internal control unit depending on the mode of operation and engine selected for start.

Normally the GBX is driven by the RH engine. However, if the LH engine RPM exceeds the RH engine RPM by 7 % or more, a freewheel clutch will select the LH engine and drive the GBX instead.

All accessories except the AC generator are gear-driven by the GBX with a fixed transfer ratio. The AC generator is driven by a constant speed torque converter.

INDICATIONS AND WARNINGS

Malfunctions in the GBX detected by the engine fault detection unit are issued to the warning equipment.

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	GBX FIRE	Fire in GBX.
VIWAS	"BORDNUMMER, FEUER IM KSA" (message will be paged twice)	

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	OIL GBX	Oil pressure in the GBX is below the min value for at least 20 sec.
AEKRAN	OIL PRESS ACCRY GBX	
VIWAS	"SCHMIERSTOFFDRUCK IM KSA ZU GERING" "AUFGABE ABBRECHEN"	

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
AEKRAN	ACFT ACCRY GBX VIBR	Vibration level of the GBX exceeds a preset level by 35 % for at least 2 to 3 sec.
VIWAS	"VIBRATION IM KSA" "AUFGABE ABBRECHEN"	

ENGINE GEARBOX

Two ENG GBX are installed, one for each engine to drive engine accessories. Refer to figure FO-7.

For normal operation, torque is transferred to the GBX via an angle drive. The angle drive receives torque from the GBX to start-up the engine.

The following engine accessories are mounted on the ENG GBX:

A low pressure fuel pump, an ECP, a nozzle high-pressure pump, an AB fuel pump with NH sensor unit, a lube oil unit, an oil centrifugal breather, a fuel filter unit and an additional NH sensor unit.

AIRCRAFT FUEL SYSTEM

Fuel is carried internally in five interconnected internal fuselage tanks and two internal wing tanks.

External fuel is carried in a single 1 500 l fuselage-mounted centerline (CL) tank on station seven.

All internal tanks may be refueled on the ground through a single pressure refueling point, located in the left main gear well.

The CL tank must be refueled individually through a pressure refueling point, located in the front section of the tank.

The fuselage tanks are arranged so that tank 1 is behind the aft bulkhead of the cockpit. Tank 2, the engine feed tank, tank 3 and the two tanks 3A complete the fuselage tank group. A fuel accumulator is installed in tank 3 to supply engine fuel during near-zero-g flight. Tanks 1 and 2 are arranged so that fuel will gravity-flow into tank 2 if a transfer pump failure occurs. Flapper valves prevent reverse fuel flow. The two internal wing tanks are installed in the wing roots, one at each side. Refer to figure 1-18.

All fuel is transferred to the engine feed tank, tank 2, and from there fed to the engines. Check valves within the transfer lines prevent reverse fuel flow in all aircraft attitudes. If the transfer rate to tank 2 is higher than the fuel consumption, a pressure relief valve opens and fuel is dumped to tank 1. A safety relief valve between tank 2 and 3 permits dumping of fuel to tank 3.

Both tanks 3A and the internal wing tanks contain a jet pump to transfer fuel to tank 3. Tank 3 contains a jet pump to transfer fuel to tank 1 and additionally a turbo pump to transfer fuel to the engine feed tank 2. Tank 1 contains a turbo pump to transfer fuel to tank 2.

The fuel transfer sequence is automatically controlled by a hydro-mechanical system. Regulated engine bleed air is used to transfer fuel from the external tanks to the internal tanks. Internal fuel transfer is accomplished by transfer pumps.

Air pressure or nitrogen pressure is used to maintain positive pressurization in all internal tanks, air pressure only is used for pressurization of the centerline tank.

Fuel is also used as a cooling medium to cool hydraulic and lube oil as well as the cooling fluid for the radar equipment.

Level control valves control the fuel levels in the tanks during transfer operations. Fuel gaging units supply quantity and flow data to the indication system.

AFTER MODIFICATION WITH WING DROP TANKS

Additional external fuel is carried in two 1 150 l wing drop tanks. The wing drop tanks are suspended by pylons, mounted to the wing stations one and two. Refueling of the wing drop tanks is accomplished individually through external filler points.

Regulated engine bleed air is used for wing drop tank pressurization and fuel transfer.

FUEL TANKS ARRANGEMENT

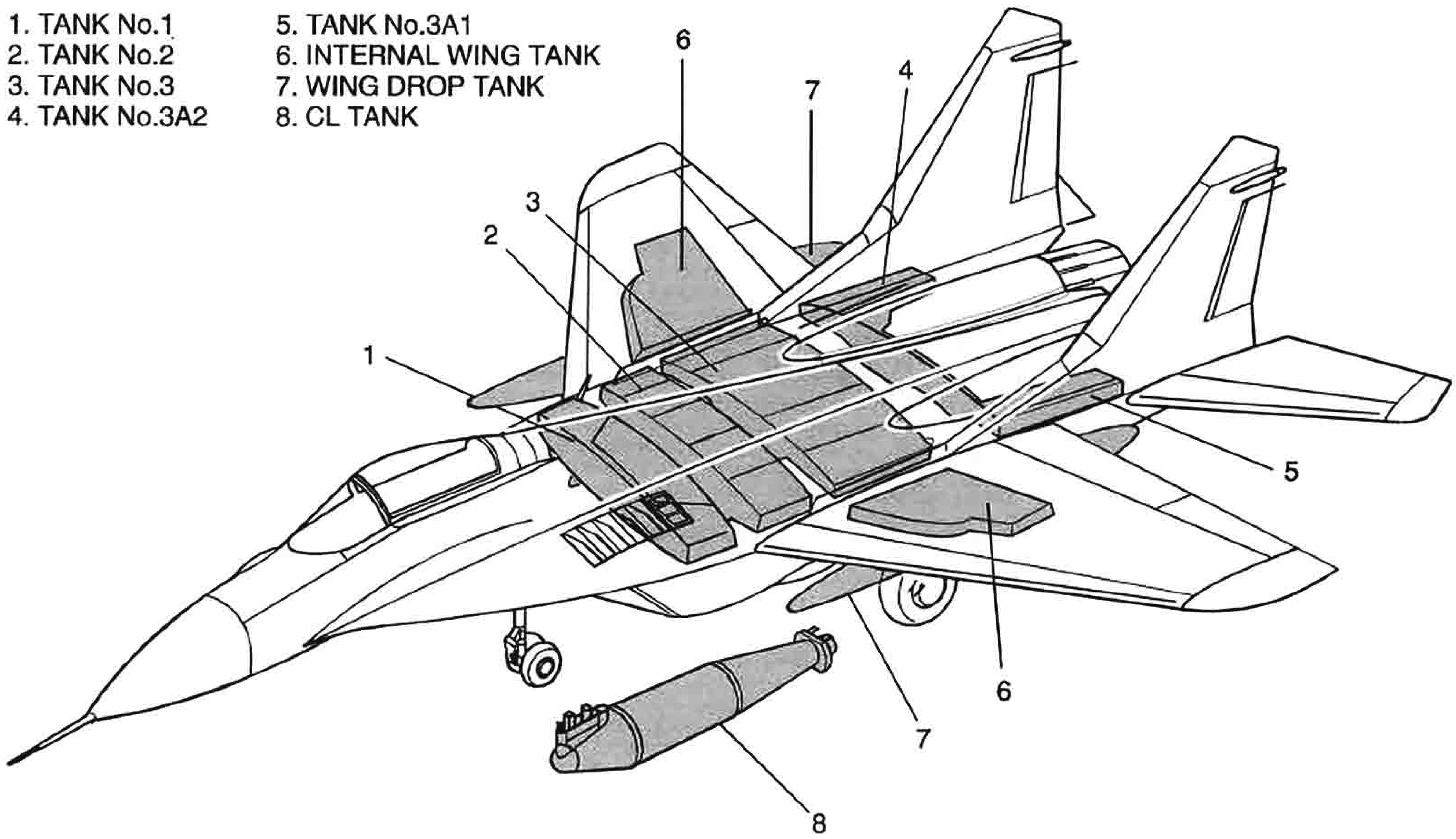


Figure 1-18

TRANSFER SYSTEM

The transfer system serves two purposes:

- transfer all fuel to the engine feed tank,
- maintain CG variation due to fuel consumption within limits.

The transfer system is a self-sustaining system. The accessory gearbox drives a centrifugal-type pump for active fuel. Active fuel is used to drive / control the associated components:

- level / pump control valves installed in tanks 1, 2, 3 and the internal wing tanks at certain levels,
- empty sensors in tanks 1 and 3 and the centerline tank,
- electromagnetic valves for control of external fuel transfer,
- turbo-type transfer pumps in tanks 1 and 3,
- jet-type transfer pumps in tanks 3, 3A and the internal wing tanks.

Upon switchover to internal power supply, the electromagnetic control valve for CL tank fuel transfer opens. After engine start, the turbo-type transfer pumps in tank 1 are running to transfer fuel to the engine feed tank, i.e. tank 2. The CL tank fuel shut-off valve and the transfer valve open, allowing fuel to transfer. However, CL tank fuel is not transferred at power settings below 80 % RPM. Above 80 % RPM, all CL tank fuel is transferred to tank 1. As soon as all fuel has been transferred, a sensor sends a tank empty signal to the fuel indicator and to an electromagnetic check valve. The check valve interrupts control pressure to the transfer valve, causing the valve to close. During negative g flight, an inertia switch associated with the check valve causes the transfer valve and the shut-off valve to close momentarily to prevent pressurized air from entering tank 1 and forcing fuel into the drain lines.

NOTE

At altitudes above 30 000 ft, centerline tank fuel may or may not transfer due to design limitations.

After transfer of a total of approximately 250 l fuel from tank 1, the transfer pumps in tank 3 start transferring fuel to tank 1 and to the engine feed tank.

Upon depletion of approximately 200 l from tank 3, the jet-type transfer pumps in tanks 3A and the internal wing tanks start transferring fuel to tank 3. Tanks 3A keep feeding until they are empty, however, there is no indication for the exhaustion of tanks 3A.

When the internal wing tanks are empty, a sensor sends a tank empty signal to the fuel indicator. When all fuel is transferred from the internal wing tanks and the tanks 3A, and approximately 600 l are depleted from tank 3, the transfer pumps of tank 3 are shut down.

Consecutively another 280 l are transferred from tank 1 to the engine feed tank.

Transfer from tank 3 is resumed and when tank 3 is empty, the empty sensor signals depletion to the fuel indicator. The turbo-type transfer pump in tank 3 and the transfer pumps in tank 3A as well as the internal wing tanks are shut down, the jet-type transfer pump in tank 3 continues to pump active fuel to tank 1, from where it is transferred to the engine feed tank.

When tank 1 is empty, the empty sensor causes the respective caption on the fuel indicator to illuminate.

When the amount of fuel in the engine feed tank has diminished to 550 kg, the caption 550 KG REMAIN on the TLP illuminates.

After complete depletion of the engine feed tank, fuel is forced from the fuel accumulator to the engine feed line. A pressure differential sensor causes the AEKRAN to indicate NO BOOST when the entire fuel has been consumed.

Figure 1-19 illustrates the fuel transfer sequence.

AFTER MODIFICATION WITH WING DROP TANKS

With wing drop tanks installed, the transfer sequence is essentially the same. However, since the pressurization system has been modified, CL tank fuel is transferred immediately, regardless of engine RPM.

When 70 l have been transferred from the internal wing tanks, the transfer valves of the wing drop tanks open and fuel is transferred to the internal wing tanks. Upon completion of fuel transfer from the wing drop tanks, an empty sensor signals fuel depletion to the fuel indicator and to an electromagnetic check valve. The check valve interrupts control pressure to the transfer valve causing the valve to close. During negative g flight, an inertia switch associated with the check valve causes the transfer valve to close momentarily to prevent pressurized air from entering the internal wing tanks.

After depletion of all fuel from the wing drop tanks, the transfer sequence continues the same way as with no wing drop tanks installed.

Figure 1-19A illustrates the fuel transfer sequence.

GT:

After engine start, fuel is transferred from tank 1 to the engine feed tank.

When approximately 50 l of fuel are transferred, the CL tank transfer valve opens, allowing fuel to transfer to tank 1 as soon as RPM is increased above 80 %. Upon completion of fuel transfer, an empty sensor signals fuel depletion to the fuel indicator, the transfer valve closes.

After depletion of another 500 l from tank 1, the transfer pumps in tank 3 start transferring fuel to tank 1 and to the engine feed tank.

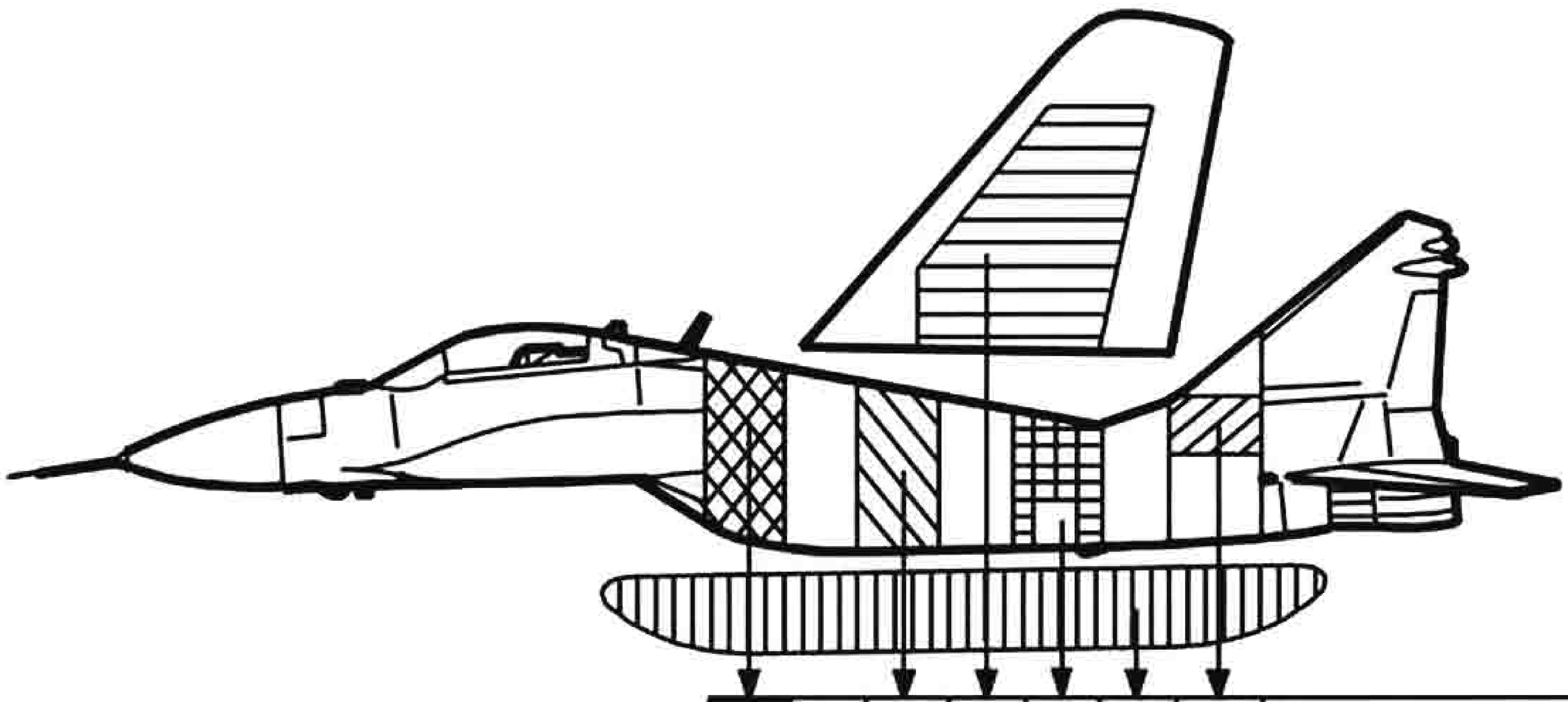
Upon depletion of approximately 50 l from tank 3, the jet-type transfer pumps in tanks 3A and the internal wing tanks start transferring fuel to tank 3. When the internal wing tanks are empty, an empty sensor signals fuel depletion to the fuel indicator.

Upon completion of fuel transfer from tanks 3 and 3A, an empty sensor signals fuel depletion to the fuel indicator, shuts off the jet-type transfer pumps in tanks 3, 3A and the internal wing tanks, and closes the transfer valves in tanks 3A and the internal wing tanks. The turbo-type transfer pumps in tank 3 continue running.

The transfer sequence is continued with tank 1, 2 and the accumulator tank with the respective indicator / warning captions illuminating at the appropriate fuel level.

Figure 1-19 illustrates the fuel transfer sequence.

FUEL TRANSFER SEQUENCE



G:

	TANK NO.	TRANSFER
	1	50 L DURING GND OPERATION
	CL	COMPLETE
	1	250 L APPROX.
	3	200 L APPROX.
	3A	COMPLETE
	INT. WING	COMPLET E
	3	600 L APPROX.
	1	280 L APPROX.
	3	COMPLETE
	1	COMPLETE
550 KG REMAINING	2	COMPLETE
	ACCUM.	COMPLETE
		125 L UNUSABLE FUEL

GT:

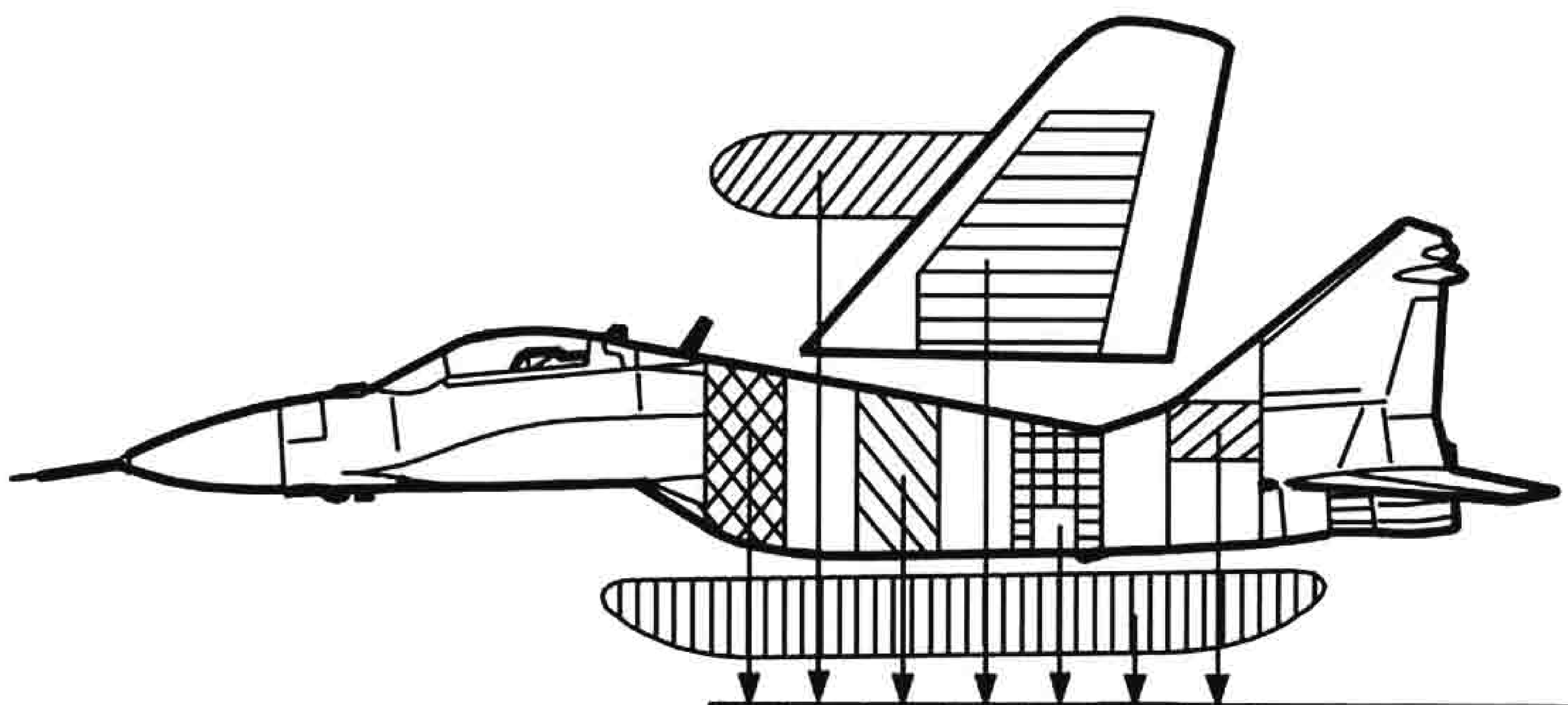
	TANK NO.	TRANSFER
	1	50 L
	CL	COMPLETE
	1	500 L APPROX.
	3	50 L APPROX.
	3A	COMPLETE
	INT. WING	COMPLETE
	3	COMPLETE
	1	COMPLETE
550 KG REMAINING	2	COMPLETE
	ACCUM.	COMPLETE
		125 L UNUSABLE FUEL

FUEL TRANSFER

Figure 1-19

FUEL TRANSFER SEQUENCE

AFTER MODIFICATION WITH WING DROP TANKS



								FUEL TRANSFER	
								TANK NO.	TRANSFER
					■			CL	COMPLETE
■								1	250 L APPROX.
				■				3	200 L APPROX.
			■			■		INT. WING	70 L APPROX.
	■					■		3A	COMPLETE
	■							WING DROP TANKS	COMPLETE
			■					INT. WING	COMPLETE
				■				3	600 L APPROX.
■								1	280 L APPROX.
				■				3	COMPLETE
■								1	COMPLETE
		■					550 KG REMAINING	2	COMPLETE
				■				ACCUM.	COMPLETE
									125 L UNUSABLE FUEL

Figure 1-19A

FUEL PRESSURIZATION AND VENTILATION SYSTEM

The pressurization and ventilation system uses air or nitrogen to pressurize the internal tanks. Regulated engine bleed air is used to pressurize the accumulator tank and all external tanks.

Both subsystems are interconnected by check valves to permit pressurization of the internal tanks by engine bleed air when the nitrogen supply is exhausted or when the electro-pneumatic nitrogen control valve is closed.

Four pressure bottles contain the nitrogen at a pressure of 32 MPa. During engine start, at 55 % RPM, the electro-pneumatic nitrogen control valve opens and nitrogen is supplied via a pressure reduction valve to the venting unit at a pressure of 0.8 MPa. The venting unit controls pressurization of the internal tanks between 3 kPa and 25 kPa, depending on aircraft altitude. It also provides pressure relief, whenever a predetermined value is exceeded.

The circuitry for the nitrogen control valve is routed through the left main gear scissors switch to prevent closure of the valve in case of an engine failure during flight.

As soon as engine bleed air from the low pressure compressor is available, the fuel accumulator and the external tanks are pressurized. Engine bleed air is delivered at a pressure of 50 Pa to 500 Pa and reduced by safety relief valves to deliver a constant pressure of 55 ±5 Pa to the fuel accumulator and 90 ±10 Pa to the external tanks for pressurization and fuel transfer purposes.

NOTE

At high altitudes and / or power settings below 80 % RPM, bleed air pressure may be insufficient to maintain fuel transfer from the CL tank, resulting in intermittent illumination of the AEKRAN indication DROP TANK NO USAGE.

If the nitrogen supply is exhausted or the nitrogen control valve closed, engine bleed air is supplied to the vent unit through the check valve for pressurization of the internal tanks.

AFTER MODIFICATION WITH WING DROP TANKS

Prior to the modification, pressure from the fuel pressurization and ventilation system may have been insufficient to transfer external fuel at low power settings and high altitude. To maintain positive pressurization for fuel transfer, the fuel pressurization and ventilation system has been modified to use compressed air from the cabin pressurization system during high altitude and low engine RPM conditions.

FUEL BOOST SYSTEM

The fuel boost system supplies fuel for the following purposes:

- Engine operation during positive and negative g flight,
- operation of the active fuel system,
- cooling of the radar cooling fluid and,
- cooling of engine oil.

Two turbo-type boost pumps are located in the engine feed tank 2. The pumps are arranged so that at least one pump remains submerged in fuel regardless of flight attitude to deliver fuel to the engine low pressure fuel pumps during positive and negative g flight.

A part of the fuel is pumped to the fuel accumulator in tank 3. The accumulator consists of two chambers, separated by a membrane. The upper chamber is pressurized by compressed air from the fuel pressurization and ventilation system. During positive or negative g flight, fuel is transferred to the lower chamber, the membrane is bent toward the upper chamber. The compressed air in the upper chamber forces the fuel from the lower chamber of the accumulator into the fuel supply lines in case pressure in these lines is insufficient during near-zero g flight conditions.

ACTIVE FUEL

A part of the fuel is routed to the active fuel pump. The centrifugal-type pump is mounted to the accessory gearbox. It increases fuel pressure depending on engine speed to drive the transfer pumps and the boost pumps.

The active fuel is also used to control the level control valves, the transfer pumps and the external tank pressurization valves and transfer valves.

COOLING FUEL

Fuel from the lower boost pump is used to cool the cooling fluid of the radar equipment cooling system. After passing the heat exchanger, the fuel is routed to the entrance of the upper boost pump.

Active fuel is routed to the oil cooler and returned to the fuel supply line.

Two temperature controlled valves open a return line to the engine feed tank when the fuel temperature after the engine low pressure fuel pump exceeds 105° C. The resulting increased fuel flow causes the temperature to decrease.

STARTER FUEL PUMP

The motor driven starter fuel pump supplies fuel to the APU and to the engine supply lines during engine start. The pump remains in operation as long as either engine is running. Therefore it provides a back-up for fuel supply in case of an active fuel system and a resulting boost pump failure.

FUEL PUMP Switch

The FUEL PUMP switch is located on the electric power panel on the RH console. Refer to figure 1-22. It is used to activate the starter fuel pump.

EXTERNAL TANK JETTISON SYSTEM

The CL tank jettison button is located on the control stick. The button is safety covered to prevent inadvertent actuation. Pressing the button causes an electromagnetic lock to open and releases the tank. The system is DC powered by an external source, the DC generator or the batteries.

WARNING

Jettison on the ground is possible with DC power available.

AFTER MODIFICATION WITH WING DROP TANKS

The EMERG RELEASE button located on the front panel is used to jettison the wing drop tanks. Refer to figure 1-19B. The button is safety covered and secured to prevent inadvertent actuation. Pressing the button causes the release cartridges to fire. A safety device automatically activates the firing circuit of the remaining wing drop tank upon separation of either tank to prevent jettisoning of a single wing drop tank which would result in a severe asymmetry.

WARNING

The jettison circuit will be hot as soon as DC power is available.

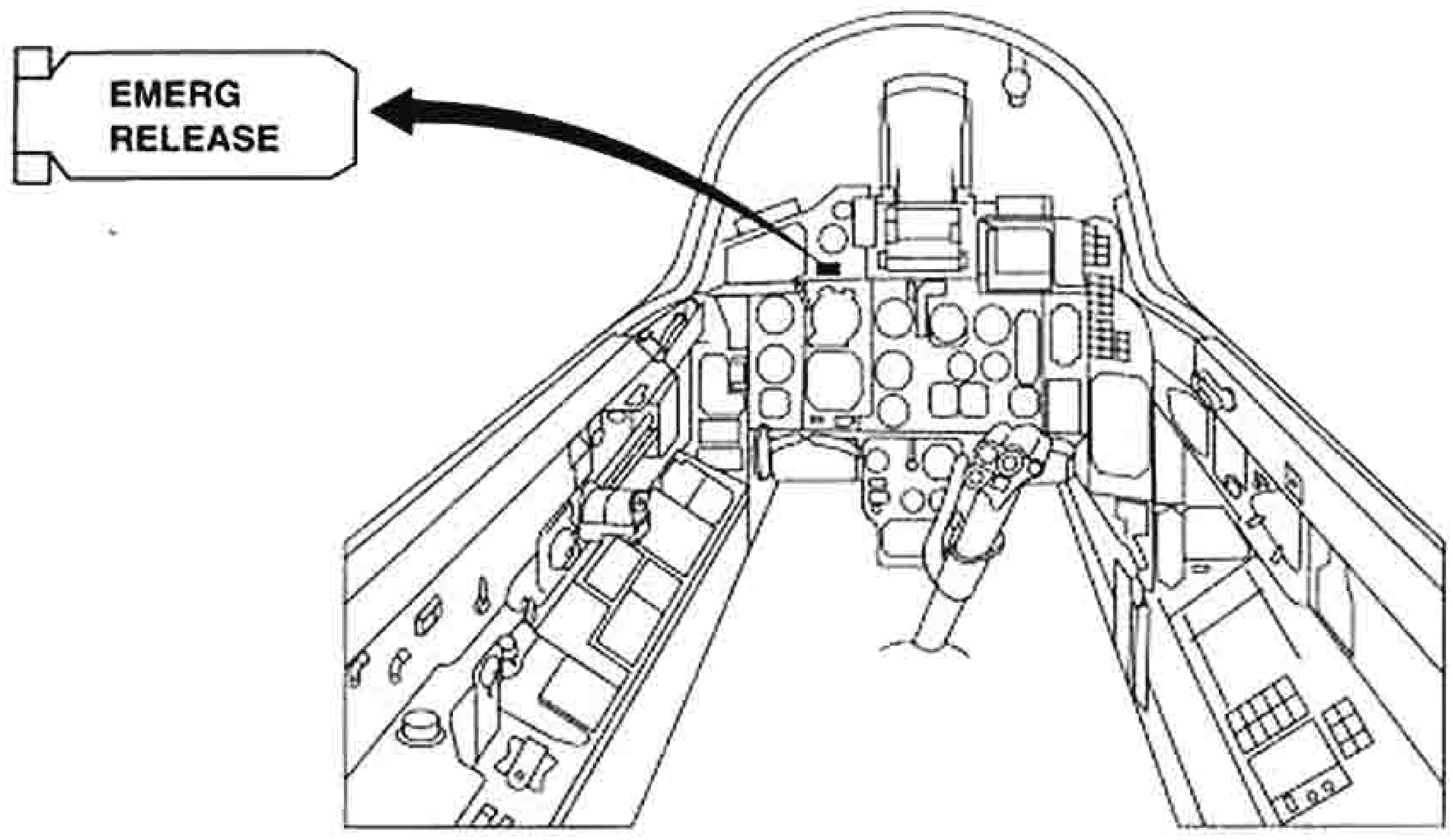


Figure 1-19B

GROUND REFUELLING / DEFUELLING

All internal tanks can be refueled via a single pressure refueling point or through external filler points.

The single point refueling receptacle is located in the left main wheel well. External filler points are installed on tanks 1 and 3. The CL tank will not be replenished during single point refueling and must be refueled individually.

The central refueling process is controlled from the refueling panel located in the left main gear bay. Following selections are available:

- 50 % Partial load, for refueling of tank 2 and partial refueling of tank 1 and 3;
- 100 % full load, for refueling of all internal tanks;
- 130 % full load plus centerline tank;
- CL centerline tank.

The refueling process automatically stops when the selected fuel load is reached and the caption REFUEL COMPLETE illuminates.

When refueling of internal tanks plus centerline tank is selected, the central refueling process automatically stops when the internal tanks are refueled completely, however, the caption REFUEL COMPLETE does not illuminate until the CL tank is full.

To refuel the CL tank, the respective position has to be selected. After completion of centerline tank refueling, selection of the position for full load plus centerline tank is mandatory for adjustment of the indication system.

If the refueling vehicle is equipped with two connectors for single point refueling, simultaneous refueling of the internal tanks and the centerline tank is permissible.

The internal tanks are defueled through a drain valve located near tank 3. The CL tank is defueled through its own drain valve. Remaining fuel can be drained by opening several drain plugs.

AFTER MODIFICATION WITH WING DROP TANKS

The wing drop tanks are always refueled last. They can only be refueled manually through filler caps. To obtain the correct fuel amount, the fuel indicator has to be adjusted manually for the amount of fuel in the wing drop tanks (approximately 1 800 kg) with the MAN adjustment knob on the fuel signals calibration panel in the rear of the cockpit.

The wing drop tanks are defueled through the filler caps.

FUEL QUANTITY DATA TABLE

	FULLY SERVICED			USABLE FUEL		
TANK	LITER	F-34 / kg *1)	TS-1 / kg *2)	LITER	F-34 / kg *1)	TS-1 / kg *2)
TANK 1	650	522	507			
TANK 2	900	723	702			
TANK 3	1 800	1 445	1 404			
TANKS 3A	300	241	234			
INTERNAL WING TANKS	650	522	507			
TOTAL INTERNAL FUEL	4 300	3 453	3 354	3 899	3 131	3 041
CL TANK	1 500	1 205	1 170	1 370	1 100	1 068
TOTAL INTERNAL FUEL PLUS CL TANK	5 800	4 658	4 524	5 269	4 231	4 110
WING DROP TANKS	2 300	1 847	1 794	2 129	1 709	1 661
TOTAL INTERNAL FUEL PLUS WING DROP TANKS	6 600	5 300	5 148	6 028	4 840	4 702
MAXIMUM FUEL LOAD TOTAL INTERNAL PLUS ALL EXTERNAL TANKS	8 100	6 505	6 318	7 398	5 940	5 770

*1) F-34 at 15° C, specific weight 0.803
*2) TS-1 at 20° C, specific weight 0.780

NOTE: Usable fuel is calculated by subtracting 7 % of fully serviced fuel, and additionally the fuel remaining trapped in the tanks: Internal 100 L, CL tank 25 L, WDTs 10 L.

Figure 1-19C

REFUELING PANEL

The refueling panel is located in the left main wheel well. Refer to figure 1-19D. It is used for refueling the aircraft, adjustment of the fuel indication, and verification of oil and hydraulic fluid quantities.

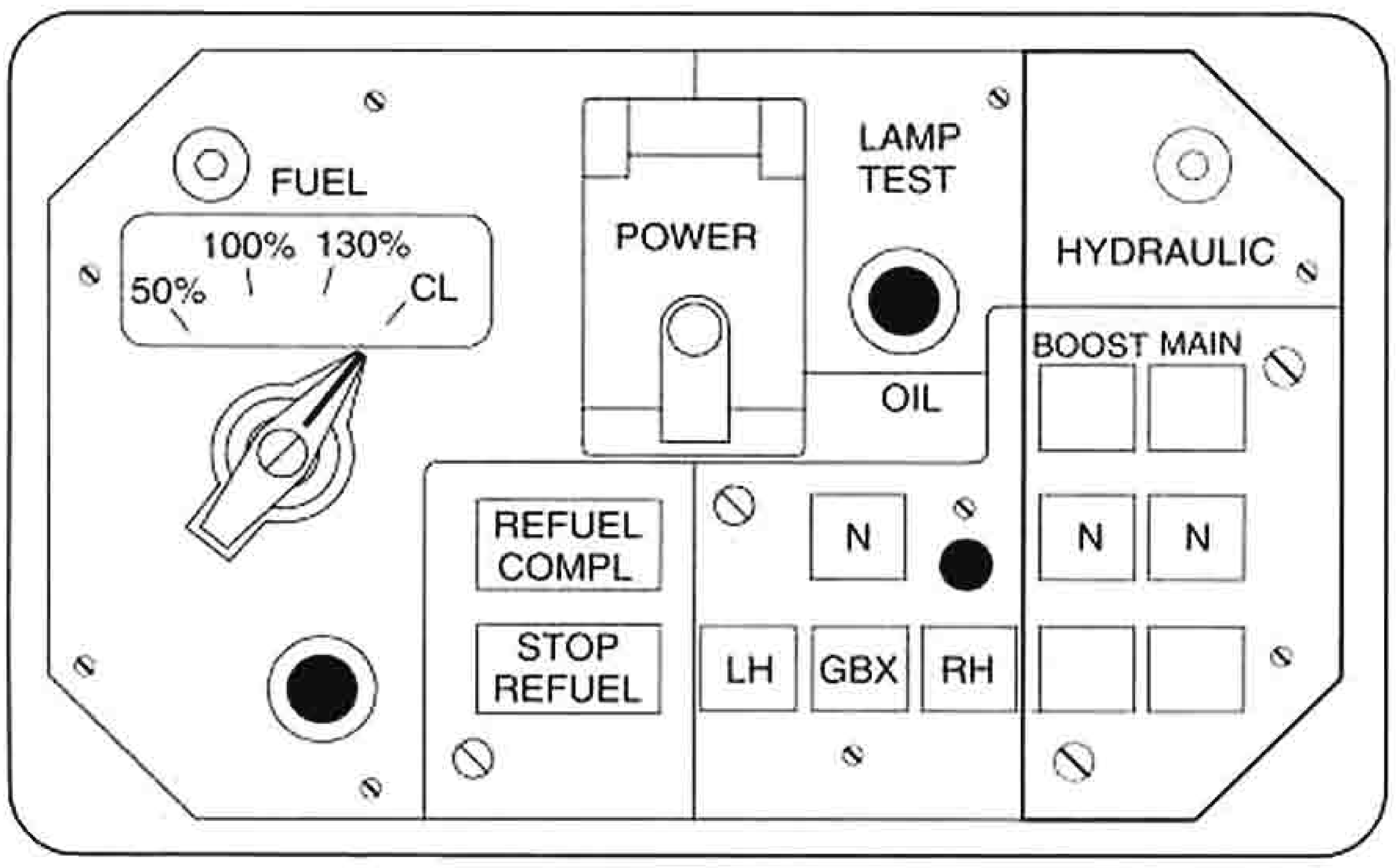


Figure 1-19D

POWER switch

The POWER switch is used to connect DC power to the refueling panel, provided the BAT GND SUPPLY switch is in the ON position.

LAMP TEST button

A LAMP TEST button is provided to check the light captions prior refueling.

FUEL selector

The FUEL selector is a rotary knob with the following functions:

- 50 % for refueling of tank 2 and partial refueling of tank 1 and 3;
- 100 % for refueling of all internal tanks;
- 130 % for refueling of all internal tanks plus centerline tank and;
- CL for refueling of centerline tank only.

NOTE

When the CL tank is refueled, the fuel selector knob has to be positioned to 130 % to obtain correct indications on the fuel indicator.

STOP REFUEL pushbutton

The STOP REFUEL pushbutton is used to stop the refueling procedure manually.

REFUEL COMPL caption

The caption illuminates when refueling is finished in accordance with the selection of the FUEL selector knob.

OIL captions

One green and three red captions indicate oil servicing requirements. The green caption illuminates when the oil level is within limits. Individual red captions indicate servicing requirements for the accessory gearbox, the LH and the RH engine.

HYDRAULIC captions

Three captions each indicate servicing requirements for the MAIN and the BOOST hydraulic system. The green caption marked N indicates no servicing required, the red caption indicates a servicing requirement and the yellow caption indicates that the system is overfilled.

FUEL SIGNALS CALIBRATION PANEL

The fuel signals calibration panel is located in the rear of the cockpit beneath the power distribution panel. Refer to figure 1-19E. It is used for adjustment, calibration and checkout of the fuel indication system. AC power must be available for operation.

NOTE

After refueling has been accomplished with battery power, the PTO switch must be set to ON before the indication system can be adjusted. Due to the limited capacity of the batteries, operation of the PTO is limited to 30 sec.

AFTER MODIFICATION WITH WING DROP TANKS

A rotary selector knob has been added to the panel, refer to figure 1-19F, to select a voice warning for bingo fuel. The fuel selector knob has been replaced by a three-position toggle switch.

GT:

The fuel signals calibration panel, refer to figure 1-19E, is located in the rear cockpit beneath the instrument panel in front of the control stick.

G / GT

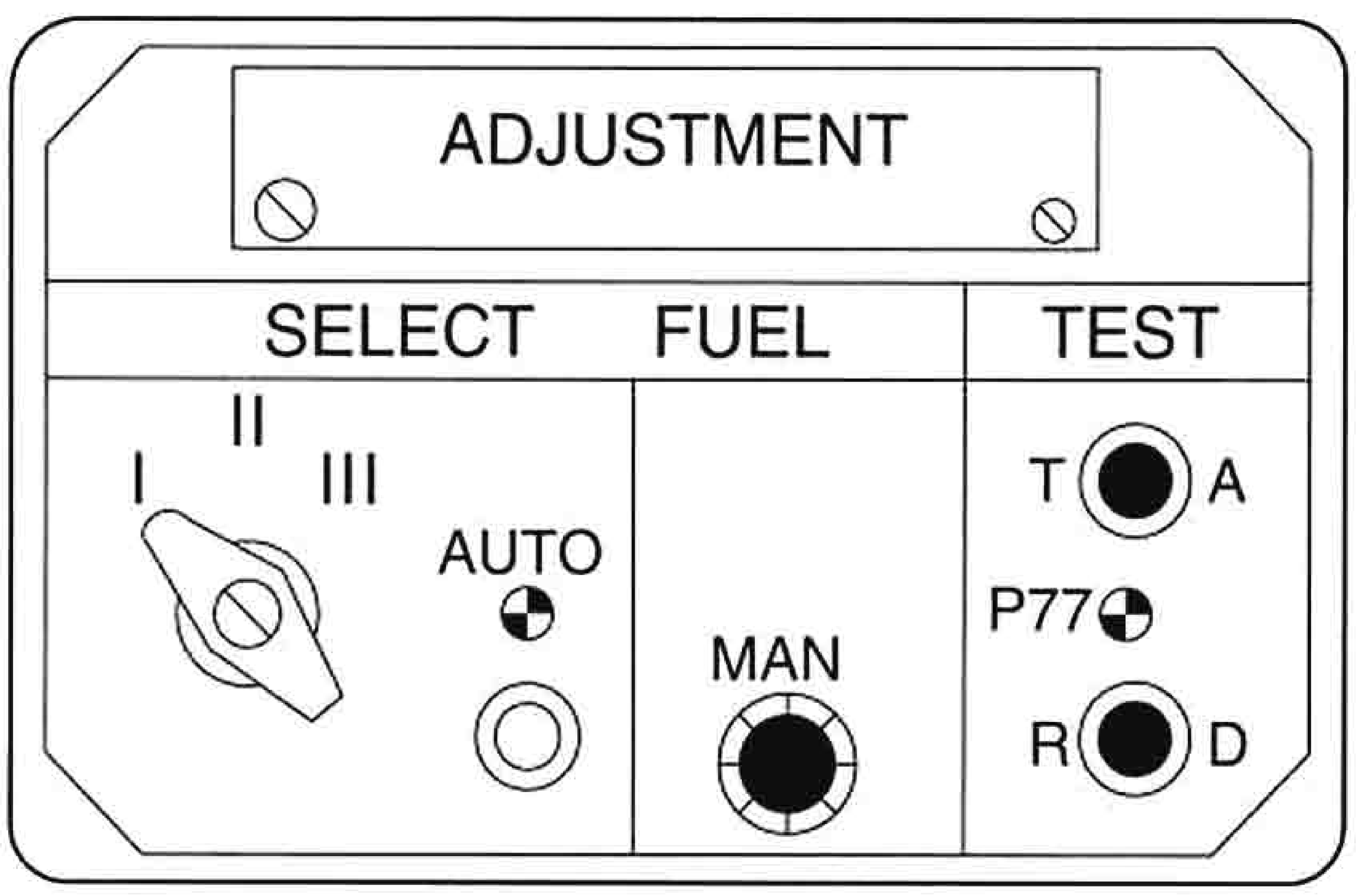


Figure 1-19E

AFTER MODIFICATION WITH WING DROP TANKS

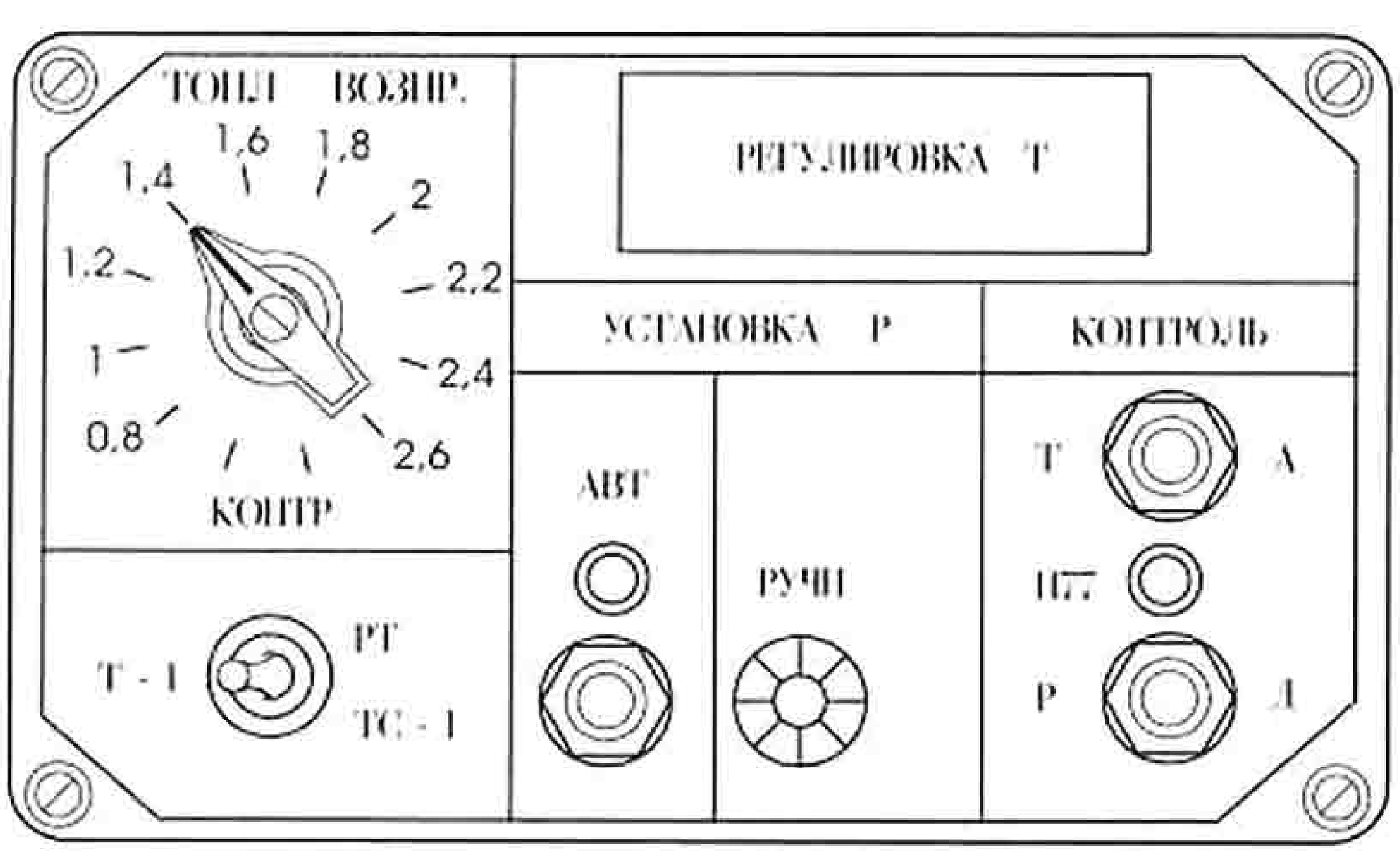


Figure 1-19F

Fuel Selector Knob

The fuel selector knob has three positions, labelled with Roman numerals, to select the type of fuel used. The fuel computer receives the specific weight data for calculation of the remaining fuel quantity and remaining flight distance.

AFTER MODIFICATION WITH WING DROP TANKS

The rotary knob has been replaced by a three-position toggle switch labelled T-1 (I) – TC-1 / PT(II).

РЕГУЛИРОВКА (ADJUSTMENT)

Covered adjustment screws are used for calibration of the fuel indicator.

АВТ (AUTO) button

Pressing the pushbutton inputs fuel quantity to the computer in accordance with the selection on the refuelling panel. The indicator light illuminates as soon as the quantity is displayed on the indicator.

РУЧН (MAN) adjustment knob

The manual adjustment knob can be used for random adjustment of the fuel quantity. Prior to adjustment, the FUEL COUNTER circuit breaker has to be pulled and the position P has to be selected with the T/P switch on the fuel indicator.

AFTER MODIFICATION WITH WING DROP TANKS

The button is used to add the fuel quantity of the wing drop tanks in kg to the fuel indication system.

Test buttons

Two test buttons and a LED are used for maintenance checkout. The buttons are labeled T-A and P(R)-Д(D), the LED is labeled П77(P77).

ТОПЛ ВОЗНР (FUEL RETURN) selector

The fuel return selector is used to select the desired bingo fuel. A VIWAS warning is issued when the selected fuel amount is reached.

CAUTION

Displaying of the VIWAS warning may differ from the adjusted value up to 190 kg.

CONTROL AND TEST PANEL

A new test button, labelled TEST WDT has been added on the control and test panel on the right console, replacing the FEEL UNIT OK caption, refer to figure 1-19G. Illumination of the TEST WDT caption on the telelight panel indicates a valid system check when the test button is pressed.

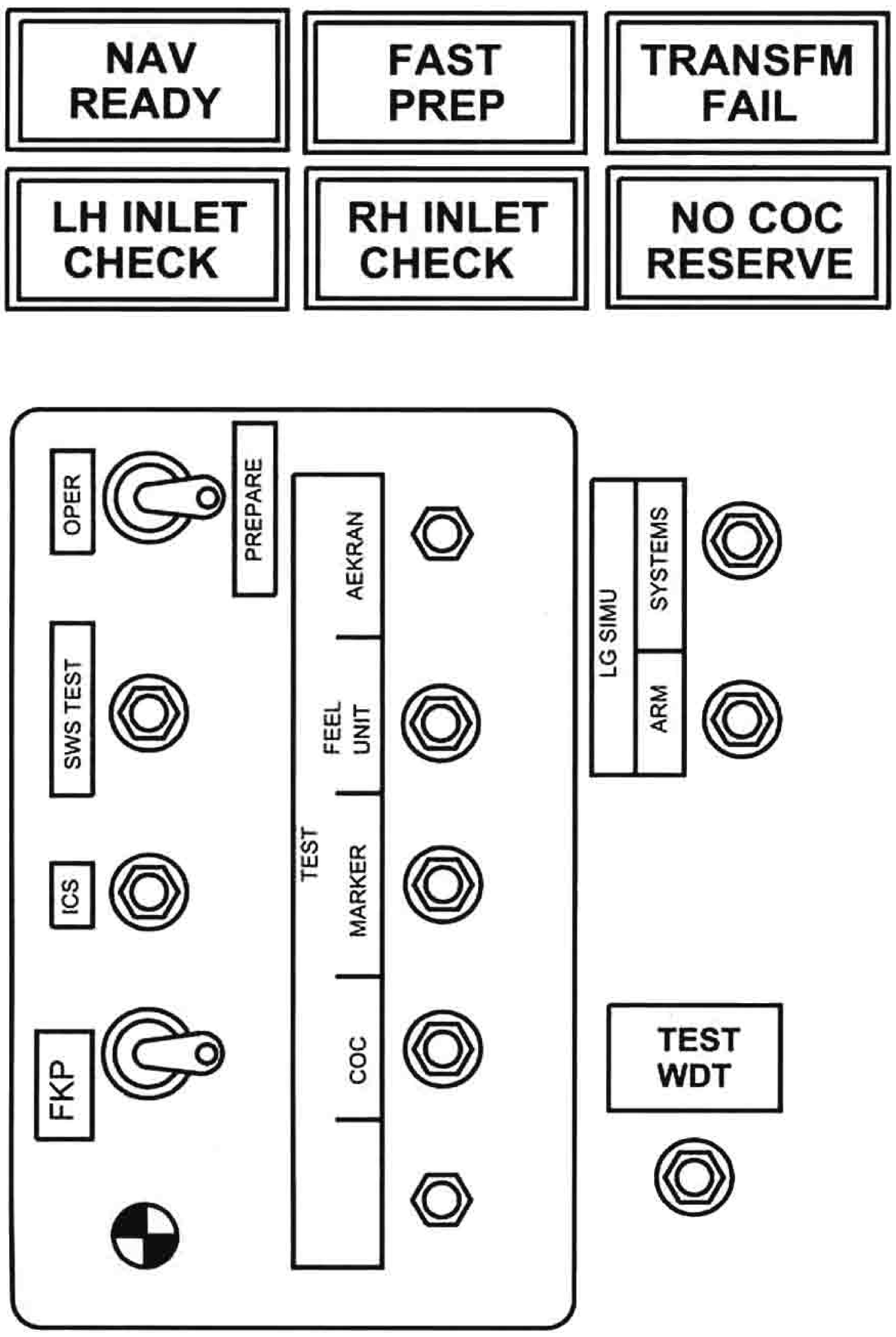


Figure 1-19G

FUEL INDICATION SYSTEM

The fuel indication system consists of a remaining fuel quantity computer, a remaining distance computer, a fuel consumption computer and an indicator, refer to figure 1-20.

The system is interconnected with the refuelling panel and the fuel signals calibration panel. Basic calculations such as total fuel depend on settings on the refuelling panel and indicator calibration. Fuel consumption calculations depend on settings on the fuel signals calibration panel and inputs from the fuel flow sensors.

The system includes capacitance type gaging units in tanks 1, 2 and 3, level control valves, full- and empty sensors, fuel flow sensors and temperature sensors.

The fuel indicator displays the following data:

- Remaining fuel quantity in kg;
- remaining flight distance in NM and ;
- tank empty signals.

The initial fuel quantity is computed using inputs from the capacitance type gaging units plus the quantities for the remaining internal and external tanks based on refuelling information from the refueling panel and density information from the fuel signals calibration panel.
Consecutive fuel quantity data computation is based on fuel flow corrected for density and temperature.
When the CL tank is jettisoned, the computer corrects the remaining fuel quantity accordingly.

NOTE

Jettisoning of the wing drop tanks may lead to a false, normally higher than actual, fuel quantity indication.

The remaining flight distance computer computes the remaining flight distance based on actual fuel consumption and data from the ADC. This distance is not corrected for wind.

FUEL INDICATION SYSTEM

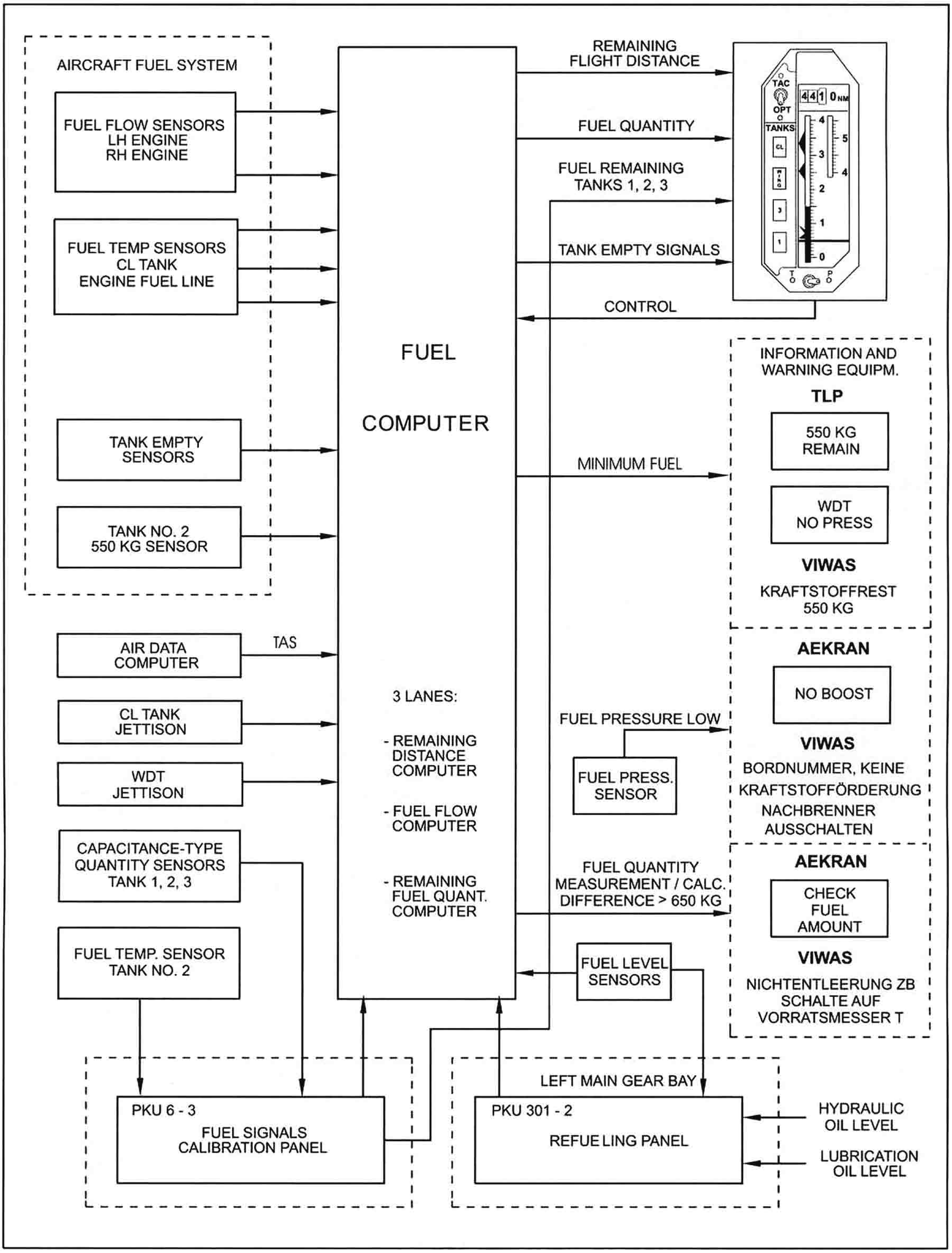


Figure 1-20

FUEL INDICATOR

The fuel indicator consists of a tape-type quantity scale, a remaining flight distance counter and tank empty captions. A two-position toggle switch marked T and P with indicator lights for selection of measured or calculated fuel is located at the bottom of the scale. Triangular shaped markers along the scale complete the indicator.

In position T the scale indicates fuel quantity sensed by the capacitance-type gages in tanks 1, 2 and 3. In position P, total fuel quantity is displayed as calculated by the fuel computer.

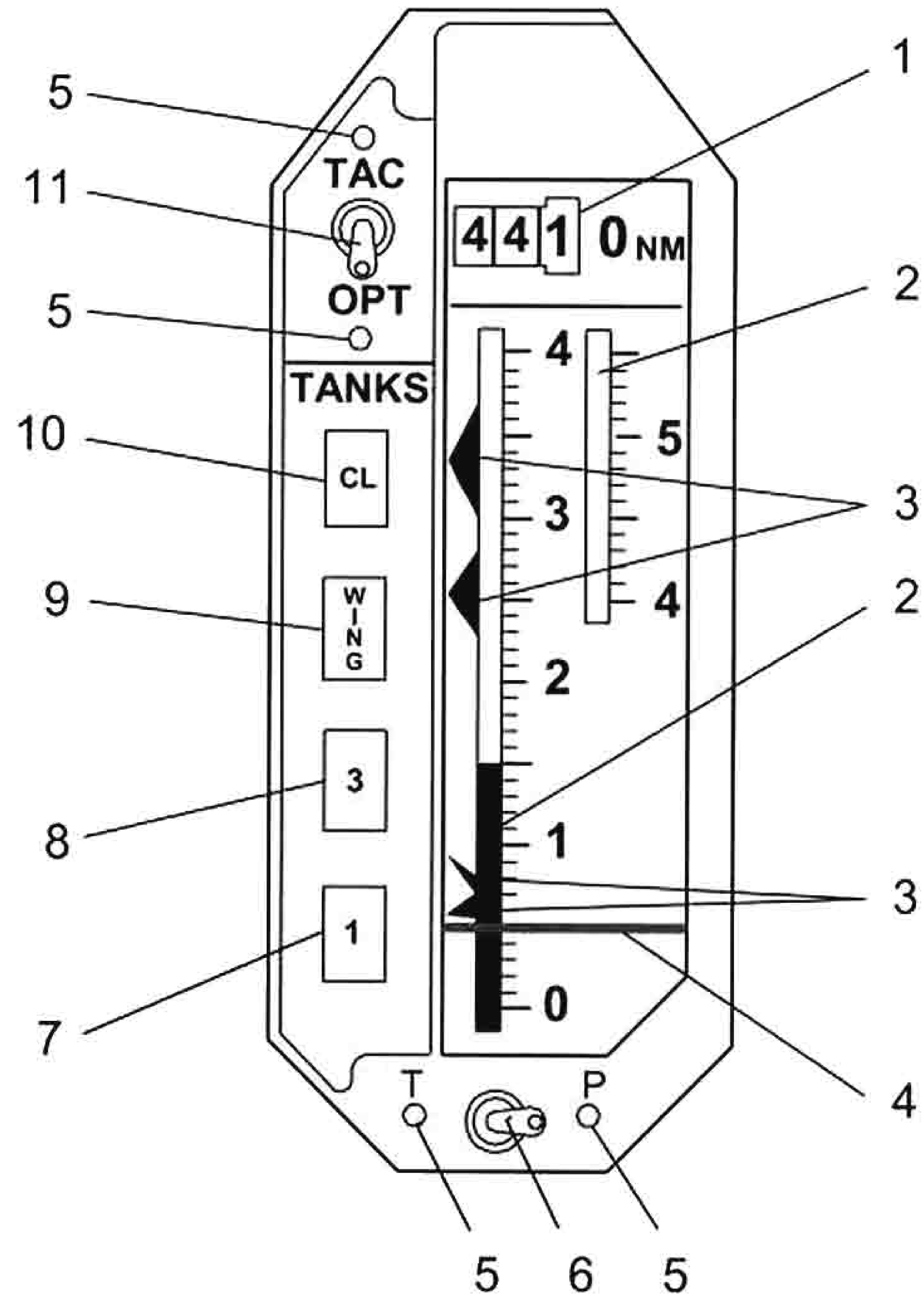
Four triangular shaped markers along the scale indicate the tolerable remaining fuel quantity when the particular tank empty caption illuminates. The markers point in the direction of the corresponding indicator caption.

Illumination of a tank empty caption with the fuel quantity above or below the applicable marker indicates a transfer system malfunction or a fuel leak.

The counter at the upper part of the indicator indicates the remaining flight distance based on actual or optimum flight conditions. A switch next to the counter selects actual or optimum range. Refer to figure 1-21.

NOTE

The fuel indicator has a tolerance of ±3 % of the maximum scale value.



1. REMAINING DISTANCE COUNTER
2. TOTAL QUANTITY SCALE
3. REMAINING QUANTITY MARKERS
4. MINIMUM QUANTITY MARKER (550 kg)
5. LED
6. T/P SWITCH
7. TANK 1 EMPTY LIGHT CAPTION
8. TANK 3 EMPTY LIGHT CAPTION
9. WING TANKS EMPTY LIGHT CAPTION
10. CL TANK EMPTY LIGHT CAPTION
11. DISTANCE COMPUTER MODE SWITCH
    TAC = WITH CURRENT FUEL CONSUMPTION
    OPT = WITH AN OPTIMIZED CALCULATED FUEL FLOW FOR MAXIMUM RANGE FLIGHT

Figure 1-21

REMAINING FUEL QUANTITY

TANK EMPTY	REMAINING FUEL IN KG	ILLUMINATED CAPTION
CL TANK	3 000 TO 3 700	CL
WING TANKS	2 300 TO 2 800	WING
TANK 3	700 TO 850	3
TANK 1	550 TO 700	1
TANK 2	470 TO 630	TLP: 550 KG REMAIN
	0 TO 100	NO BOOST

Figure 1-21A

AFTER MODIFICATION WITH WING DROP TANKS

The tape-type quantity scale has been modified to indicate the additional fuel available. Refer to figure 1-21B. Two markers are available for indication of the tolerable fuel quantity remaining when the CL tank is empty. The triangular marker on the left scale is applicable with no wing drop tanks installed, the marker lines on the right scale are applicable when carrying wing drop tanks.

NOTE

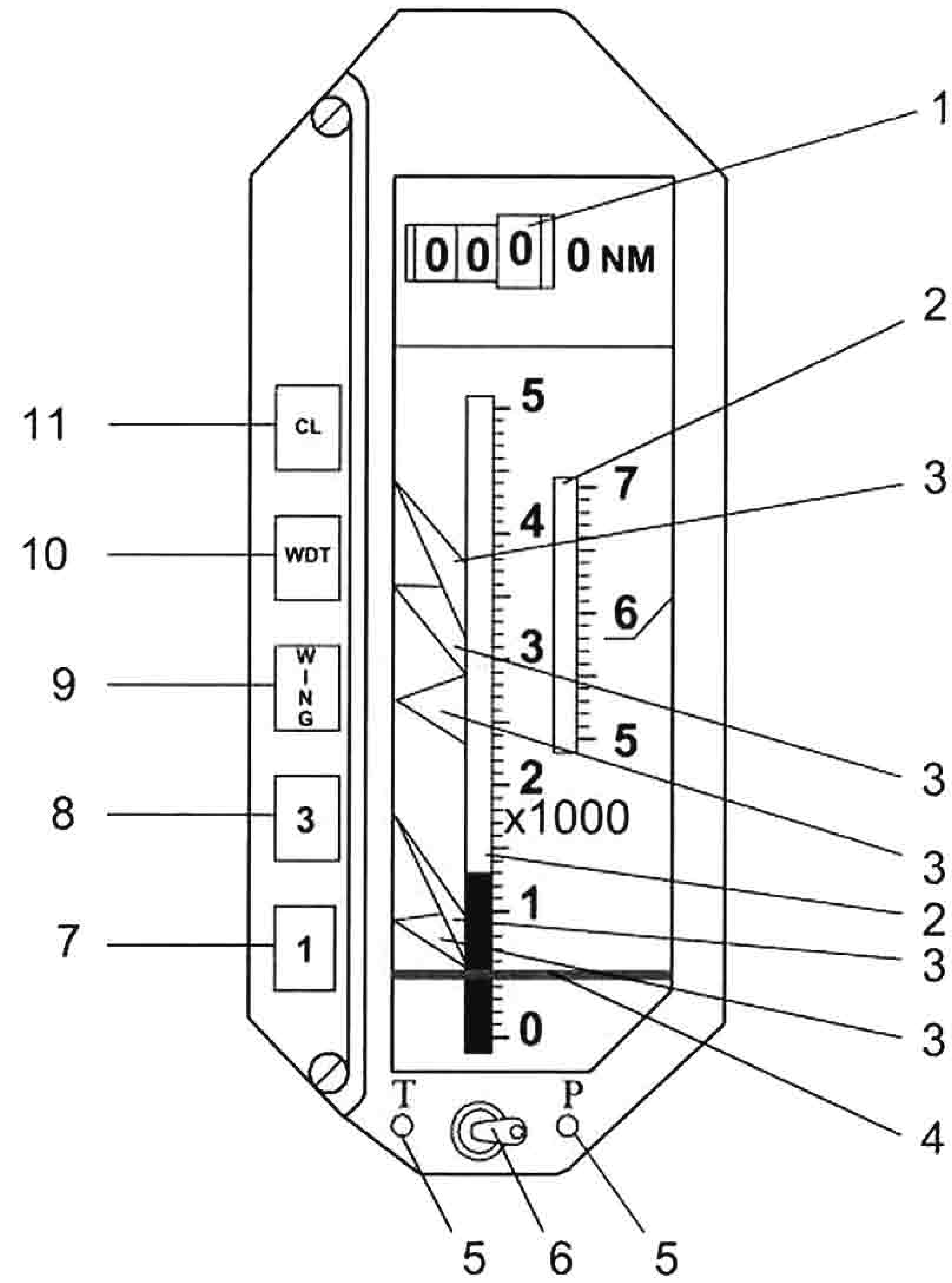
- During level, unaccelerated flight without afterburner, the tank 1 empty caption must not illuminate prior to illumination of the tank 3 empty caption.
- After all fuel from the wing drop tank has been transferred, the wing drop tank empty light caption may flash.

The counter at the upper part of the indicator shows the remaining flight distance based on actual flight conditions / current fuel flow.
The selector switch for optimum or actual range available has been removed.

Two LEDs next to the T/P switch illuminate when the respective position is selected with the T/P switch.

NOTE

The fuel indicator has a tolerance of ±3 % of the maximum scale value.



1. REMAINING DISTANCE COUNTER
2. TOTAL QUANTITY SCALE
3. REMAINING QUANTITY MARKERS
4. MINIMUM QUANTITY MARKER
5. LED
6. T/P SWITCH
7. TANK 1 EMPTY LIGHT CAPTION
8. TANK 3 EMPTY LIGHT CAPTION
9. WING TANKS EMPTY LIGHT CAPTION
10. WING DROP TANK EMPTY LIGHT CAPTION
11. CL TANK EMPTY LIGHT CAPTION

Figure 1-21B

REMAINING FUEL QUANTITY

TANK EMPTY	REMAINING FUEL IN KG	ILLUMINATED CAPTION
CL TANK	5 000 TO 5 700	CL
WING DROP TANKS	2 800 TO 3 500	WDT
INTERNAL WING TANK	2 300 TO 2 800	WING
TANK 3	550 TO 1 000	3
TANK 1	550 TO 1 000	1
TANK 2	470 TO 630	TLP: 550 KG REMAIN
	0 TO 100	NO BOOST

Figure 1-21C

INDICATIONS AND WARNINGS

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
AEKRAN	NO BOOST	Fuel pressure drop at the input of the fuel pumps. or: Total fuel quantity below 100 kg.
VIWAS	"BORDNUMMER, KEINE KRAFTSTOFFÖRDERUNG" "NACHBRENNER AUSSCHALTEN"	

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	550 KG REMAIN	Total fuel quantity below 550 kg.
VIWAS	"BORDNUMMER, KRAFTSTOFFREST 550 kg" "NACHBRENNER AUSSCHALTEN"	

The red caption 550 KG REMAIN on the TLP may illuminate occasionally during negative g flight or when the tank depletion sequence is disrupted.

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
AEKRAN	CHECK FUEL AMOUNT	After a delay of 35 to 50 seconds the indication is displayed when: measured fuel quantity is below 1 800 kg and the internal wing tanks are not empty; or: measured fuel quantity is below 1 800 kg and a difference of 550 kg to 750 kg between calculated and measured fuel exists; or: a difference of 200 kg to 350 kg exists between measured and calculated fuel quantity with tanks 1 and 3 empty.
VIWAS	"KONTROLLIERE KRAFTSTOFFVORRAT" "SCHALTE AUF VORRATSMESSER T"	

NOTE

Application of g-forces along the longitudinal axis with a duration in excess of 35 sec may cause the indication CHECK FUEL AMOUNT to illuminate until the g-forces subside.

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
AEKRAN	DROP TANK NO USAGE	After a delay of 20 to 35 seconds the indication is displayed when: the CL tank is not pressurized; or: when 120 kg to 360 kg of fuel have been used and CL tank fuel is not transferred.
VIWAS	"NICHTENTLEERUNG ZB" "SCHALTE AUF VORRATSMESSER T"	

AFTER MODIFICATION WITH WING DROP TANKS

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
AEKRAN	DROP TANK NO USAGE	After a delay of 20 to 35 seconds the indication is displayed when: the CL tank is not pressurized; or: when 120 kg to 360 kg of fuel have been used and CL tank fuel is not transferred.
VIWAS	"NICHTENTLEERUNG RUMPFZUSATZBEHÄLTER" "SCHALTE AUF VORRATSMESSER T"	

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	**WDT NO PRESS**	Wing drop tank pressurization/transfer failure.
VIWAS	"KEINE BELÜFTUNG DER TRAGFLÜGELZUSATZBEHÄLTER"	

If a BINGO FUEL has been selected with the fuel return selector knob on the fuel signals calibration panel, the following announcement is displayed when the selected fuel quantity is reached.

VIWAS	"BINGO FUEL" - "BINGO FUEL" - "BINGO FUEL"

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	**WDT TEST**	Indicates a valid system check if the WDT TEST button on the control and test panel on the right console is pressed without engines running.

ELECTRICAL POWER SUPPLY SYSTEM

The electrical power supply system consists of the power generating system and the control and monitoring system. The power generating system is composed of the AC generator, the DC generator, the converter, the batteries and the distribution network.

The control and monitoring system interacts with various other aircraft systems, e.g., engines, fuel system, fire extinguisher system etc.

AC ELECTRICAL POWER

One three-phase 115 / 200 V, 400 Hz constant frequency generator is the primary source of AC power. The generator is equipped with a regulator and overload protection device. Refer to figure FO-9. It is attached to the aircraft GBX. The AC generator is regulated by a hydrodynamic constant speed drive to ensure a stable RPM, regardless of engine RPM and generator load.

Maximum power output of the generator is 30 kVA. A transformer supplies three-phase 36 VAC power at a maximum output of 1.5 kVA. In case of a generator failure or a shut-down triggered by the regulator and protection device or in case of a transformer failure, the systems essential for flight are powered by the DC / AC converter, PTO.

DC ELECTRICAL POWER

One DC generator 28.5 V ±0.5 V, 400 A, 12 kW is the primary source of DC power. The DC generator is equipped with a regulator and overload protection device. Refer to figure FO-10.

This regulator and overload protection device ensures a DC regulated power output and protects the generator and the DC buses against overvoltages as well as short circuits in the generator itself.

DC / AC CONVERTER PTO

A DC / AC converter supplies emergency power in case of an AC generator failure or a transformer failure. The PTO is capable of delivering 115 VAC at a minimum output of 1.5 kVA and three-phase 36 VAC at a minimum output of 1 kVA. If PTO operation is caused by a transformer failure, the LRF, the angular rate sensors of the flight control system and course and glidepath indications of the approach mode render inoperative. In case of an AC generator failure, all systems not essential for flight are automatically disconnected. Refer to electrical system failures section 3.

BATTERIES

Two silver-zinc batteries serve as an emergency power source. Nominal voltage of the batteries is 27.6 V, however, under load (200 A), a minimum of 22 V is delivered.

NOTE

Since the charging voltage of the silver-zinc batteries is higher than the generator voltage, the batteries will not be charged when the generator power is available.

The batteries supply power to the DC bus during:

- Engine start with internal power.
- When the DC generator voltage drops below battery voltage.
- DC generator failure.

EXTERNAL POWER SUPPLY

External DC and / or AC power may be supplied to the aircraft for start-up or alignment purposes. Regardless of whether external power is connected or not, the BAT-GND SUPPLY switch must be switched ON to energize the DC bus of the aircraft. The voltmeter will indicate a voltage in excess of 22.5 V in case the external power is connected and the engines are off.

CONTROLS AND INDICATORS

The controls for the electrical power supply system are located on the electrical power panel. Refer to figure 1-22. The DC power can be checked with a DC voltmeter located on the lower center part of the vertical panel. It measures the voltage of the DC source actually connected with the DC bus.

The charge of the batteries can be checked on the Ah counter in the nose section.

ELECTRICAL POWER PANEL

The electrical power panel is located on the RH console. It contains the switches for the electrical power system and for essential engine control components.

The two-position toggle switches have the following functions:

BAT-GND SUPPLY	Engagement of the batteries.
DC GEN	Engagement of the DC generator.
AC GEN	Engagement of the AC generator.
PTO	Engagement of the DC / AC converter.
ENG SYS	Engagement of various engine control components. Refer to engine controls and indicators in this chapter and figure FO-10.
FUEL PUMP	Activation of the engine fuel pump, refer to fuel boost system in this chapter.
ANTI SURGE	Engagement of the engine anti-surge system, refer to engine controls and indicators in this chapter.

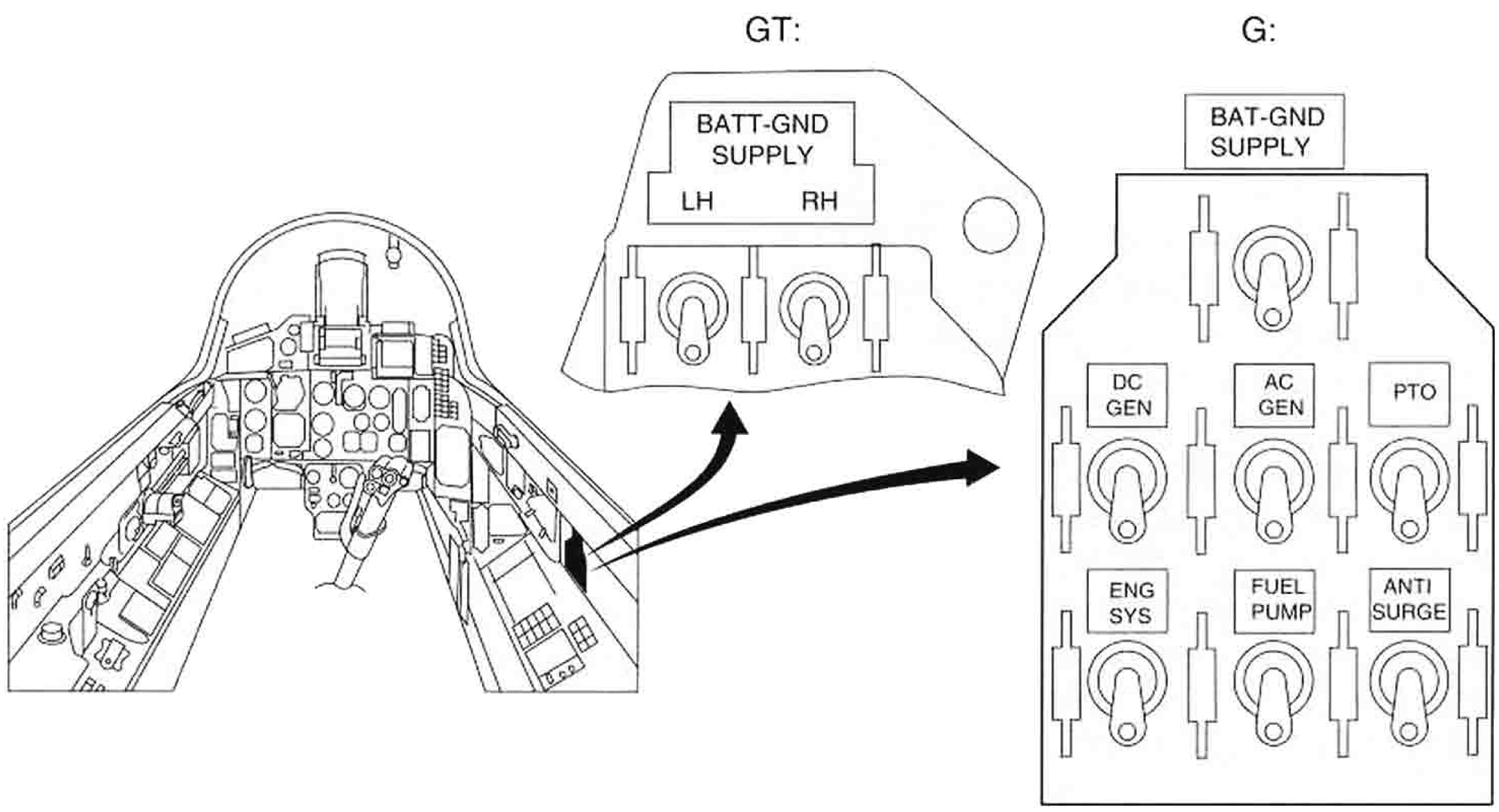


Figure 1-22

INDICATIONS AND WARNINGS

In case of an AC generator failure or a shut-down triggered by the regulator and overload protection device, the most important consumers will be powered by the DC / AC converter.

To prevent a converter overload, the following consumer are automatically shut off:

- Heaters for the main and standby inertial navigation platform
- Radar system and the optical sight, with the exception of the navigation system
- IRSTS / LRF and HMS
- LH AOA probe heater and side slip vane heater
- Windshield heaters, upper and central section
- AFCS system
- The external armament
- IFF altitude encoder

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
AEKRAN	AC GEN	AC generator failure or RPM of both engines below 55 %.

The constant speed drive of the AC generator is a hydrodynamic transmission with the generator drive shaft running at 12 000 RPM. An electromagnetic clutch connects the drive shaft to the AC generator and provides for automatic or manual disconnect in case of an constant speed drive failure:

Automatic	If the drive shaft speed is increased to 14 300 to 14 500 RPM, equivalent to a frequency of 465 to 480 Hz.
Manual	If the oil pressure of the constant speed drive drops below 10 kp/cm^2 (1 MPa) or the oil temperature is to high, with the GEN DRIVE EMERG OFF switch.

Illumination of the indication DISCON GEN DRIVE indicates the need for a disengagement of the constant speed drive. Successful cut-off is indicated by illumination of the AC GEN indication.

CAUTION

To avoid serious damage or destruction of the AC generator, the constant speed drive has to be disconnected manually as soon as possible if the DISCON GEN DRIVE indication is displayed. The GEN DRIVE EMERG OFF switch on the emergency panel must be held to the spring-loaded position for a maximum of 25 seconds for generator cut-off and for avoidance of demage to electrical circuits.

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
AEKRAN	DISCON GEN DRIVE	Oil pressure of the constant speed drive is below 10 kp/cm^2 (1 MPa) or the oil temperature is to high or generator drive shaft speed has increased to 14 300 to 14 500 RPM, equivalent to a frequency of 465 Hz to 485 Hz.
	AC GEN	Successful automatic or manual disconnect in conjunction with a constant speed drive failure.

If overvoltage or short circuit conditions occur in the DC power supply system, the regulator and overload protection device will disconnect the generator from the DC bus. This fail condition can be reset by the ground crew only.

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
AEKRAN	DC GEN WATCH TIME	DC generator failure.
VIWAS	"AUSFALL GLEICHSTROMGENERATOR" "DÄMPFUNG AUS, NB AUS, SN-29 AUF HAND"	

If both generators fail, the voice information and warning system will give a corresponding message as well.

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
AEKRAN	TWO GEN WATCH TIME	AC and DC generator failure.
VIWAS	"AUSFALL GLEICHSTROMGENERATOR" "DÄMPFUNG AUS, NB AUS, SN-29 AUF HAND"	

HYDRAULIC POWER SUPPLY SYSTEM

The aircraft hydraulic system consists of two independent systems, the main hydraulic system and the hydraulic boost system. Refer to figure FO-11.

The main hydraulic system provides:

- Operation of the second chamber of the hydraulic actuators of the tailerons, the ailerons, the rudders, and of the hydraulic actuator of the AOA limiter system (COC).
- Extension and retraction of landing gear, flaps / LEF and speedbrakes.
- Control of the variable air intake ramps, the actuator of the APU exhaust door, the nose wheel steering and damper system and the hydraulic actuator of the rudder feel force system.

The hydraulic boost system provides:

- Operation of the first chamber of the dual chamber hydraulic actuators of the tailerons, the ailerons, the rudder and the hydraulic actuator of the AOA limiter system (COC).

HYDRAULIC FLUID RESERVOIRS

The hydraulic fluid reservoirs are located inside fuel tank 3 and ensure normal operation of the pumps under all flight conditions. Due to the arrangement of the two-chamber configuration, they will feed the pumps even during negative g flight.

The reservoirs are filled-up as a closed system through filler caps in the return lines.

To avoid cavitation at the inlet of the hydraulic pumps, the reservoirs are pressurized with low-pressure compressed air.

The necessary compressed air is attained from the main pneumatic system through a pressure regulator valve.

Safety relief valves are incorporated to prevent overpressurization.

HYDRAULIC PUMPS

Both systems are powered by variable volume piston pumps, one for each system, flanged to and driven by the GBX. The pumps operate at a pressure between 190 kp/cm^2 (19 MPa) and 220 kp/cm^2 (22 MPa).

A pressure limiter and isolation valve is installed in the high-pressure lines of the main system. If the pressure drops below 130 kp/cm^2 (13 MPa), it will disconnect all systems except the flight controls and supply only one chamber of tailerons, ailerons, rudders, flaps / LEF and AOA limiter actuator.

To protect the hydraulic lines, pressure relief valves are installed in the main and the boost hydraulic system. The valves open at a pressure of 240 kp/cm^2 (24 MPa) and discharge hydraulic fluid into the return lines.

An emergency hydraulic pump installed in the hydraulic boost system supports basic control functions of the aircraft at an RPM of both engines of at least 85 %. Although the pump delivers hydraulic pressure of up to 240 kp/cm^2 (24 MPa), the delivery rate permits only severely degraded aircraft control. This pump is driven by fuel pressure and is activated automatically whenever the system pressure of both systems drops below 100 kp/cm^2 (10 MPa). It can also be activated manually with the EMERGENCY HYDR PUMP switch located on the control panel next to the CHUTE JETTISON button.

HYDRAULIC ACCUMULATORS

Accumulators in the main and boost systems store a supply of high pressure fluid to damp pressure surges caused by sudden variations in flow demands.

Both accumulators are charged with nitrogen at 80 kp/cm^2 (8 MPa) which corresponds with the marking P_{AK} on the combined pressure indicator (CPI).

HYDRAULIC COOLING

Installation of the hydraulic reservoirs in the fuel tank 3 ensures adequate cooling of the hydraulic fluid due to heat exchange with fuel.

CONTROLS AND INDICATORS

Both hydraulic fluid reservoirs are equipped with level transmitters connected to level indicators installed in the central refueling control panel located in the left main wheel well.

The actual hydraulic pressures of the main system as well as of the boost system are displayed on the CPI. The indicator utilizes 115 VAC power from the AC generator.

COMBINED PRESSURE INDICATOR

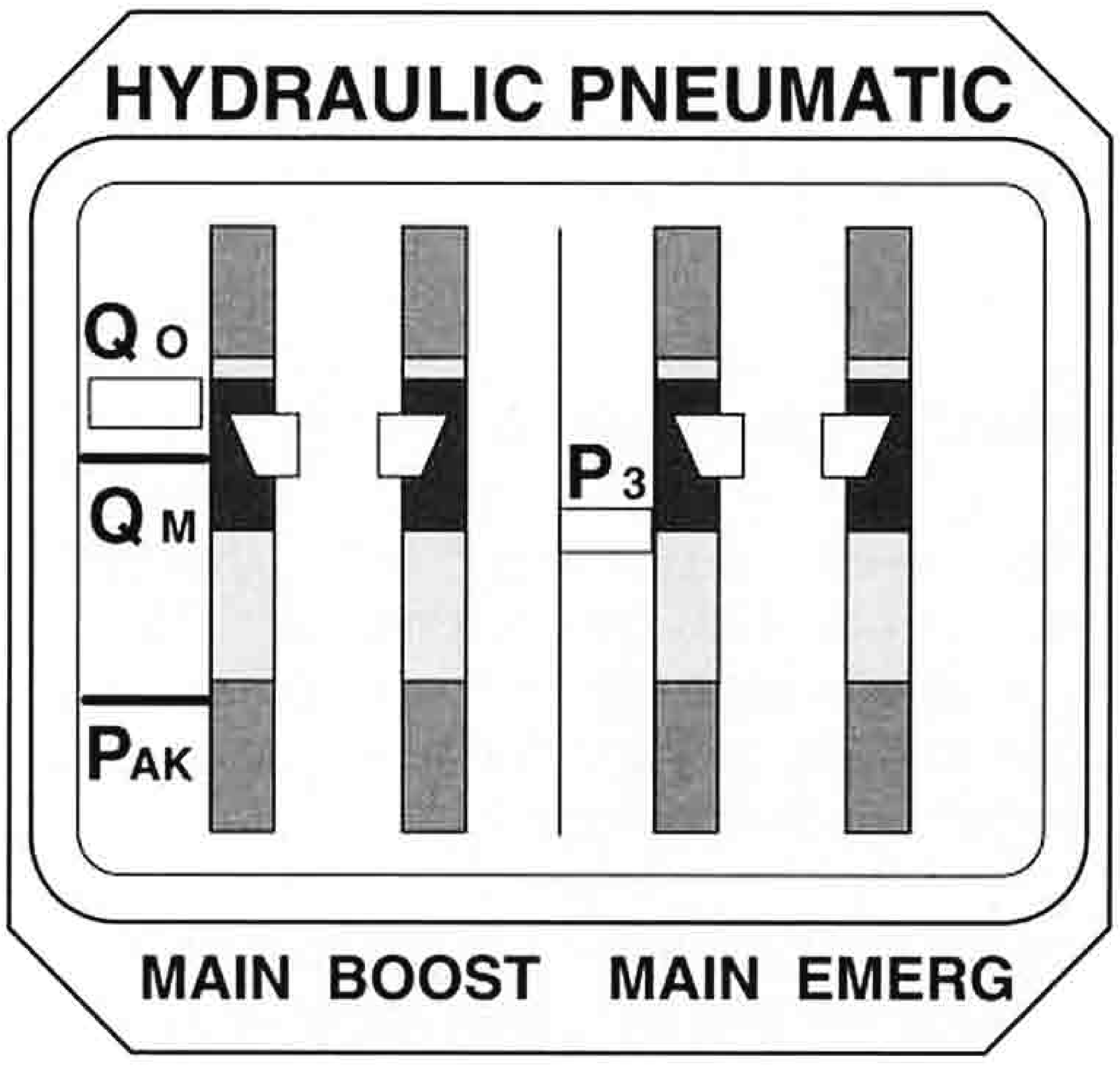


Figure 1-23

The transmitters for this indicator are connected to the nitrogen gas side of the accumulators. Therefore the CPI will indicate nitrogen pressure P_{AK} when the hydraulic system pressure is zero.

Whenever one engine has reached a minimum of 20 % RPM, the hydraulic pressures must be within the green areas on the indicator.

These green areas contain two markers, Q_O and Q_M which indicate hydraulic pump performance.

Q_O is the marker for the system pressure with none of the actuators moving, i.e. zero delivery of the pumps.

Q_M is the marker for the system pressure with the hydraulic actuators moving with maximum velocity.

NOTE

During engine start at low ambient temperatures and increased hydraulic fluid flow rate, the hydraulic pressure indication may drop into the yellow area.

INDICATIONS AND WARNINGS

Whenever the pressure of one of the hydraulic systems drops below 100 kp/cm^2 (10 MPa), the respective warning indications are displayed.

NOTE

If hydraulic system pressure is regained, the respective warning indications extinguish.

The AFCS is disengaged if the pressure in both systems drops below 100 kp/cm^2 (10 MPa) until pressure is regained.

With the emergency hydraulic pump running, pressure in the boost hydraulic system may increase up to 240 kp/cm^2 (24 MPa) at zero delivery rate. Whenever a pressure above 100 kp/cm^2 (10 MPa) is regained, the DOUBLE HYD SYS indication on the TLP and the BOOST HYD SYS indication on the AEKRAN extinguish.

WARNING

With the emergency hydraulic pump running, boost hydraulic system pressure may be regained at low or zero delivery rates. Therefore, extinguishing of the double hydraulic system indications must be considered temporary and the situation continued to be treated as a double hydraulic system failure.

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	DOUBLE HYD SYS / EMERG HYD PUMP ON	Both hydraulic systems below 100 kp/cm^2 (10 MPa), emergency hydraulic pump activated.
AEKRAN	BOOST HYD SYS / MAIN HYD SYS	
VIWAS	"AUSFALL HYDRAULIKVERSTÄRKERSYSTEM" "NOTAUSFAHREN FAHRWERK BEI 500" "AUSFALL HYDRAULIKHAUPTSYSTEM" "NOTAUSFAHREN FAHRWERK BEI 500"	

PNEUMATIC POWER SUPPLY SYSTEM

The pneumatic power supply system consists of two independent systems, the main and the emergency power supply system. Refer to figure 1-24.

The main system supplies nitrogen pressure to operate the:

- Wheel brakes
- Canopy and canopy seals
- Drag chute
- Fuel shut-off valves
- Radio and radar equipment pressurization
- Venting of the hydraulic reservoirs

GT: Periscope system

The emergency system supplies nitrogen pressure to ensure the:

- Emergency gear extension
- Emergency braking of the main wheels

PNEUMATIC RESERVOIRS

Several pressure bottles are charged to 150 kp/cm^2 (15 MPa) gaseous nitrogen. To ensure drag chute operation, an additional pressure bottle charged to 63 kp/cm^2 (6.3 MPa) is located close to the drag chute compartment.

A common charging valve and a pressure indicator are located in the left main landing gear bay. Crossfeed between the systems is prevented by check valves in the interconnecting lines.

NOTE

The pneumatic power supply system is not charged during flight.

CONTROLS AND INDICATORS

Pressure indicators are integrated in the CPI located at the right side of the main vertical cockpit console.

Two scales display main and emergency pneumatic system pressure. For normal operation, the pressure must remain in the green area.

PNEUMATIC POWER SUPPLY SYSTEM

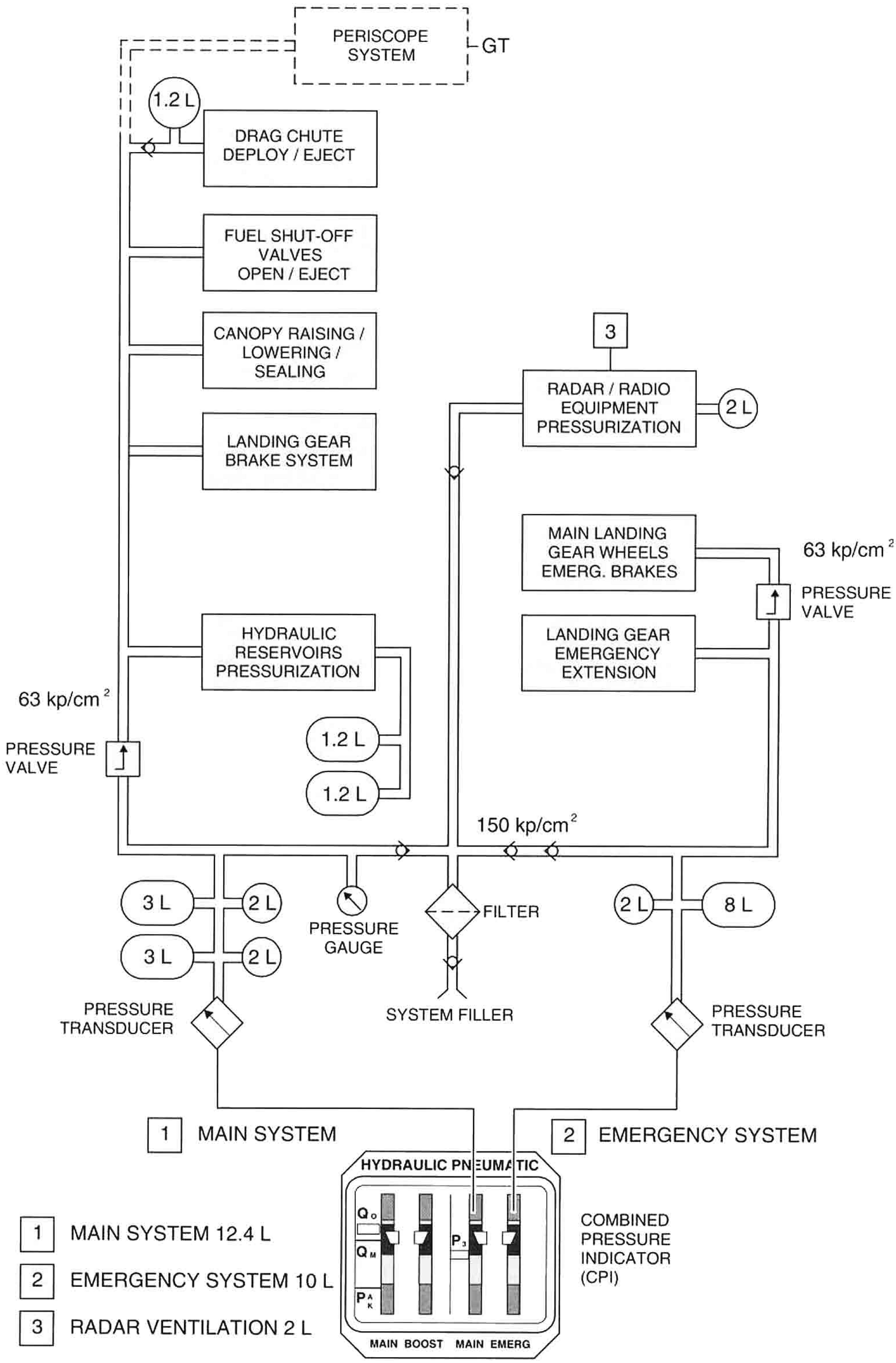


Figure 1-24

LANDING GEAR SYSTEM

The aircraft is equipped with a retractable tricycle landing gear. The gear is electrically controlled and hydraulically actuated by the main hydraulic system. DC power from the generator or the batteries is required for operation.

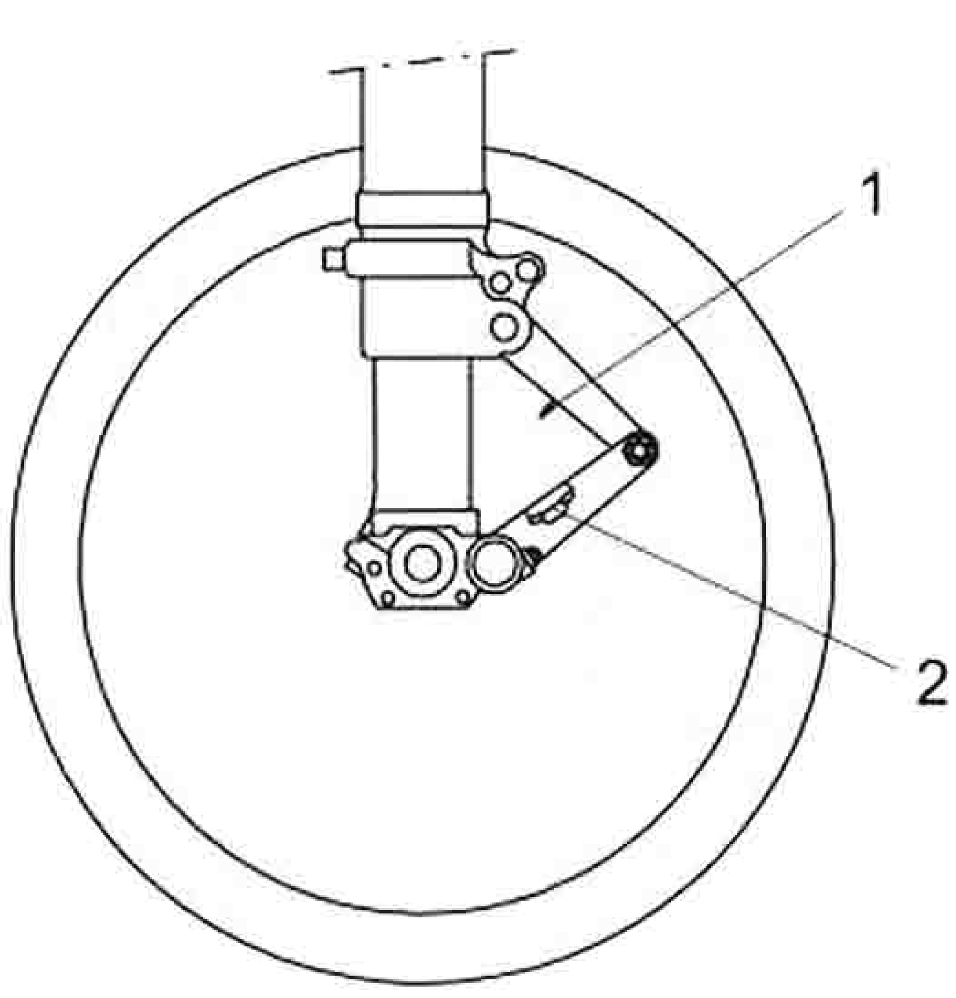
NOTE

- Retraction time for the landing gear is 9 to 10 sec, extension time is 7 to 8 sec.
- Simultaneously to the extension or retraction of the landing gear, the APU doors are opened or closed respectively.

MAIN LANDING GEAR

Each main gear is hydraulically retracted and extended. In the extended position the gear is locked down by an internal mechanical lock in each gear actuating cylinder. When the landing gear handle is in the position RETRACTED, the gear will retract. As the main gear retracts, the wheels are automatically braked to a stop. When the gear is up and locked, brake pressure is automatically released. The main gear retracts forward and is enclosed by three fairing doors. The gear is mechanically locked in the wheel wells.

The right main gear strut is equipped with a shock detector plate to indicate evidence of a hard landing.
It consists of a steel pin mounted in the upper track swivel arm and a small metal plate mounted in the lower track swivel arm of the shock absorber. Refer to figure 1-25.



1. PIN
2. METAL PLATE

Figure 1-25

The metal plate will be dented or pierced whenever the shock absorber is compressed beyond limits during a hard landing. As a result, the landing gear has to be inspected and the plate has to be replaced.

NOSE LANDING GEAR

The nose gear is hydraulically retracted and extended. The gear is locked in the down position by a mechanical lock inside the gear actuating cylinder. A mechanical lock installed in the wheel well locks the gear in the up position. The nose gear retracts aft into the fuselage. The nose gear is equipped with twin nose wheels, a nose wheel steering (NWS) and damper system and wheel brakes. As the nose gear retracts, the wheels are automatically braked to a stop and the strut is mechanically shortened. When the gear is up and locked, brake pressure is automatically released.

NOSE WHEEL STEERING SYSTEM

The NWS system provides two steering modes, a low mode and a high mode. In the low mode, used for takeoff and landing, nosewheel deflection up to 8° to either side is possible. For taxiing, the high mode may be selected allowing nose wheel deflections of up to 31° to either side.

NOTE

Activating the high mode causes the nose wheel brakes to be disabled.

Directional control is obtained by operating the rudder pedals. The nose gear is controlled mechanically, and operated hydraulically. Additionally, electrical power is needed to engage the high mode and simultaneously disengage the nosewheel brakes. A damper system prevents lateral oscillation of the nose wheels during takeoff, landing and taxiing. High mode is engaged by pressing the LOCK ON button, provided the flaps are up and the MRK EMERG OFF switch is in the normal (safety wired) position. In case the MRK EMERG OFF switch is placed to the OFF position, the high steering mode cannot be activated, however, the nosewheel brakes will be disabled as usually upon actuation of the LOCK ON button. When the nose gear is retracted, NWS is disconnected mechanically.

LANDING GEAR EMERGENCY LOWERING SYSTEM

The aircraft pneumatic system provides compressed air to extend the landing gear, regardless of the landing gear handle position. Either the nose gear only, or nose and main gear can be extended. Pulling the EMERG GEAR handle aft directs compressed air to the nose gear hydraulic actuator to extend the nose gear. A pneumatic shut-off valve closes the hydraulic lines to prevent inadvertent gear retraction in case hydraulic pressure is regained. After the nose gear is confirmed fully extended, rotating the EMERG GEAR handle 90° clockwise and pulling full aft activates the main gear hydraulic actuators to extend the main gear. The hydraulic lines are shut-off by pneumatically driven shut-off valves. Normal gear down indication on the landing system signal panel is achieved after all gears are fully extended.

CONTROLS AND INDICATORS

Landing Gear Handle

The landing gear is controlled by a handle on the left side of the instrument panel. Refer to figure 1-25A. To move the handle up or down, it has to be pulled to override a stop. Placing the handle in the RETRACTED or EXTENDED position uses DC power to actuate hydraulic valves to position the landing gear.

Emergency Landing Gear Lowering Handle

A red handle, marked EMERG GEAR is located beneath the left front panel. It is used to lower the landing gear pneumatically.

NOTE

After the landing gear has been extended with the emergency gear lowering system, normal gear retraction is not possible.

MRK EMERG OFF Switch

The MRK EMERG OFF switch is used to disable the high mode of the NWS system.

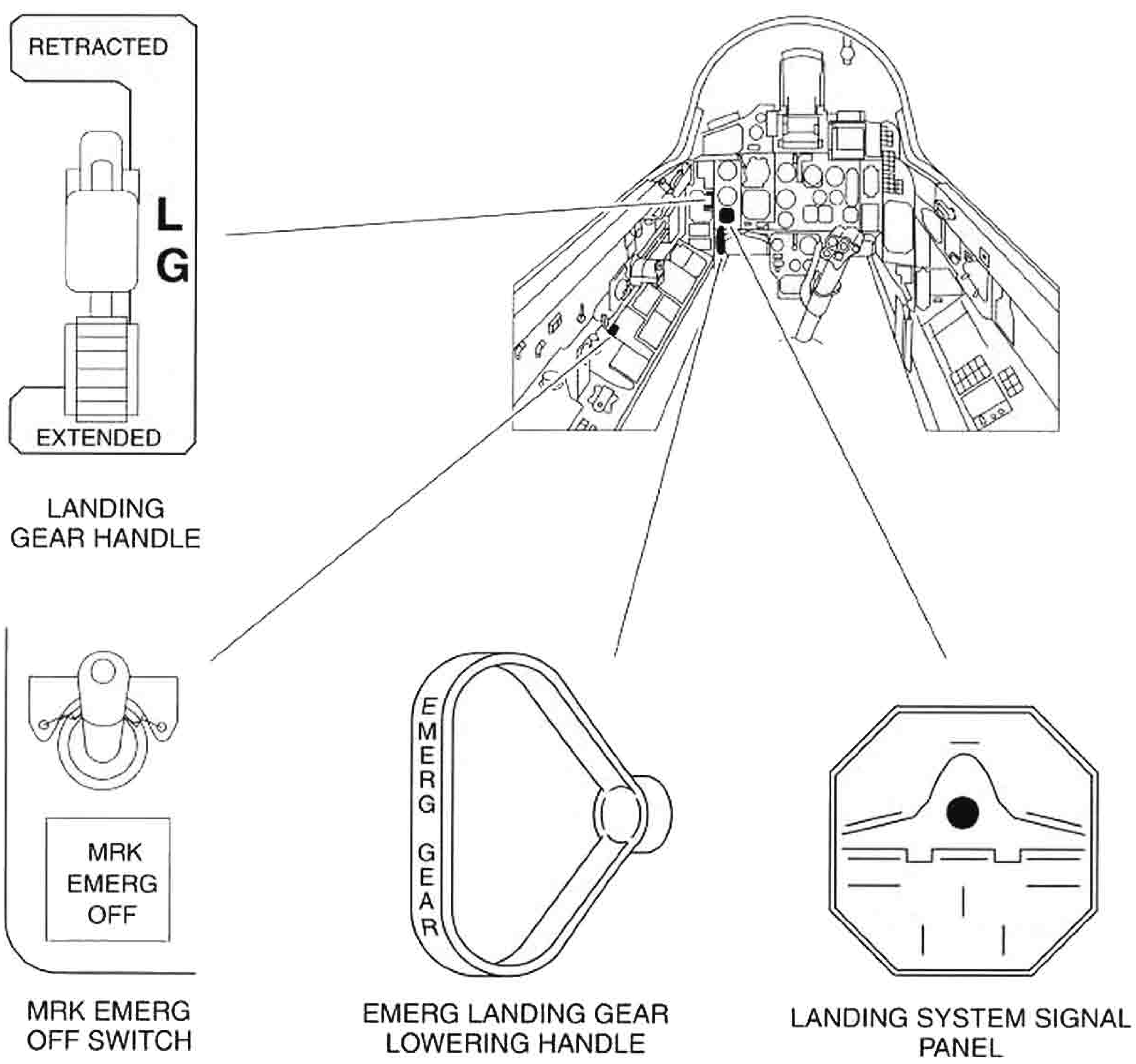


Figure 1-25A

BRAKE SYSTEM

The main wheels and the nose wheels are equipped with a pneumatically operated brake system for normal operation.

Additional features are a run-up brake and a nose wheel brake handle to disable the nose wheel brakes if required.

An emergency brake system is available for the main wheels in case of normal brake system failure.

Normal braking action is accomplished by pulling the brake lever at the control stick aft. The braking force is proportional to brake lever displacement. Differential braking is achieved by displacing the rudder pedals. Moving the right rudder pedal forward, releases brake pressure from the left main wheel brake and vice versa. For engine run-up the run-up brake lever has to be pulled together with the brake lever to achieve a higher brake force preventing the aircraft from rolling, refer to figure 1-26.

CAUTION

Under no circumstances, use the run-up brake to slow down the aircraft.

The rims of both main landing gear wheels are equipped with four fuse plugs each. Three of them, with a melting point of 126° C ±1° C, are mounted in the wheel flange, spaced 120° apart. If any of these fuse plugs has melted, it indicates an overheat condition of the brake system, requiring a system checkout. If all three plugs have melted, the entire wheel rim must be considered damaged beyond repair.

The fourth fuse plug, with a melting point of 143.5° C ±1.5° C, is mounted opposite the tire inflation valve. If this plug melts, the air from the tire is released completely and the brakes must be considered defective.

ANTI-SKID SYSTEM

The aircraft wheel brake system is equipped with an electromechanical controlled anti-skid system. It consist of two basic units, a wheel driven mechanical sensing unit and an electrically driven pneumatic valve. The units are designed to give individual anti-skid protection to each main wheel, and to both nose wheels if either one begins to skid.

The system utilizes DC power from the generator or the batteries. It is activated by placing the BAT-GND SUPPLY switch to on.

Whenever a wheel starts to skid, the wheel driven mechanical sensor closes an electrical switch, which causes the electrically driven pneumatic valve to release pneumatic air pressure from the adjacent brake. Once the wheel has regained its speed, the sensor reopens the switch and braking action is resumed.

CAUTION

Applying brake pressure during touchdown will cause the anti-skid system to be inoperative momentarily. Brakes should not be applied until all gears, including the nose gear, have touched down.

EMERGENCY BRAKE SYSTEM

Pulling the emergency brake handle disables the normal braking system and directs compressed air from the aircraft pneumatic system to the main wheel brakes only. Braking action is degraded approximately 40 % compared to normal braking. Differential braking is not possible. The pressure applied to the brakes is linearly proportional to the displacement of the handle. Releasing the handle relieves brake pressure, refer to figure 1-26.

CAUTION

The anti-skid system is inoperative when the EMERG BRAKE handle is pulled.

NOTE

Emergency brakes are not to be used for normal taxiing.

BRAKE SYSTEM CONTROLS AND INDICATORS

BRAKE PRESSURE INDICATOR

The pneumatic control pressures for the left and right main wheel brakes are monitored by a double pointer instrument. Actual brake pressure is three times higher than the indicated control pressure.

NOSE WHEEL BRAKE HANDLE

The nose wheel brake handle is located on the front panel and is moved to the OFF position to disable the nose wheel brake system. Nose wheel brake pressure is twice as high as the control pressure.

EMERGENCY BRAKE HANDLE

The red emergency brake handle is located on the upper left side of the instrument panel and is labeled EMERG BRAKE. Pulling the handle activates the emergency brake, refer to figure 1-26.

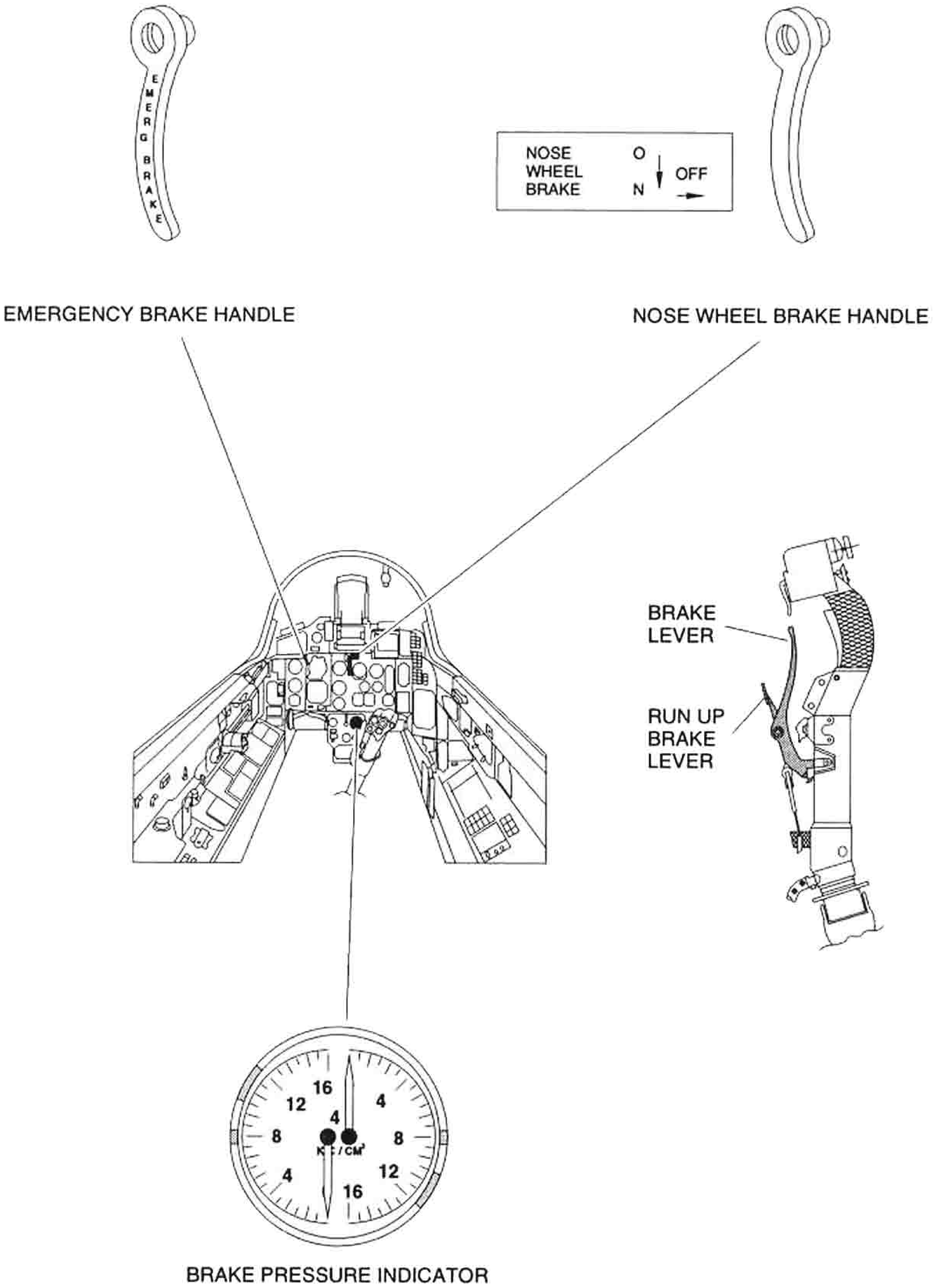


Figure 1-26

DRAG CHUTE SYSTEM

A drag chute, contained in the aft section of the fuselage between the speedbrake doors, reduces landing roll distance. A red control pin is visible whenever the jaws of the attaching mechanism are open. It is pulled into the airstream by a pilot chute when the electrically controlled, pneumatically operated compartment door is opened.

In case of main pneumatic system failure, the system is buffered by a reservoir of 1.2 l, refer to figure 1-24.

If the compartment door opens inadvertently during flight, the chute is allowed to separate from the aircraft by means of a shear bold connecting the chute to the attaching mechanism.

NOTE

The drag chute will separate from the aircraft when exceeding a speed of 175 KIAS.

DRAG CHUTE OPERATION

The drag chute is deployed by pressing the CHUTE DEPLOY button beneath the left canopy rail. Pushing the button activates a pneumatic valve to open the chute compartment door. The spring-loaded pilot-chute pops out and pulls out the drag chute. The drag chute is jettisoned by pressing the CHUTE JETTISON button on the left side panel. To prevent unintentional chute release, the CHUTE JETTISON button is deactivated until the CHUTE DEPLOY button has been depressed.

Use of drag chute is mandatory for:

- Landing immediately after takeoff
- Landing on a wet RWY
- Short field landings
- Landing without LEF
- Abort after nose wheel lift off
- Feel unit is in the position heavy

WING FLAP SYSTEM

The flap system provides an automatic LEF configuration for in-flight maneuvering and a selective flap configuration for takeoff and landing. Each wing has two independent LEF, the root section consisting of three interconnected segments, and the unique end section.

A single slotted flap is mounted on the trailing edge, adjacent to the fuselage. The LEF and the flaps are electrically selected and operated by the main hydraulic system.

The LEF incorporate hydraulic locks, which lock them in either the in or out position.

The flaps are locked in the up position only. The extended position depends on hydraulic pressure only, and as airspeed increases, the flaps are partially blown up by the airstream.

SELECTIVE FLAPS

For takeoff and landing, LEF and the flaps operate together. Anytime the flaps are selected down, the LEF extend automatically. However, if the landing gear is extended, the LEF are extended, regardless of the flaps position.

MANEUVERING LEF

With the flaps in the position UP, the LEF operate automatically as a function of AOA and airspeed. When the AOA is increased to 8.7° or above and the airspeed is below M $0.8^{+0.1}_{-0.05}$, the LEF extend automatically. The LEF retract when the AOA decreases to 7° or airspeed increases above M $0.8^{+0.1}_{-0.05}$. The exact mach number is dependent on the switching point of the mach sensor installed.

CONTROLS AND INDICATORS

Flap operation is controlled by three pushbuttons on the left console. Two are marked FLAPS DOWN, one FLAPS UP. Pushing either FLAPS DOWN button extends all flaps. Pushing the FLAPS UP button will retract the flaps and the LEF, provided the gear is up, refer to figure 1-27.

The position of each flap/LEF is indicated individually by the corresponding light on the landing system signal panel, refer to figure 1-25.

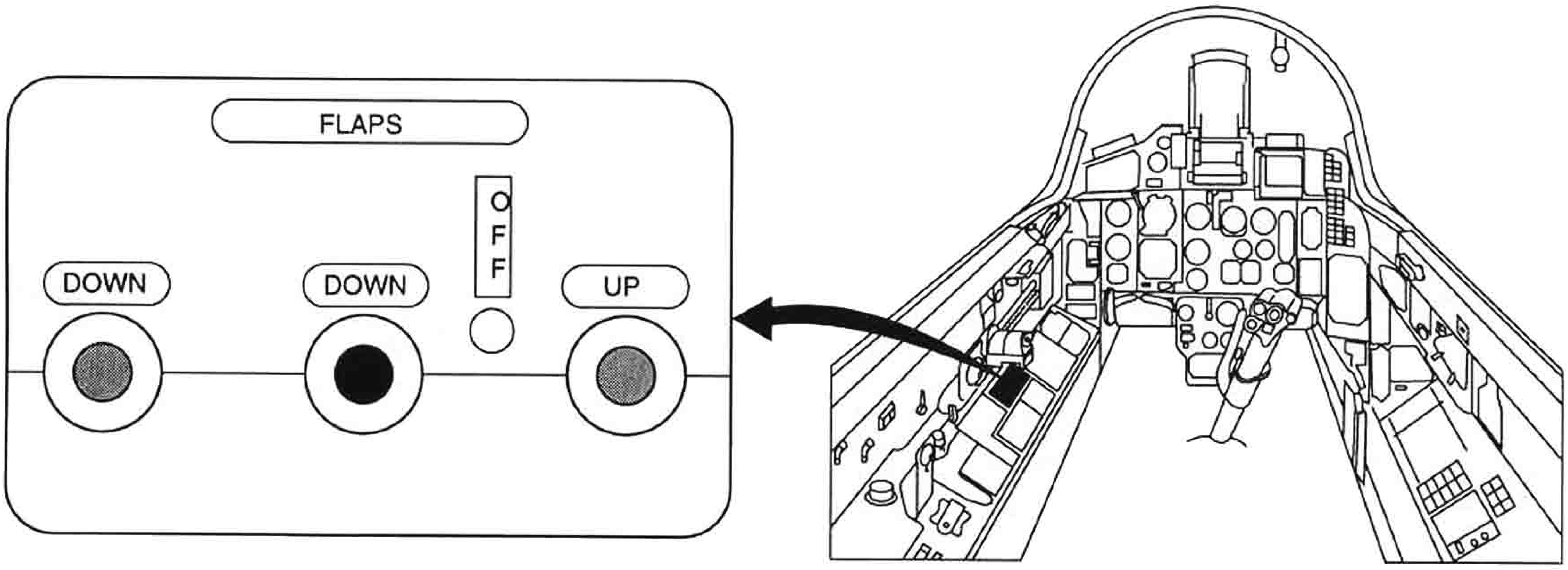


Figure 1-27

SPEEDBRAKE SYSTEM

Electrically controlled, hydraulically operated speedbrakes are mounted above and beneath the drag chute compartment.

The two surfaces are operated simultaneously, but they are not synchronized.

A blow-back feature is incorporated, providing structural protection of the actuators and speedbrake surfaces at airspeeds above 540 KIAS.

The speedbrakes are operated by a spring-loaded switch located on the RH throttle, refer to figure 1-11. It returns automatically to the IN-position upon release. Full extension is achieved within 3 sec.

To protect the operation of the taileron, the speedbrakes are electrically deactivated if main or boost hydraulic pressure decreases. If the boost hydraulic pressure drops below 100 kp/cm^2 (10 MPa), the speedbrakes are retracted automatically by the main hydraulic system. If the main hydraulic pressure drops below 100 kp/cm^2 (10 MPa), the speedbrakes are pushed in by the air stream.

NOTE

Speedbrake operation is not possible with the centerline tank installed, or the gear extended. If total electrical failure occurs, the speedbrakes will retract automatically.

FLIGHT CONTROLS

The aircraft primary flight controls consist of the tailerons, rudders and ailerons. Artificial feel systems provide simulated aerodynamic forces to control stick and rudder pedals. Secondary controls are LEF, flaps and speed brakes. Mechanical linkages transmit control inputs to dual irreversible hydraulic actuators mounted next to the corresponding control surface. All primary flight controls are operated by the main and boost hydraulic system. Full control command is retained if the main or boost hydraulic system fails.

If both systems fail, an emergency hydraulic pump supplies pressure to the boost system, provided hydraulic fluid is still available, refer to FO-11.

WARNING

- If the emergency hydraulic pump has to be used, control will be severely degraded
- In order to maintain aircraft control as long as possible, flight control inputs must be smooth and kept to a minimum.

TAILERONS

Longitudinal control is provided by synchronized deflection of the tailerons. Partial lateral control is achieved by differential deflection. Maximum pitch authority is 15° nose down and 35° nose up. Differential taileron deflection is limited to ± 5°. Taileron authority is varied as a function of airspeed and altitude. It is limited to a minimum of 5°45' nose down and 17°45' nose up near ground level at speeds between 470 KTAS and 650 KTAS. Differential taileron is disabled when the LEF are extended.

AILERONS

Lateral control is provided mainly by the ailerons, assisted by the tailerons if the LEF are IN, and by the rudders at high AOA, provided the stability augmentation system is operating. Maximum aileron deflection is 25° up and 15° down from the neutral position. To prevent excessive yaw during rolls, the neutral position is 5° up from the aircraft horizontal reference line. For the same reason, aileron authority is reduced above 18° AOA.

RUDDERS

The aircraft vertical stabilizers are equipped with small rudders, deflecting 25° to either side. The rudder feel gradient is increased by 30 kp at a rudder pedal deflection of 24 mm from the trimmed position
G: at airspeeds above M 0.8
GT: with gear up.

It is strongly recommended not to override the artificial stop.

CAUTION

To prevent overstress of the vertical stabilizers it is prohibited to override the rudder artificial stop at airspeeds
G: >485 KIAS,
GT: >432 KIAS.

ARTIFICIAL FEEL SYSTEM

Artificial feel is provided by a system of springs. The artificial feel applies centering forces to the stick and the rudder pedals towards the trimmed position.

FEEL CONTROL UNIT

The pitch feel control unit utilizes signals from the air data computer to control an electric actuator gearbox. This gearbox varies the length of a rod by up to 50 mm, to change the stick to taileron linkage ratio. This results in alteration of the taileron deflection range and required stick force with respect to aerodynamic forces, i.e. airspeed and altitude. The FEEL UNIT TO / LD light on the TLP illuminates whenever the feel control unit is in the easy position, e.g. during takeoff and landing. Refer to figure 1-29.

At altitudes below 3000 ft, the length of the variable rod depends on airspeed only. At 215 KTAS, the rod starts to retract. It is fully retracted at 470 KTAS and stays fully retracted up to 650 KTAS. At 650 KTAS, the rod starts to extend again and is fully extended at 810 KTAS, the FEEL UNIT TO / LD light illuminates again.

At altitudes above 3000 ft, rod retraction becomes smaller with increasing altitude. Above 30 000 ft, it is always fully extended. For detailed operation see figure 1-28.

If the feel control unit fails, it can be controlled manually with the FEEL UNIT control switch on the left console. Refer to figure 1-29.

WARNING

If the feel control unit fails, stick movements should be minimized to prevent PIOs.

FEEL CONTROL UNIT SCHEDULE (PITCH)

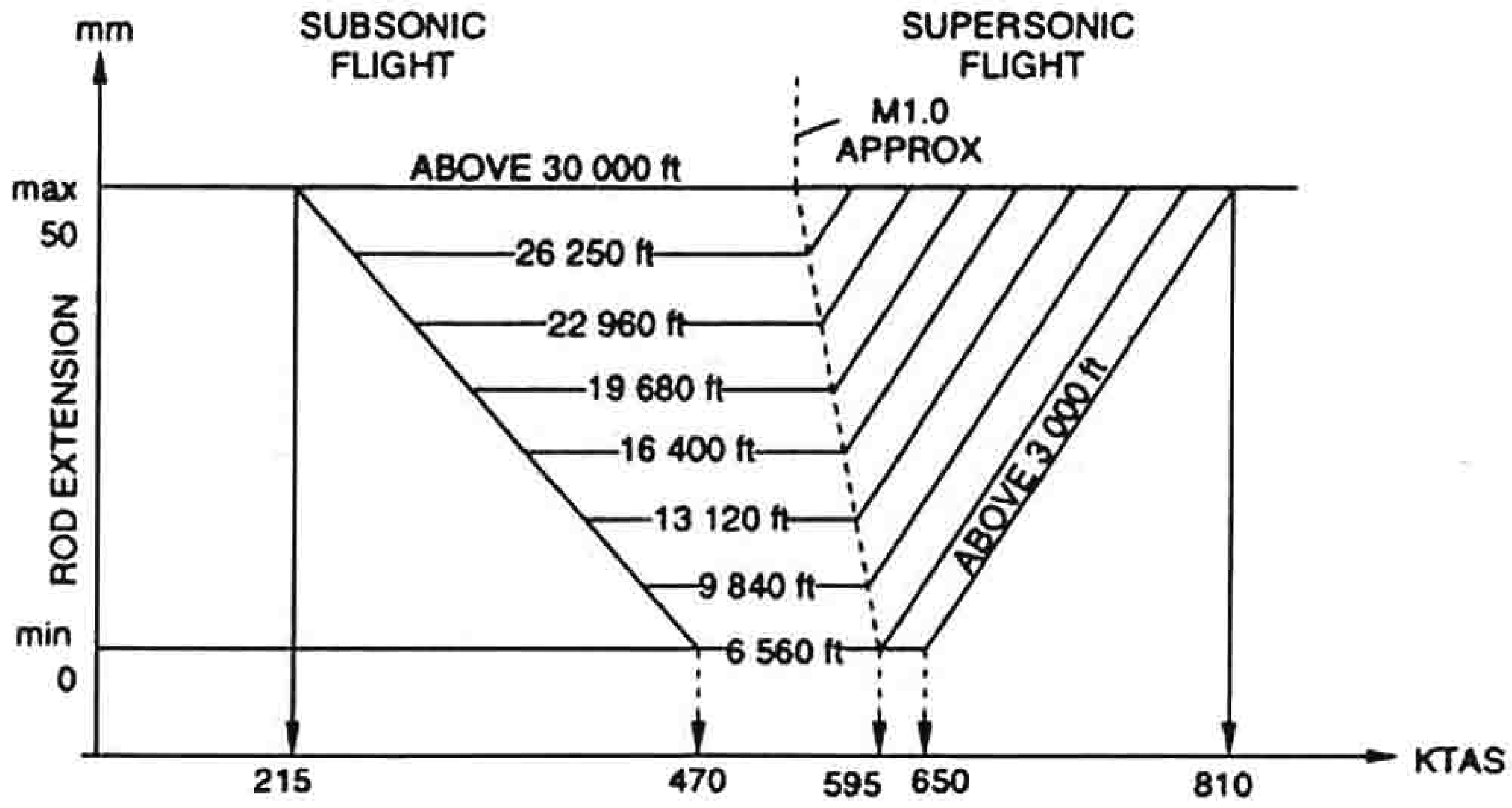


Figure 1-28

INDICATION AND WARNINGS

	INDICATION	FAULT/EFFECT
AEKRAN	FEEL CONT UNIT	Feel control unit out of limit
VIWAS	"AUSFALL ARU" "VOR LANDUNG AUF STELLUNG LEICHT"	

ROLL

Roll feel forces are generated by a mechanical spring unit with linear force characteristics. Non linearities are induced with the stability augmentation system engaged to provide low sensitivity for small control inputs and high sensitivity for large control inputs.

YAW

Pedal feel forces are generated by a spring unit system with linear force characteristics. When the gear is retracted, a hydraulic actuator adds additional centering forces at airspeeds higher than M 0.8 and rudder pedal displacements of more than 24 mm, equivalent to 6° rudder deflection.

TRIM SYSTEM

The trim system is used to relieve control stick pressure. Actuating a trim switch causes the appropriate trim actuator to move either in yaw, roll or pitch.

PITCH TRIM

Pitch trim is affected by a trim actuator incorporating an electric motor. When operated, the trim actuator varies the translation ratio of the taileron linkages, which in turn provides a new stick center position. Trim authority is 80 % of available taileron deflection.

ROLL TRIM

Roll trim is similarly affected by an electric motor driving a trim actuator. Trim authority is 60 % of available aileron deflection.

YAW TRIM

Yaw trim is similarly affected by an electric motor driving the trim actuator. Trim authority is 60% of the available rudder deflection.

CONTROLS AND INDICATORS

Trim Button Unit

The trim button unit on the control stick grip consists of a pyramid cap which houses two toggle switches. It provides trim control in the pitch and roll axis. The trim button is springloaded to the center and can be moved forward, aft, left and right.

Rudder Trim Switch

The rudder trim switch is located on the left vertical panel.

Trim Indicators

Three lights on the TLP indicate the neutral position of the corresponding trim actuator.

FEEL AND TRIM CONTROLS AND INDICATORS

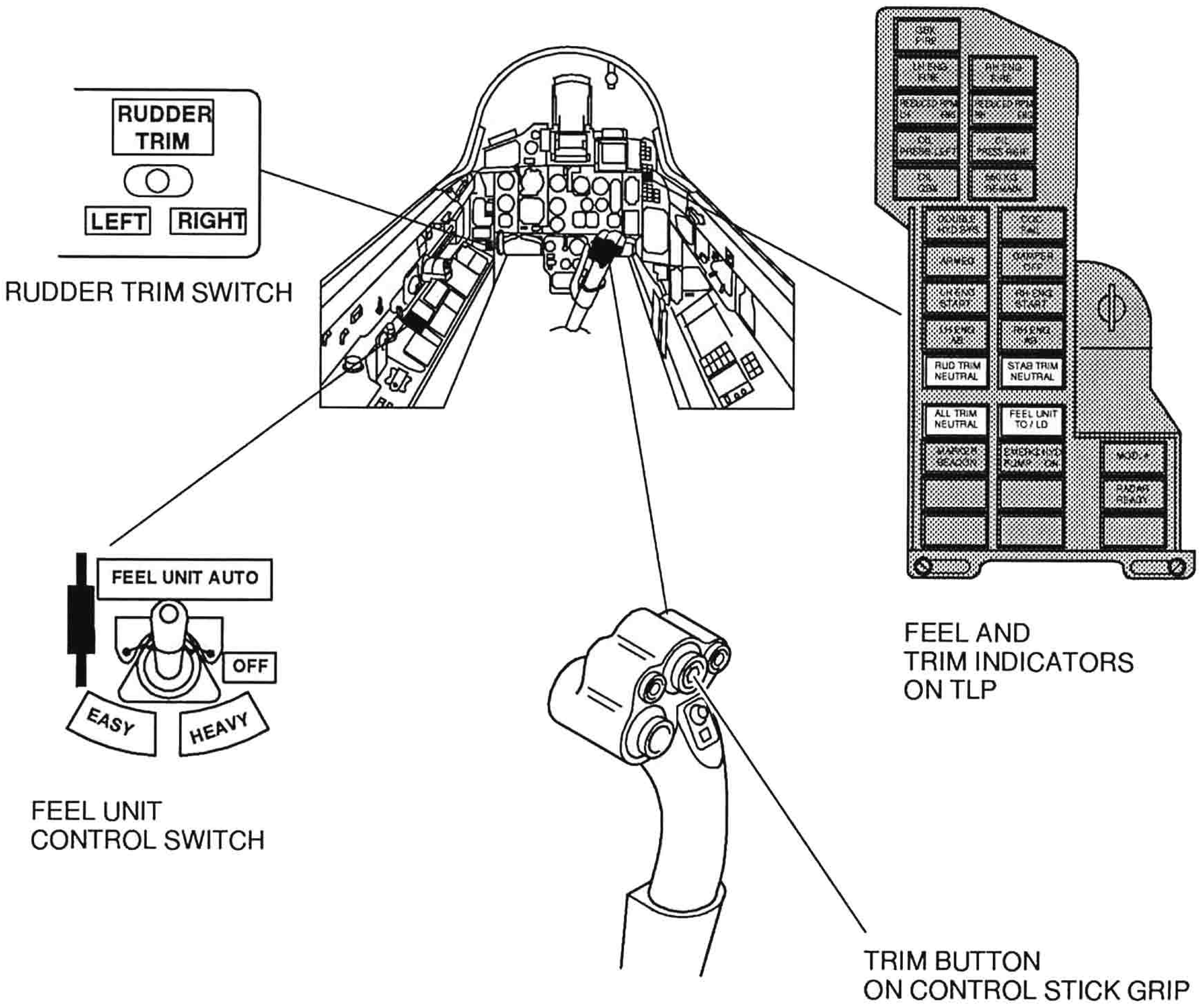


Figure 1-29

CONTROL STICK

The control stick consists of a grip with an adjustable hand rest and the following functions:

1. Gun trigger
2. Missile trigger
3. AP cut-out lever
4. Brake lever
5. Run-up brake lever
6. CL tank jettison button
7. Trim button
8. Levelling button
9. AFCS MODES OFF button
10. Target acquisition symbol button
11. Break-lock button
12. Rudder pedal adjustment handle

The AFCS MODES OFF button interrupts the power supply to the automatic flight control system. The AP cut-out lever disables all automatic flight modes of the AFCS as long as it be pressed.

CONTROL STICK

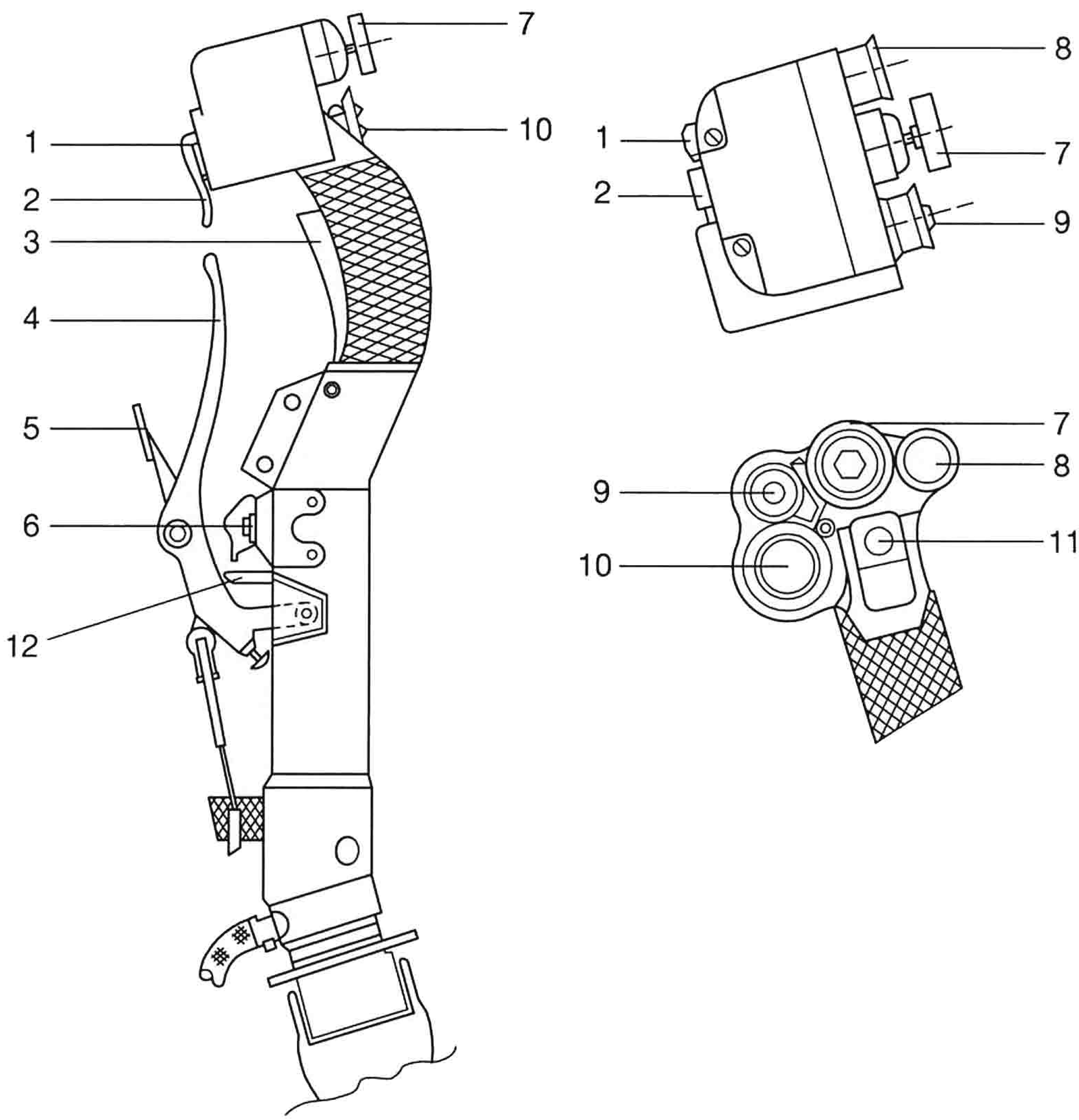


Figure 1-30

RUDDER PEDALS

Primary controls for the rudders consist of conventional rudder pedals mechanically connected to hydraulic actuators.

During ground operation, differential braking is controlled by the rudder pedals. The rudder pedals can be adjusted by the ring-type rudder adjustment handle on the control stick.

AOA / G CONTROL SYSTEM

The AOA / G control system (COC) measures and indicates angle of attack (AOA) and g-forces, it controls automatic LEF operation and prevents inadvertent stalls by moving the control stick forward. The system is powered by 28.5 VDC and 2 phases 115 VAC.

The system consists of the AOA / G computer, the combined AOA / G meter as well as warning and indicator lights. It utilizes inputs from the AOA vanes, Mach sensors, the g-sensor and the LEF down limit switches to perform the following tasks:

- Display of actual and maximum g-forces
- Display of actual AOA
- Automatic LEF operation considering pitch velocity
- Computation of the maximum AOA, considering LEF position and pitch rate
- Operation of the pitch kicker considering pitch rate and AOA
- Display of system malfunctions

The g-sensor measures g-forces between -2 g and +10 g. The signals are amplified in the computer and displayed on the g-scale of the combined AOA / G meter at a rate of at least 5 g per second with an accuracy of ±0.3 g (±0.4 g under extreme weather conditions).

Actual AOA is measured by the LH and the RH AOA vanes from -1.5° to +29°. The computer selects the higher value, amplifies the signals and displays the AOA on the AOA scale of the combined AOA / G meter at a rate of at least 20° per second with an accuracy of ±1° (±1.5° under extreme weather conditions).

AOA LIMITER

The AOA / G computer utilizes signals from the AOA vanes and the Mach sensors to position the LEF and to actuate the pitch kicker. The system is disabled when the nose landing gear is not up and locked.

The LEF extend at an AOA of 8.7° at Mach numbers below M $0.8^{+0.1}_{-0.05}$. Depending on pitch rate, the LEF may extend prior to reaching 8.7° AOA.

The signals from the LEF down limit switches are utilized to switch the AOA / G computer from the low AOA value of 15° (GT: 14°) maximum to the high AOA value of 26° (GT: 24°) maximum.

The pitch kicker is designed to prevent inadvertent stalls by moving the control stick forward of neutral when either pitch rate or AOA, or a combination of both, reaches the critical value. The computer triggers solenoid valves to operate the hydraulic actuators, which cause the taileron to assume an aircraft nose-down deflection and the control stick to move forward. Thus the pilot is immediately made aware of an approaching stall condition of the aircraft. A force of 17 kp, in addition to normal control forces, applied on the stick can override the pitch kicker.

NOTE

Full aft pitch trim reduces the forward force on the control stick considerably. Under this condition, caution should be used when reapplying backstick pressure.

CAUTION

- Due to extremely reduced stability margin at high AOA, an AOA reduction of approximately 4° is strongly recommended prior initiating any roll maneuver.
- Overriding the pitch kicker intentionally is prohibited.

System redundancy is achieved by using dual actuators operated by the main and the boost hydraulic system and by duplicating the computer channels.

The AOA limiter system contains continuos BIT. It monitors the heating system of the AOA vanes, AOA signal inputs and DC electrical power. The heating system of the AOA vanes will operate with reduced power when the pitot heat switch is positioned to ON, however, full heating power is automatically provided when weight is off the RH main gear, regardless of switch selection.

A test button on the control and test panel can be used for initiating an extended self test for maintenance purposes.

AOA / G CONTROL SYSTEM

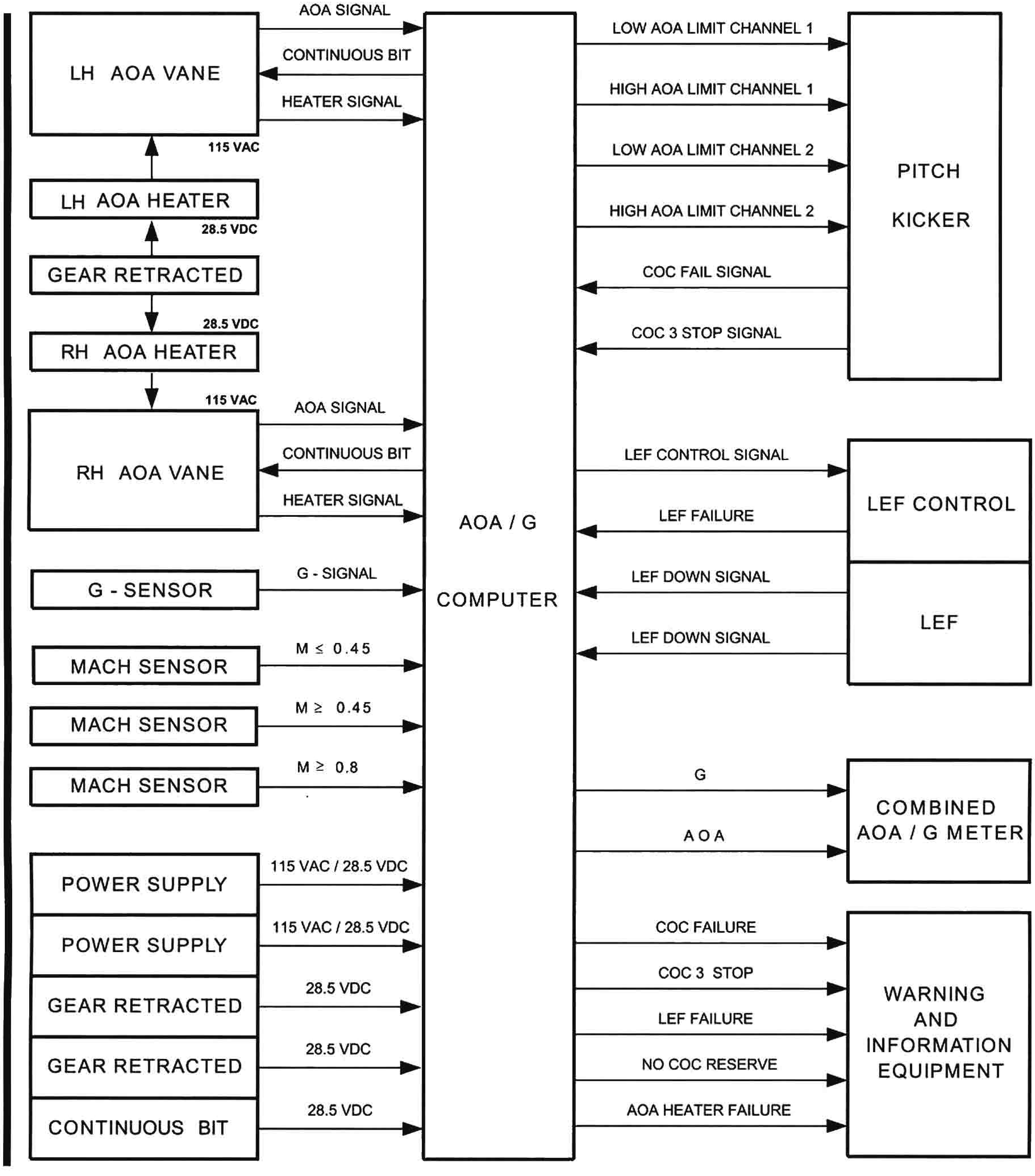


Figure 1-30A

INDICATIONS AND WARNINGS

A complete COC failure is indicated on the TLP, when both AOA limiter channels of the AOA/G computer have failed.

CAUTION

With a complete COC malfunction, pitch kicker warning is not available. Extreme care should be used when operating near the allowable AOA limit.

	INDICATION	FAULT/EFFECT
MASTER CAUTION	LIGHT FLASHING	
TLP	COC FAIL	Complete AOA limiter failure

A COC 3 STOP indication on the AEKRAN during landing gear extension indicates a failure of the pitch kicker hydraulic actuator. A forward force of 17 kp is added to normal control forces when the landing gear is extended and airspeed is below M 0.45.

	INDICATION	FAULT/EFFECT
AEKRAN	COC 3 STOP	Pitch kicker hydraulic actuator failure.

If the LEF down limiter switches do not signal LEF down when an AOA of 12.5° is reached, the signal LEAD EDGES NOT EXTEND is displayed on the AEKRAN. The extension signal for the LEF is deleted in this case.

	INDICATION	FAULT/EFFECT
AEKRAN	LEAD EDGES NOT EXTEND	LEF fail to extend, maximum AOA limited to 15° (GT: 14°) AOA.

A single AOA limiter channel failure of the AOA/G computer is indicated on the control and test panel.

	INDICATION	FAULT/EFFECT
CONTROL AND TEST PANEL	NO COC RESERVE	Single channel AOA limiter failure, no effect on maximum allowable AOA.

AUTOMATIC FLIGHT CONTROL SYSTEM

The automatic flight control system (AFCS) is an electro-hydraulic system designed to provide automatic, semiautomatic and manual flying modes without interfering with manual control. The system requires DC power and 36 VAC power.

It consists of the AFCS computer, the stability augmentation system, and the trim actuators, refer to FO-13.

The AFCS computer receives signals from the gyro platforms, the accelerometers, the g-meter, the control stick position and the navigation system. The signals are corrected for altitude, mach number, AOA and pitot pressure.

The inputs are processed into control signals for the dampers, the trim actuators and the flight instruments.

The automatic flight control system is capable of performing the following modes of operation:

- DAMPER (stability augmentation)
- ATT HOLD (attitude hold)
- AUTO RECOVER (level off mode)
- APPROACH
- ALT HOLD (altitude hold)
- Automatic landing approach control
- Levelling (automatic unusual attitude recovery)

OPERATING MODES

DAMPER

Stability augmentation is obtained for pitch, roll and yaw by placing the AFCS switch to the ON position. The AFCS computer controls electro-hydraulic actuators within the mechanical control linkages to compensate any tendency of the aircraft to oscillate in roll, yaw and pitch. Aileron-rudder-interconnect is provided at high AOA by the stab aug system to improve lateral stability.

The automatic longitudinal stability control adjusts the taileron actuator to counteract pitch moments during LEF operation. The automatic longitudinal stability control is disengaged when the LEF are in, the flaps are down or the dampers are off.

The dampers are disengaged if

- Hydraulic supply to the actuators fails
- Any phase of the 200 V, 400 Hz electric power fails
- The AFCS computer fails
- The AFCS MODES OFF button on the control stick is pressed

If the AFCS MODES OFF button is pressed less than 3 sec, the stab aug system is reengaged automatically upon release. If the button is pressed more than 3 sec, the stab aug system disengages completely. The DAMPER pushbutton on the AFCS mode control panel must be pressed to reengage the system.

ATTITUDE HOLD MODE

The Attitude Hold mode is designed to maintain the airplane's attitude. At bank angles from 7° to 80° and pitch angles of ±80°, the aircraft attitude is maintained. At bank angles below 7° and pitch angles of ±40°, attitude and heading are maintained.

GT: At bank angles from 7° to 70° and pitch angles of ±50°, the aircraft attitude is maintained.

To engage the attitude hold mode, the ATT HOLD button on the autopilot control panel is pressed. If the damper system is off, it will automatically be engaged. The ATT HOLD light will flash until the autopilot AP cut-out lever is released. Releasing the lever starts mode operation, and the light becomes steady.

Aircraft attitude is changed by pressing the A/P cut-out lever, attaining new attitude and releasing the lever. Stick forces have to be trimmed to balance prior releasing the lever.

Engaging the attitude hold mode automatically disengages the levelling mode. The attitude hold mode is automatically disengaged if the levelling mode or auto recover mode are engaged.

Trim condition may be slightly out of balance when disengaging the attitude hold mode.

AUTO RECOVER MODE

Auto recover mode is designed to recover the aircraft to a minimum altitude set on the radar altimeter during flights below 3 000 ft above ground level. It is engaged by pressing the AUTO RECOVER pushbutton on the AFCS control panel. However, engagement should not take place when flying below the preset minimum altitude. Operation is indicated by illumination of the AUTO RECOVER light.

If the aircraft descends below the preset altitude, the low altitude warning light on the radar altimeter illuminates. The AEKRAN displays ALT ALERT and VIWAS says "Gefährliche Höhe". The aircraft starts an 8° climb, initiated with 1.5 to 5.0 g. The wings are set level and the levelling button light on the control stick illuminates simultaneously. Upon reaching the preset altitude, the levelling mode is engaged automatically. When the aircraft is in level flight, ALT HOLD mode is engaged automatically.

If the AP cut-out lever is pressed, or trim is applied during the descent, the aircraft does not recover automatically. However, the low altitude warnings are displayed and an aft stick force is applied within 3 to 4 sec.

Auto recover mode is disengaged by pressing the AUTO RECOVER button again on the AFCS control panel or by pressing the AFCS MODES OFF button for more than 3 sec. In this case the dampers disengage simultaneously and must be reengaged by pressing the DAMPER button on the AFCS control panel.

NOTE

- The auto recover mode is restricted to a minimum altitude of 600 ft AGL, a max bank angle of 30° and a max descent rate of 2 000 ft per min.
- During AUTO RECOVER, flying with the parameters mentioned above, an altitude loss of up to 300 ft has to be anticipated.

At descent rates of less than 200 ft per min with wings level, no altitude restrictions apply.

WARNING

Due to radar altimeter restrictions, the AUTO RECOVER mode is not reliable at bank angles in excess of 30°.

APPROACH MODE

The approach mode provides ILS information and command steering. It can be engaged by pressing the APPROACH button on the AFCS control panel as soon as reliable ILS signals are received for course and glideslope.

Operation is indicated by the illumination of the APPROACH light and the disappearance of the pitch and course OFF-flags on the ADI.

CAUTION

To prevent (violent) control transients, the AFCS mode ATT HOLD must be disengaged prior to engagement of the APPROACH mode.

Bank and pitch commands are provided by the course and pitch steering bar on the ADI and the command circle in the HUD. Bank steering commands are based on a roll rate of 5 to 8° per sec. The pitch steering bar provides steering towards the engagement altitude of the APPROACH mode until glide path interception. Centering the pitch and course steering bar, as shown by the command circle in the HUD, ensures proper steering.

Failure of the glide slope indication will cause a level-off command by the pitch steering bar on the ADI. The pitch OFF-flag on the ADI and the ILS glide slope OFF-flag on the HSI will appear.

WARNING

When any OFF-flag on HSI or ADI appears, level off immediately and execute a missed approach.

Approach mode can be deselected by pressing the AFCS MODES OFF button for less than 3 sec.

ALTITUDE HOLD MODE

The altitude hold mode is designed to maintain the aircraft at a specific barometric altitude. Altitude hold is engaged by pressing the ATT HOLD button first and then the ALT HOLD button on the AFCS control panel. Illumination of both lights indicates proper operation.

To engage the mode, the pitch attitude must not exceed ± 5°. If the pitch angle exceeds 5°, the aircraft stabilizes at the given angle and the ALT HOLD light flashes until the angle is decreased below 5°.

After correction to the engagement altitude, the aircraft is stabilized in bank angle and altitude. If the bank angle was less than 7° during engagement, heading and altitude are stabilized.

Attitude hold can be cut out intermittently by pressing the AP cut-out lever, which is indicated by flashing of the ATT HOLD and ALT HOLD lights. Pressing the AFCS MODES OFF button for less than 3 sec disengages the altitude hold mode.

After recovery to level flight with the levelling mode, ALT HOLD is engaged automatically.

NOTE

- Altimeter fluctuations while accelerating through the transsonic range will produce transient fluctuations, which, although not violent, may cause the reference altitude to slip.
- Do not use ALT HOLD at altitudes below 300 ft AGL.

AUTOMATIC LANDING APPROACH CONTROL

The automatic landing approach system can be engaged after the approach mode has been selected and the pitch and course steering bars on the ADI have been centered.

For smooth operation, airspeed should be below 215 KIAS. Automatic throttle adjustment is not available.

To engage automatic landing approach control, the ATT HOLD button has to be pressed in addition to the APPROACH button on the AFCS control panel. Illumination of the ATT HOLD and the ALT HOLD lights as well as the disappearance of the OFF-flags on the ADI/HSI indicate proper system operation.

Level flight is maintained until glide slope interception. Upon glide slope interception, the ALT HOLD light extinguishes and the aircraft begins the descent. Course corrections are performed with bank angles up to 5°, glide slope corrections with pitch angles up to 2°.

The automatic landing approach system can be temporarily disengaged by pressing the A/P cut-out lever and is automatically reengaged as soon as this lever is released.

The automatic landing approach has to be discontinued if the pitch or course steering bar on the ADI indicate a difference to the course or glide slope indication on the HSI, or the minimum altitude of 150 ft is reached.

The automatic landing approach control is disengaged by pressing AFCS MODES OFF button for less than 3 sec or by engaging the AFCS levelling mode.

WARNING

The autopilot may trim the aircraft considerably out of balance. Therefore, when disengaging the automatic landing approach mode, be prepared to counteract large control transients.

LEVELLING MODE

Levelling mode is designed to recover the aircraft to straight and level flight in case of pilot's spatial disorientation. Pressing the levelling button on the right side of the trim button disengages all other AFCS modes, and engages the levelling mode, provided the AP cut-out lever is released. If the dampers are OFF, they are automatically engaged. Levelling operation is indicated by steady illumination of the button light.

At bank angles below 80°, bank and pitch attitude are recovered to level flight simultaneously. At bank angles of more than 80° bank is recovered to below 80° before simultaneous recovery. The recovery rate varies from 10° to 45° per sec in bank and -1 g to +5 g in pitch, depending on altitude, attitude and airspeed.

Once the aircraft is recovered to ±7° AOB and ±5° of pitch or below, the ALT HOLD mode is automatically engaged within 3 to 4 sec as indicated by the ALT HOLD light.

If the pilot interferes during levelling operations by pressing the AP cut-out lever or using the trim button, the levelling mode disengages momentarily which is indicated by flashing of the levelling light.

Levelling mode is disengaged either by pressing the AFCS MODES OFF button for less than 3 sec, or by engaging the attitude hold mode.

NOTE

- During levelling operations, the rudder pedals must be neutralized and maintained in neutral position.
- Throttle adjustments may be required according to airspeed and altitude.
- If a forward trim condition exists prior to engagement of levelling, negative g may be experienced momentarily before positive recovery.
- During recovery from negative pitch angles, or bank angles of 40° to 50°, bank may increase up to 70°, not resulting in additional loss of altitude.

AFCS BITE

Placing the AFCS switch to ON, initiates the BIT, provided weight is on the nose gear, the inertial platforms are ready and hydraulic pressure is available.

Prior to engagement, the control surfaces must be trimmed to neutral, as shown by the indicator lights.

During the BIT, controls must be released to allow unrestricted movements. The DAMPER OFF light illuminates on the TLP and the MASTER CAUTION light flashes. The DAMPER light flashes continuously at a rate of 1.5 to 2.0 cycles per second and all other lights on the AFCS control panel may illuminate temporarily, except for the MISSED APPROACH light.

Upon completion of the BIT, the DAMPER OFF light on the TLP extinguishes and the DAMPER light illuminates steadily. It indicates satisfactory BIT completion and the stab aug system engaged. If the control stick is out of the neutral position it must be trimmed back to neutral.

An AFCS malfunction is indicated by flashing of all lights on the AFCS control panel, except for the MISSED APPROACH light. If the malfunction is not within the stab aug system, the dampers can be engaged by pushing the DAMPER button. In this case only the damper function is usable, none of the other AFCS modes may be engaged.

A normal test cycle lasts 90 sec. However, the dampers can be engaged after 40 sec. In this case, the BIT is interrupted and none of the other AFCS modes can be engaged. After neutralizing the control stick the BIT may be reinitiated by pressing the AFCS MODES OFF button momentarily.

NOTE

With the ICAO II modification implemented, the course pointer must be set to a vertical position before the AFCS self test is initiated.

CONTROLS AND INDICATORS

AFCS Switch

The AFCS switch is located on the system power panel. Refer to figure 1-31. It is used to switch the AFCS ON and OFF.

AFCS CONTROL PANEL

The AFCS control panel is located in front of the left console and has five pushbuttons to select the AFCS modes. When a mode is selected, the corresponding green light illuminates.

The button MISSED APPROACH is used to obtain steering information from the navigation system.

AFCD CONTROL PANEL

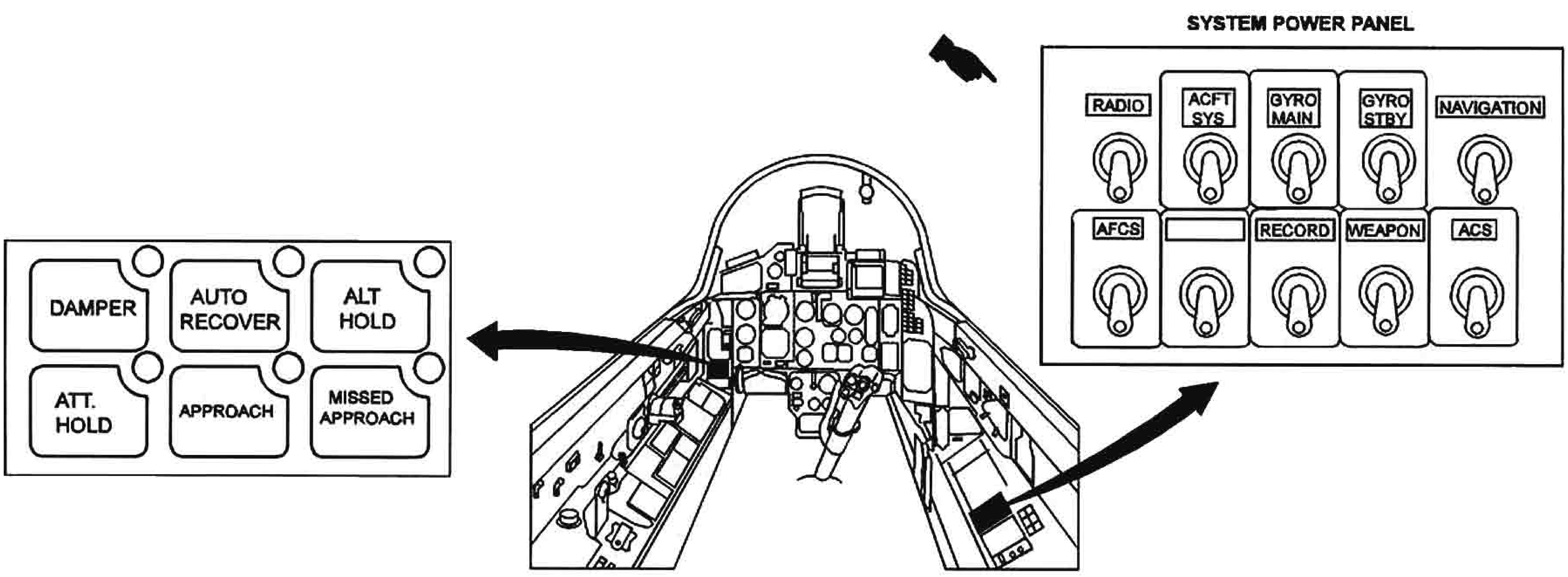


Figure 1-31

STICK CONTROLS

Refer to figure 1-30.

AP Cut-Out Lever

The AP cut-out lever is integrated into the control stick grip and disables the AFCS flying modes while pressed. The associated AFCS modes indicator lights begin to flash.

AFCS MODES OFF Button

The AFCS MODES FF button on the left side of the trim button disengages all AFCS modes. However, the stab aug system will be reengaged if the button is pressed for less than three seconds. If it is pressed for more than three seconds, the system is switched off and the DAMPER OFF light on the TLP illuminates. The AFCS is disengaged completely.

Levelling Button

The levelling button with an integrated indicator light located to the right side of the trim button is used to engage the unusual attitude recovery mode of the AFCS.

WARNING

When an AFCS mode is selected, be prepared to manually counteract any abrupt control movements in the event of an AFCS malfunction.

PITOT STATIC SYSTEM

Two pitot booms, a main and a emergency pitot boom, supply impact and static pressures to various flight instruments and aneroid switches.

The main pitot boom provides pressure to the IAS/TAS indicator, the VVI, the ADC and the IFF.

The emergency pitot boom provides pressure for the Mach transducer, the LEF controls, the ECU, the AFCS, the engine intake ramp control and the ejection seat.

If the main system fails, the emergency pitot boom can be selected to provide pressure to the main system users, for details see figure 1-32.

PITOT STATIC SYSTEM

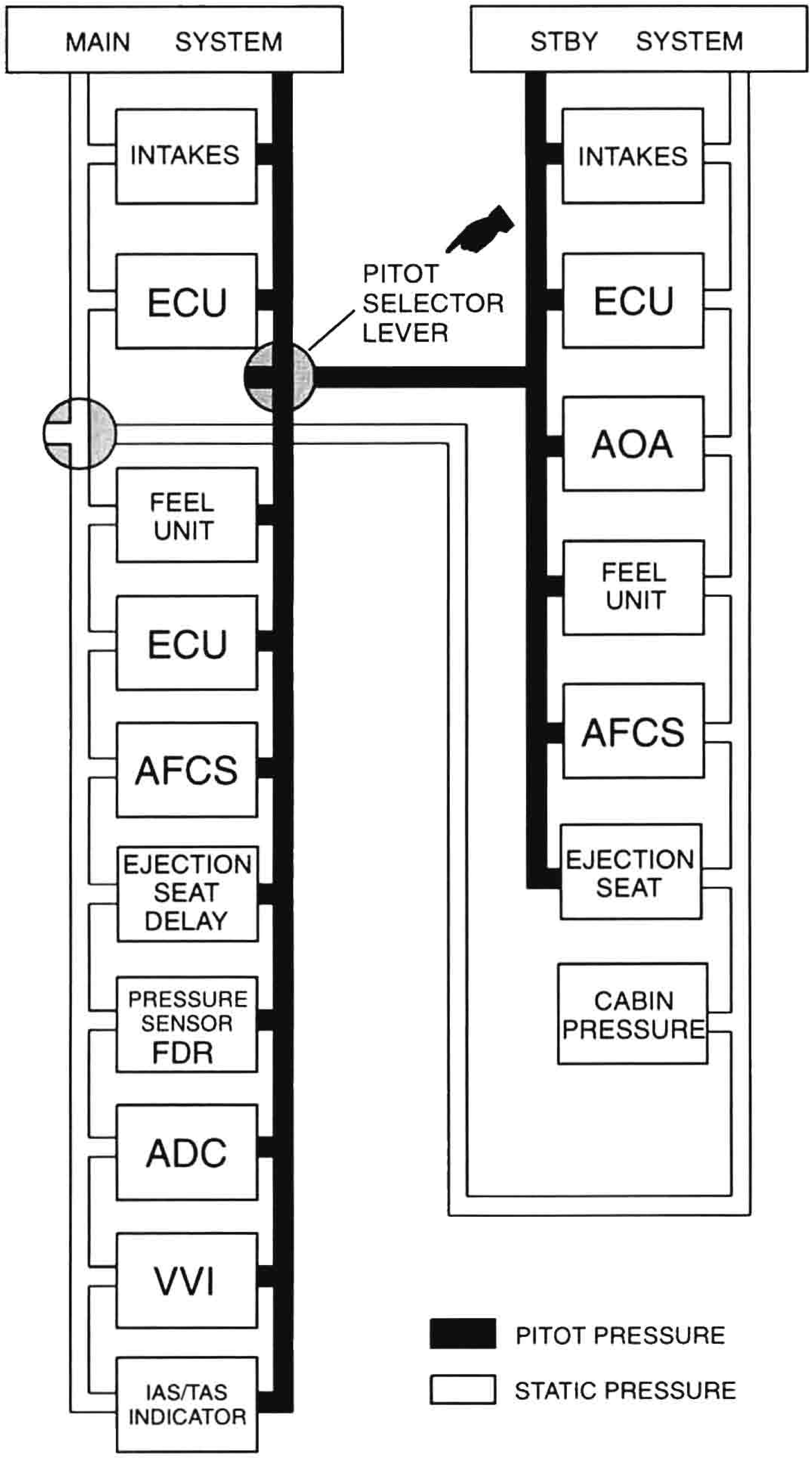


Figure 1-32

CONTROLS AND INDICATORS

Pitot Selector Lever

The pitot selector lever is located at the pedestal panel with the positions MAIN and STBY.

Pitot Heat Switch

The pitot heat switch is located on the RH side wall. It controls operation of the heating elements in both pitot-booms, AOA probes and the windshield. The heater elements are energized any time 28 VDC power is available and the pitot heat switch is in the ON position. AOA probe and windshield heater elements are energized when the weight is off the wheels and 115 VAC / 400 Hz power is available.

CAUTION

Pitot heat should not be used for more than one minute during ground operations to prevent damage to the system.

PITOT STATIC SYSTEM CONTROLS

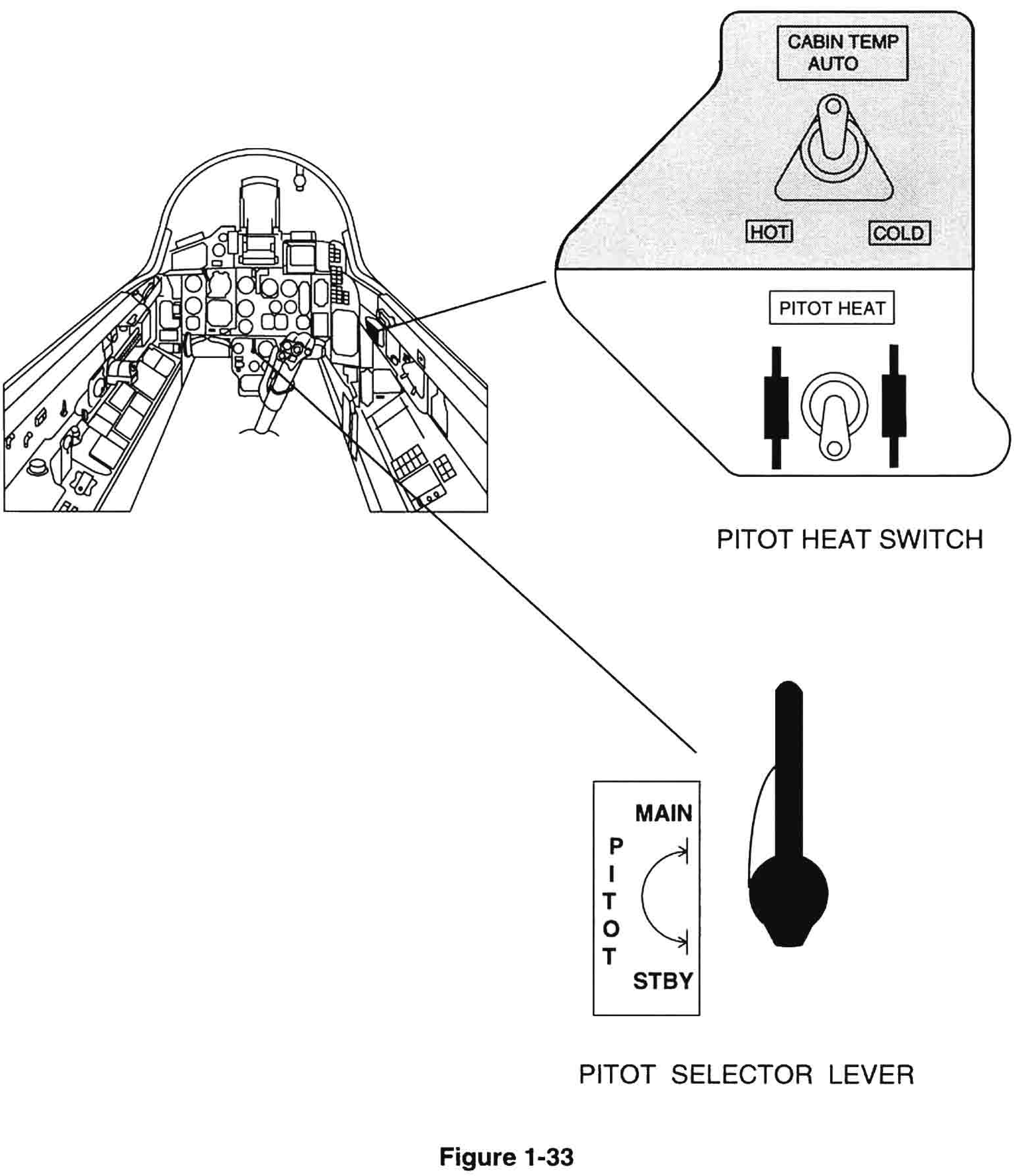


Figure 1-33

AIR DATA COMPUTER

The air data computer (ADC) is part of the pitot static system and consists of:

- Air data system
- Pressure altimeter
- TAS indicator

It utilizes inputs of pitot and static pressure from the main pitot system and OAT.

The computer provides electrical outputs representing:

- TAS
- Mach number
- Pressure altitude
- Density altitude

Data are supplied to the navigation system, the AFCS, the fire control system and the fuel indication system. The ADC utilizes DC power from the generator or the batteries, 36 VAC and 115 VAC.

A build in test is provided to check system readiness during preflight preparation and check-out for maintenance purpose.

AIR DATA COMPUTER

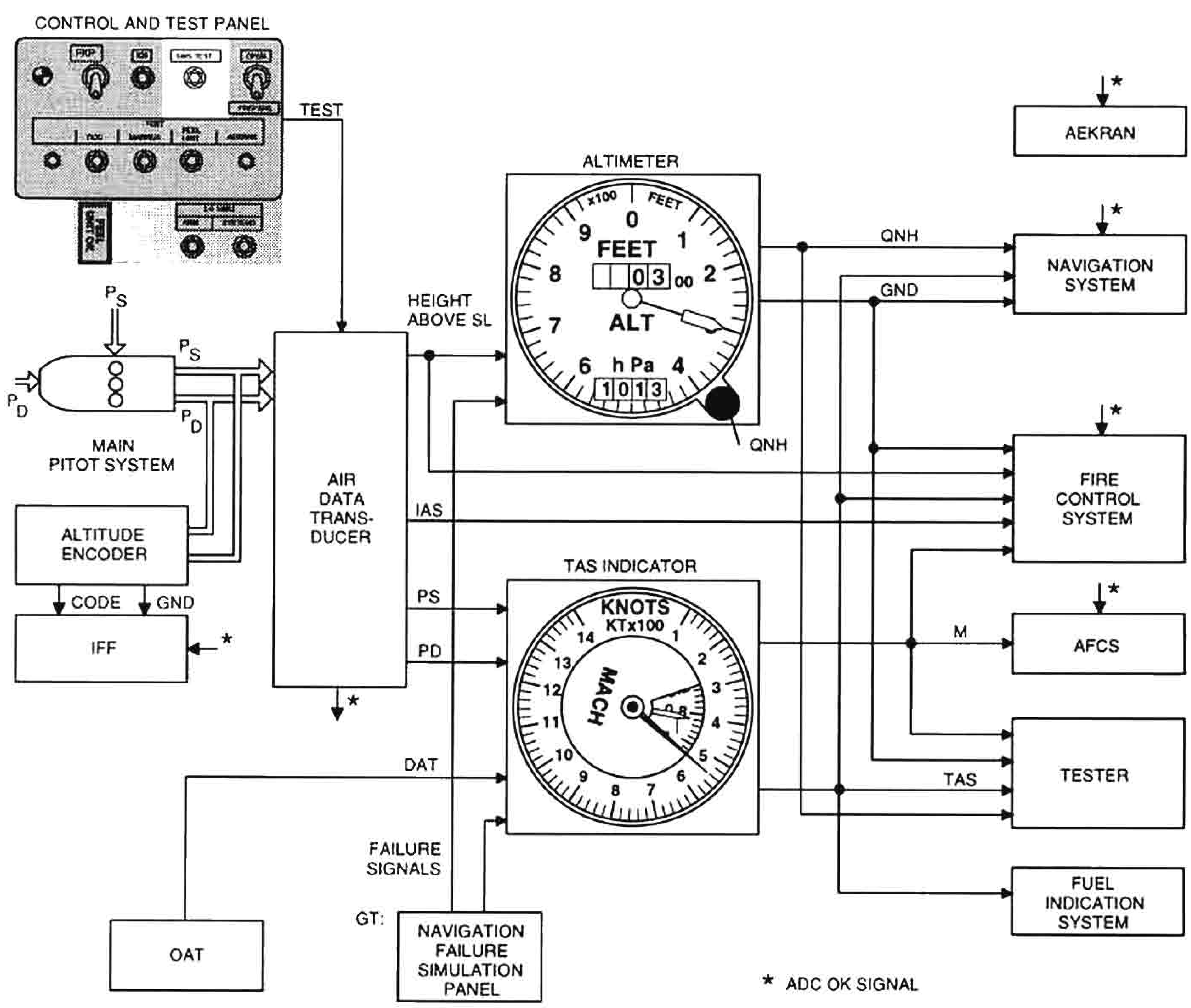


Figure 1-34

INDICATIONS AND WARNINGS

	INDICATION	FAULT/EFFECT
AEKRAN	AIR DATA SYS	Unreliability of airspeed, altitude and vertical velocity
VIWAS	"AUSFALL SWS" "AP AUS, V_{max} 600"	

INSTRUMENTS

IAS INDICATOR

The IAS indicator USM-2AE displays indicated airspeed. A single pointer indicates airspeed values between 0 and 800 kts on a non-linear scale. Mach number is indicated on the inner scale.

The pneumatic inputs for the indicator are impact and static pressure supplied by the main or the backup pitot tube.

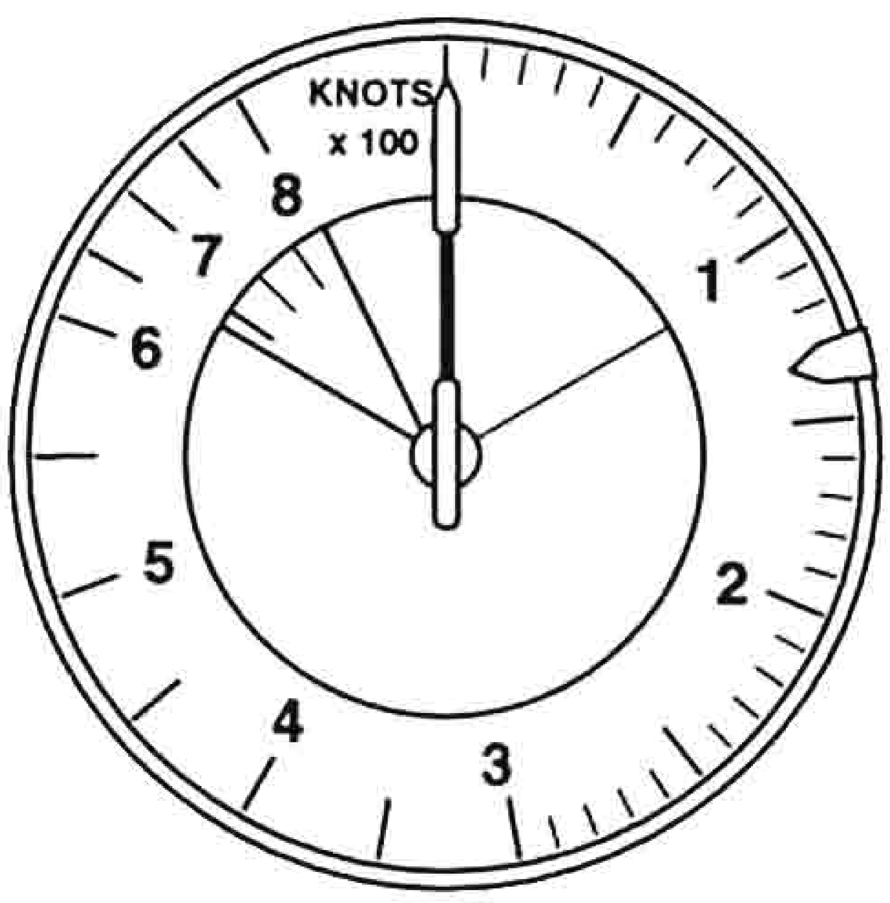


Figure 1-35

NOTE

Since the indicator is directly driven by the pneumatic outputs of one of the pitot tubes, the indicated mach number may differ from the real value by as much as M 0.05 due to non-linearities of those pitot tubes.

TAS INDICATOR

The TAS indicator UMS-2,5-2U provides a combined display of TAS and mach number. Electrical power is supplied from the ADC, since it is part of the system. The longer pointer rotates at the linear outer scale to indicate values between 100 and 1 400 kts. The shorter pointer traverses the inner mach number scale. The input signals for the indicator are supplied by the ADC and the ambient air temperature sensor.

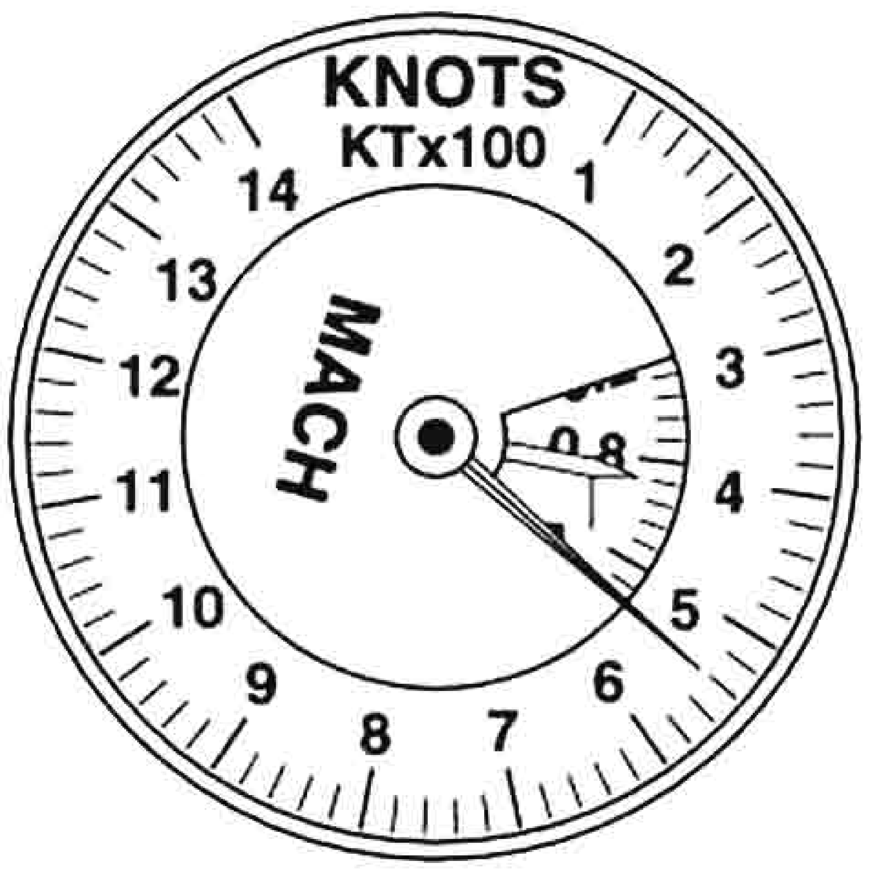


Figure 1-36

GT, R/C:

The TAS indicator UIS-1250 AE provides the display of true airspeed between 100 and 1 250 kts.

NOTE

Since the indicator is directly driven by the pneumatic outputs of one of the pitot tubes, the indicated mach number may differ from the real value due to non-linearities of those pitot tubes.

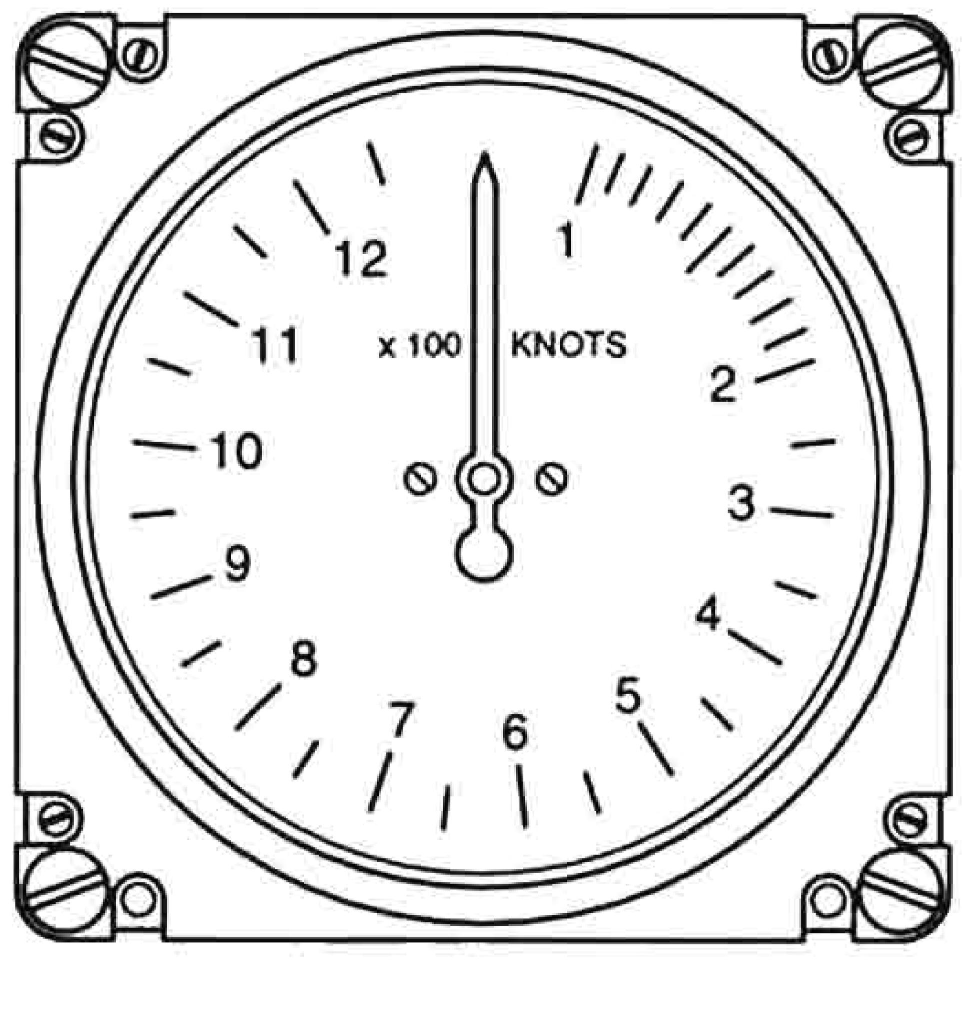


Figure 1-37

ALTIMETER

The altimeter is an electrically operated instrument, indicating from 0 to 100 000 ft. Electrical power is supplied by the ADC system, since the altimeter is an integral part of this system.

An adjustable barometric scale is provided so that the altimeter may be set at the corrected sea level pressure.

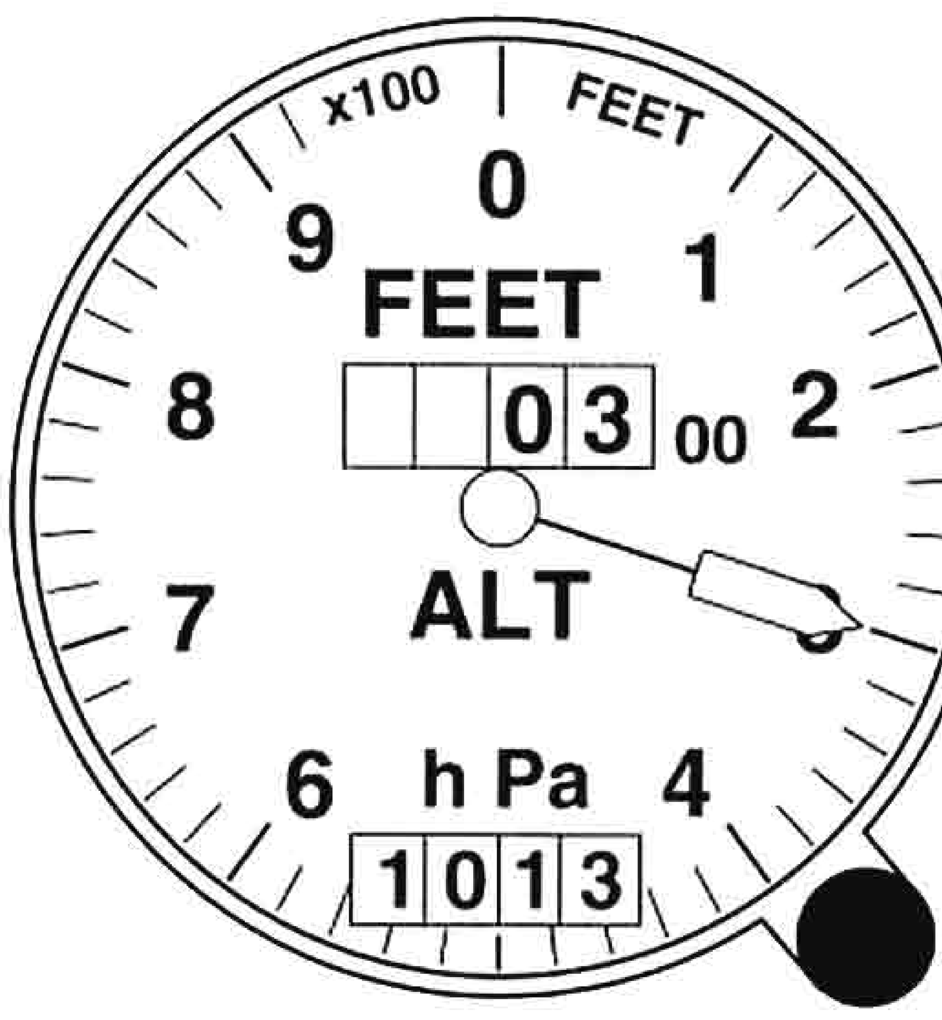


Figure 1-38

GT, R/C:

The altimeter in the rear cockpit is a pneumatically operated instrument, indicating from 0 to 100 000 ft. The altimeter incorporates a vibrator which reduces mechanical friction of gear trains and linkages of the mechanical assembly. The vibrator is powered by 28.5 VDC from the generator, or in case of failure by the batteries.

A barometric setting knob adjusts the barometric setting on the hPa counter of the altimeter between 700 and 1 080 hPa.

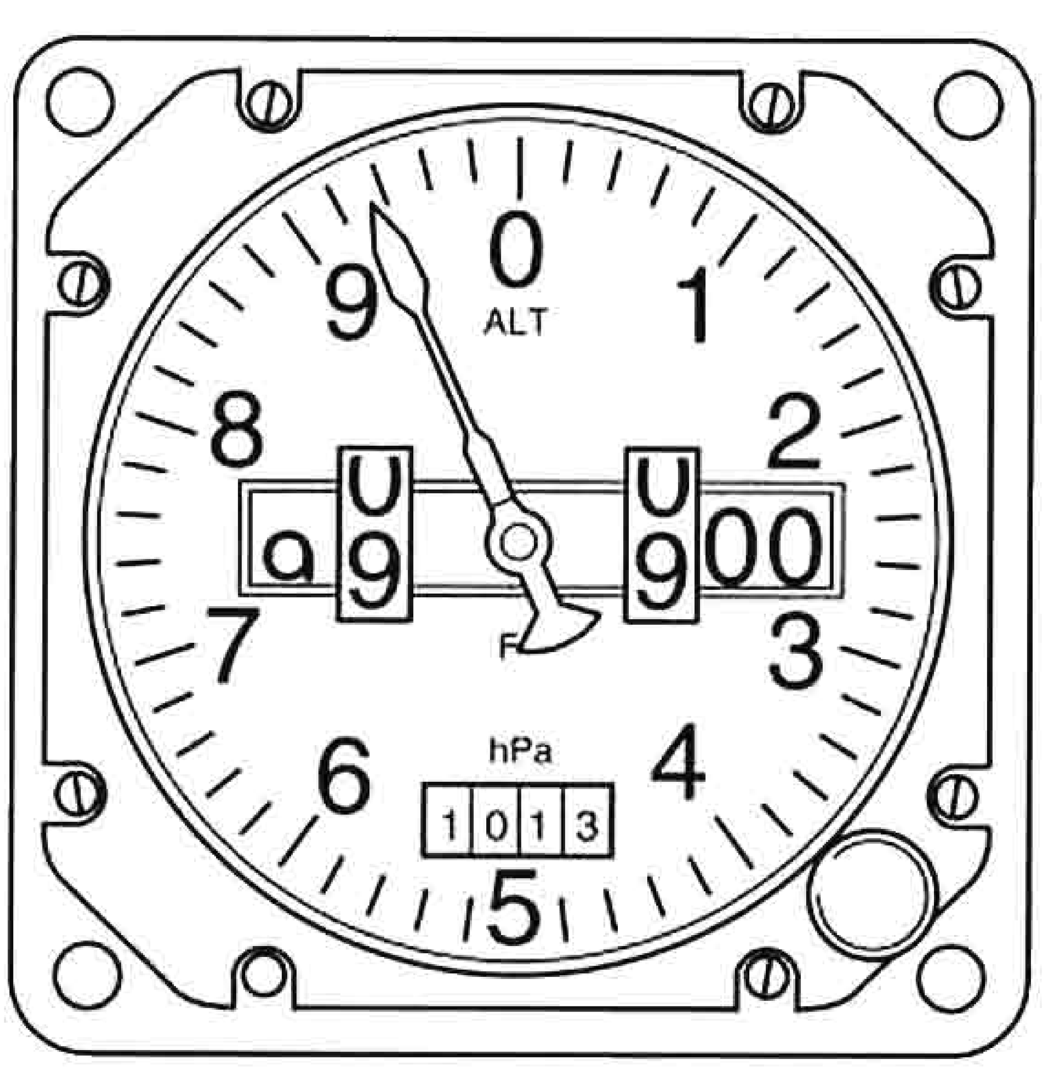


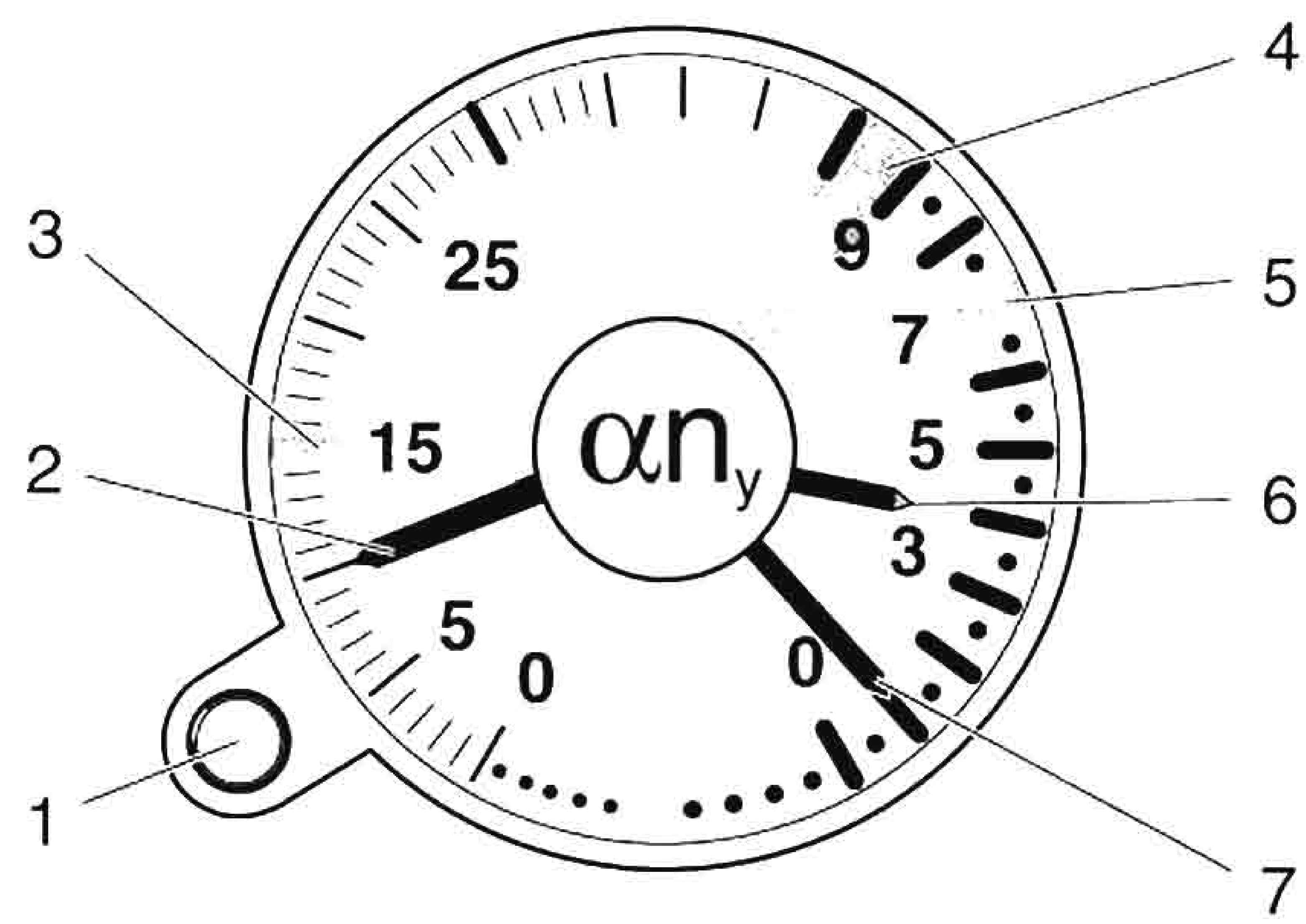
Figure 1-39

COMBINED AOA / G METER

The angle of attack and g meter is a combined instrument with two different indicating systems. Since it is part of the COC, it receives electrical power via this system.

The AOA pointer, moving along the left scale, is electrically connected to AOA probes located at the left and right forward section of the aircraft.

G-loads are indicated on the right scale by a main pointer indicating instantaneous g-loads and an index tab driven by the main pointer. The index tab remains on it's maximum indication until resetted. The g meter is electrically connected to an external g-sensing transducer.

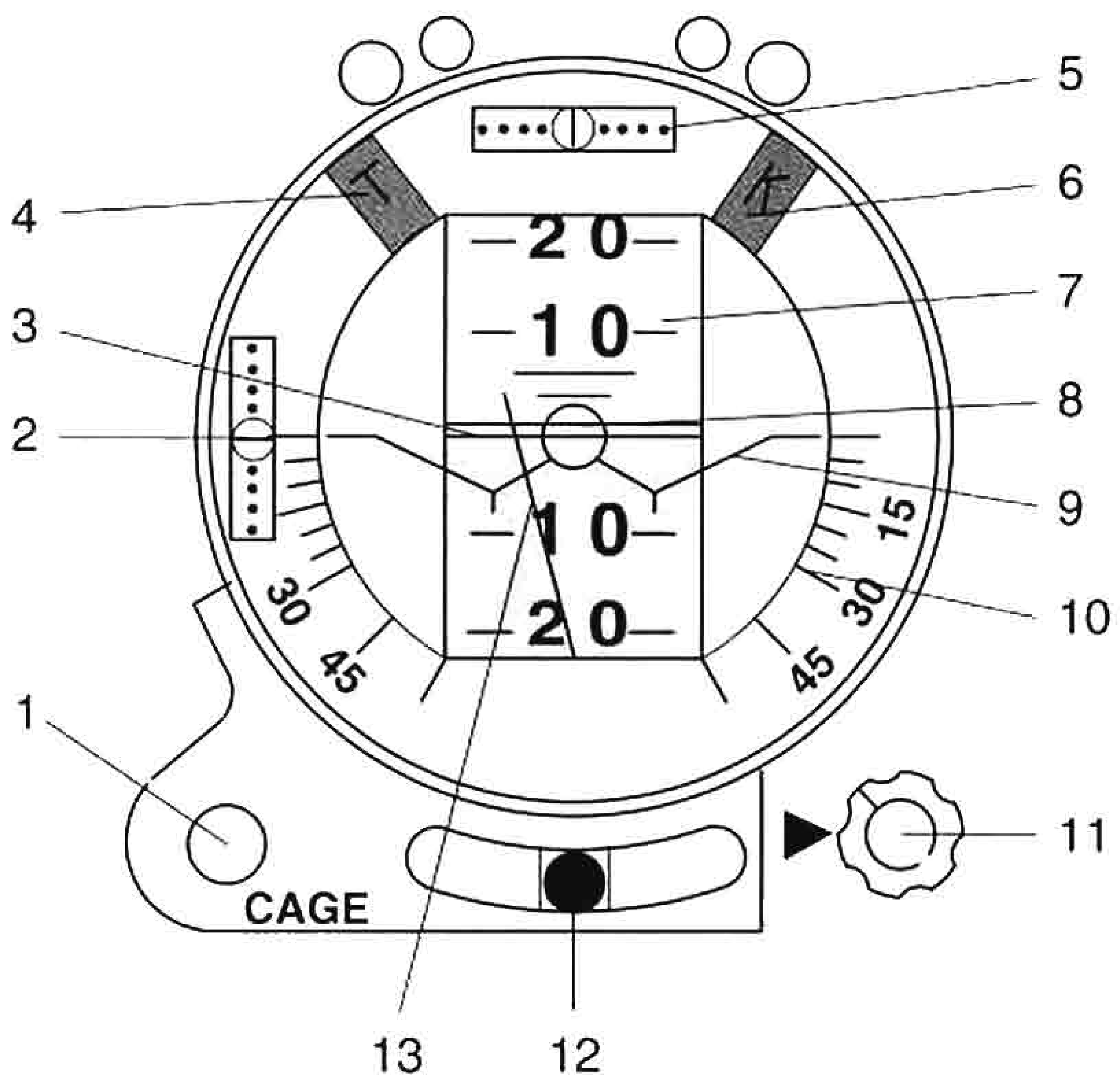


1. RESET BUTTON FOR G-INDEX TAB
2. AOA POINTER
3. 15° MARKER (RED)
4. RED REGION
5. 7 G-INDEX (RED)
6. MAX G-INDEX TAB
7. G POINTER

Figure 1-40

ATTITUDE DIRECTOR INDICATOR

The attitude director indicator (ADI) provides a pictorial display of the aircraft's attitude in pitch, roll and turn. It utilizes 36 VAC and 28.5 VDC received from the gyro system. A slip indicator is integrated at the bottom part of the instrument. Attitude is indicated by the aircraft symbol in relation to the horizon, pitch is indicated by the attitude sphere.



1. GYRO CAGE BUTTON / GYRO FAIL LIGHT
2. GLIDE SLOPE DEVIATION INDICATOR
3. PITCH STEERING BAR
4. PITCH OFF FLAG
5. COURSE DEVIATION INDICATOR
6. AZIMUTH OFF FLAG
7. ATTITUDE SPHERE
8. CENTER DOT
9. AIRCRAFT SYMBOL
10. BANK SCALE
11. AIRCRAFT SYMBOL SETTING KNOB
12. SLIP INDICATOR
13. COURSE STEERING BAR

Figure 1-41

Pitch and Bank

The pitch angle of the aircraft is displayed by the pitch scale on the spheroids surface and the center of the aircraft symbol. The vertical position of the aircraft symbol can be adjusted with the aircraft symbol setting knob.

The bank angle is displayed by the aircraft symbol rolling on the spheroids surface to indicate aircraft its bank on the bank scale.

Glide Slope Deviation Indicator

During the NAV mode RETURN, a 7° glide slope is displayed on the glide slope deviation indicator for a extended runway centerline interception at 2 000 ft AGL.

During the NAV mode MISSED APPROACH, deviation from an altitude of 2 000 ft AGL is displayed. In the AFCS mode LANDING, deviation from the ILS glide path is displayed.

Course Deviation Indicator

During the NAV modes RETURN and MISSED APPROACH, the index on the course deviation indicator provides steering for extended runway centerline intercept.

In the AFCS mode LANDING, deviation from the ILS course is displayed.

Command Steering

During an ILS approach, with the AFCS mode APPROACH selected, command steering information is available by the pitch steering bar and the bank steering bar. When the aircraft is exactly on the desired flight path, the intersection of the two bars coincide with the center dot. The bars will be parked in the center also when the system is not engaged. However, the pitch and azimuth OFF flags will be visible.

OFF Flags

Two red flags are incorporated in the upper left and right part of the instrument. The pitch OFF flag marked T and the azimuth OFF flag marked K will be visible when either channel of the ILS system is failed or not activated. As soon as the AFCS mode APPROACH is selected and no malfunctions exist, both OFF flags will disappear.

NOTE

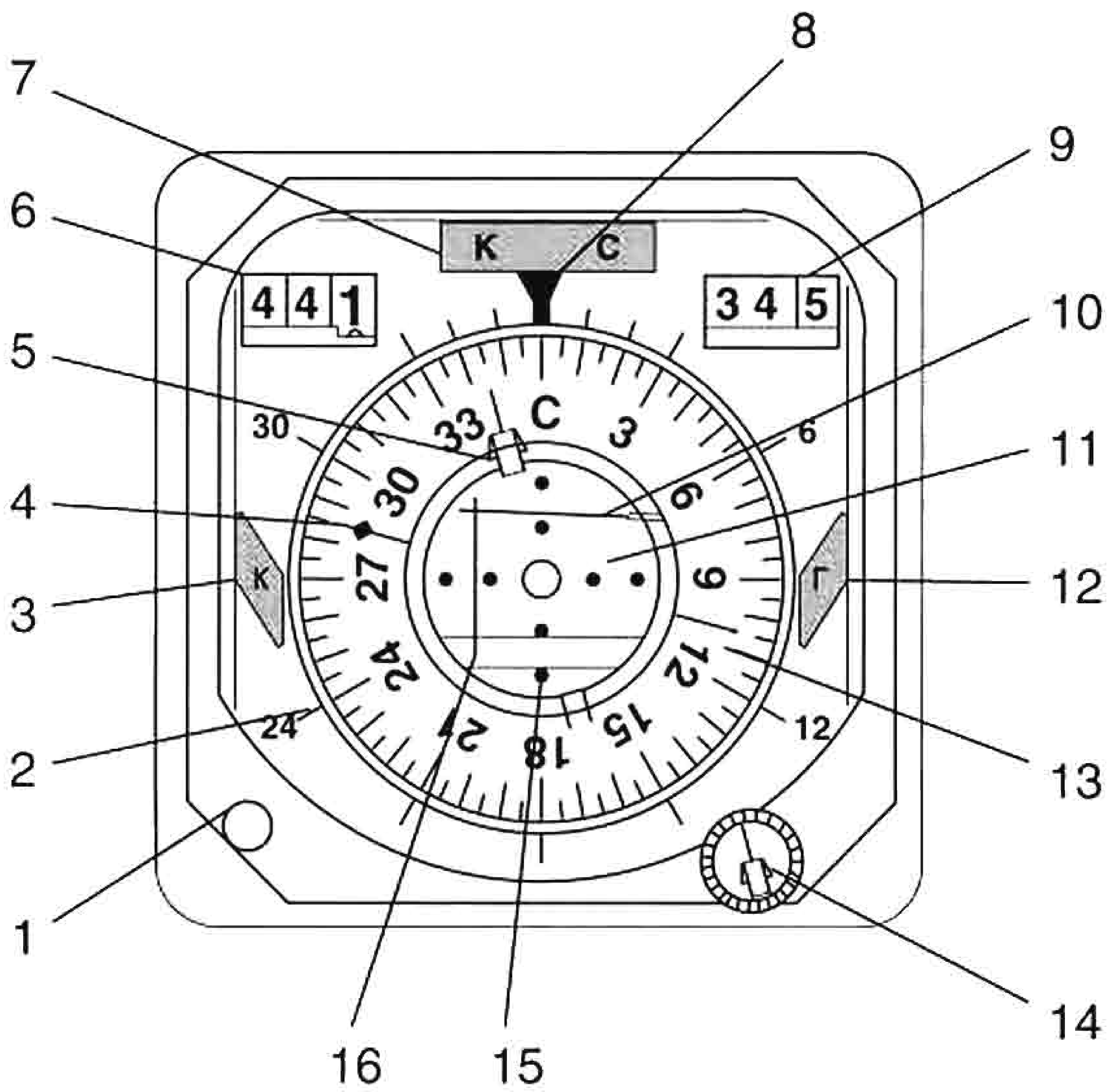
During an automatic landing approach, the appearance of either OFF flag initiates an automatic level off by the AFCS.

Slip Indicator

Aircraft slip is indicated by a ball inside a tubular case located at the lower part of the instrument face.

HORIZONTAL SITUATION INDICATOR

The horizontal situation indicator (HSI) provides a horizontal view of the aircraft with respect to the navigation situation. The compass card rotates so that the aircraft heading is always under the course index. Three OFF flags provide warnings for course and glide path indicator failures and navigation or gyro / platform malfunctions. Depending on the selection on the NAV panel, indications on the HSI vary. The HSI utilizes 36 VAC power.



1. TEST BUTTON
2. FIXED COMPASS CARD
3. ILS COURSE OFF FLAG
4. BEARING POINTER
   BEARING (BRG) POINTER (YELLOW)
5. COURSE POINTER (WHITE)
6. RANGE (RNG) COUNTER
7. HSI OFF FLAG
8. COURSE INDEX
9. BRG COUNTER
10. ILS GLIDE SLOPE INDICATOR
11. ILS COURSE DEVIATION SCALE
12. ILS GLIDE SLOPE OFF FLAG
13. COMPASS CARD
14. COURSE SELECTOR KNOB
15. ILS GLIDE SLOPE DEVIATION SCALE
16. ILS COURSE DEVIATION INDICATOR

Figure 1-42

CLOCK

A mechanical clock allows determination of normal daytime, elapsed (mission) time and has a stopwatch feature. The indications as mentioned may be read on three individual scales:

- Normal daytime is displayed on the outer scale with an hour and a minute pointer.
- Elapsed time is displayed in hours and minutes on the upper inner scale.
- Elapsed time is displayed in minutes and seconds (stopwatch) on the lower inner scale.

A red winding and setting knob is provided on the left lower corner of the clock. Rotating the knob counter-clockwise winds up the clock. Pulling the knob permits setting the clock. Pushing the knob starts the small elapsed time scale on the upper portion of the face, a small status indicator window within the scale changes color from white to red. Pushing the knob again stops the small clock, the indicator turns red / white. Pushing the knob a third time resets the elapsed time and the status indicator turns white.

A setting knob on the lower right corner of the clock is used to start and stop the seconds pointer of the normal daytime scale and to operate the stopwatch on the lower portion of the clock face.

Rotating the knob clockwise stops the seconds pointer, rotating the knob counter-clockwise starts it again. Pushing the knob starts the stopwatch located at the lower part of the face, pushing it again stops it and pushing a third time resets both pointers to the zero position.

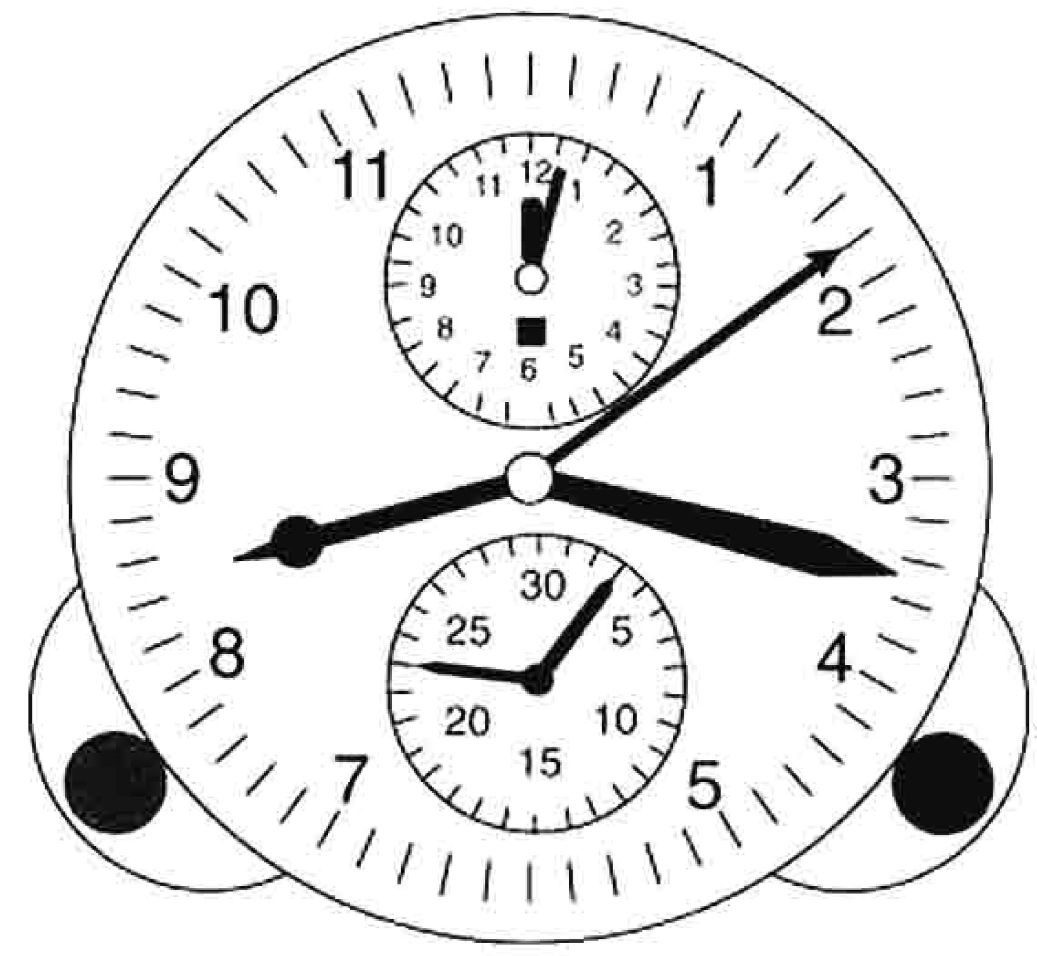


Figure 1-42A

HSI INDICATION

MODE SETTING	COURSE POINTER	BEARING POINTER	BRG COUNTER	RANGE COUNTER	COURSE INDEX	ILS COURSE INDICATOR	ILS GLIDE SLOPE INDICATOR
NAV	Course to selected navigation point	Bearing to selected TACAN / NDB	Course to selected navigation point	Distance to selected navigation point	Aircraft HDG	-	-
NAV RETURN	An offset point to intercept final	Bearing to TACAN / NDB	Aerodrome reference point	Distance to aerodrome	HDG	-	-
NAV RETURN without TACAN update	Course to the aerodrome reference point	Unreliable	Course to the aerodrome reference point	Distance to the aerodrome reference point	HDG	-	-
NAV landing approach	Final course	Bearing to selected NDB	Final course	Distance to touchdown	HDG	Deviation from final course, max deviation is indicated 0.5° (4 dots) on the course deviation scale	Glide slope deviation, max deviation is indicated 0.5° (4 dots) on the glide slope deviation scale
NAV missed approach	Course to an offset point to intercept final approach	Bearing to TACAN / NDB	Course to the aerodrome reference point	Distance to the aerodrome reference point	HDG	-	-
MANUAL TACAN / RSBN for non programmed aerodrome	Course set by the course selector knob	Bearing to selected station	Course selected	Distance to selected station	HDG	-	-
BIT initiated by pressing the TEST button	Will rotate 20° ±5° CCW	-	-	Indicates 43 ±2.5 NM	Will rotate 20° ±5°	-	-

Figure 1-43

VERTICAL VELOCITY INDICATOR

The vertical velocity indicator (VVI) indicates the rate of climb or descent of the aircraft. The indicator is connected to the static pressure system and actuation of the pointer is controlled by the rate of change of the atmospheric pressure. It can register a rate of gain or loss of altitude which would be too small to cause a noticeable change in the altimeter reading.

TURN AND SLIP INDICATOR

A turn and slip indicator is incorporated in the VVI. The turn needle indicates direction of turn but does not provide accurate turn rate. The instrument receives 3 phases 36 VAC from the gyro system.

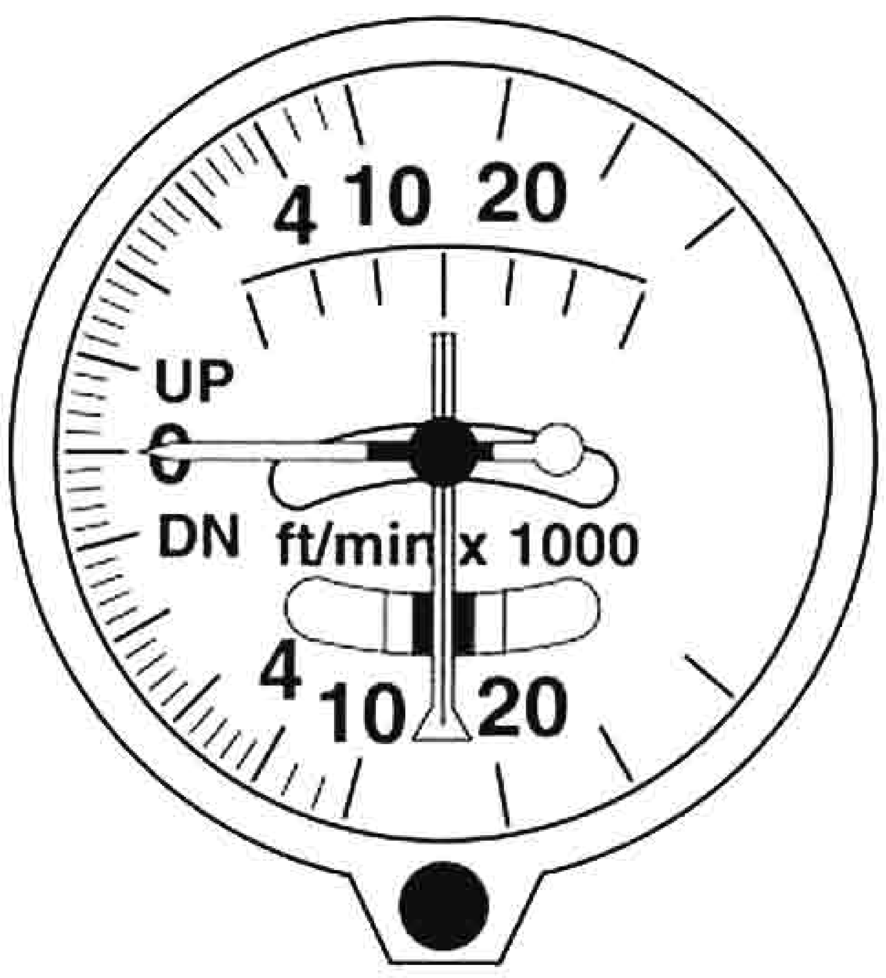


Figure 1-44

RADAR ALTIMETER

The continuous wave radar altimeter (RAD ALT) measures height above surface. It supplies information to the avionics equipment and the radar altimeter indicator. The height marker may be set to the desired minimum height. If the aircraft is below this height, the radar altimeter forwards inputs to AFCS, AEKRAN, VIWAS and the RAD ALT indicator warning light. Accuracy of the min. height selected is 1.5 ft from 0 to 60 ft and ±3 % above 60 ft.

The radar altimeter is switched on with the ACFT SYS switch. After the warm-up period, the indicator OFF flag disappears and an altitude of 0 ±3 ft is indicated.

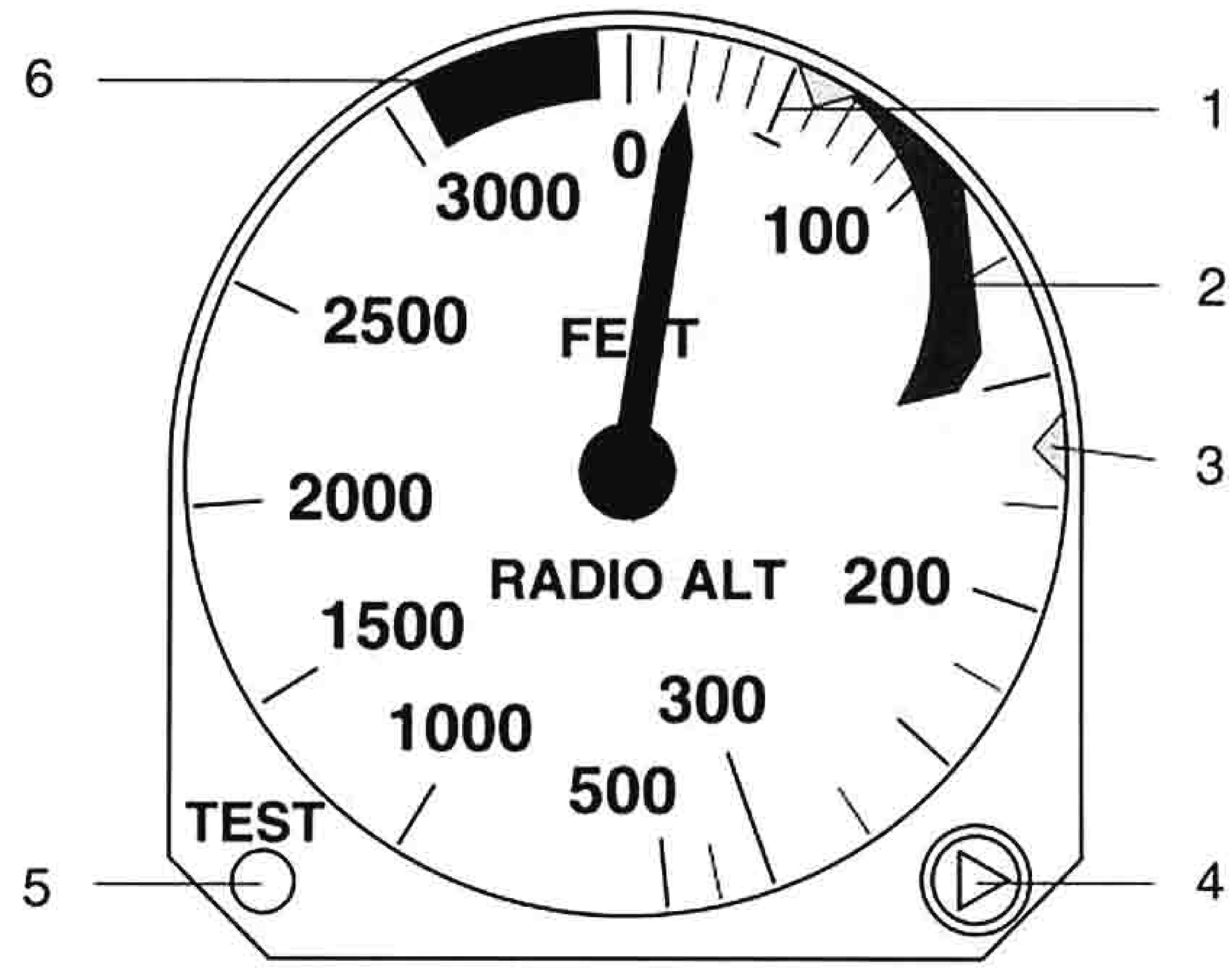
The RAD ALT contains continuous BIT. It is powered by 28.5 VDC and 115 VAC. If the DC generator fails, power is supplied from the aircraft batteries and if the AC generator fails, from the PTO.

The system provides height information from 0 to 3 000 ft AGL within bank angles up to 15°. The system accuracy is:

±3 ft from 0 to 30 ft
±10 % above 30 ft.

WARNING

At bank angles above 30° or over rough terrain, RAD ALT may give false height information. At bank angles from 15° to 30° system accuracy decreases, above 30° bank it should be considered unreliable.



1. 45 FT TEST MARK
2. OFF FLAG
3. MINIMUM HEIGHT SET MARKER
4. MINIMUM HEIGHT SET / WARNING LIGHT
5. TEST BUTTON
6. BLACK SECTOR

Figure 1-45

At heights above 3 000 ft an OFF flag appears on the indicator and the pointer rotates to the black sector. Identical indications occur with a malfunction.

A test button is provided to check the instrument. When the button is pressed, the pointer moves to the test mark at 45 ft.

INDICATIONS AND WARNINGS

	INDICATION	FAULT / EFFECT
AEKRAN	ALT ALERT	Descent below set minimum height.
VIWAS	"GEFÄHRLICHE HÖHE"	

HUD / HDD

The HUD / HDD consists of the head up display (HUD), the head down display (HDD), units for image processing, synchronization and power supply. The system is powered by 28.5 VDC and 115 / 200 VAC power.

It displays information originating from the navigation system and the fire control system (FCS). Signals from the AFCS, AOA and side slip vanes, and from the radar altimeter are processed.

HUD

The HUD projects information in symbolic and numeric form into the pilot's field of view. This source of information provides steering commands in the navigation mode and constitutes the primary source of information during attack phase.

The image processing unit receives inputs from the navigation system, the FCS and additionally from the AFCS, AOA and sideslip vanes and from the radar altimeter. It generates symbols which are displayed on a cathode ray tube (CRT) and projected into the pilot's line of sight by means of a collimator and a combining glass. The collimator focuses the HUD picture to infinity.

The combining glass projects the symbology within a space of 13° in azimuth and 18° in elevation resulting in a circular field of view of 24°. A light filter may be raised to ensure readability of the HUD display against a bright background.

HUD CONTROLS

Filter Operating Handle

The filter operating handle on top of the left mirror unit of the helmet mounted sight is used to erect the light filter to the vertical position.

BRIGHTNESS Control Knob

The BRIGHT knob is used to adjust the brightness of the HUD.
In addition, a light dependent resistor (LDR) on the front side of the HUD display unit automatically adjusts the brightness of the display depending on ambient light conditions. The brightness of the image as seen by the pilot is the result of the setting of the BRIGHT knob and the intensity of ambient light.

HUD Selector

The HUD selector has three positions:

NIGHT	The color of the HUD image is amber.
DAY	The color of the HUD image is green.
RETICLE	The HUD image is blanked off, a fixed reticle is displayed for A/A weapon employment.

TEST Button

A TEST button is provided for equipment test. When pressing the button, the boresight cross appears in a square on the HDD, on the HUD, identical crosses appear in the center of each quadrant of the display additionally, indicating system readiness.

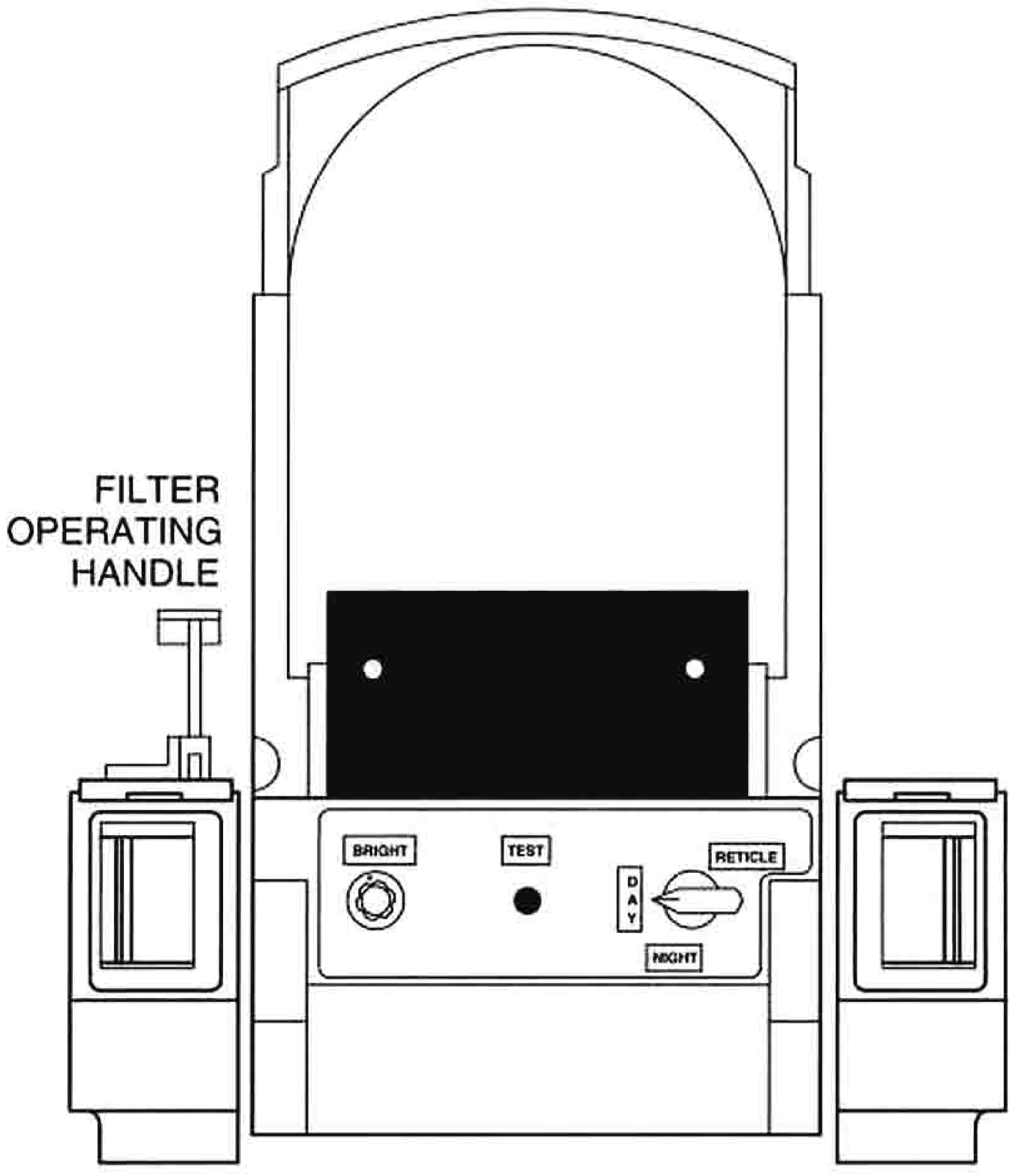
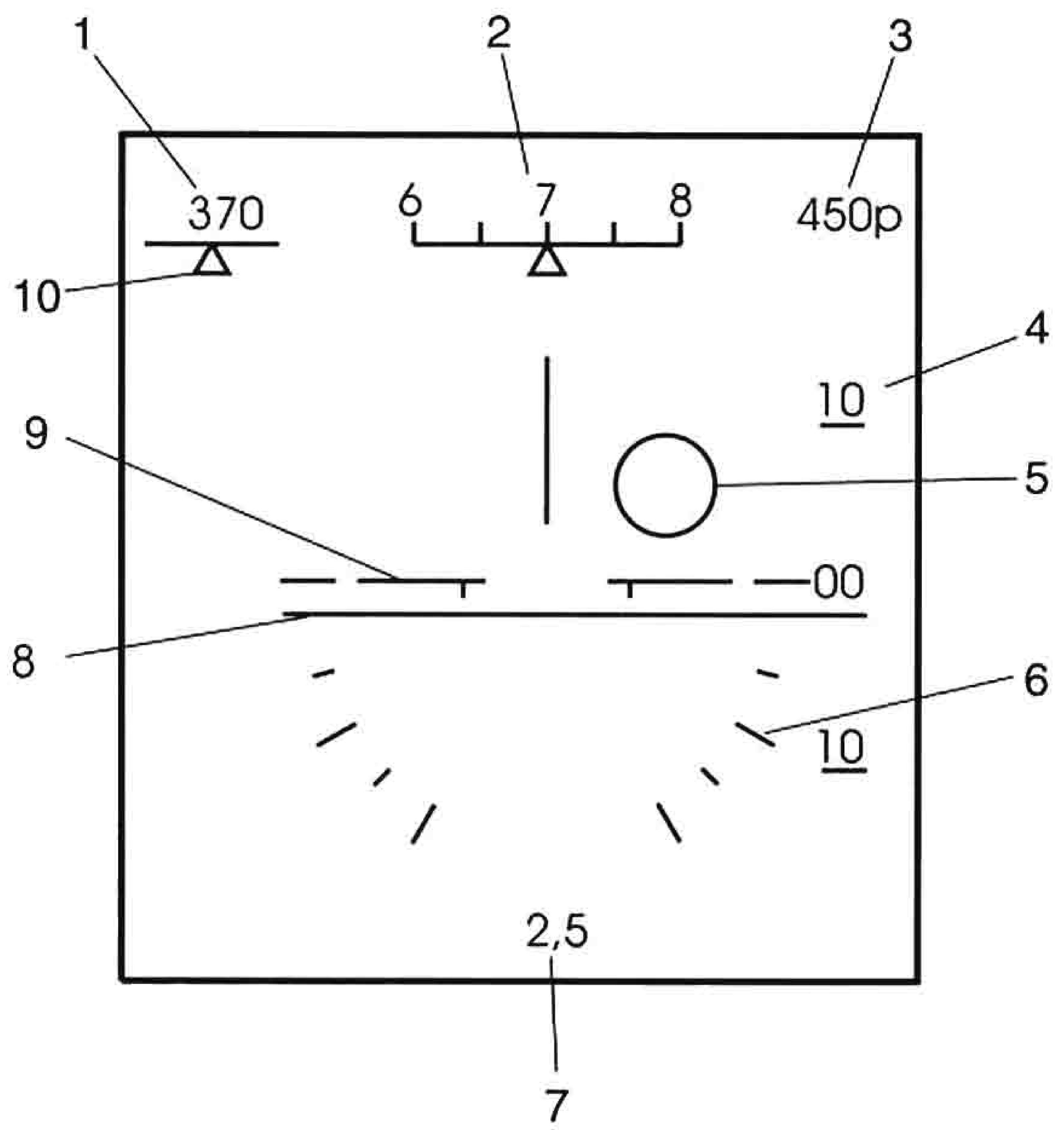


Figure 1-46

HUD Display

Following navigational symbols can be displayed on the HUD.



1. INDICATED AIRSPEED
2. HEADING REFERENCE
3, PRESS ALT / RAD ALT
4. PITCH ANGLE
5. STEERING CIRCLE
6. BANK ANGLE
7. NAV RANGE
8. ARTIFICIAL HORIZON
9. AIRCRAFT SYMBOL
10. IAS TREND INDEXER

Figure 1-47

HUD Weapon Employment Display

For weapon employment see GAF T.O. 1F-MIG29-34-1.

HDD

The HDD is a TV monitor on the right side of the instrument panel. It displays essentially the same picture as the HUD. A light dependent resistor (LDR) in the lower right corner of the front panel automatically adjusts the brightness of the display depending on ambient light conditions. In combination with the setting of the CRT brightness control knob, it renders the displayed information readable even in direct sunlight.

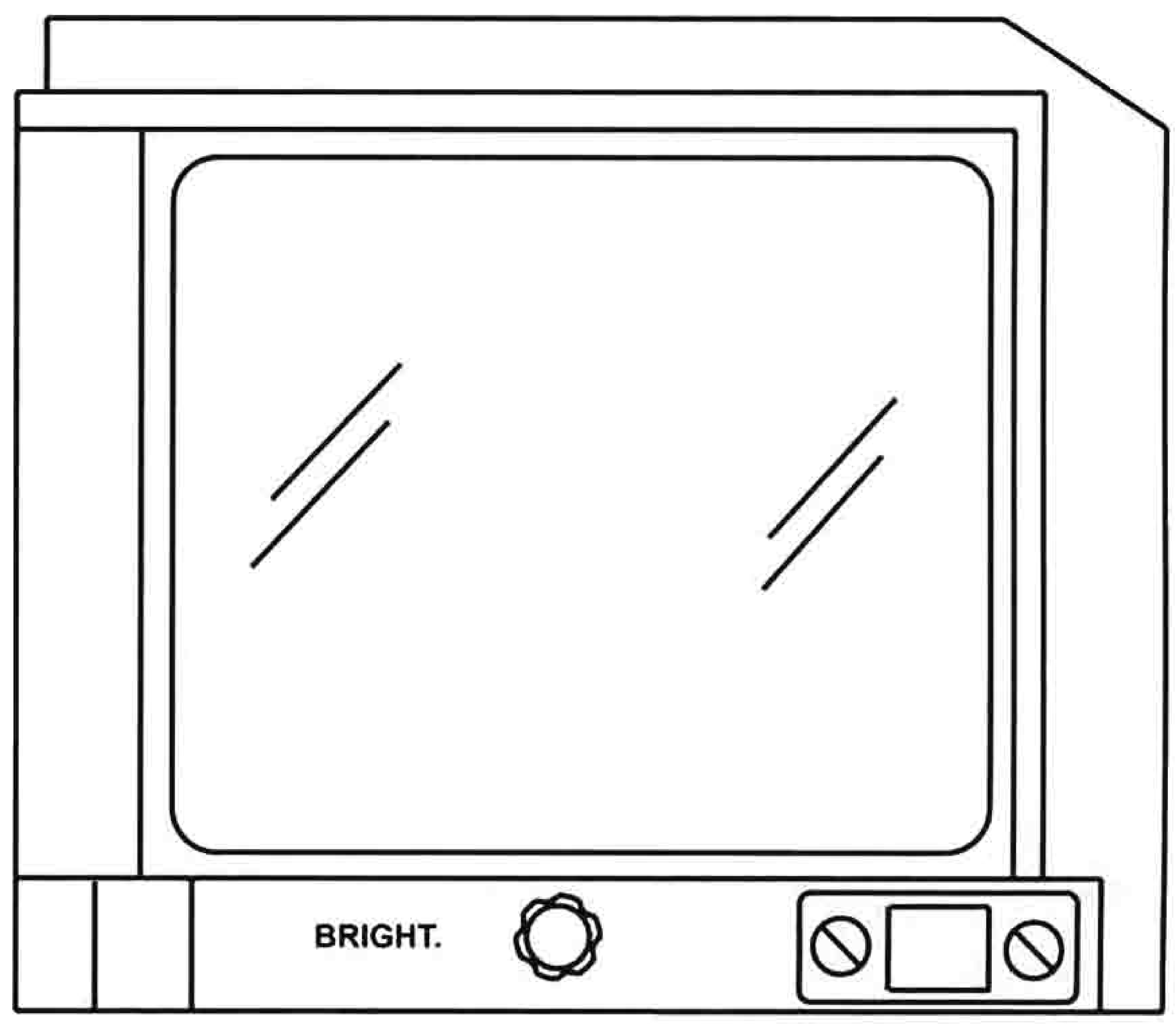


Figure 1-48

CANOPY

The canopy consists of a rigid curved front section, the windshield, fixed to the fuselage and a section that can be raised which is hinged aft.

CANOPY OPERATION

Normal canopy operation is controlled by the canopy control handle and powered by the pneumatic power supply system. Refer to figure 1-49.

The pneumatic canopy operation system ensures:

- Raising and lowering of the canopy corresponding to the canopy control handle position.
- Downlock of the canopy in the closed position.
- Sealing of the canopy.

INTERNAL CANOPY CONTROL HANDLE

The internal control handle has three position detents:

Open, taxi, and closed.

To open the canopy, first set the handle in the taxi position and then in the open position.

When the handle is in the taxi position, the seal is deflated and the canopy will be released and lifted approximately two inches above the cockpit rim.

When moving the handle further into the open position, the canopy will be raised and held by a pneumatic actuator.

CAUTION

- Taxiing with the canopy in the open position is prohibited.
- Max speed for taxiing is 16 kts with the canopy in taxi position.

GT:

Identical internal control handles are located in both cockpits. The rear cockpit canopy control handle is safety wired to the close position, since normal canopy operation is performed from the front cockpit.

Opening the rear cockpit canopy control handle to the taxi or open position overrides the front cockpit canopy control handle and positions the front cockpit canopy control handle accordingly.

When the canopy control handle in either cockpit is moved from the open position towards the taxi or the closed position, the other handle is automatically positioned accordingly. The canopy is lowered to the taxi position or the closed position respectively.

EXTERNAL CANOPY OPERATING HANDLE

The external canopy operating handle is mechanically linked to the internal handle and is located on the LH front fuselage. It is used to open or close the canopy from the outside.

CANOPY OPERATION WITHOUT PNEUMATIC PRESSURE

To open the canopy without pneumatic pressure available, the control handle has to be set in the open position (to disengage the locks), the canopy has to be raised manually and held in the open position with the canopy retaining rod.

CANOPY JETTISON

The canopy emergency jettison system provides release and separation from the cockpit:

- If the emergency jettison handle on the right cockpit sill is pulled.
- Automatically, if ejection is initiated.

A pyro-mechanical system is used to jettison the canopy. Explosive cartridges are used to open the locks.

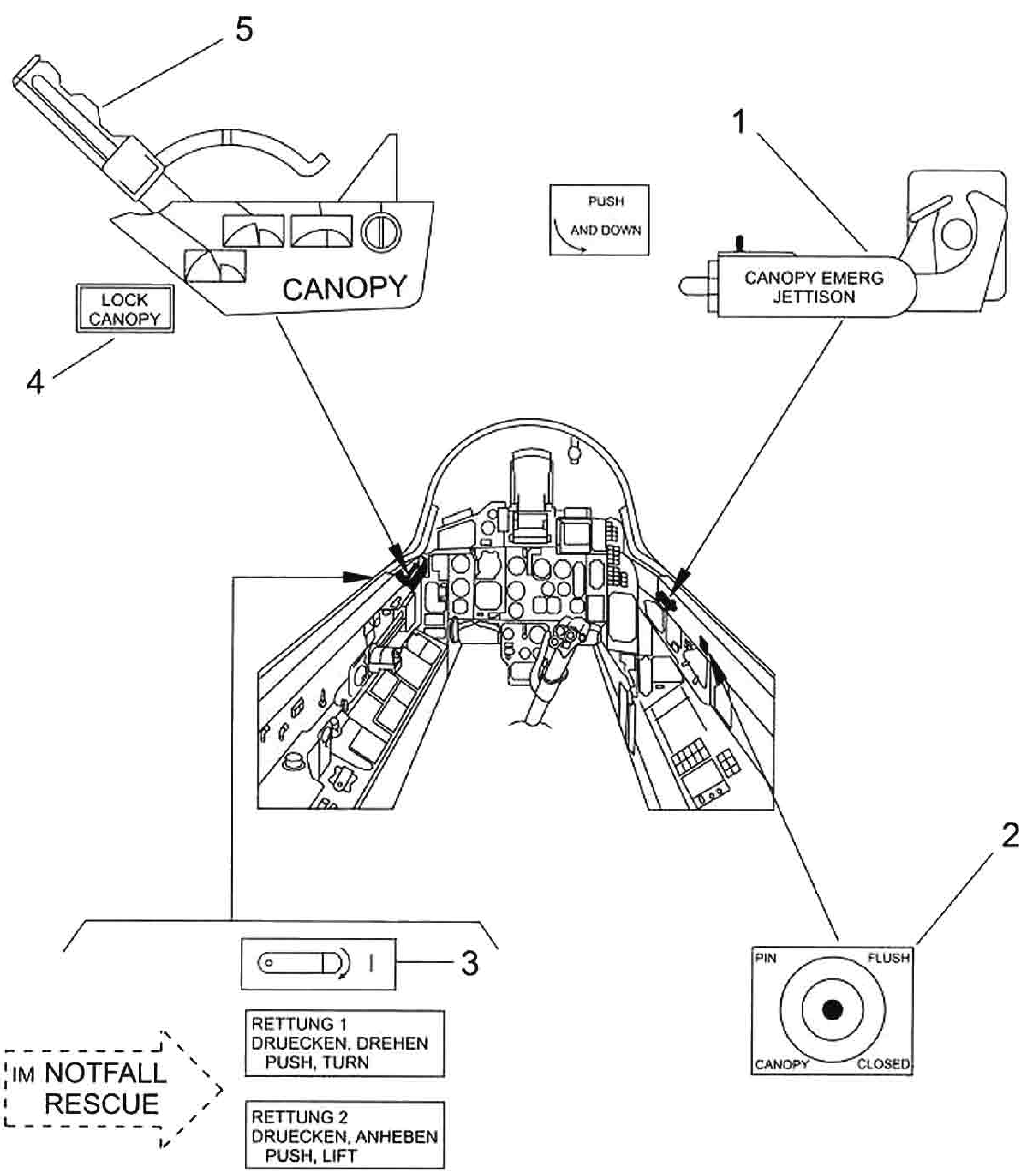
GT:

Canopy emergency jettison handles are located in both cockpits on the right cockpit sill. Pulling either handle jettisons the canopy.

INDICATIONS AND WARNINGS

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
CANOPY WARNING LIGHT	LOCK CANOPY	Canopy downlock failure.
CANOPY LOCK PIN	Pin protruding	
AEKRAN	LOCK CANOPY	
VIWAS	"KABINENDACH SCHLIESSEN" (message will be paged twice)	

CANOPY INTERNAL AND EXTERNAL CONTROLS



1. INTERNAL CANOPY EMERGENCY JETTISON HANDLE
2. CANOPY CLOSED CONTROL PIN
3. EXTERNAL CANOPY OPERATING HANDLE
4. LOCK CANOPY CONTROL LIGHT
5. INTERNAL CANOPY CONTROL HANDLE

Figure 1-49

EJECTION SEAT SYSTEM

The K-36 DM ejection seat system provides the pilot with a safe escape from the aircraft under various combinations of aircraft altitude, speed, attitude and flight path.

The seat is propelled from the aircraft by a cartridge-operated twin barrel ejection gun assisted by a rocket motor, both located at the back of the seat. The ejection system is designed to function at all altitudes at airspeeds up to 700 kts. However, during ground operation, a minimum of 40 kts (80 kts for the trainer version) is required for safe canopy separation.

Pulling the ejection handle initiates the ejection sequence, causing the canopy to be jettisoned and the ejection gun to fire. Caution should be used to maintain a continuous pull until full travel of the ejection handle is reached and the seat fires.

The ejection sequence continuous until a normal parachute descent of the occupant is accomplished. After the initial firing of the seat, seat operation is completely automatic and requires no additional action by the occupant.

NOTE

Canopy jettison malfunctions will not interfere with the seat firing system. Should the canopy fail to jettison after ejection has been initiated, the seat fires through the canopy after a delay of 1 sec.

SAFETY FEATURES

WARNING

The escape system is a potential source of danger and inadvertent operation may cause fatal injuries. Upon completion of the flight, the pilot must ensure that the seat is in the 'safe for parking' condition.

Safety pins are provided to various components of the escape system to prevent inadvertent initiation. Refer to figure FO-18.

EJECTION SEQUENCE

Ejection is initiated by pulling the seat firing handle. The sequence is electrically controlled until firing of the ejection gun. A mechanical backup provides fail-safe operation. As the main cartridge of the pyro-mechanical system is fired, gases are ducted to the canopy lock-down mechanism to jettison the canopy, to retract and lock the shoulder harness, and lap belt by means of the retraction units, and to activate both leg raisers and arm protectors.

Simultaneously, a backup system is activated to open the canopy locks after 0.5 seconds in case of a main breech unit failure. It allows the seat to fire through the canopy after another 0.5 seconds.

As the canopy is jettisoned, the canopy firing cable is pulled, allowing the twin barrel ejection gun to fire and to accelerate to at least 13.6 m/s. As the seat rises along the cockpit rails, the emergency oxygen supply is tripped, a body windshield is activated above 485 kts, and the leg restraint lines are retracted. The rocket motor fires to propel the seat to a greater height.

The seat is stabilized and decelerated by two rotating drogues on telescopic struts during descent through the upper atmosphere with the occupant securely restrained in the seat.

Automatic operation of the delay-release-mechanism occurs after reaching the barostat altitude (16 000 ft) or, in ejections below this altitude, when the seat is decelerated to parachute-opening speed.

The headrest / parachute container is fired from the seat to pull out the parachute. The recoil produced is also used during the process of man / seat separation.

EJECTION SEAT

The ejection seat is mounted on the guide rails and the telescopic ejection gun. The firing handle is connected to an electromagnetic ignitor unit which starts the ejection sequence.

Electromagnetic ignitor cartridges are installed for initiating the ejection gun, activating the retraction units and raising the windshield.
Percussion cartridges are installed for the ejection gun, the drogue gun and the firing mechanism of the rocket motor.

EJECTION SEQUENCE

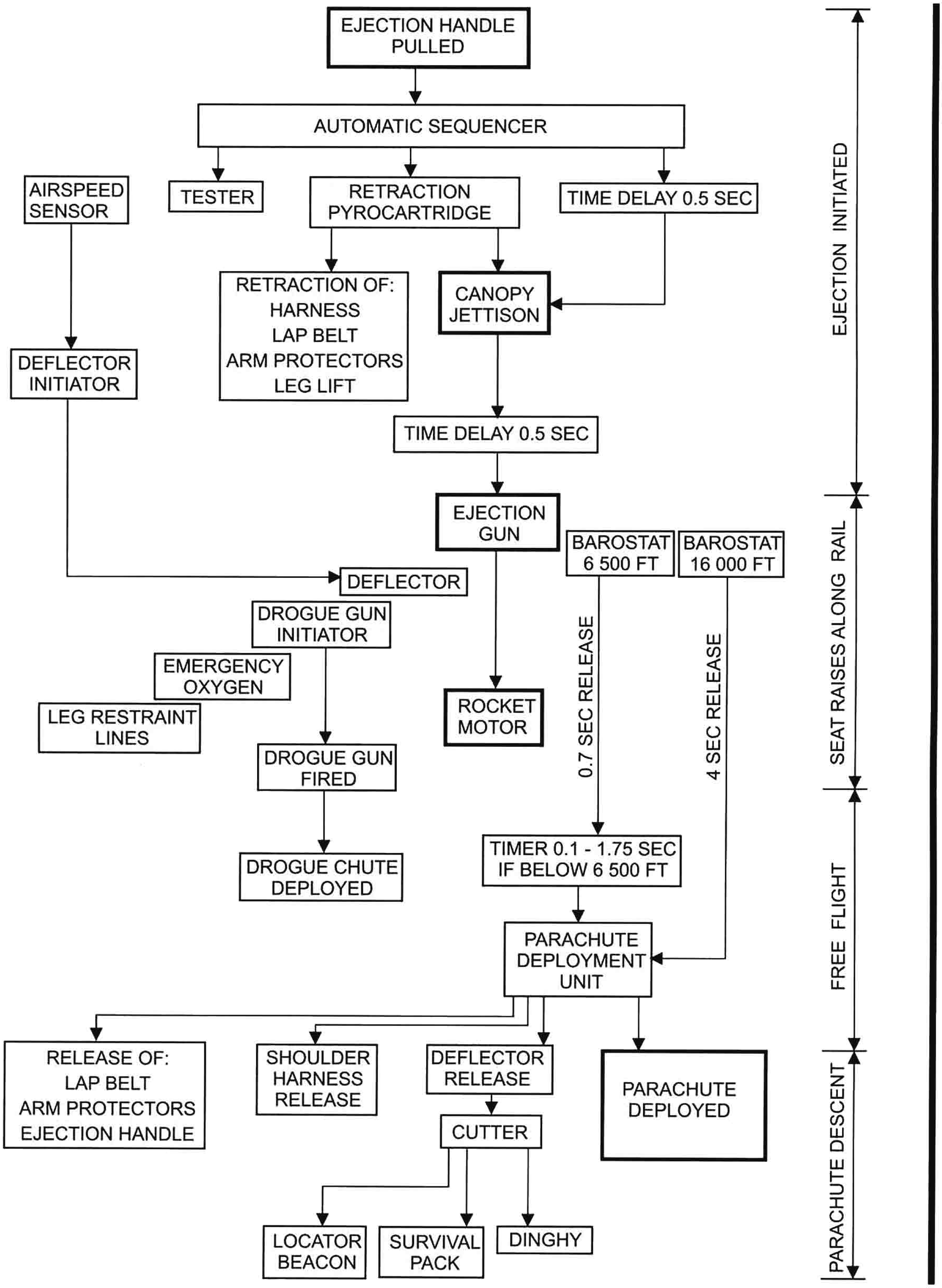


Figure 1-50

EJECTION GUN

The ejection gun provides the initial power for seat ejection by means of a single percussion-fired cartridge. The gun consists of three major assemblies, which are the breech firing mechanism, a fillet transition piece and a telescopic launch tube.

The gun assembly is mounted to the seat structure, except for the inner barrel of the launch tube, which is mounted to the bulkhead of the cockpit.

After ignition, the gun develops thrust for 0.2 sec. As the seat raises along the guide rails, it extends an initiator cable which fires the rocket motor when the seat has been raised between 104.5 and 107.5 cm.

ROCKET MOTOR

The thrust of the ejection gun will be sustained by the rocket motor, located under the seat pan, and is ignited as the seat leaves the ejection gun. A static line, incorporated in the rocket firing unit, cocks and triggers a firing pin to fire the ignition cartridge.

The gas pressure, generated by this cartridge ignites the rocket propellant. The rocket motor develops a thrust of 3 300 kg.

DROGUE GUN

Two drogue arms are mounted on the right and left side of the head rest. They consist of a firing mechanism and a telescopic rod with built-in drogue chute.

The unit is triggered as the ejection seat raises along the guide rails. The gas pressure of the cartridge extends each rod aft at an angle of 15°, and deploys the drogue chute of 0.06 m². The drogue chutes are ribbon-type chutes with opposite direction of rotation, thus actively stabilizing the seat.

DELAY UNIT

Two independent time release mechanisms are installed to the right and left side of the main beam assembly. Their function is to delay deployment of the personal parachute and separation of the occupant from the seat until it has descended from high altitude and / or slowed enough to prevent excessive opening shock of the personal parachute.

A barostat assembly prevents operation of the time delay above a preset altitude. The barostat of one delay unit is set to an altitude of 16 000 ft MSL, 19 500 ft when flying above 13 000 ft mountainous terrain. The associated time release unit is set to 4 sec.

The barostat of the second delay unit is set to an altitude of 6 500 ft MSL, 10 000 ft when flying above 18 000 ft mountainous terrain. The associated time release unit is set to 0.7 sec.

Although both time delay units operate independently, each unit triggers the parachute deployment mechanism of both units.

At altitudes below 6 500 ft, the time release mechanism of the second delay unit is adjusted for the airspeed at time of ejection. Depending upon the speed, the time is readjusted from 0.1 sec at 375 KTAS up to 1.75 sec at 750 KTAS.

HARNESS POWERED RETRACTION UNIT

The ejection seat contains a powered inertia lock which provides a velocity (g-sensing) system (inertia lock) and a power retraction system. The inertia lock provides safe restraint during violent aircraft maneuvers. Restraint is accomplished by a g-sensing mechanism functioning in accordance with acceleration (2 g). Manual locking of the inertia reel lock can be accomplished by the shoulder harness release handle on the left forward side of the seat bucket. The powered retraction system provides automatic retraction of the shoulder harness for ejection. The device is gas-powered and functions only when the ejection handle is pulled.

LAP RETRACTION UNIT

A lap arrestment unit provides the pilot with a safe hold in the seat during normal flight. Manual adjustment can be accomplished by the lap adjustment lever on the right side of the seat bucket. A gas-powered retraction system is automatically activated when the ejection handle is pulled.

ARM PROTECTION SYSTEM

To prevent flailing injuries to the arms during ejection, an arm protection system has been fitted. The system consists of two metal blades mounted to the side of the backrest. As ejection is initiated, the arm protectors are rotated down to a horizontal position. The device is gas-powered and operated simultaneously with the shoulder harness and the lap retraction.

WINDSHIELD

A windshield is mounted to the front of the seat bucket to protect the pilot from the windblast during high speed ejection. The unit consists of two telescopic rods which extend and raise a shield of Kopran ribbons in front of the pilot. The system is designed to extend even when one of the telescopic rods fails.

The windshield is activated above 485 KTAS only and is raised by an electro-pyrotechnic charge.

LEG RESTRAINT LINES

The leg restraint lines are routed along the cockpit sidewall, the instrument panel and the control stick casing. The restraint lines are fixed into position with clips.

Paddings are fixed to the section of the restraints which actually retract the legs.

When the seat is ejected, the occupants legs are firmly pulled against the seat bucket. Simultaneously the thighs are lifted to optimize body position during ejection.

PERSONAL PARACHUTE

A 60 m^2 personal parachute is packed into the headrest container, located on top of the seat beam.

Upon release from the delay units, two cartridges are fired to separate the container and pull out the parachute.

The gases of the cartridge are also used to operate cutters for simultaneous man / seat separation and activation of the emergency locator beacon and the survival pack.

The personal parachute is connected to both shoulder harnesses by canopy quick release connectors. The quick release connectors can be opened by pressing the latches on both sides simultaneously after a safety guard, located between these latches has been pulled forward. The purpose of the safety guard is to prevent inadvertent operation of the quick disconnect.

EMERGENCY OXYGEN SYSTEM

An emergency oxygen bottle is installed in the ejection seat bucket. Activation of the oxygen bottle is accomplished automatically upon ejection. The emergency oxygen can be activated manually by pulling upon the emergency knob (red mushroom).

The pressure bottle contains 0.7 l of compressed oxygen at a pressure of 180 kp/cm^2, indicated on the pressure gage.

The bottle supplies 100 % oxygen for about 6 min during emergency descent, 3 to 4 min during high altitude ejection and 3 min at low altitude.

EMERGENCY OXYGEN KNOB (Red Mushroom)

The emergency oxygen knob is on the right side of the seat bucket. Once the emergency oxygen knob is pulled, it cannot be shut off.

SEAT POSITION SWITCH

The seat may be adjusted vertically only. Positioning is accomplished by actuating a momentary contact switch located on the right side of the seat bucket. The seat can be adjusted (up or down) through a total range of 135 mm. It is not necessary to adjust the seat height before ejection.

SURVIVAL PACK

The survival pack contains the survival equipment, the emergency locator beacon, the emergency ration, the first aid kit and the distress signaling kit.

It is stored in the seat pan, side by side with the dinghy. A cushioned profile seating face, designed and shaped to give maximum support to the crewmember covers the equipment.

The survival pack will be released automatically after man / seat separation thereby inflating the self inflating floating device of the emergency locator beacon and the dinghy automatically. Dinghy bottom and the spray deflector can be inflated through rubber tubes after landing.

The profile seating face will be retained, and the dinghy, emergency locator beacon and survival pack remain attached to the life vest by a lowering line. If the dinghy fails to separate completely, tearing up the sewed up portion of the dinghy lowering line abruptly will cause the lowering line to extend completely separating the dinghy from the seating face.

NOTE

- Should the dinghy fail to unfold before landing, it can be inflated manually by pulling a handle on the CO_2 bottle.
- Should the self inflating floating device of the emergency locator beacon fail to inflate completely, it can be inflated manually.

DUAL EJECTION

As soon as either the F/C or the R/C ejection handle is pulled, the ejection sequence starts. The rear seat always ejects first, followed by the front seat after 1 sec.

AIR CONDITIONING AND PRESSURIZATION SYSTEM

The air conditioning and pressurization system (refer to figure FO-14) consists of two major systems, one for the cockpit and one for electronic equipment compartments. The cockpit air is conditioned so that it will have a defined temperature and pressure. The air conditioning system for the avionics provides cooled air for the various equipment compartments.

Engine bleed air for both systems passes through a common line to a pair of identical pressure reducer valves, arranged in series for fail-safe operation.

It is routed through a parallel arrangement of two air-air coolers and an evaporator cooler. Behind the evaporator cooler, the airstream is divided for equipment cooling and cockpit air conditioning.

The air for equipment cooling is passed through a turbo cooler, and as a cooling medium through a heat exchanger / dehumidifier for the cockpit air before being supplied to the equipment compartments.

The air for the cockpit is passed through the heat exchanger / dehumidifier and cooled down in a second turbo cooler. After being mixed with hot air from the pressure reducer valves, it enters the cockpit through several manifolds.

Hot bleed air used for windshield defogging is taken from the pressure reducer valves and routed to a motor driven valve which remains open at airspeeds below M 0.8.

CABIN TEMPERATURE CONTROL

The temperature of the air which is supplied to the cockpit and / or canopy is regulated by regulating the mixing ratio of cold and hot air. Normally the ratio is adjusted automatically to maintain the selected cockpit temperature. However, manual adjustment is also possible. DC power is required for temperature control.

CABIN PRESSURIZATION MIG-29G

Pressure in the cockpit is controlled by a cabin pressure control valve. When the aircraft is below 6 500 ft, the control valve automatically maintains a pressure difference of 0.05 kp/cm^2 (50 hPa) or less. From 6 500 ft up, differential pressure increases up to 40 000 ft. The differential pressure of 0.29 to 0.31 kp/cm^2 (300 ±10 % hPa) obtained between 30 000 and 40 000 ft is maintained constant at higher altitudes. Refer to figure 1-51.

CABIN PRESSURIZATION SCHEDULE MIG-29G

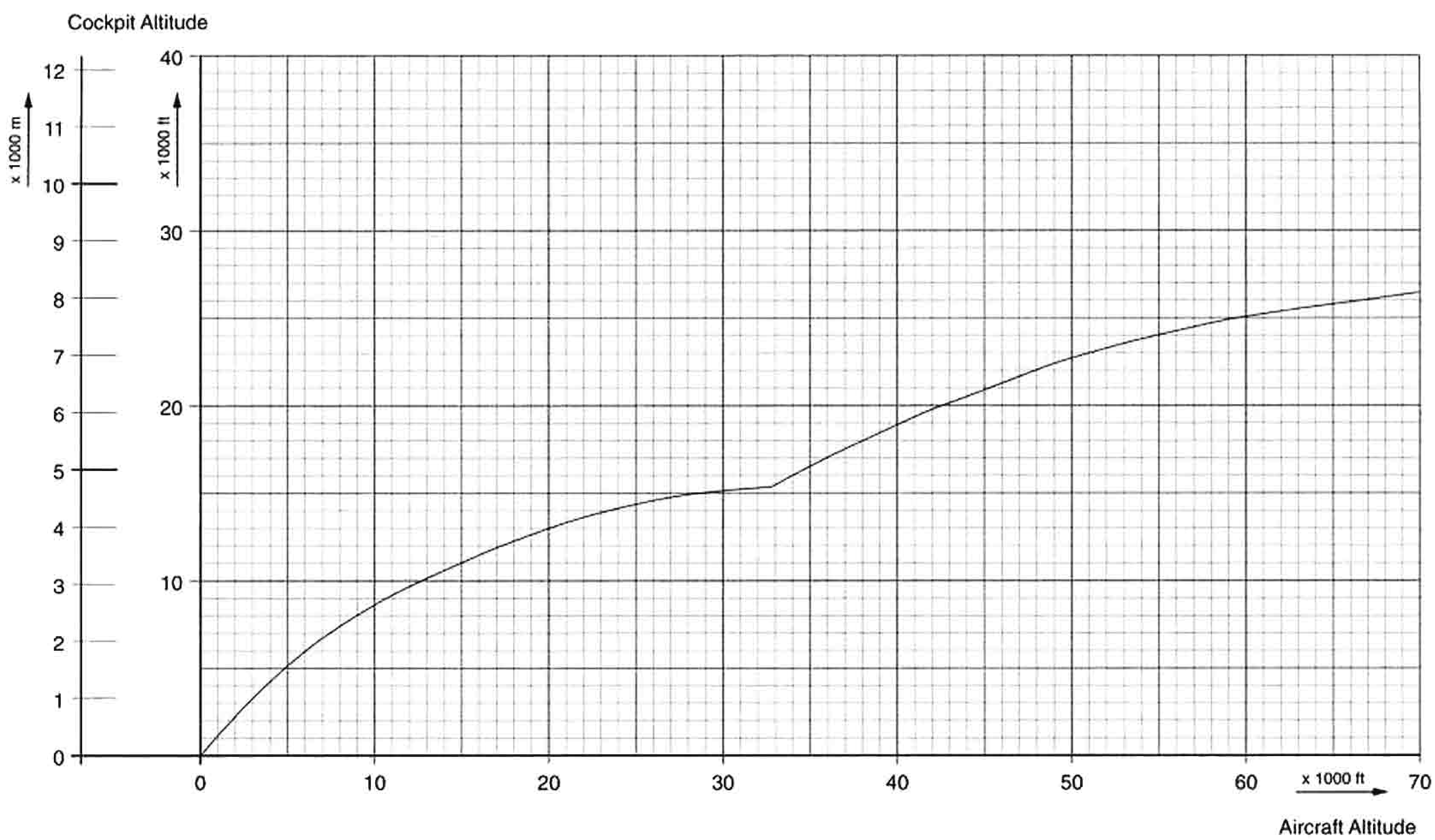


Figure 1-51

A safety relieve valve controls the cabin pressure at a nominal 0.33 kp/cm^2, 33 kPa above ambient pressure in case the cabin pressure control valve fails. If ambient pressure exceeds cabin pressure, a vacuum valve opens to allow pressure compensation.

CABIN PRESSURIZATION MIG-29GT

Cabin pressure is controlled by a cabin pressure control valve. Prior to takeoff and immediately after landing, the cockpit is not pressurized. Inflight, the cockpit is pressurized to a cockpit altitude equivalent to an altitude below mean sea level immediately after takeoff. As altitude increases, differential pressure increases so that a cockpit altitude equivalent to mean sea level is maintained up to approximately 4 000 ft. From 4 000 ft up, the pressure differential increases continuously until a differential pressure of 290 hPa is reached at 12 000 ft MSL. Above 12 000 ft a differential pressure of 290 hPa is maintained. Refer to figure 1-51A.

A safety relieve valve controls the cabin pressure at a nominal 315 hPa to 340 hPa above ambient pressure in case the cabin pressure control valve fails. If ambient pressure exceeds cabin pressure, a vacuum valve opens to allow pressure compensation.

CABIN PRESSURIZATION SCHEDULE MIG-29GT

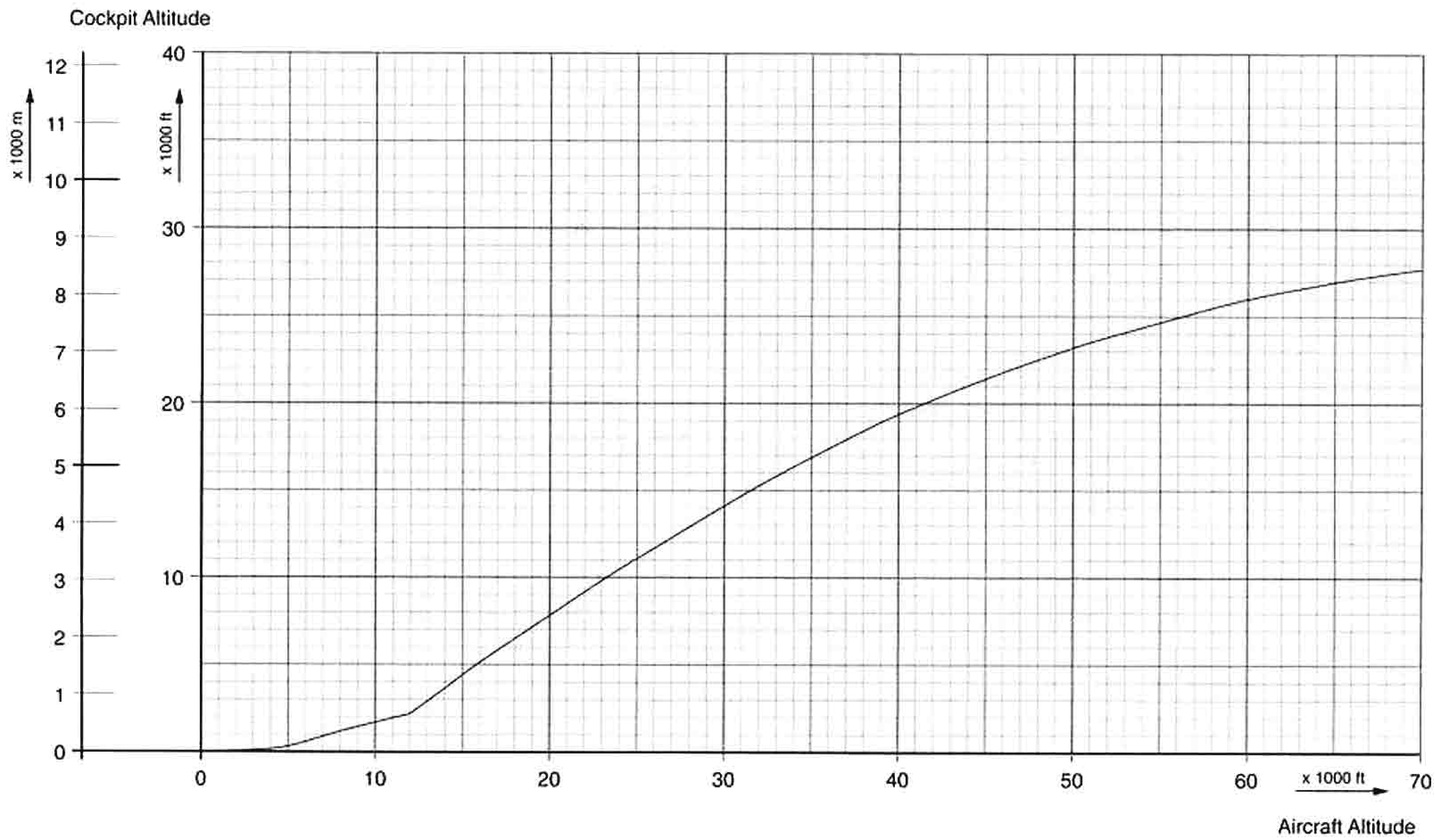


Figure 1-51A

WINDSHIELD DEFOGGING

Fogging of the windshield is prevented by heating the inside surface of the glass with hot air. Air from the pressure reducer valve is delivered through defogging manifolds at airspeeds below M 0.8.

Cold air is automatically mixed with the hot air to prevent inconvenient cockpit temperature. When the mixing valve is fully open, windshield defogging is automatically shut off to prevent an overtemperature. Manual shut down is possible with the CABIN AIR lever.

INDICATIONS AND WARNINGS

	INDICATION	FAULT / EFFECT
AEKRAN	CABIN LIMIT PRESS DESCEND	Cabin altitude above 42 650 ft ±1 640 ft.

ANTI G VALVE

The anti g valve controls air delivery to the anti g suit. Air is tapped from downstream of the air-air coolers, passed through a regulator valve, an anti g valve, and delivered to the suit via the PEC. Below 2.5 g no pressure passes through the suit. Above 2.5 g, the anti g valve controls the suit pressure in proportion to the g-forces experienced.

NOTE

For proper inflation of the anti g suit, the pressure regulator must be set to min. Any position other than min will result in premature pressurization.

VENT SUIT VALVE

The ventilation suit valve controls air delivery to a ventilation suit. Air is tapped from downstream of the turbo cooler, passed through an ejector valve and mixed with hot air, and delivered via the PEC.

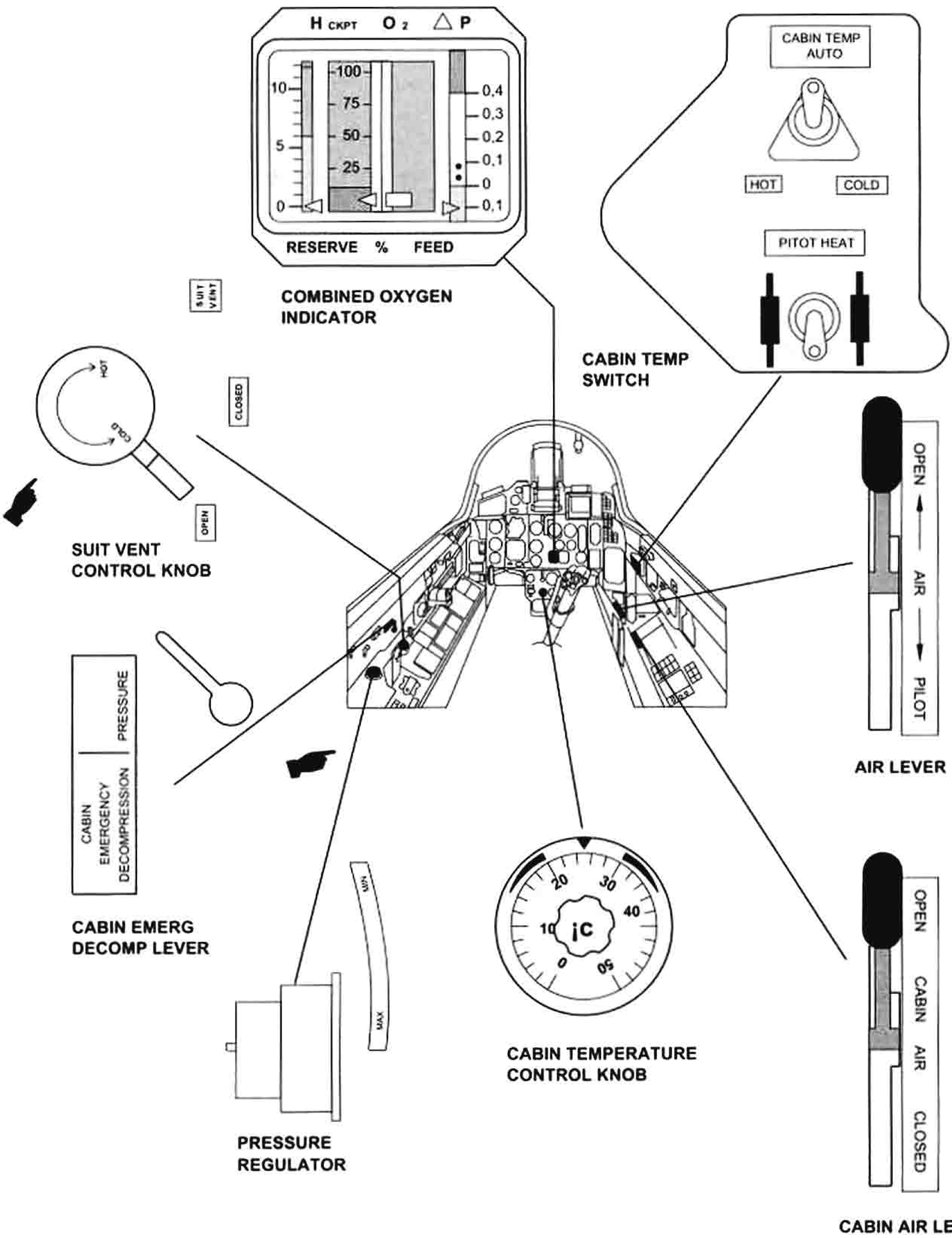


Figure 1-52

CABIN TEMP SWITCH

The cabin temp switch is located on the RH side wall. The four position switch permits selection of automatic temperature control system or manual adjustment of cockpit temperature.
The switch positions are:

AUTO	Cockpit temperature is automatically adjusted in accordance with the setting of the cabin temperature control knob on the main vertical console.
Center position	It is the neutral position and removes the electrical supply from the mixing valve and as a result freezes the valve in it's last position.
HOT / COLD	The mixing valve is driven in the appropriate direction. The switch should only be bumped to the position momentarily to prevent the valve from driving to an extreme temperature position.

CABIN TEMPERATURE CONTROL KNOB

The cabin temperature control knob permits selection of the desired cockpit temperature, provided the CABIN TEMP switch is in the AUTO position.

CABIN AIR LEVER

The cabin air lever regulates the volume of air delivered from the air conditioning system. In the CLOSED position, windshield defogging is manually shut down.

AIR LEVER

The air lever routes the air either to the manifolds directed towards the pilot in the position pilot or the ones towards the canopy in the position open.

PRESSURE REGULATOR

The pressure regulator controls volume and pressure of the air used for inflation of:

- Anti g suit in MIN.
- Partial pressure suit in MAX.

SUIT VENT CONTROL KNOB

The suit vent control knob adjusts temperature and flow of the air routed to the ventilation suit.

COMBINED OXYGEN INDICATOR

The utmost right scale of the combined oxygen indicator displays differential pressure between cockpit pressure and outside air pressure. See oxygen system in this section.

INDICATIONS AND WARNINGS

	INDICATION	FAULT / EFFECT
AEKRAN	NO COOLING	Temperature of the air towards the equipment compartments exceeds +80° C.
VIWAS	"AUSFALL KÜHLUNG GERÄTESEKTION" "VERRINGERE TEMPERATURREGIME"	

NOTE

A reduction of the temperature in the equipment compartment can only be achieved by reducing the airspeed.

LIGHTING SYSTEM

The lighting system consists of the external and internal lighting equipment. Refer to figure 1-53.

LIGHTING SYSTEM CONTROLS

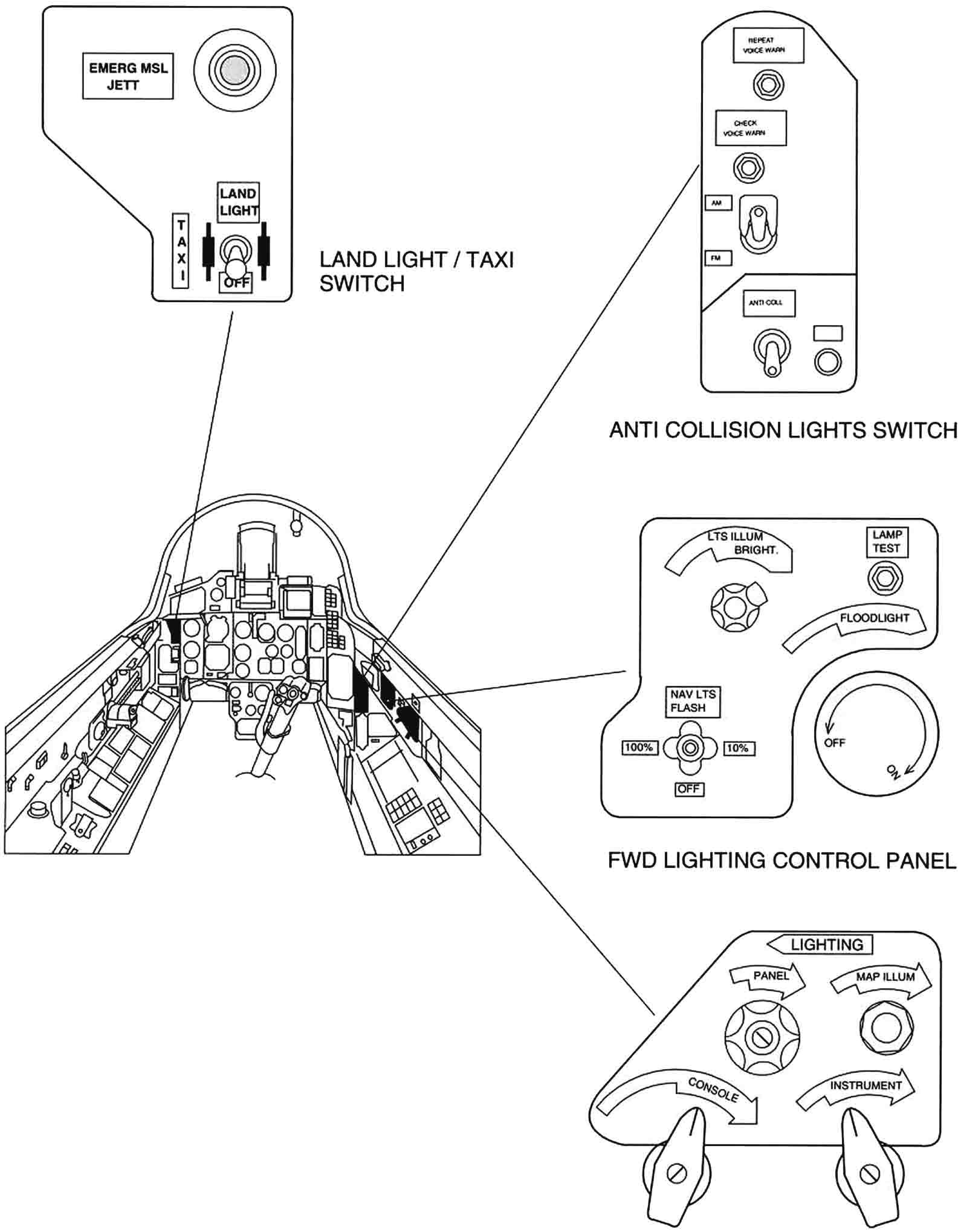


Figure 1-53

EXTERNAL LIGHTING

The external lights include navigation lights, anti collision lights, landing and taxi lights. All external lights are powered by the DC system.

NAVIGATION LIGHTS

Two position lights are installed at the wing tips, a green one right, a red one left and a white one at the left vertical stabilizer.

All navigation lights are controlled by the NAV LTS switch located on the forward lighting control panel.

The selectable modes are: OFF, 100 %, 10 %, and FLASH.

On the ground, the navigation lights will be in the 10 % intensity mode regardless of mode switch setting.

ANTI COLLISION LIGHTS

Anti collision lights are installed behind the cockpit and on the left engine bay. These lights are controlled by the ANTI COLL switch.

LANDING / TAXI LIGHTS

Two landing lights are installed, one on each main landing gear door, and a single taxi light on the nosewheel strut.

The light beam of the right landing light is angled down 10° with respect to the horizon, and offset by 12° to the left with respect to the aircraft's center line.

The light beam of the left landing light is angled down 8° and offset 14° left.

The light beam of the taxi light is aligned parallel to the horizon and to the center line of the aircraft.

The landing as well as the taxi lights are powered by the DC supply system and are controlled by the LAND LIGHT / TAXI switch located on the left side of the instrument panel.

With the switch in position TAXI, only the taxi light is on, whereas both taxi and landing lights are on when the switch is in the LAND LIGHT position.

The landing lights are disabled when the landing gear is retracted regardless of the position of the control switch.

INTERNAL LIGHTS

The internal lighting equipment comprises console panel lights, instrument lights, console flood lights, spot lights, map reading light and associated controls. The control panels and indicators are powered by the AC system, and the console floodlights for the panels, instruments and the map reading light, by the DC system.

The instruments are illuminated with shielded light fixtures located adjacent to each indicator. The major left and right console control panels are indirectly illuminated. The lights are controlled by rheostat-type switches located on the aft lighting control panel.

The control knobs are assigned to the various illumination systems as follows:

- The PANEL control knob has a dual function. If pushed in, it allows manual intensity control of all cockpit information and warning lights except AEKRAN. If pulled out, intensity is automatically controlled by a photo diode according to ambient brightness.
- The MAP ILLUM control knob switches and dims the map reading light located near this panel.
- The instrument illumination is switched and dimmed with the INSTRUMENT control knob.
- The CONSOLE control knob switches and controls the intensity of the indirect illumination of various switches and control knobs.

Two more control knobs are located on the forward lighting control panel to control the floodlights and the brightness of the landing system signal panel illumination.

OXYGEN SYSTEM

The oxygen system is a pressure demand system, and consists of a main system, located in the fuselage, an engine supply system and an emergency system on the ejection seat. Refer to figure FO-15. The oxygen supply for the main oxygen system and for the engine supply is replenished through one single charging connection. Emergency oxygen supply is charged directly to the bottle.

MAIN OXYGEN SYSTEM

The main oxygen system supply consists of three 4 liter high pressure gaseous oxygen bottles (MiG-29GT seven bottles), charged at 150 kp/cm². Further components are a oxygen flow valve, a pressure reduction valve, an oxygen flow regulator, a PEC, a pressure regulator, the combined oxygen indicator and the oxygen control panel. The system is mechanically controlled, however, 115 VAC is required for indicator operation.

ENGINE SUPPLY

Refer to ENGINE STARTING SYSTEM in this section.

EMERGENCY OXYGEN SYSTEM

Refer to EJECTION SEAT SYSTEM in this section.

OXYGEN SYSTEM CONTROLS AND INDICATORS

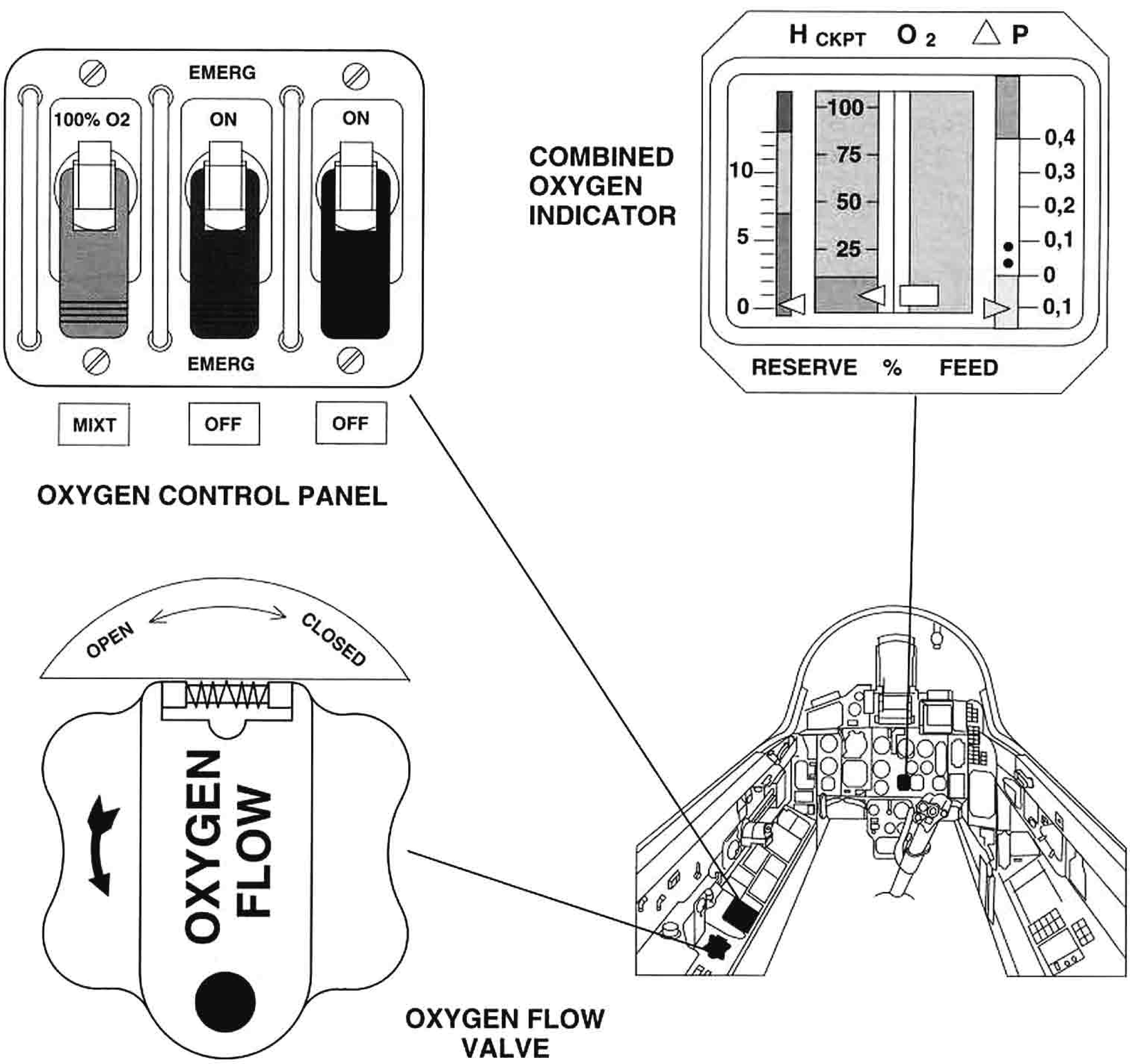


Figure 1-54

OXYGEN FLOW VALVE

The oxygen flow valve is a rotary knob marked OPEN / CLOSE on the LH console. It controls the supply of oxygen to the system.

OXYGEN CONTROL PANEL

MIX - 100 % switch

The blue MIX - 100 % switch allows the selection of either an oxygen / air mixture or pure oxygen.

EMERGENCY ON - OFF switch

The red EMERGENCY ON - OFF switch permits selection of 100 % oxygen with positive pressure or normal oxygen supply. The switch should remain in OFF position at all times, unless an unscheduled pressure increase is required. Moving the switch to EMERGENCY ON provides 100 % oxygen with continuous positive pressure to the face mask.

NOTE

When EMERGENCY ON is selected, use of oxygen is 2 to 3 times higher than normal. Quantity remaining must be continuously monitored.

HELM VENT ON - OFF switch

The black HELM VENT ON - OFF switch is provided to activate the helmet ventilation system.

OXYGEN PRESSURE REGULATOR

The pressure regulator comprises an airmix and a 100 % oxygen demand type regulator. With MIX selected, the air / oxygen ratio is determined by an air inlet valve and thus varies according to cabin altitude. Below 6 600 ft, pure cabin air is delivered. Above 6 600 ft, the air inlet valve reduces the air percentage until 100 % oxygen is delivered at 26 000 ft. Above 40 000 ft, pressure breathing is introduced with pressure increasing with altitude. Between 0 and 40 000 ft, with 100 % selected, the 100 % regulator delivers 100 % oxygen. Above this level, pressure breathing is introduced with pressure increasing with altitude.

COMBINED OXYGEN INDICATOR

Operation of the oxygen system can be monitored on the combined indicator located in the center of the front panel.

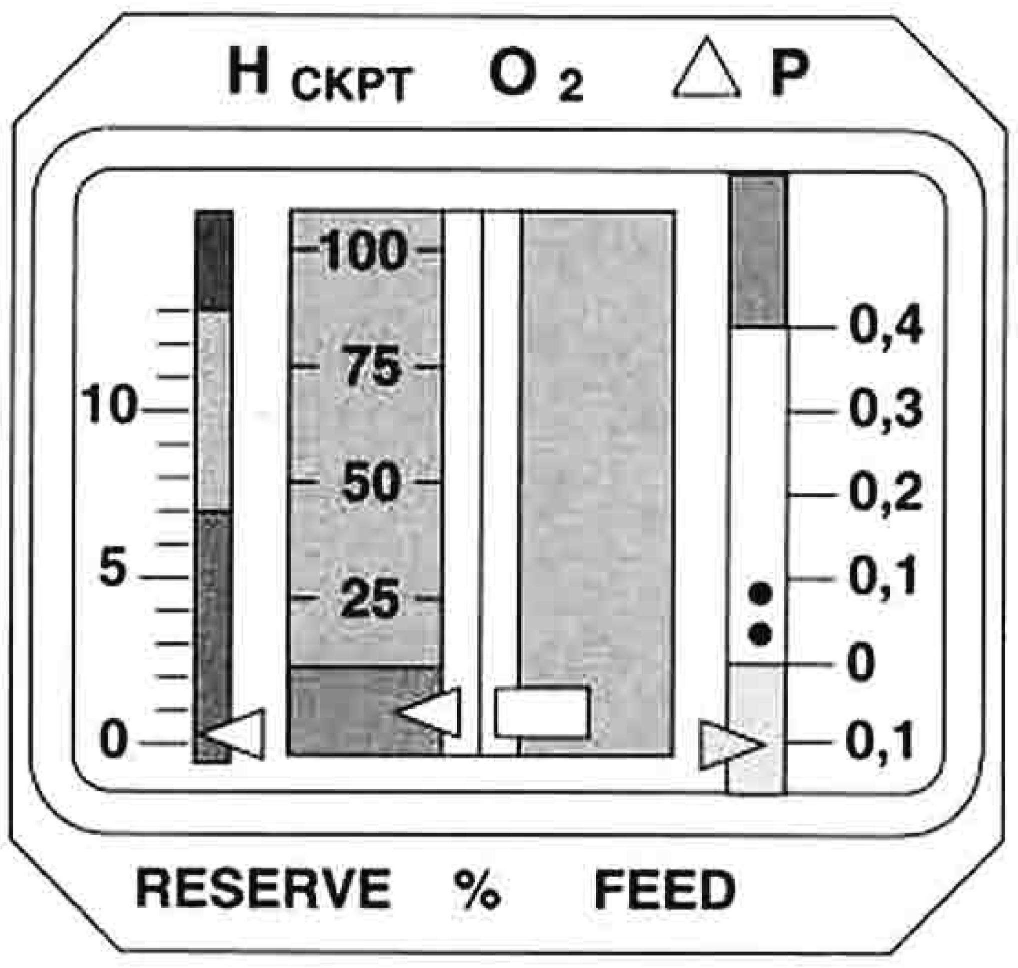


Figure 1-55

The instrument has three thermometer-type scales with triangular pointers and one rectangular pointer for oxygen flow.
The following parameters can be monitored:

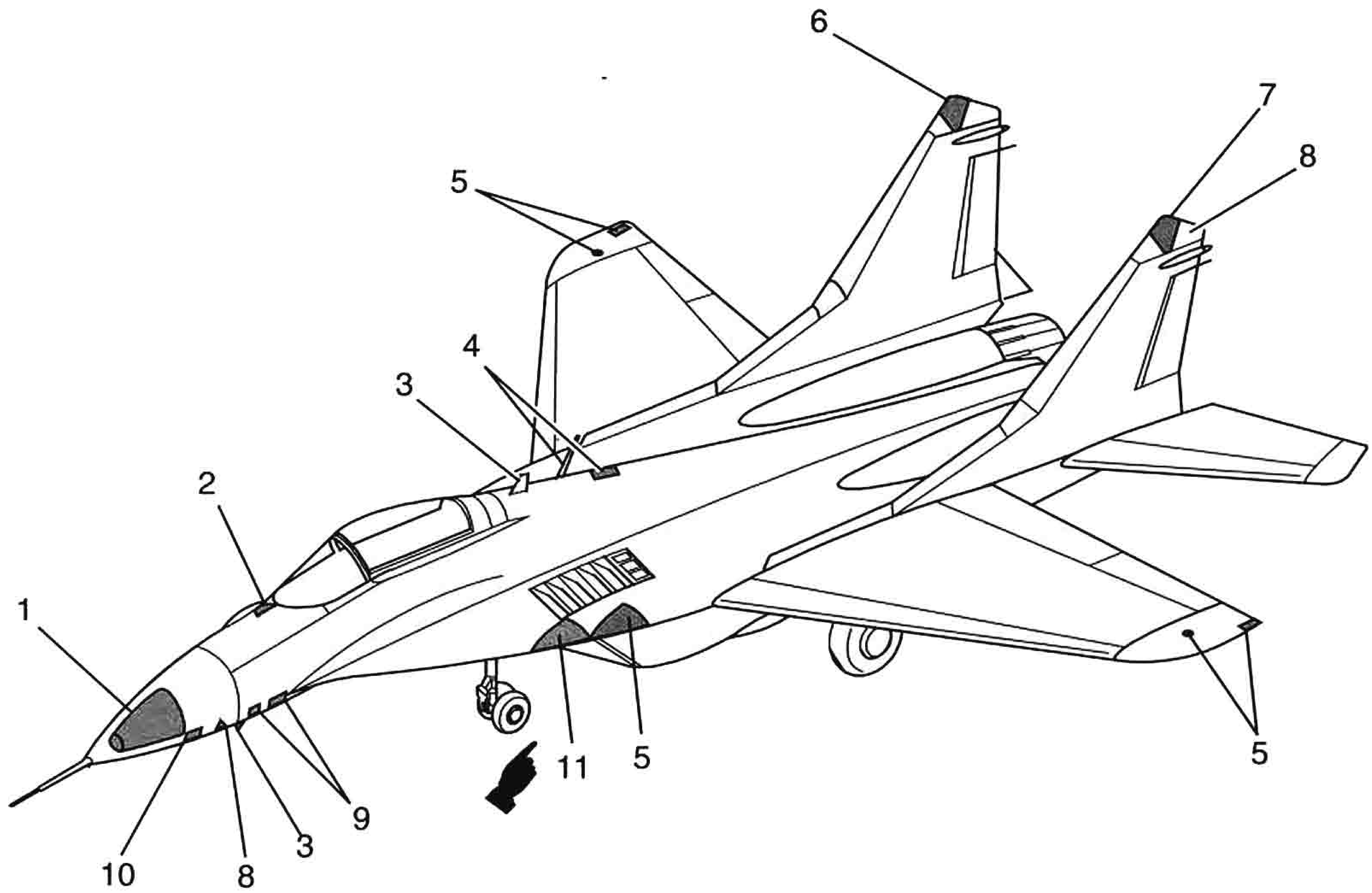
- Cabin altitude:
  The scale is calibrated in km. As long as the cabin is not pressurized, the pointer indicates actual flight altitude.
- Oxygen quantity:
  100 % is indicated with a pressure of 150 kp/cm^2 in the oxygen bottles. As the pressure decreases indication drops proportionally.
- Oxygen flow:
  During normal oxygen flow, the pointer moves up during inhalation and moves down during exhalation. A steady pointer indicates that no oxygen is supplied (e.g. with MIX selected at low altitude).
- Cabin pressure differential.

INDICATIONS AND WARNINGS

	INDICATION	FAULT / EFFECT
AEKRAN	EMERG OXYGEN RESERVE	Oxygen quantity 15 %.
VIWAS	"SAUERSTOFFVORRAT MINIMAL"	

COMMUNICATION AND AVIONIC EQUIPMENT

ANTENNA SYSTEM



1. RADAR	5. RHAW	9. RAD ALT
2. IRSTS	6. UHF / VHF RADIO	10. MARKER BEACON
3. TACAN	7. XT-2000	11. NOT USED
4. ADF	8. IFF / SIF	

Figure 1-56

VHF / UHF RADIO

Voice communication is provided by the VHF / UHF radio. 2 000 VHF and 7 200 UHF frequencies can be used. 29 preset frequencies are available, with UHF guard channel monitoring.

NOTE

Due to the wide-band antenna location in the right fin-tip, communication may be interrupted momentarily during turns exceeding 45°AOB.

RADIO Switch

The radio switch is located on the right console. Power to operate the VHF / UHF radio is supplied by the DC generator or the batteries.

VHF / UHF FREQUENCY CONTROL PANEL

A VHF / UHF frequency control panel (refer to figure 1-57) is installed on the LH console. The function of each control is:

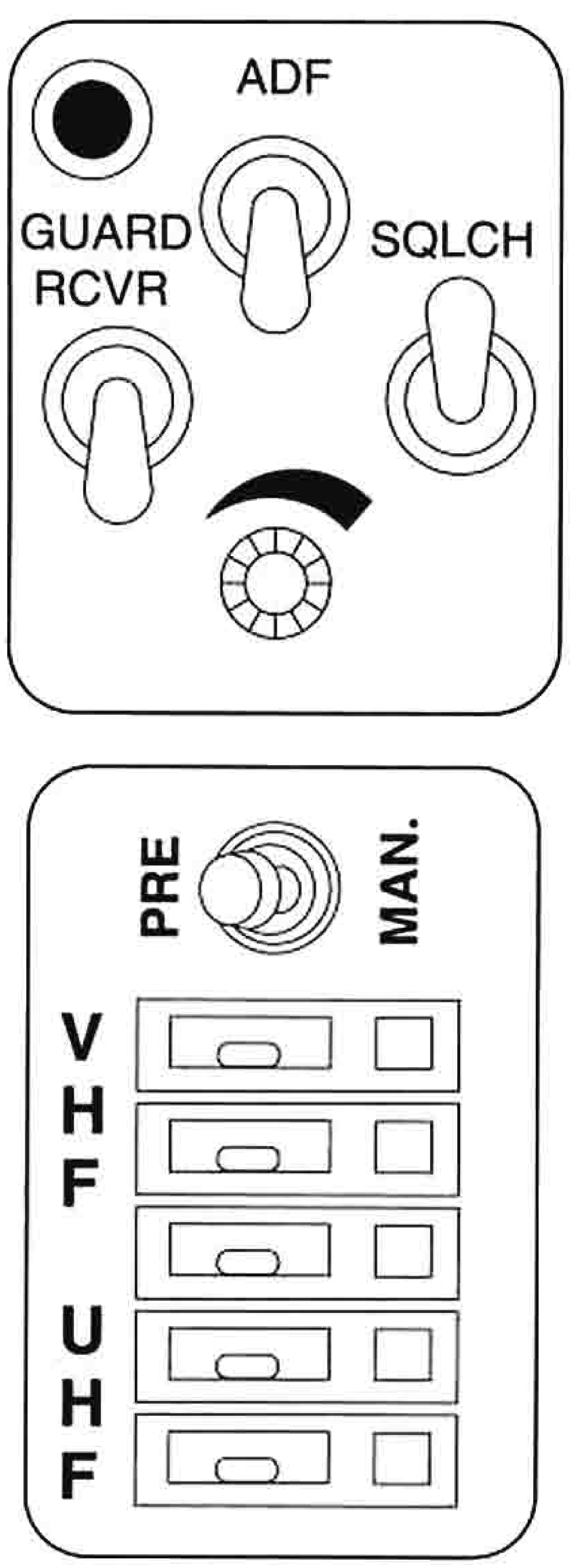


Figure 1-57

Communication Frequency Toggle switches

With the preset / manual switch on MAN., five toggle switches are used to select the desired frequency. Physically, frequencies from 100.000 to 399.975 MHz in increments of 0.025 MHz can be selected. However only frequencies from 100.000 to 149.975 MHz and 220.000 to 399.975 MHz can be used. The selected frequency is displayed on the VHF / UHF radio indicator panel. However, the last digit (0 or 5) is not displayed. If an unusable frequency is selected, the indication flashes.

Preset / Manual Switch

This switch controls the frequency selection method. In MAN., the frequency is selected with the toggle switches , in PRE, frequency is selected with the channel selector knob on the indicator panel.

VHF / UHF RADIO PANEL

A VHF / UHF radio panel (refer to figure 1-57) is installed on the LH console. The function of each control is:

Volume Control Knob

Clockwise rotation of the volume control knob increases the communication receiver volume.

Guard Receiver Select Switch

With GUARD RCVR selected, UHF guard frequency is monitored.

Squelch Switch

The squelch switch enables and disables communication receiver squelch.

Guard Receiver Control Light

The light illuminates when guard transmissions are received even if the guard receiver is deselected.

ADF Switch

With ADF selected, transmissions on the selected ADF frequency can be monitored.

VHF / UHF INDICATOR CONTROL PANEL

A VHF / UHF indicator control panel (refer to figure 1-58) is installed on the instrument panel. The function of each control is:

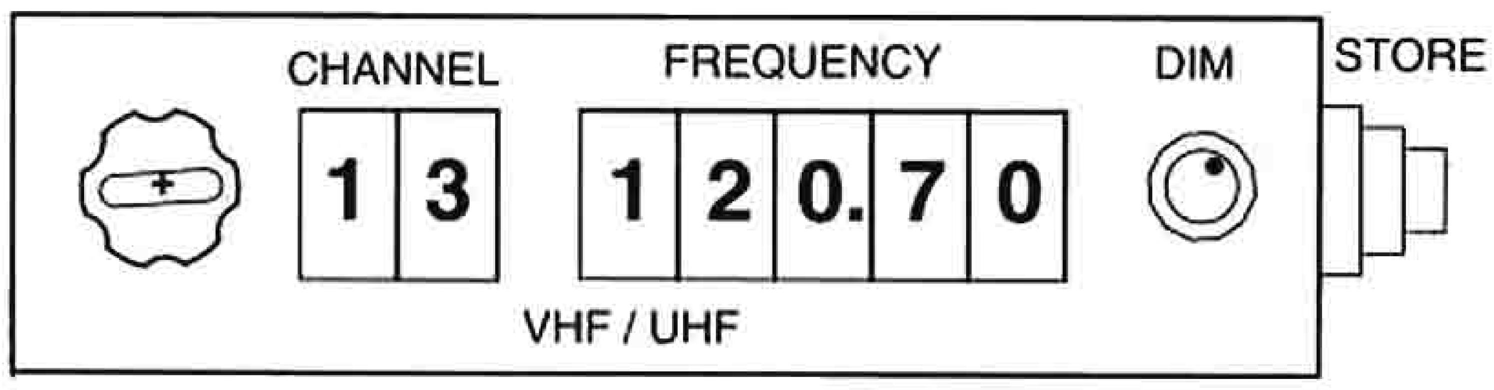


Figure 1-58

Channel Selector Knob

With the preset / manual switch in PRE, 29 preset frequencies can be selected.

Channel Display

The selected channel is displayed.

Frequency Display

With the preset / manual switch in PRE, the frequency of the selected channel is displayed. In MAN., the manually selected frequency is displayed, except the last digit.

DIM Knob

The DIM knob is used to adjust the brightness of the channel and frequency display.

Store Pushbutton

Pushing the store button enters the frequency selected with the toggle switches to the indicated channel.

VHF / UHF RADIO OPERATION

The VHF / UHF equipment is activated by switching the RADIO switch to ON. Transmission is accomplished by pressing the throttle-mounted microphone button. The receiver signal can be controlled with the volume knob on the control panel.

Manual Frequency Selection

Set preset / manual switch to MAN. Enter a six-digit frequency via the toggle switches. The last two digits (00-25-50-75) are selected with one toggle switch.

Channel Selection

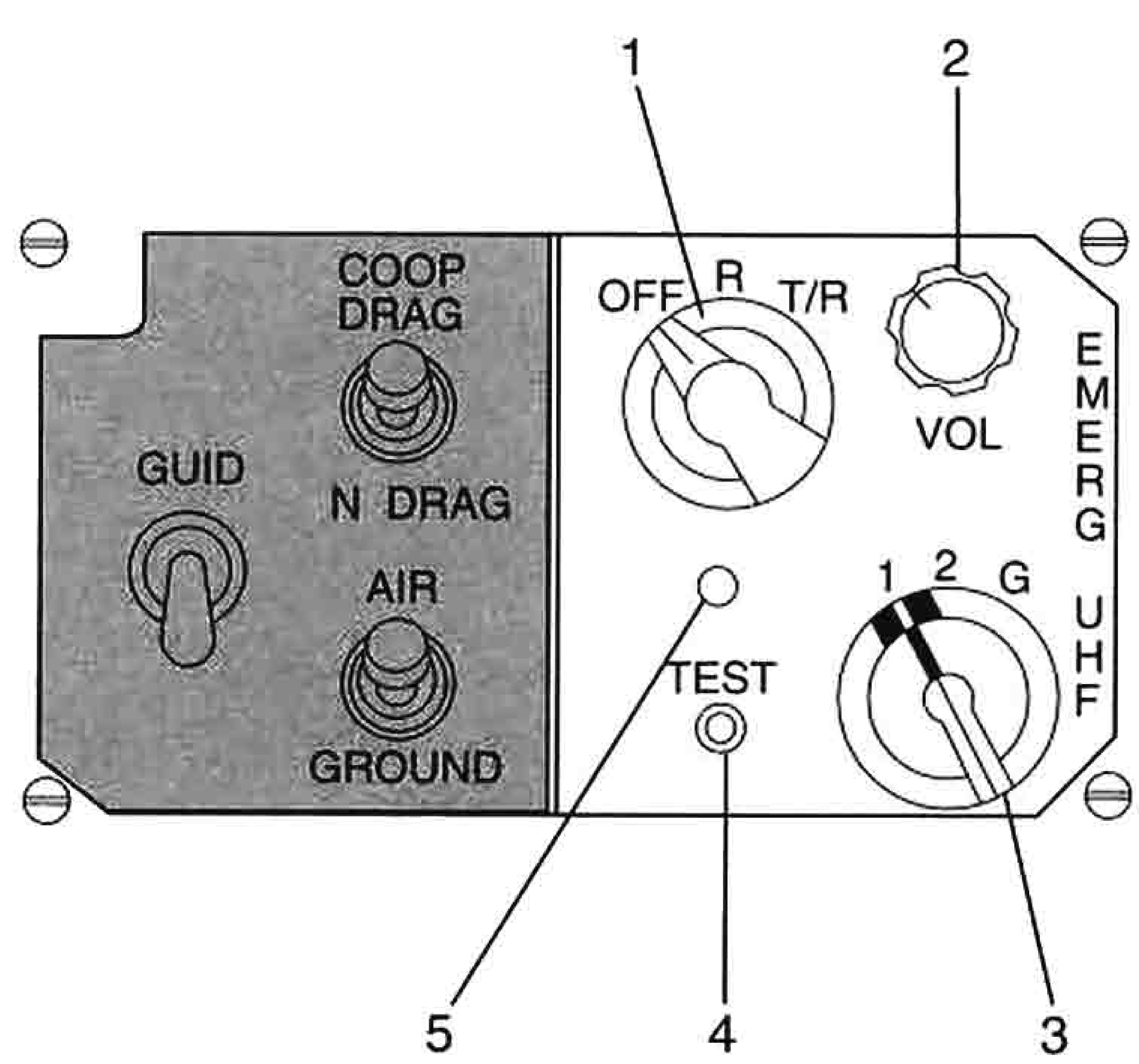
Set preset / manual switch to PRE. Select desired channel with the channel select knob. Channel number is displayed on the channel display, and the corresponding frequency on the frequency display.

Channel / Frequency Loading

If a stored frequency needs to be changed, select the desired channel. Insert the new frequency via the toggle switches. After pressing the STORE button, the new frequency is displayed on the frequency display and stored simultaneously.

EMERGENCY UHF RADIO (XT-2000)

The emergency UHF radio provides air-to-air and air-to-ground communication on the UHF preset distress frequency of 243.0 MHz, and two further preset channels. The radio can also be used as guard or auxiliary receiver in the airborne communication system. The radio is powered by 28.5 VDC and utilizes the emergency UHF antenna in the left vertical fin.



1. **FUNCTION SELECTOR**
2. **VOLUME CONTROL KNOB**
3. **CHANNEL SELECTOR**
4. **TEST PUSHBUTTON**
5. **TEST INDICATOR LIGHT**

Figure 1-59

CONTROLS AND INDICATORS

The emergency UHF control panel is installed on the forward LH console.

Function Selector

The function selector is a rotary knob with positions marked OFF, R and T/R.

OFF	Electrical power is disconnected
R	The system can only receive
T/R	The system is able to transmit and receive

Volume Control Knob

The volume control knob adjusts the audio level of the receiver.

Channel Selector

The channel selector, which is labeled CHAN, is a rotary knob with positions 1, 2 and G.

1, 2	Select one of two preset channels
G	Select the UHF distress frequency (GUARD) of 243.0 MHz

Test Pushbutton and Indicator Light

The test pushbutton and indicator light is used for performing built-in test to determine if the system is operating properly.

Built-In Test

1. Function selector - T/R
2. Channel selector - To desired channel
3. Test pushbutton - Press and hold

The indicator light illuminates and remains illuminated as long as the pushbutton is held, and a short tone is heard if the system is operating properly. If there is a system malfunction, the indicator light flashes momentarily and then extinguishes, and the audio tone is not heard.

EMERGENCY UHF RADIO OPERATION

Use as guard or auxiliary receiver :

1. Function selector - R
2. Channel selector - To desired channel
3. Test pushbutton - Press
4. Volume control knob - To desired audio level

Use as a transmitter / receiver :

1. Function selector - T/R
2. Channel selector - To desired channel
3. Test pushbutton - Press
4. Volume control knob - To desired audio level
5. PTT button - Press for transmission

INTERCOM SYSTEM

The intercom (I/C) system enables communication between the ground crew and the pilot.

GT: The I/C enables additionally communication between the two cockpits.

All audio warnings produced by various aircraft systems and identification signals from radio and navigation equipment are routed to the pilot's headset. Volume of the radio and navigation equipment can be adjusted by the relevant volume control. Aircraft warnings are transmitted at an audio level high enough to attract attention. Ground crew / pilot communication is adjusted at a level to provide understanding. Audio signals which may be heard over the I/C are listed with the relevant system, VIWAS signals in section 3.

RECORDERS

HUD CAMERA

Control signals from the fire control system triggers camera operation, refer to GAF T.O. 1F-MIG29-34-1.

FLIGHT DATA RECORDER

The flight data recorder (FDR)/TESTER records flight parameters and the operation of important aircraft systems.

Conservation of recorded data is assured under the following conditions:

- Impact with g-forces up to 1 000
- Temperatures up to 1 000° C during 15 minutes
- Exposure to sea water up to 5 days
- Exposure to fuel up to two days

The recorder is powered by 28.5 VDC. The data recording is made of the last three hours of the aircraft operation.

Controls and Indications

The recorder is activated manually with a switch labeled RECORD on the RH console.

If not activated manually, automatic operation starts at or above 85 % engine RPM with the trailing edge flaps down, or if the weight is off the right main gear at any RPM.

Operation

Data from the flight data recorder are transferred to the ground evaluation system without extracting the tape, at a rate 8 - 12 times faster than the recording. They are used to make an express analysis of aircraft and system operation.

The express analysis contains following information on a data sheet:

- Aircraft number
- Flight number
- Date of flight
- Sequence number of the malfunction
- Channel number for recorded data and extreme values
- Start and end time of occurrences or malfunctions

In addition, the following possibilities are available:

- Display in a graphical form of the coded values as recorded by the FDR
- A print-out of all malfunctions

List of Recorded Parameters

Aircraft velocities, rates and control surface positions:

- Aircraft velocity
- TAS
- Barometric altitude
- Altimeter setting
- G-forces in all axes
- True course.
- AOB
- Pitch angles
- AOA
- Deflection angle of tailerons
- Deflection angles of the rudders
- Control stick deflection
- Pedal position
- Aileron position
- Mach number

Engine parameters:

- RPM of engine HP compressors
- RPM of engine LP compressors
- Air temperature at the intake of the engines
- Fuel pressure at the first stage of the engines
- Fuel quantity
- Oil pressure in the ENG GBX
- Oil pressure of the engines
- Position of the exhaust nozzle flaps at the critical cross section
- Pressure at the exhaust of both engines turbines
- Pressure at the intakes of both engines
- Temperature at the exhaust of the engine turbines
- Throttle positions
- Vibration of the ENG GBX
- Vibration of the engine turbines

AFCS system:

- Stroke of the yaw damper actuator
- Stroke of the longitudinal damper actuator
- Stroke of the feel unit actuator
- Stroke of the aileron trim drive
- Stroke of the pitch trim drive
- Discrete signals of the AFCS system

Electrical power supply:

- AC bus voltage 115 V, 400 Hz
- DC bus voltage 22 to 28.5 V

Discrete signals:

- Failure of the main hydraulic system
- Failure of the hydraulic booster system
- Backup system of the LH engine
- Backup system of the RH engine
- LH engine overheated
- RH engine overheated
- No fuel flow
- Reduce RPM of LH engine
- Reduce RPM of RH engine
- Surge of RH engine
- No oil pressure in the GBX
- Fire
- Speedbrakes out
- Landing flaps down
- Landing gear retracted
- Eject
- Engine surge LH
- Engine surge RH
- Jettison of the canopy
- Louvers open/closed
- AB of the LH engine
- AB of the RH engine
- Position of LEF
- Signal of the marker receiver
- PTT button depressed
- 550 kg fuel remaining
- Trigger operation
- External stores loaded

Service parameters:

- Calibration voltage index for self-check
- Mission time in seconds
- Mission time in minutes
- Initialization sequence number of the FDR
- Overflow voltage
- Aircraft number

NAVIGATION SYSTEM

The aircraft navigation system consists of the gyro platform reference system, navigation computer, air data computer (ADC) and radio navigation equipment. The gyro platform reference system consists of a main and a standby platform, the radio navigation system of TACAN, ADF and marker beacon receiver. All systems are closely interfaced to supply complete navigational information throughout all flight phases.

NAV ARCHITECTURE

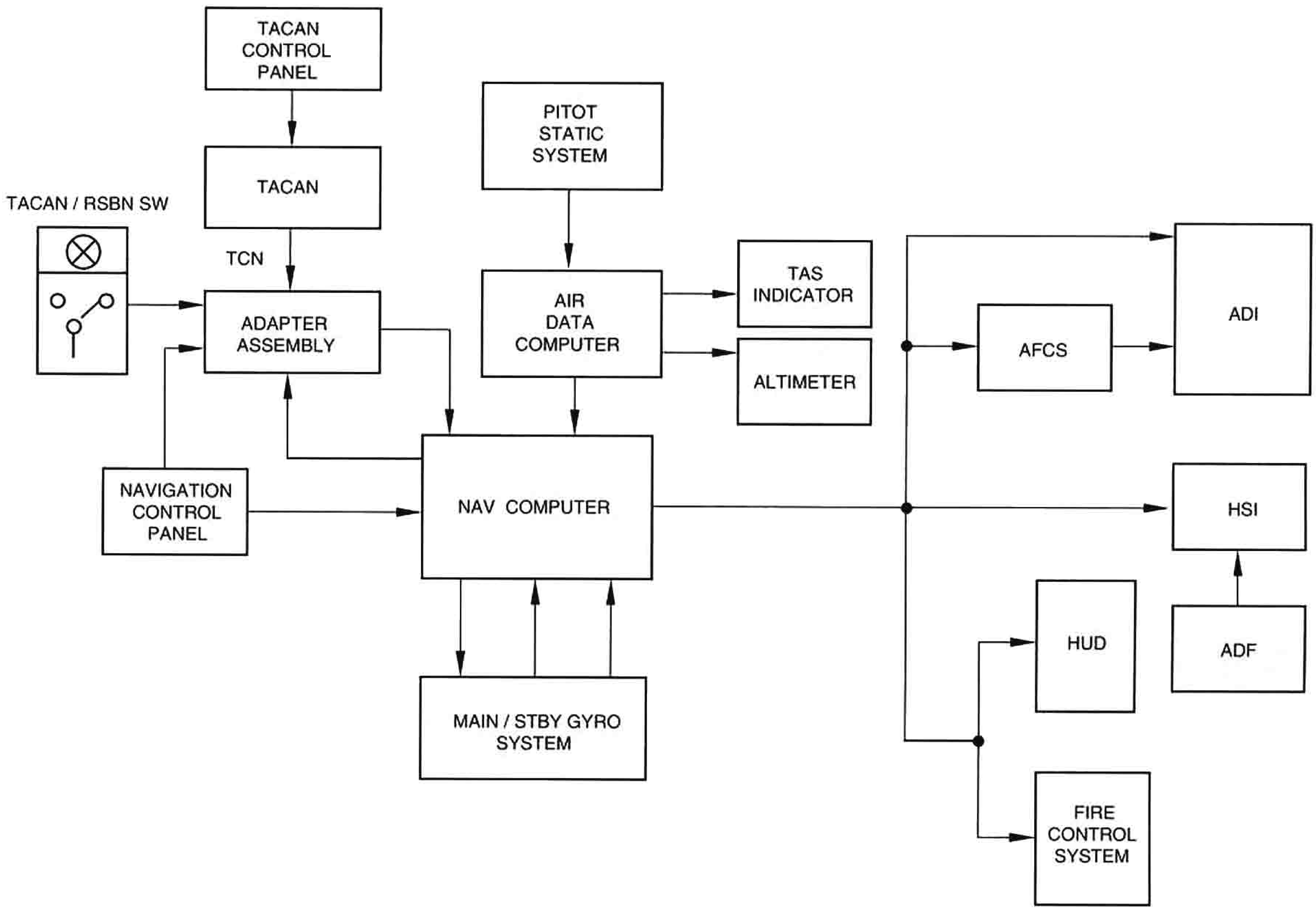


Figure 1-60

NAVIGATION SYSTEM CONTROLS

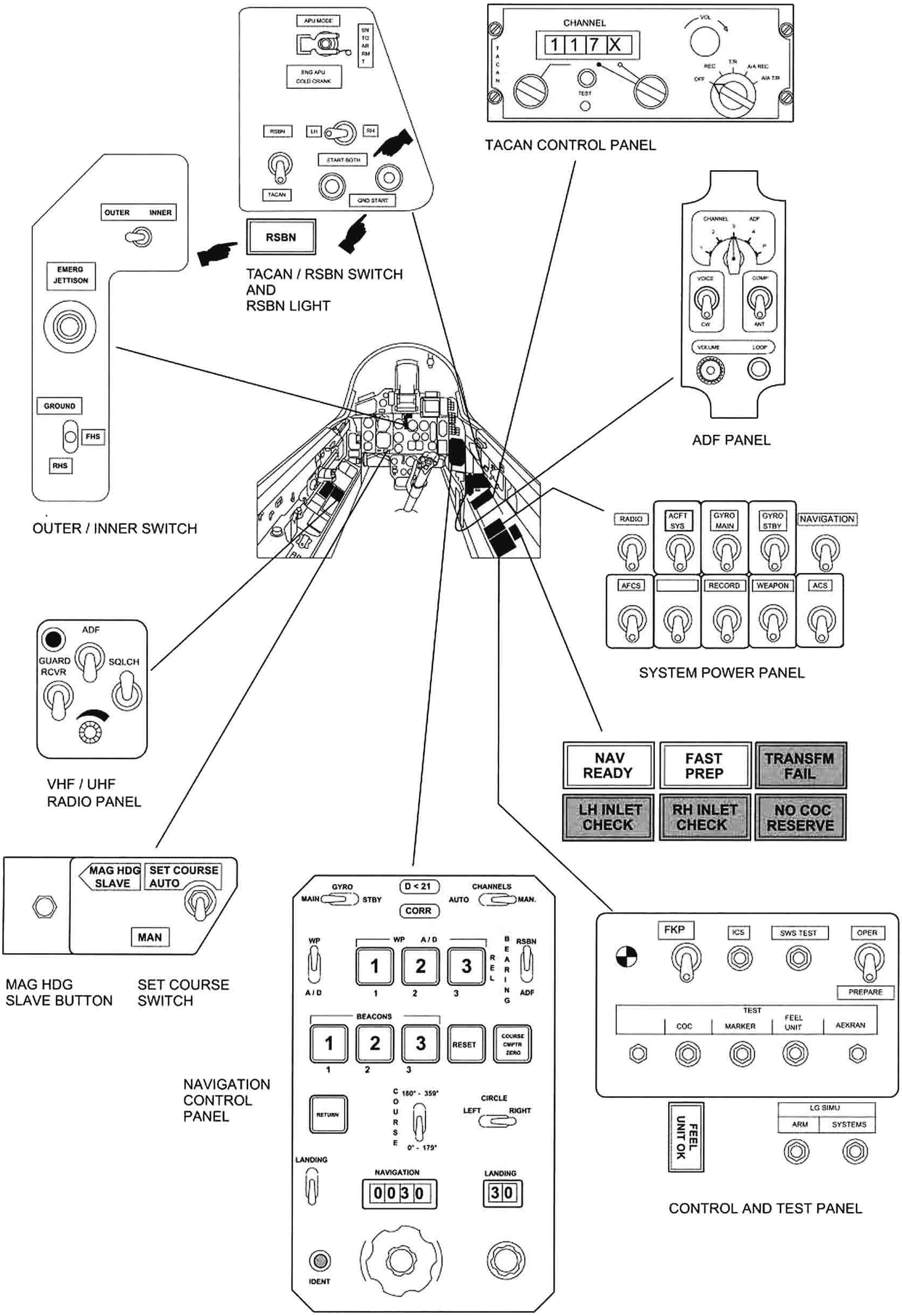


Figure 1-61

GYRO SYSTEMS

The gyro systems are used to measure bank and pitch angles, course and acceleration along the axis of the platform. They supply the primary azimuth and attitude reference, and additionally supply direction, velocity, and distance inputs to the navigation computer. The system utilizes DC power from the generator or the batteries. 115 VAC and 3 phases 36 VAC are supplied by the AC generator and the transformer, or by the PTO. However, the PTO capacity is insufficient to supply heating power to the gyro system.

The unit consists of the main and of the standby gyros, an analog / digital computer, operating controls for variation (behind the ejection seat) and latitude, a flux valve and BITE.

Each system uses a gyro-stabilized platform upon which three accelerometers are mounted. With the platform stabilized in pitch and roll by gyros and oriented along the aircraft axis, the accelerometers sense acceleration in any direction. This acceleration is processed by the analog / digital computer to provide course reference, attitude information, main gyro platform stabilization and signals for the navigation computer. A circuitry corrects for apparent precession, based on preset latitude.

GYRO ALIGNMENT

With the BAT-GND SUPPLY switch and the generator and navigation switches placed to ON, power is applied to bring the gyro platforms to operating temperature. MAIN and STBY switches are selected for stabilization of the platforms. Prior to the alignment, the appropriate aerodrome has to be selected.

Fast Alignment

The alignment cycle is started by placing the MAIN and STBY gyro switches to ON, PREPARE / OPERATE switch in PREPARE.

After approximately 30 to 40 sec, heading reference is inserted to the analog / digital computer by pressing the mag heading slave button and COURSE CMPTR ZERO pushbutton simultaneously for 10 to 15 sec. The PREPARE / OPERATE switch has to be placed to OPERATE within 90 sec after switching both gyros to ON. After a total time of 3 minutes, the fast prepare light on the right console rear panel illuminates, indicating completion of the alignment and system readiness. The light distinguishes during T/O when the weight is off the main landing gear.

Long Alignment

The PREPARE / OPERATE switch is left in the PREPARE position until the NAV READY light illuminates on the right console rear panel after 15 minutes, indicating completion of the alignment. Alignment may last up to 20 minutes at temperatures between -30° C and -60° C. When switching the PREPARE / OPERATE switch to OPERATE, the light extinguishes, indicating system readiness.

NOTE

If long alignment has been selected, and circumstances dictate switchover to fast alignment, at least 5 minutes should elapse prior switching from PREPARE to OPERATE.

GYRO OPERATION

The gyro system can be operated either with the main gyro platform or in the standby gyro platform. However, the main gyro platform is more accurate since digital integration is provided to the main gyro platform only. In case of a main gyro system failure, switchover from main to standby has to be accomplished manually.

INDICATIONS AND WARNINGS

	INDICATION	FAULT / EFFECT
MASTER CAUTION	LIGHT FLASHING	
AEKRAN	MAIN DIR VERT GYRO or STBY DIR VERT GYRO or TWO DIR VERT GYRO	Failure of corresponding gyro system(s).
ADI	Gyro fail light	

NAVIGATION COMPUTER

The navigation computer is the central unit of the navigation system. It processes data from the inertial navigation unit and from the air data computer (ADC) to compute the present position and to correct it according to TACAN signals, to compute azimuth and distance information to a selected, programmed navigation point, and altitude deviations as well as ground track information.

Additionally, the navigation computer produces discrete control signals for automatic control of the complete navigation system.
The navigation computer is provided with 28.5 VDC, 115 VAC and 3 phases 36 VAC.

To reduce required computational capacity, a relative coordinate system, restricted in latitude and longitude, is used. The zero point of the system is in the lower left corner. Required navigation points are entered into the navigation computer via the navigation computer programming panel, located on the left side of the nose section.

Two different types of navigation computers are available, CWU A-340-071M version 2204 and version 2205.

Computational capability of the CWU version 2204 is restricted in latitude and longitude to an area of 36° and one set of coordinates, while the CWU version 2205 is restricted to an area of 40° and two sets of coordinates.

OPERATING MODES

Four operating modes are possible. Normal operating mode is dead reckoning with TACAN update.

DEAD RECKONING Mode

This mode is available after fast alignment, only analog integration of both gyro platforms is performed. Accuracy is minimal: 4 % of distance traveled per hour of circular error probability, 1.5° precession per hour. However, since no aerological wind information data are processed, large computational errors may be present.

INERTIAL NAVIGATION Mode

The inertial navigation (IN) mode is available after long alignment with the main gyro platform. Since precise alignment was performed, maximum error is 4.5 NM per hour of circular error probability, precession 1° per hour.

UPDATE Mode

Automatic continuous navigation computer update is available in both operating modes. System inaccuracy is reduced to ±0,2 NM +0,1 % of distance to the station used for update. Three individual TACAN stations can be programmed prior flight for update purposes. The channel select switch must be placed to AUTO, the REL BEARING switch to RSBN, and the landing switch to OFF. TACAN is selected on the RH console. The CORR light illuminates if proper signals are received and automatic update is performed. For update with a TACAN station, the programmed station must be selected on the TACAN control panel additionally.

Visual Update

The navigation computer can be manually updated by pressing the COURSE CMPTR ZERO illuminated pushbutton and releasing it upon overflight of a selected and programmed waypoint (WP) or airfield.

NOTE

Update is not possible if a discrepancy of more than 21 NM exists between the present position computed and the actual position.

NAVIGATIONAL OPTIONS

Various navigational options are operated according to the setting on the navigation control panel:

- Point-to-point navigation
- Return
- Landing approach
- Traffic reentry (missed approach)
- Manual station select

POINT-TO-POINT NAVIGATION

Six navigation points can be programmed and are selected by setting the WP-A/D switch and selection of one of the three WP-A/D illuminated pushbuttons.

The course to the selected coordinate is displayed by the course pointer and the course window, distance is displayed by the range indicator on the HSI and on the HUD. As the aircraft closes on the selected coordinate the D < 21 NM light illuminates. Passing the coordinate, the lost bearing indication will be shown until exceeding 3.2 NM distance outbound.

RETURN

Pressing the RETURN illuminated pushbutton provides bearing information to a lead point for the nearest 9.2 NM final intercept to the selected airfield, provided the correct landing direction is selected with the COURSE switch and the update function is operating. Slant range is indicated to the selected A/D coordinates.

If automatic navigation computer update is inoperative, course and distance to the aerodrome reference point are provided.

During the approach, glide path information is displayed on the ADI for a 7° glide slope to the final intercept point at 3 700 ft AGL or QFE.

TRAFFIC RE-ENTRY (MISSED APPROACH)

If the missed approach button is pressed, the navigation computer supplies steering information for a traffic pattern, provided the landing select switch is off and TACAN correction is operative. Steering is provided for a 5.4 NM downwind leg and final intercept.

Pattern direction left or right hand is selected by placing the circle left-right switch to the corresponding direction. Glidepath information is displayed for a pattern altitude of 2 000 ft AGL or QFE.

MANUAL STATION SELECT

With the channels MAN. / AUTO switch in MAN., bearing and distance to a selected TACAN station is displayed. Navigation computer update is not provided.

NOTE

With the CHANNELS switch in manual, the final course must be dialed in on the HSI to receive steering commands.

NAVIGATION CONTROL PANEL

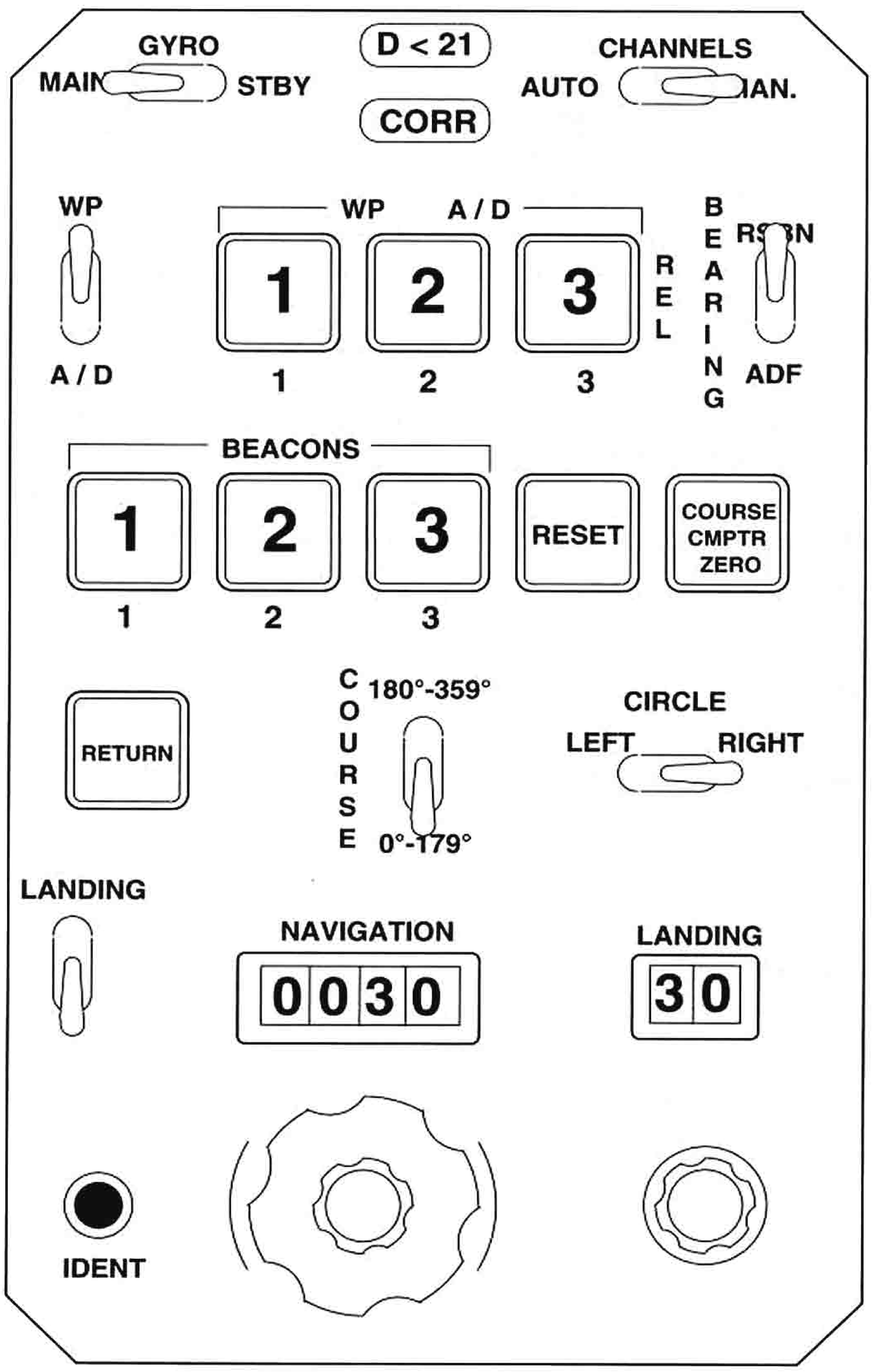


Figure 1-62

NAVIGATION CONTROL PANEL

GYRO Switch

A two position toggle switch, positions marked MAIN-STBY, selects the appropriate gyro system.

CHANNELS Switch

A two position toggle switch with following functions:

AUTO- Beacon selected for NAV system update

MAN. - NAV system not updated

WP-A/D Switch

A two position toggle switch, marked WP-A/D, selects appropriate function of the corresponding pushbuttons.

REL BEARING Switch

A two position toggle switch, positions marked RSBN-ADF, selects TCN / RSBN or ADF display on the HSI.

COURSE Switch

A two position toggle switch, positions marked 0 - 179° and 180 - 359°, selects appropriate hemisphere for RWY in use.

CIRCLE Switch

A two position toggle switch, positions marked LEFT-RIGHT, selects direction of LDG pattern.

LANDING Switch

A two position toggle switch to select the ILS if not automatically switched in.

IDENT Button

This button is not in use.

WP A/D Buttons

Three combined pushbutton and indicator lights, marked 1, 2, 3, to select a navigation point or an aerodrome.

BEACONS Buttons

Three combined pushbutton and indicator lights, marked 1, 2, 3, to select a beacon for NAV system update.

RESET Button

A combined pushbutton and indicator light to deselect the previously selected BEACON.

COURSE CMPTR ZERO Button

A combined pushbutton and indicator light to reset the NAV computer.

RETURN Button

A combined pushbutton and indicator light activates RETURN.

NAVIGATION Channel Window

Manually selected RSBN channel is displayed.

RSBN Channel Selector Knob

Selects desired RSBN channel in MAN.

LANDING Channel Window

Manually selected ILS channel is displayed.

ILS Selector Knob

Selects desired ILS channel.

NAVIGATION COMPUTER PROGRAMMING

The navigation computer programming panel is located on the lower left side of the nose section. Refer to figure 1-62A. It is used to program following data:

- Reference latitude for the coordinate square
- Relative coordinates for 3 airfields
- Relative coordinates for 3 waypoints
- Relative coordinates for 3 beacons
- RWY direction for 3 airfields between 0° to 179.9°
- Heading for 4 visual heading reference points (not in use)
- Reference heading for parking spot (not in use)
- 3 RSBN channels
- 3 ILS channels.

A second set of data has to be programmed with the computer CWU version 2205, restricted to a 40° area, however, it cannot be selected from the cockpit.

Programming is performed with an eight-digit code, consisting of a two-digit address number, a single-digit prefix number and the five-digit programming code. Refer to figure 1-62B.

REFERENCE LATITUDE

Reference latitude has to be entered into the computer to adjust the relative coordinate system. In the Northern hemisphere, the lateral geographical coordinate complemented by three zeros. In the Southern hemisphere, the lateral geographical coordinates is subtracted from 360 and the result complemented by two zeros.

HEADINGS

The programming code for headings consists of the geographical heading with the arc minutes and arc seconds expressed in hundredth of degrees.
For programming of runway directions, the value between 0.00 and 179.99 has to be selected.

COORDINATES

For programming of coordinates, the relative coordinate is calculated to an accuracy of four decimals, rounded to the third decimal and expressed as an unfragmented number.

The relative coordinates are obtained subtracting the reference coordinates of the relative coordinate system from the geographical coordinates.

Sample problem:

Geographical coordinate: 54°20'22"N, 07°34'56"E.
Reference coordinate for the relative coordinate system: 30°N, 10°W.

Latitude:	Longitude:
54°20'22"N	07°34'56"E
- 30°00'00"N	- 10°00'00"W
24°20'22"	17°34'56"

To receive the five-digit programming code, the relative coordinate has to be converted to a decimal number by the following formula:

$$\text{arc degrees} + \frac{\text{arc min}}{60} + \frac{\text{arc sec}}{3600} = \text{five - digit code}$$

Sample relative latitude:

$$24 + 0.\overline{3333} + 0.006\bar{1} = 24.3394$$

Five-digit code 24339

Sample relative longitude:

$$17 + 0.56\overline{66} + 0.01\overline{55} = 17.5822$$

Five-digit code 17582

The five-digit code has to be combined with the address and the prefix number listed in figure 1-62B.

Sample:

Waypoint 3, relative latitude	24°20'22"
Address number for waypoint 3 latitude	41
Prefix number	0
Five-digit code for 24°20'22"	24339
Programming code	41024339

RSBN / ILS CHANNELS

RSBN and ILS channels are entered in coded format for navigation computer update. Since RSBN and ILS are presently not in use, the appropriate code list is not published in GAF T.O. 1F-MIG29-1. However, a random code has still to be entered to permit nav computer update by use of TACAN stations.

ENTERING DATA SET

To enter the data set, DC and AC power must be available and the NAVIGATION switch on the system power panel has to be switched to the ON position.

The navigation computer programming panel is switched on with the ВКЛ ОТКЛ (ON) button.

Illumination of the data input display indicates system readiness.

Pressing the СБРОС (RESET) pushbutton on the keyboard sets the data input display to zero.

Before entering the data set in random sequence, the opening code 77127777 has to be entered. After entering each eight-digit code, the ЗАЛ (ENTER) button has to be pressed for data input.

After resetting the system with the СБРОС (RESET) pushbutton, the next code can be entered. The СБРОС (RESET) pushbutton can be used to delete the displayed code in case of an input failure. After input of the complete data set, the closing code 77139999 has to be entered to complete data input.

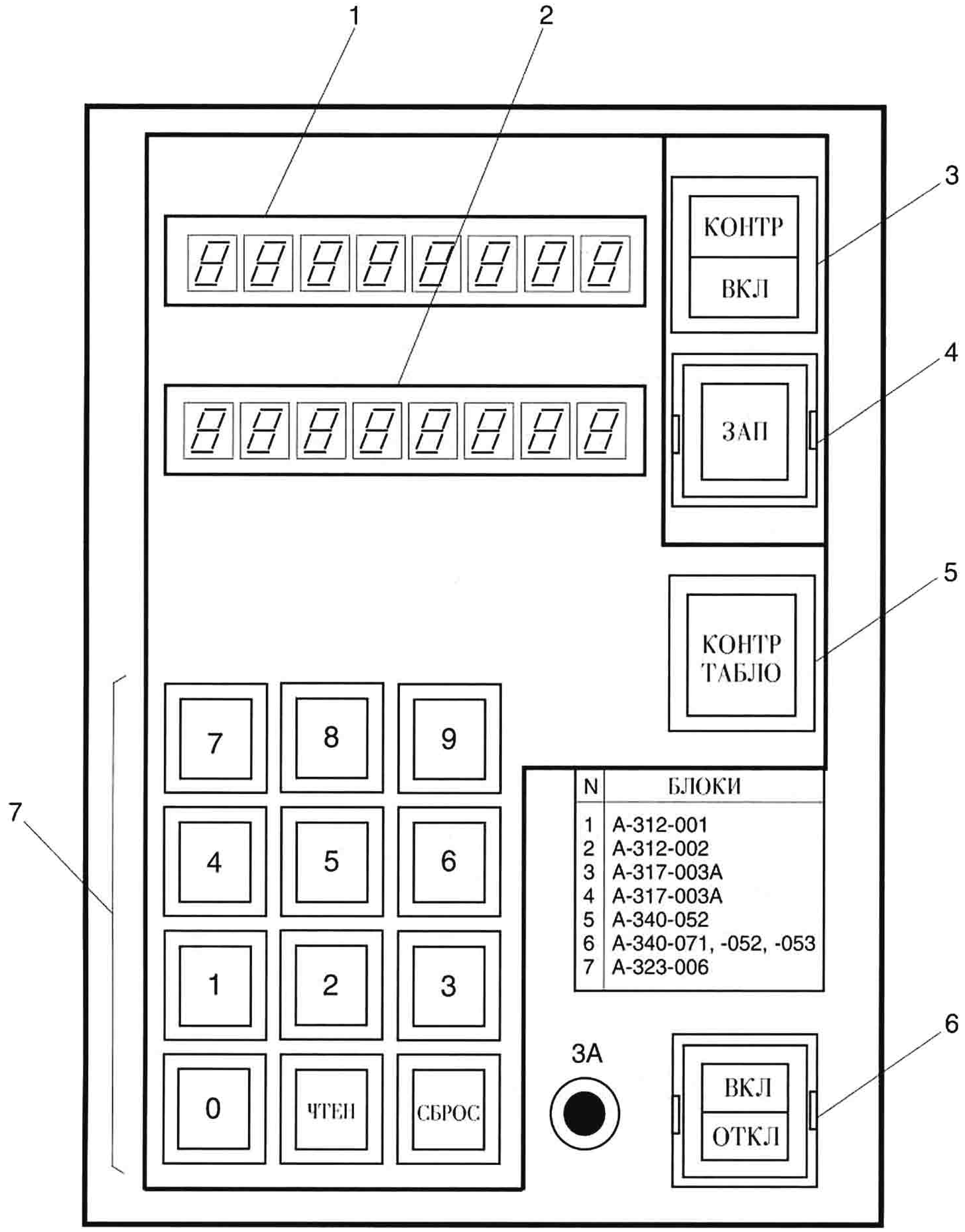
When the computer CWU version 2205 is in use, the code 25100001 has to be entered after the opening code for input of the first data set, and code 25100002 has to be used for input the second data set. The same code is used for switching sets between two sorties.

The ЧТЕН (READOUT) button can be used to check out valid codes. After entering the two-digit address number and the prefix number and pushing the ЧТЕН button, the valid code is displayed on the data input display.

INDICATIONS AND WARNINGS

	INDICATION	FAULT / EFFECT
MASTER CAUTION	○ LIGHT FLASHING	
AEKRAN	NAVIG COMPUTER	No NAV system update.

NAVIGATION COMPUTER PROGRAMMING PANEL



1. DATA INPUT DISPLAY
2. REPEATER DISPLAY
3. BIT
4. ENTER BUTTON
5. LAMP TEST BUTTON
6. ON / OFF BUTTON
7. KEYBOARD

Figure 1-62A

ADDRESS AND PREFIX NUMBERS

Reference		Address number	Prefix	Code
Reference latitude		07	0	latitude complemented by zero's
A/D 1	relative latitude	11	0	thousands of degrees
A/D 1	relative longitude	12	0	"
A/D 2	relative latitude	31	0	"
A/D 2	relative longitude	32	0	"
A/D 3	relative latitude	51	0	"
A/D 3	relative longitude	52	0	"
WP 1	relative latitude	01	0	"
WP 1	relative longitude	02	0	"
WP 2	relative latitude	21	0	"
WP 2	relative longitude	22	0	"
WP 3	relative latitude	41	0	"
WP 3	relative longitude	42	0	"
BEACON 1	relative latitude	44	0	"
BEACON 1	relative longitude	14	0	"
BEACON 2	relative latitude	45	0	"
BEACON 2	relative longitude	15	0	"
BEACON 3	relative latitude	46	0	"
BEACON 3	relative longitude	16	0	"
RWY direction A/D 1	geographical heading	10	0	hundredth of degrees
RWY direction A/D 2	geographical heading	30	0	"
RWY direction A/D 3	geographical heading	59	0	"
Heading reference 1	geographical heading	27	0	"
Heading reference 2	geographical heading	37	0	"
Heading reference 3	geographical heading	47	0	"
Heading reference 4	geographical heading	57	0	"
Parking position	geographical heading	55	0	"
BEACON 1 channel	code	04	1	not published in GAF T.O. 1F-MIG29-1
BEACON 2 channel	code	05	1	
BEACON 3 channel	code	06	1	
ILS 1 channel	code	13	1	
ILS 2 channel	code	33	1	
ILS 3 channel	code	53	1	

Figure 1-62B

TACAN

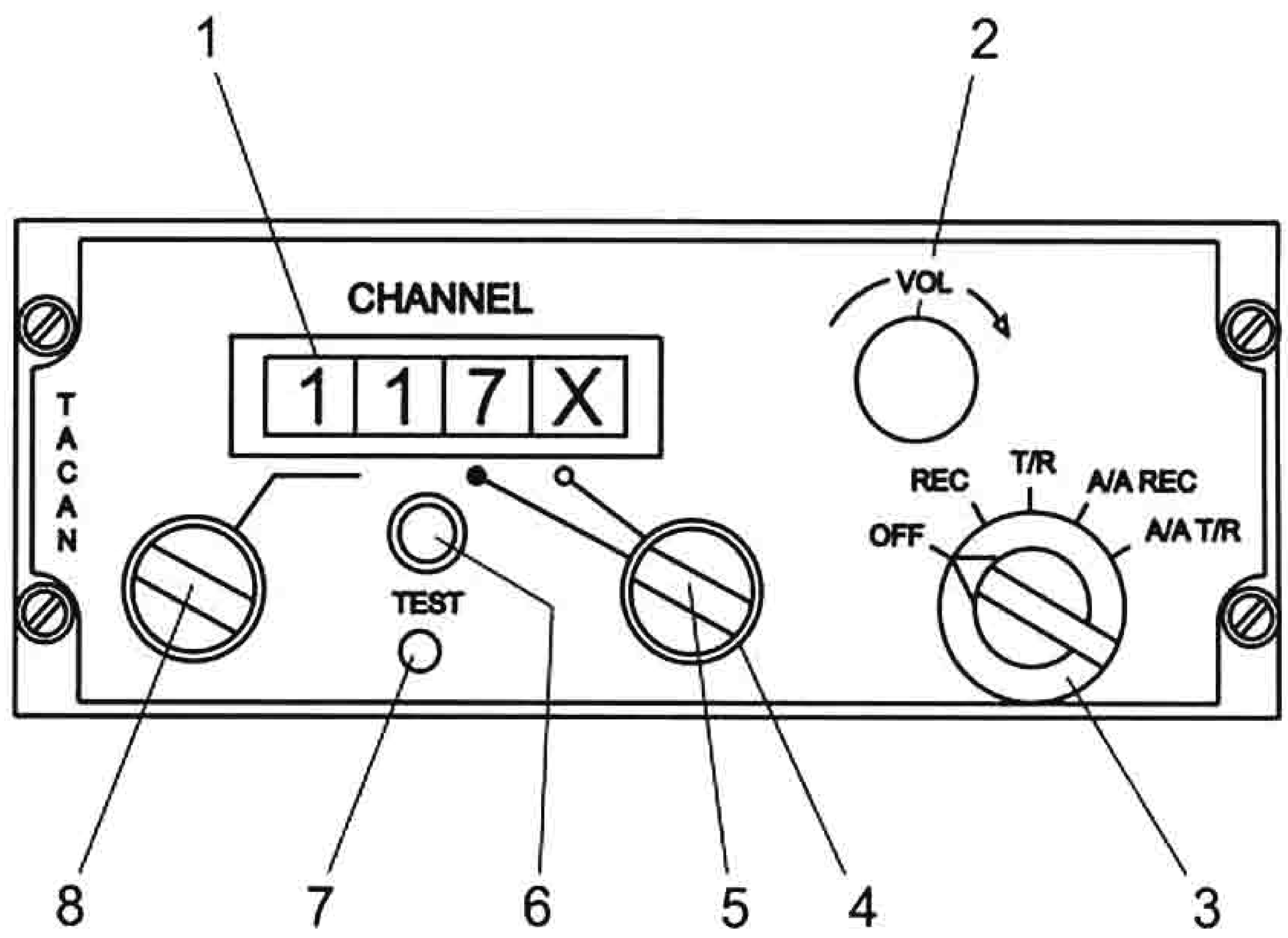
The TACAN system provides magnetic bearing and slant range to the selected ground station, or to a suitably equipped cooperating aircraft (air-to-air). A suitably equipped cooperating aircraft is one equipped with bearing transmitting equipment. The TACAN system can transmit only distance information when interrogated in the A/A, T/R mode. The TACAN system determines the identity of the transmitting station and indicates the dependability of the transmitted signal.

The system utilizes radio navigation frequencies, the propagation of which is virtually limited to line of sight distances. In case of co-channel interference in T/R mode, the interfering channel identifier is garbled. When a temporary loss of signals occurs, a memory keeps range tracking for 15 seconds and bearing tracking for 3 seconds. The TACAN automatically self-tests after a temporary signal loss and displays its status on the control panel. The TACAN is supplied with 115 VAC by the AC generator or the PTO.

TACAN ANTENNAS

The TACAN system uses an upper and lower antenna to receive TACAN signals. Antenna switching to the antenna with greatest signal strength is automatic. For antenna location, refer to figure 1-56.

TACAN CONTROLS AND INDICATORS



1. NAV CHANNEL WINDOW
2. VOLUME CONTROL
3. TACAN FUNCTION SELECTOR KNOB
4. CHANNEL SELECTOR CONTROL KNOB (X/Y SELECTION)
5. CHANNEL SELECTOR CONTROL KNOB (UNIT DIGIT SELECTION)
6. TEST INDICATOR LIGHT
7. TEST BUTTON
8. CHANNEL SELECTOR CONTROL KNOB

Figure 1-63

REL BEARING Switch

A two position toggle switch on the NAV panel, selects the appropriate system for navigational display.

TACAN Function Selector Knob

OFF	Power disconnected
REC	In the receive mode, no interrogation pulse is transmitted. The bearing identification of a selected ground station are received. Range information is not available.
T/R	In the transmit / receive mode, the transmitter and receiver are active, generating range and bearing information from a selected ground station.
A/A REC	In the A/A receive mode, the system receives and decodes bearing information from a suitably equipped cooperating aircraft. The channel of the receiving aircraft must be either 63 channels above or 63 channels below the suitably equipped, cooperating aircraft channel but must be within the 1 through 126 X or Y channel range. Both aircraft must be either in the X or Y channel.
A/A T/R	In the A/A transmit / receive mode, the TACAN system interrogates a reference aircraft and the slant range to the cooperating aircraft is displayed, see A/A REC above for channel selection. In this mode, the TACAN system provides distance replies to other aircraft when interrogated. Bearing to a suitably equipped cooperating aircraft is also displayed. The TACAN AN / ARN-118 system can transmit only distance information when interrogated in the A/A, T/R mode. The maximum number of aircraft to receive range information simultaneously is limited to 5.

NOTE

- In the air-to-air modes, to prevent possible interference from IFF or transponder signals, channels 1 thru 11, 58 thru 74, and 121 thru 126 should not be used.
- To reduce the possibility of DME interference, the use of Y channels is recommended if the suitably equipped, cooperating aircraft is equipped with Y channel capabilities.

Channel Selector Control

The two rotary knobs are used to set the desired TACAN channel. The left knob selects the tens and hundreds digits of the operating channel. The right knob selects the units digits of the operating channel and contains an outer knob which selects the X or Y channel. Placing the knob to X provides capability for 126 channel operation. Placing the knob to Y adds an additional 126 channel capability to the TACAN system. The dial system is numbered 0 to 129, each number from 1 to 126 represents a specific pair (transmitting and receiving) of frequencies. Number 0, 127, 128 and 129 on the channel dial are not usable.

Channel Window

The selected TACAN channel is displayed on the NAV channel window, followed by a X or Y for the corresponding channel.

Navigation Volume Control Knob

The VOL control knob controls the volume of the audio identification signal received from the transmitting station.

TACAN Test Button

The TACAN test button may be used to test the TACAN as follows:

- TACAN / RSBN switch - TACAN
- REL BEARING switch - RSBN
- Function selector knob - T/R
- Allow 90 sec for warm-up
- TACAN test button - press and release

Observe the following:

- Test indicator light flashes momentarily
- (HSI) Distance indicates 0.0 ± 0.5 miles (-0.5 miles indicated as 399.5 miles)
- Bearing pointer slew to 180° ± 3°
- After about 15 sec normal TACAN lock-on

This test does not check the antenna interface. TACAN accuracy should be checked against a ground check point.

Automatic Self-Test

If the TACAN bearing signal becomes lost or is unreliable, the TACAN system switches to automatic self-test. The indications of the self-test correspond to the manual test. If there is a detected malfunction in the system, the test indicator light comes on steady. If the indicator light illuminates steady during the test cycle in both the T/R and REC modes, the bearing and distance information on the HSI is invalid. The self-test can be terminated any time by turning either the TACAN function selector knob or any of the channel selector knobs.

Inflight Confidence Test

If TACAN indicator readouts become suspect during flight, perform an inflight confidence self-test of the TACAN system by setting the TACAN function selector knob to T/R and then pressing the TACAN test button, the test indicator light flashes momentarily. If the test indicator light illuminates steady during the test cycle in both the T/R and REC modes, the bearing and distance information are invalid. If the test indicator light comes on in the T/R mode but not in the REC mode, the distance information is invalid and the bearing on the HSI is valid.

ADAPTER ASSEMBLY TACAN / RSBN

The adapter assembly TACAN / RSBN converts TACAN signals to match NAV system requirements. Since RSBN inputs to the NAV system, refer to TRUE NORTH, TACAN signals must be corrected for variation to simulate RSBN signals. DC power is required for operation.

Adaptation is limited to three programmed TACAN stations selected with the BEACON buttons on the navigation panel, CHANNEL select switch in AUTO.

Requirements for adapter assembly operation are:

TACAN / RSBN switch	TACAN
CHANNEL select	AUTO
BEACON	1, 2 or 3 select
TACAN control panel	Channel selected corresponding to the beacon.

CONTROLS AND INDICATORS

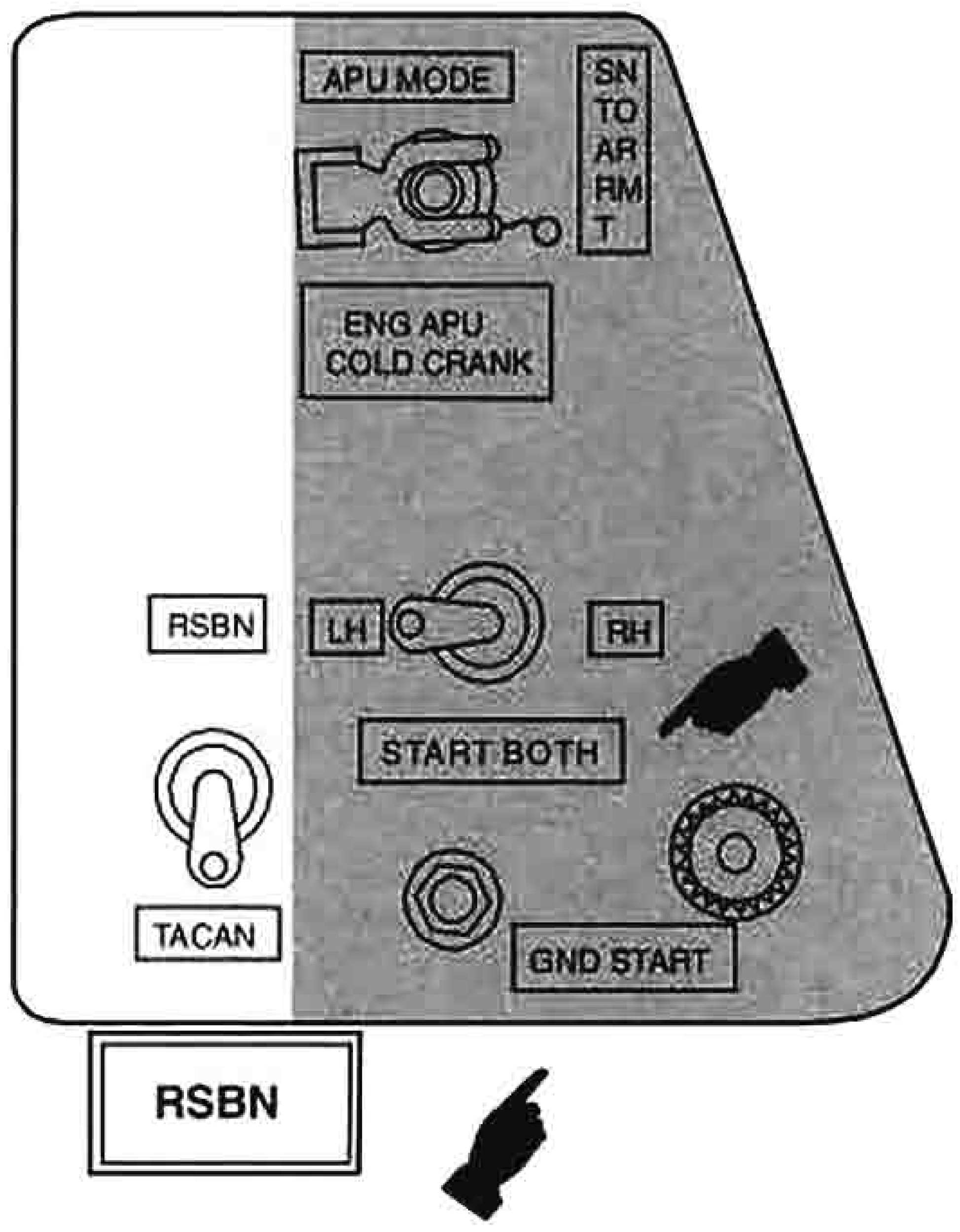


Figure 1-64

TACAN / RSBN Switch

A two position toggle switch on the RH console, selects the appropriate system as reference for the navigation system.

RSBN Light

The light illuminates when RSBN is selected.

AUTOMATIC DIRECTION FINDING

The ADF provides direction finding or radio monitoring in the HF frequency band (150 to 1299.5 kHz). Eight different stations can be channelized for inflight use. However, a standard setting is normally programmed.

The system is powered with 28.5 VDC and 36 VAC from a transformer. Battery and DC / AC converter supply the system with power in case of a generator failure.

ADF CONTROLS AND INDICATORS

The controls consist of the ADF select switch on the radio panel, the REL BEARING / ADF switch on the navigation panel, the INNER / OUTER switch and of the BEACON INNER light on the instrument panel above the radar altimeter, as well as of the ADF channel select knob, the VOICE / CW switch, the COMPASS / ANTENNA select switch, VOL control knob and LOOP pushbutton on the ADF panel.

GT: The INNER / OUTER switch and the BEACON INNER light are located on the LH side wall.

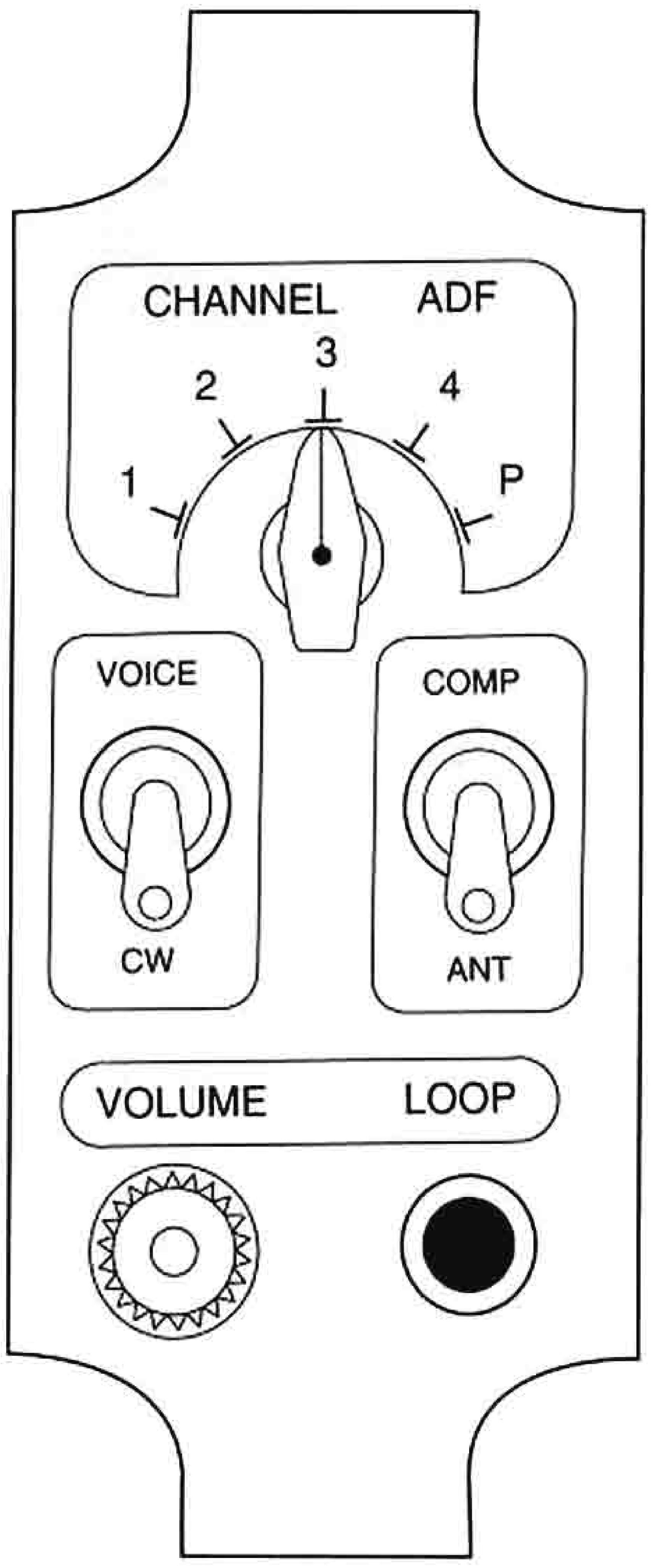


Figure 1-65

ADF Operation

The ADF operation is as follows:

- Set the ADF select switch (radio panel) to off
- Set the RSBN / ADF select switch (navigation panel) to ADF
- Set the COMPASS / ANTENNA select switch to COMP
- Set the INNER / OUTER select switch to the desired position (according to the required channel), check that the BEACON inner light is illuminated if INNER is selected
- Select desired channel with the ADF channel select knob
- Set the VOICE / CW switch as required

The ADF radio receiver operation is as follows:

- Set the ADF select switch (radio panel) to ADF
- Set the RSBN / ADF select switch (navigation panel) to ADF
- Set the COMPASS / ANTENNA switch to ANTENNA
- Select the desired channel (INNER / OUTER switch and ADF channel select knob)
- Set the VOICE / CW switch as required

ADF Self-Test

To initiate the self test:

- Set the RSBN / ADF select switch to ADF.
- Set the ADF channel select knob to position P. Check bearing pointer rotating to approximately 195°.
- Set the VOICE / CW switch to CW.
  Check for 800 Hz tone transmission.

LOOP Pushbutton

To manually rotate the ADF loop antenna press the LOOP button. The antenna will return to its original position upon release.

RADIO MARKER RECEIVER

The radio marker receiver indicates the overflight of a marker beacon.

Frequency is 75 MHz in accordance with ICAO standards.

Overflight of a marker beacon is indicated by illumination of the MARKER BEACON light on the TLP and by an audio signal of 3 000 Hz.

The marker beacon receiver is operated by 28.5 VDC from the DC generator or battery power. It is switched on by placing the ACFT SYS switch to the ON position.

NOTE

The marker beacon receiver will switch the ADF from a selected OUTER channel to INNER channel when passing a marker beacon within 15° of the final course, provided the gear is down.

IFF EQUIPMENT

The identification system STR 700 provides automatic identification of aircraft in which it is installed when challenged by surface or airborne IFF interrogation sets. It identifies the aircraft position momentarily upon request, reports the altitude of the aircraft and indicates an emergency. Altitude is given from the air data computer. In operation, the identification system receives coded interrogation signals and transmits coded response signals to the source of challenging. Five modes of operation are provided for interrogation and response to interrogation signals.

Mode 1	Security identification
Mode 2	Personal identification
Mode 3/A	Traffic identification
Mode C	Altitude reporting
Mode 4	Crypto identification

The codes for mode 1 (00-73) and 3/A can be set in the cockpit during flight, but the code for mode 2 must be set on the ground. Mode 2 and mode 3/A can be set from code 0 000 to 7 777. When mode C is selected, coded altitude information from the altitude encoder is applied to the IFF system for reply to mode C interrogation. The code represents aircraft altitude. There are no provisions to manually set mode C code.

Failure of mode 4 reply is indicated by an optical and an audio signal. The system is supplied with 28.5 VDC and 115 VAC, 400 Hz for the altitude encoder.

IFF CONTROL PANEL

The IFF control panel is located on the right console (GT: front cockpit only).The controls on the IFF panel are shown in figure 1-66. There is also a MODE 4 light on the TLP.

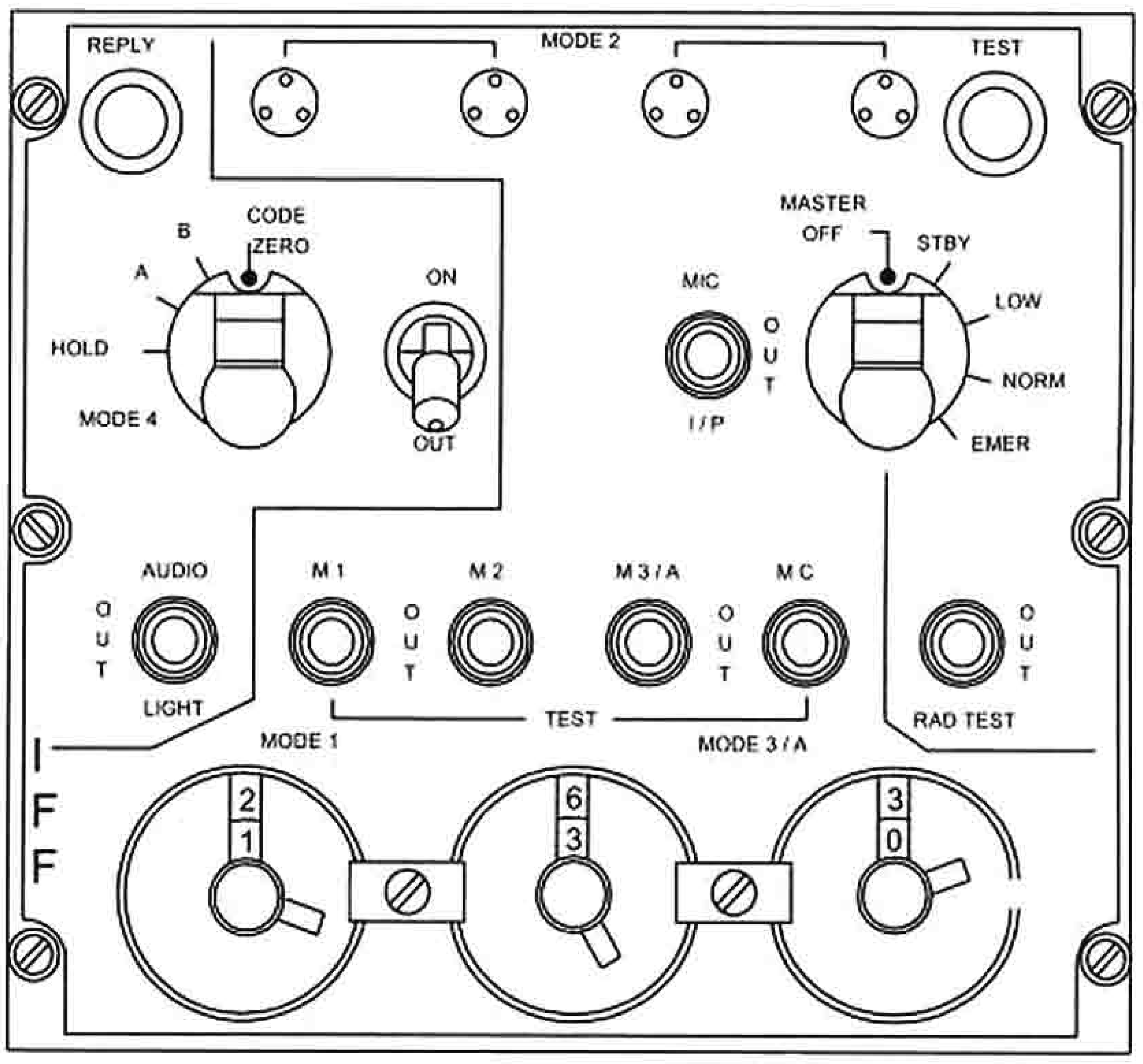


Figure 1-66

Master Function Selector Knob

A five-position rotary knob, marked MASTER, has the following functions:

OFF — Power disconnected.

STBY — Full power supplied to the system, but with interrogation replies blocked.

LOW — System operates with reduced sensitivity. However, transmitted power from the transponder is the same for both the LOW and NORM positions.

NORM — System operates at full sensitivity.

EMER — Allows the system to respond interrogations in modes 1, 2, 3/A and C. The reply for modes 1 and 2 is a special emergency signal of the codes selected on the applicable dials, while mode 3/A replies a special emergency signal of code 7 700, regardless of the selected code.

MODE Selector Switches

The four mode selector switches are three-position toggle switches for modes 1, 2, 3/A and C and control the operations as follows:

M1 - MC — Enables respective mode (M1, M2, M3/A, MC)

OUT — Disables respective mode.

TEST — Self-test position. TEST position is springloaded to the center position. The green TEST light illuminates when the respective mode is operating.

Test Light

The test light is a green press-to-test light which illuminates when the BITE check performs satisfactorily.

Position Identification Switch

The position identification switch is a three-position toggle switch used by the AC to provide momentary identification of position upon request.

I/P — Enables the system to respond with identification of position replies in modes 1, 2, 3/A and C. The response is continued for a 30 ±10 sec duration after the switch has been released.

OUT — Disables identification of position capability.

MIC — Same as positioning the switch to I/P, except that the microphone button must be pressed.

Radiation Test Switch

The radiation test switch is a two-position toggle switch used for the radiation test. The positions are OUT and RAD TEST. This switch is presently inoperative.

Mode 1 Code Selector

A rotary knob, incorporating two concentric wheels, marked MODE 1 for setting one of 32 available 2-digit codes.

Mode 2 Code Selectors

The four mode 2 code selectors are used to set the code for mode 2 operation. Each switch can be set from 0 to 7. The knob is removed upon completion of the setting on the ground.

Mode 3/A Code Selector

Two identical rotary knobs, incorporating two concentric wheels, marked MODE 3/A for setting one of 4 096 available 4-digit codes.

Mode 4 Selector Switch

ON — Enables replies to mode 4 interrogations.

OUT — Disables mode 4 replies.

Mode 4 Function Knob

This knob operates in conjunction with the master function selector knob.

A four-position rotary knob, marked MODE 4, has the following functions:

A	Interrogations from an interrogator using code A are answered.
B	Interrogations from an interrogator using code B are answered.
ZERO	To erase mode 4 code settings, provided the MASTER function selector knob is not in the OFF position. Both mode 4 codes are automatically erased after landing when the IFF system is switched off, except if the mode 4 function switch is placed to HOLD.
HOLD	This position may be used to retain mode 4 code settings, if another flight is anticipated during the coded period. HOLD must be selected at least 15 seconds before system shut-down.

Mode 4 Monitor Switch

A toggle switch with the positions AUDIO / OUT / LIGHT which enables or disables the audio / light monitor indications.

AUDIO	Interrogation and reply of mode 4, is indicated by an audio signal and the lights REPLY and IFF.
OUT	Disables all mode 4 audio / light indications.
LIGHT	Audio disabled, mode 4 interrogation is indicated by the lights REPLY and IFF only.

Mode 4 Reply Indicator Light

The green mode 4 reply indicator illuminates when mode 4 replies are transmitted, provided that the mode 4 monitor switch is in AUDIO or LIGHT. Filament press-to-test, turn-to-dim features are incorporated into the indicator.

IFF Mode 4 Warning Indicator Caption

A green MODE 4 caption is installed on the TLP. The indicator illuminates if the aircraft is interrogated by a valid mode 4 signal, but does not reply.

IFF NORMAL OPERATION

BIT Check

Prior to each flight the following check should be performed with the built-in test equipment (BITE). This will assure that the transponder system is working properly.

Test light (press-to-test) - CHECK
Master knob - STBY
(allow 2 minutes warm-up time)

Master knob - NORM
Mode 1 selector switch - TEST and hold.
Test light should illuminate. If not, the selected mode is at fault.

Test mode 1, 3/A and C respectively.

Before Takeoff

Master knob - NORM
BIT check - COMPLETED
Perform the test according to the requests of the interrogating ground station.

The above check evaluates the IFF / SIF system including the antennas.

ARMAMENT SYSTEM

The armament system consists of:

- Fire control system
- Radar system
- IRSTS / LRF
- Associated weapons

It is described in GAF T.O. 1F-MIG29-34-1.

DEFENSIVE AIDS SUBSYSTEM

The defensive aids subsystem (DASS) consist of:

- RHAW system
- FLARE dispenser

It is described in GAF T.O. 1F-MIG29-34-1.

EXTERNAL STORES

A variety of stores can be carried on the seven external stations. For stores configurations and limitations refer to section 5 of this manual. For information on armament and ECM equipment loading and operation refer to the applicable weapons delivery manual.

WARNING AND INFORMATION EQUIPMENT

To keep instruments cross-check to a minimum, warning and indicator lights are incorporated throughout the cockpit. Additional voice warning is provided for abnormal conditions.
The warning equipment consists of three independent systems:

- TLP
- AEKRAN
- VIWAS

A MASTER CAUTION light flashes whenever a warning light illuminates on the TLP (red lights) or an AEKRAN warning signal is displayed.

NOTE

All warning equipment is operated with 28.5 VDC from the generator or battery power. Refer to figure 1-67.

MASTER CAUTION LIGHT

The MASTER CAUTION light is located on the instrument panel. Whenever a warning signal is displayed on the TLP or the AEKRAN, the MASTER CAUTION light starts flashing. Brightness of the light can be adjusted by rotating the light case. Pressing the MASTER CAUTION light extinguishes the light, warnings on the TLP turn steady, AEKRAN displays are not affected.

TELELIGHT PANEL

The TLP provides immediate warning of the existence of an abnormal condition, which could affect the safety of the aircraft (red lights). Additional information lights (green) indicate system operation or condition. Refer to section 3.

A malfunction is indicated by a flashing warning light in conjunction with the MASTER CAUTION light flashing. After reset of the MASTER CAUTION light, the warning light illuminates steady until the problem is solved.

AFTER MODIFICATION WITH WING DROP TANKS

Three previous spare captions have been modified to read WDT NO PRESS, WDT TEST and FEEL UNIT OK.

The WDT NO PRESS caption illuminates prior to engine start when wing drop tanks are installed or during a wing drop tank pressurization failure. After engine start the caption extinguishes, indicating wing drop tank pressurization.

Illumination of WDT TEST caption indicates a valid system check when the TEST WDT button on the control and test panel is pressed.

The FEEL UNIT OK caption replaces the identical caption removed from the control and test panel.

Warning Light Controls

A photodiode automatically adjusts the brightness of the lights according to environmental conditions. A rheostat on the lighting panel provides for manual adjustment of brightness.

WARNING AND INFORMATION EQUIPMENT

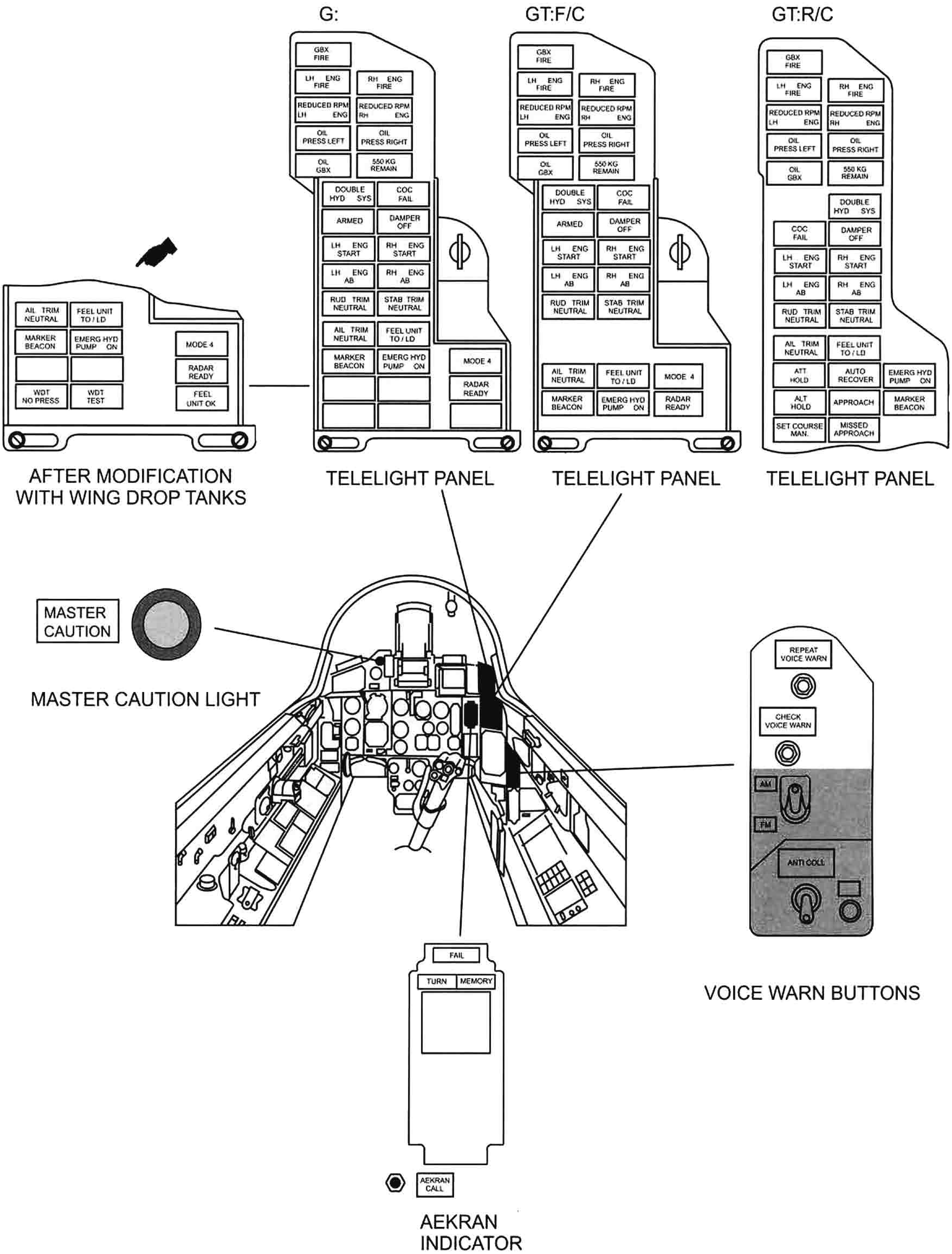


Figure 1-67

WEAPON AND ARMAMENT CONTROLS (FRONT SIDE)

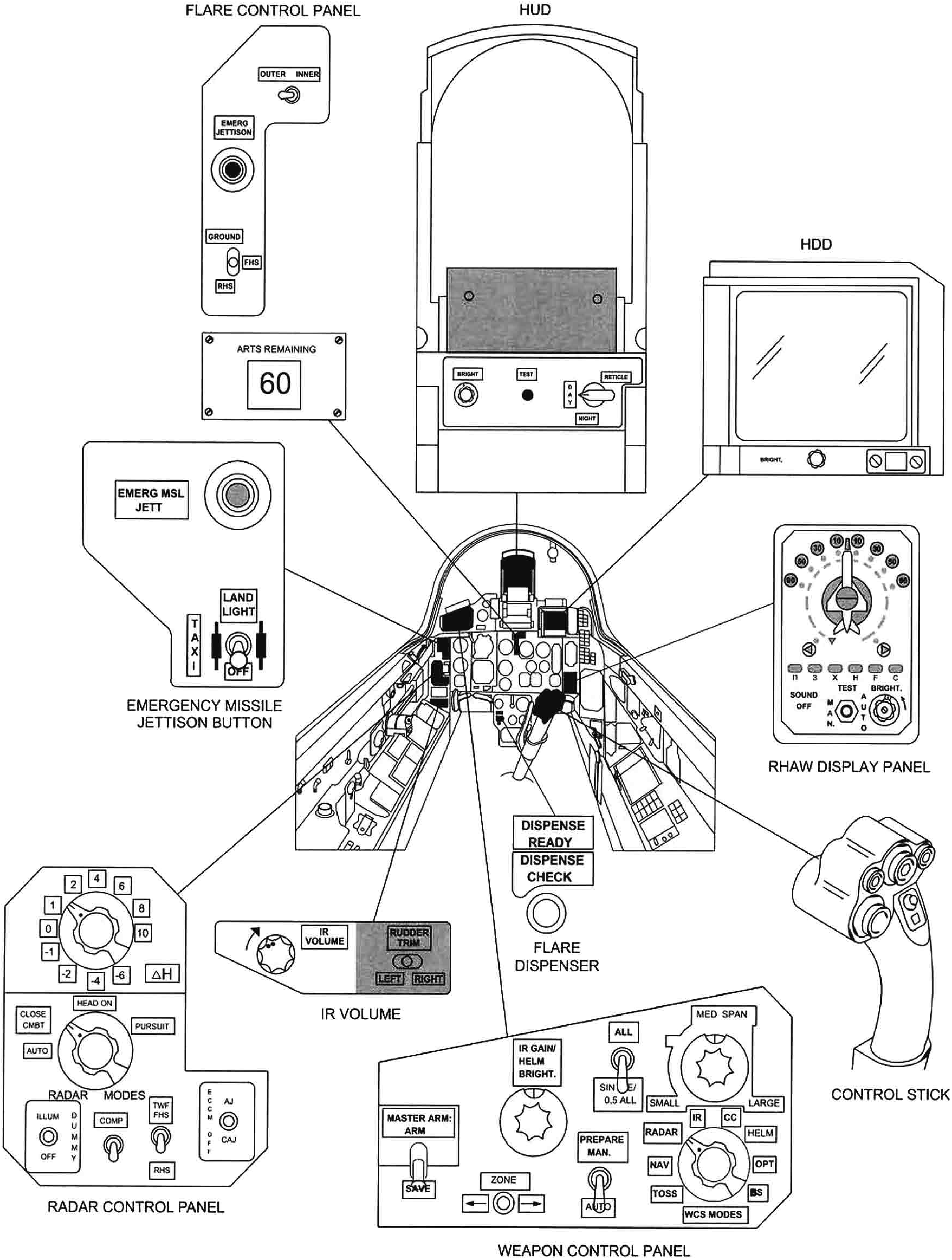


Figure 1-67A

WEAPON AND ARMAMENT CONTROLS (LH / RH SIDE)

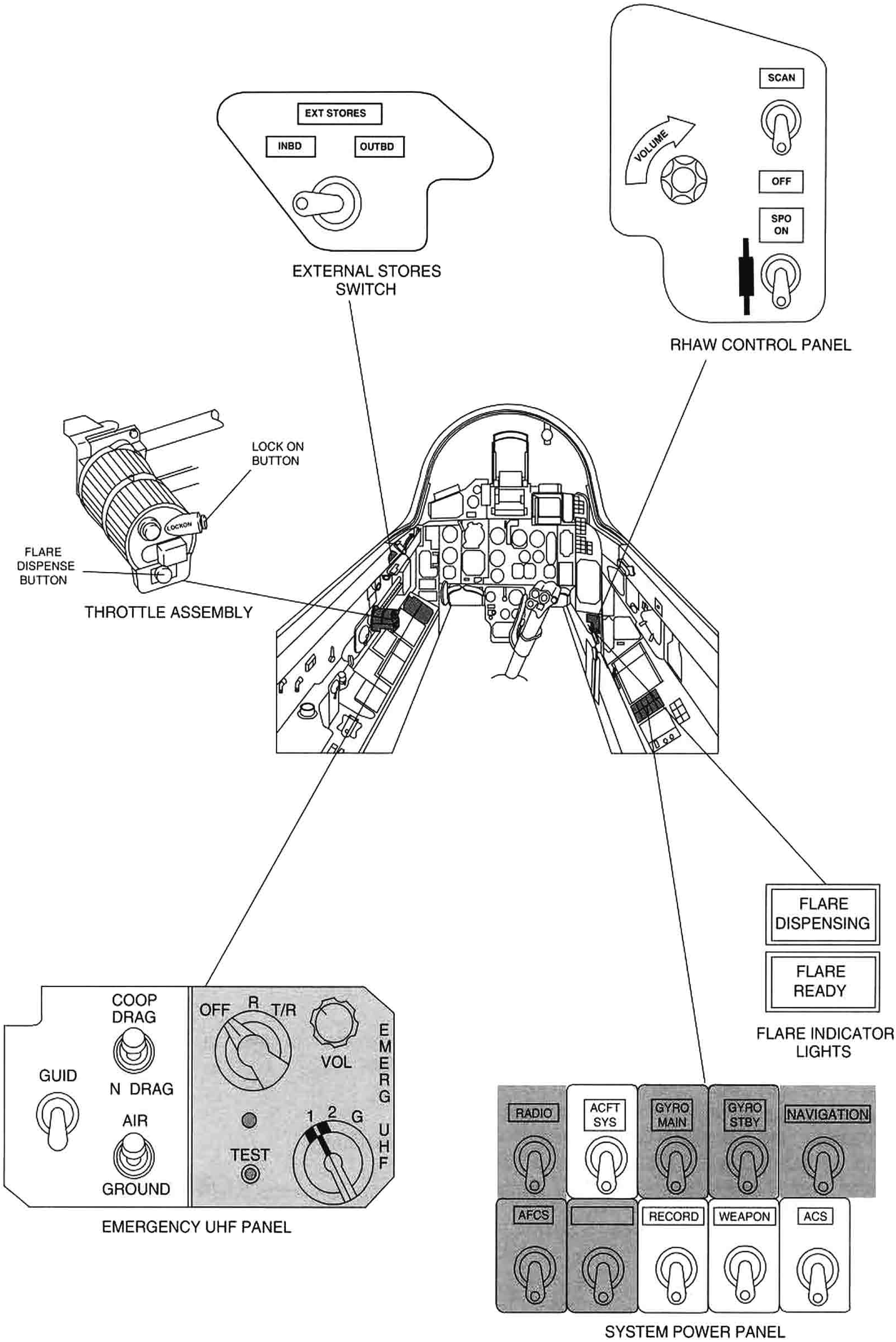


Figure 1-67B

AEKRAN

The AEKRAN system is part of the aircraft warning and recording system. It monitors and controls the operation of aircraft systems and self test equipment, and displays and records corresponding messages if malfunctions occur. After landing, additional information about aircraft systems, equipment and exceeding of limitations not relevant for the safety of flight and illumination of red warning lights are recorded by discrete signals.

AEKRAN OPERATION

The AEKRAN system is activated automatically when the NAVIGATION switch is placed to on. During engine start, both generators failure indication may be displayed until external power is disconnected.

After T/O, FLIGHT illuminates momentarily.

If a malfunction is detected, the corresponding message is displayed. Refer to figure 1-69. Simultaneously, the MASTER CAUTION light flashes and the VIWAS gives the appropriate message. If an appropriate message is not available, VIWAS signals: 'Check AEKRAN'.

The AEKRAN indication is displayed until the problem indicated is solved, a signal with higher priority is received or the AEKRAN CALL button is pressed. Pressing the AEKRAN CALL button extinguishes the indication and the MASTER CAUTION light however the signal is stored in the memory circuit indicated by a memory light.

If two or more systems fail, the turn light illuminates, the highest priority signal is displayed first. Depressing the AEKRAN CALL button stores the displayed signal in the memory circuit and permits display of the next signal in the priority sequence line. After all signals have been displayed, the turn light extinguishes. If required, the signals in the memory circuit can be repeated by pressing the AEKRAN CALL button. After landing, all stored signals are copied to a control slip, after engine shut-down, AC GEN is displayed on the AEKRAN.

SELF-TEST

Self-test is initiated by pushing the AEKRAN CALL button. The fail light must not illuminate. After 15 seconds, the SELF TEST followed by OK are displayed in the display window. A system failure may be indicated by a fail light or by a distorted SELF TEST and AEKRAN FAIL in the display window.

CONTROLS AND INDICATORS

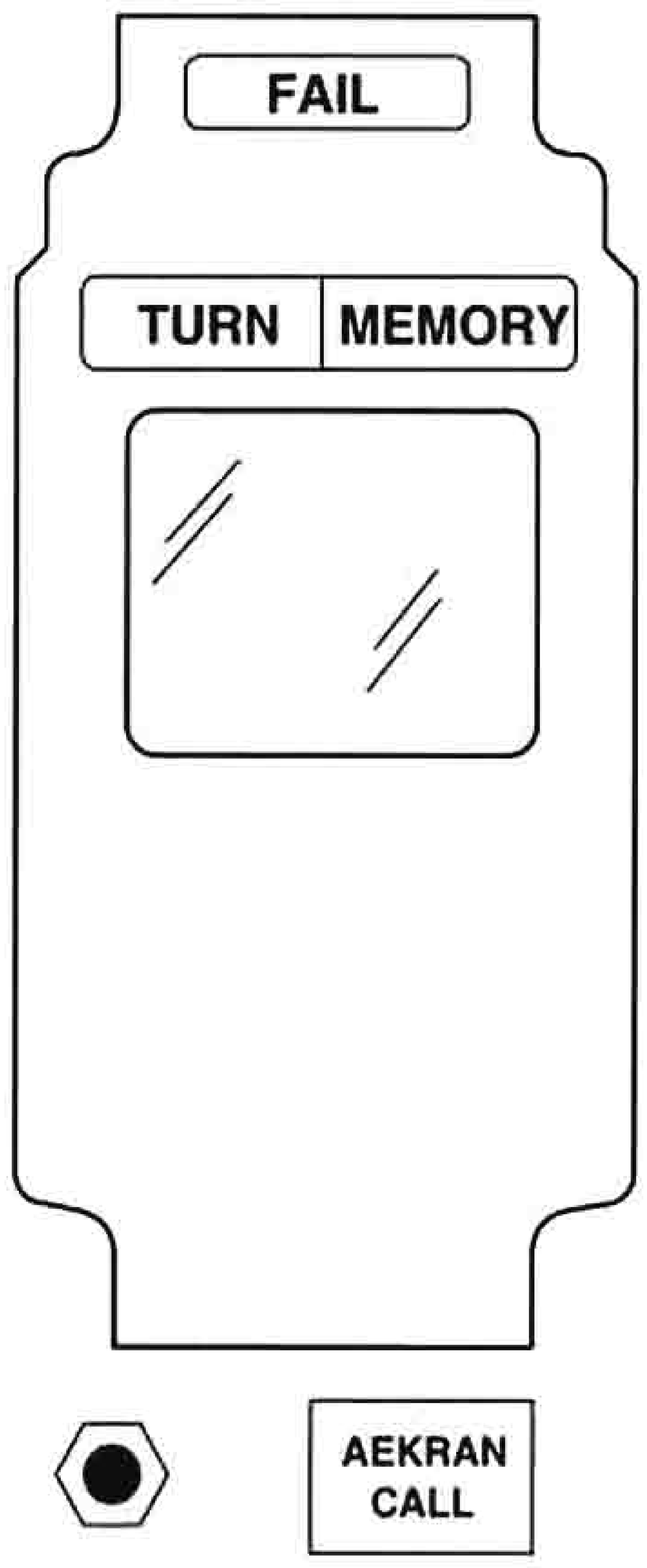


Figure 1-68

AEKRAN CALL Button

A triple use pushbutton initiates self-test, calls up signals in the priority sequence line and recalls signals in the memory circuit.

FAIL Light

Illumination of the fail light indicates a system failure.

TURN Light

Illumination of the TURN light indicates, that signals of lower priority are held in priority sequence line.

MEMORY Light

Illumination of the memory light indicates signals in the memory circuit.

AEKRAN INDICATIONS

PRIORITY	SIGNAL	PRIORITY	SIGNAL
1	START TURB CRIT CONDITNS	31	AC GEN
2	EXTEND LANDING GEAR (in air) L GEAR INDIK (on ground)	32	DISCON GEN DRIVE
3	ALT ALERT	33	LEFT AIR INTK
4	OVERHEAT LEFT	34	RIGHT AIR INTK
5	LEFT OVER SPEED	35	UPPER INLET
6	VIBR LEFT	36	LOCK CANOPY
7	FUEL PRESSURE LEFT	37	LEFT ENG STBY SYS
8	OIL PRESS LEFT	38	RIGHT ENG STBY SYS
9	OIL TEMP LEFT	39	AIR DATA SYS
10	CHIP LEFT	40	FEEL CONT UNIT
11	OVERHEAT RIGHT	41	FEEL UNIT SET EASY
12	RIGHT OVER SPEED	42	NO COOLING
13	VIBR RIGHT	43	SKIN OVERHEAT
14	FUEL PRESS RIGHT	44	COC 3 STOP
15	OIL PRESSURE RIGHT	45	EMERG OXYGEN RESERVE USE OXY (on ground)
16	OIL TEMP RIGHT	46	-
17	CHIP RIGHT	47	-
18	OIL PRESS ACCRY GBX	48	DROP TANK NO USAGE
19	ACFT ACCRY GBX VIBR	49	EXTEND FLAPS
20	TWO GENER WATCH TIME	50	LEAD EDGES NOT EXTEND
21	NO BOOST	51	NAVIG COMPUTER
22	(FUEL RETURN)	52	-
23	CHECK FUEL AMOUNT	53	RADAR NOT READY
24	CABIN LIMIT PRESS DESCEND	54	RADAR
25	BOOST HYD SYST	55	OPT SIGHT NAV SYS
26	MAIN HYD SYST	56	WEAPON CONT SYS (WCS)
27	TWO DIR VERT GYRO	57	GUN
28	MAIN DIR VERT GYRO	58	HELMET MOUNTED SIGHT
29	STBY DIR VERT GYRO	59	IR SEEKER
30	DC GEN WATCH TIME		

Figure 1-69

VOICE INFORMATION AND WARNING SYSTEM

The VIWAS provides voice warning to focus the pilots attention to a problem indicated on the warning light panel or AEKRAN. Depending on the type of emergency, advisory for initial action to be taken is added. The system is powered by 28.5 VDC.

In case of multiple malfunctions, the voice warnings are realized according to the priority list. Refer to section 3.

Warnings No. 1, 2, 3 and 5 are simultaneously transmitted over the radio.

Voice Warning Operation

The VIWAS is switched-on with the battery. Two pushbuttons are provided for operation on the RH console.

The CHECK VOICE WARN button is used to initiate a self-test. Pressing the REPEAT VOICE WARN button repeats the last warning.

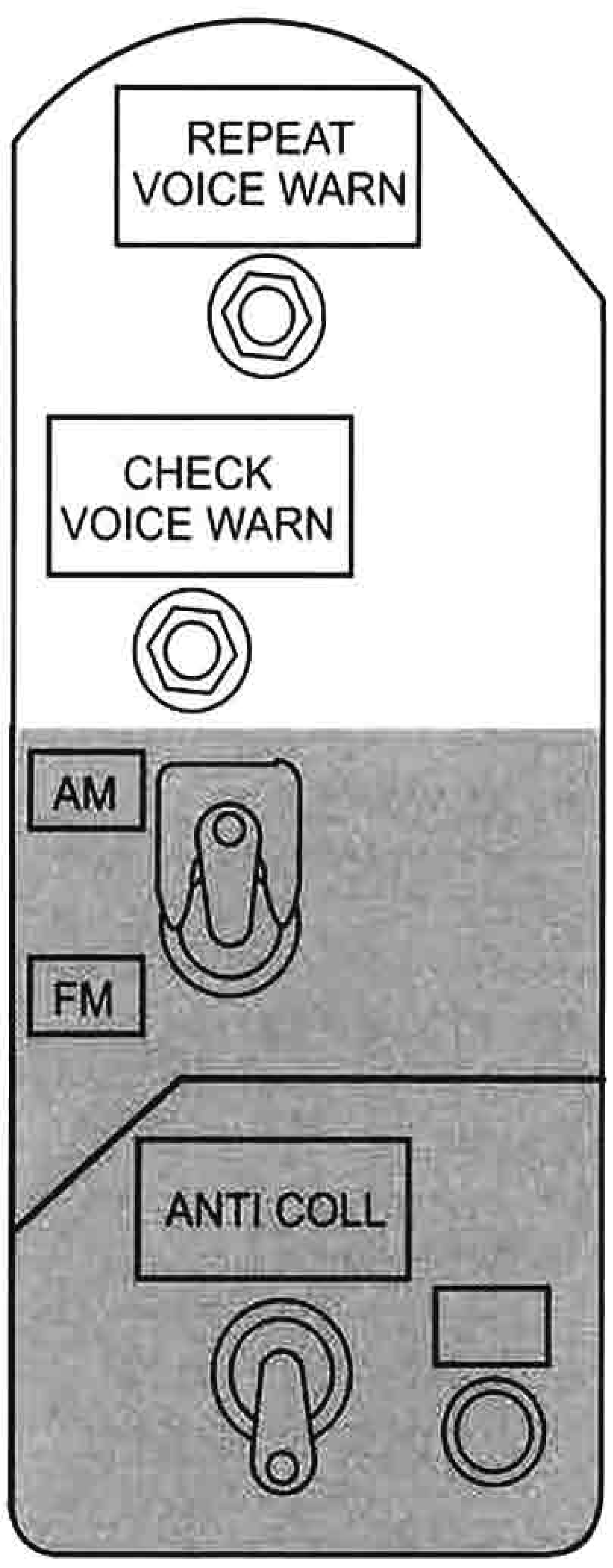


Figure 1-70

WARNING LIGHT TEST

To ensure utmost reliability of the warning system, the light bulbs of all warning and indicator lights are function tested. Pressing the LAMP TEST button on the lighting panel illuminates all lights.

SERVICING DIAGRAM

ITEM NO.	ITEM DESCRIPTION	SPECIFICATION	
1	ENGINE OIL	IPM-10	
2	GEARBOX OIL	IPM-10	
3	NITROGEN	Degree of purity 98 % Dew-point -65° C at 150 bar (15 MPa) CO_2 < 0.03 % total volume Operating pressure 150 +5 bar (15 +0.5 MPa)	
	HYDRAULIC FLUID	AMG-10 or H-515 (FH 51)	
4	RADAR COOLING	LENA-65	
5	OXYGEN, GASEOUS	GAF T.O. 15X-0-1-1000 Pressure 150 +5 bar (15 +0.5 MPa)	
6	GROUNDING CABLE	Connected	
7	COMPRESSED AIR	Pressure 150 +5 bar (15 +0.5 MPa) Temperature max +40° C Dew-point -55° C at 150 bar (15 MPa) Free of oil and grease	
8	EXTERNAL AC-POWER	117.5 / 202 VAC, 400 Hz, 3-phase	
9	EXTERNAL DC-POWER	28.5 ±0.5 VDC, 15 kW	
10	ENGINE FUEL		FUEL SELECT Position
		JP8 / NATO F-34	I
		RT	II
		TS-1	II
		T-1	I
11	GUN COOLING	H_2O (distilled water)	

Figure 1-71

SERVICING DIAGRAM

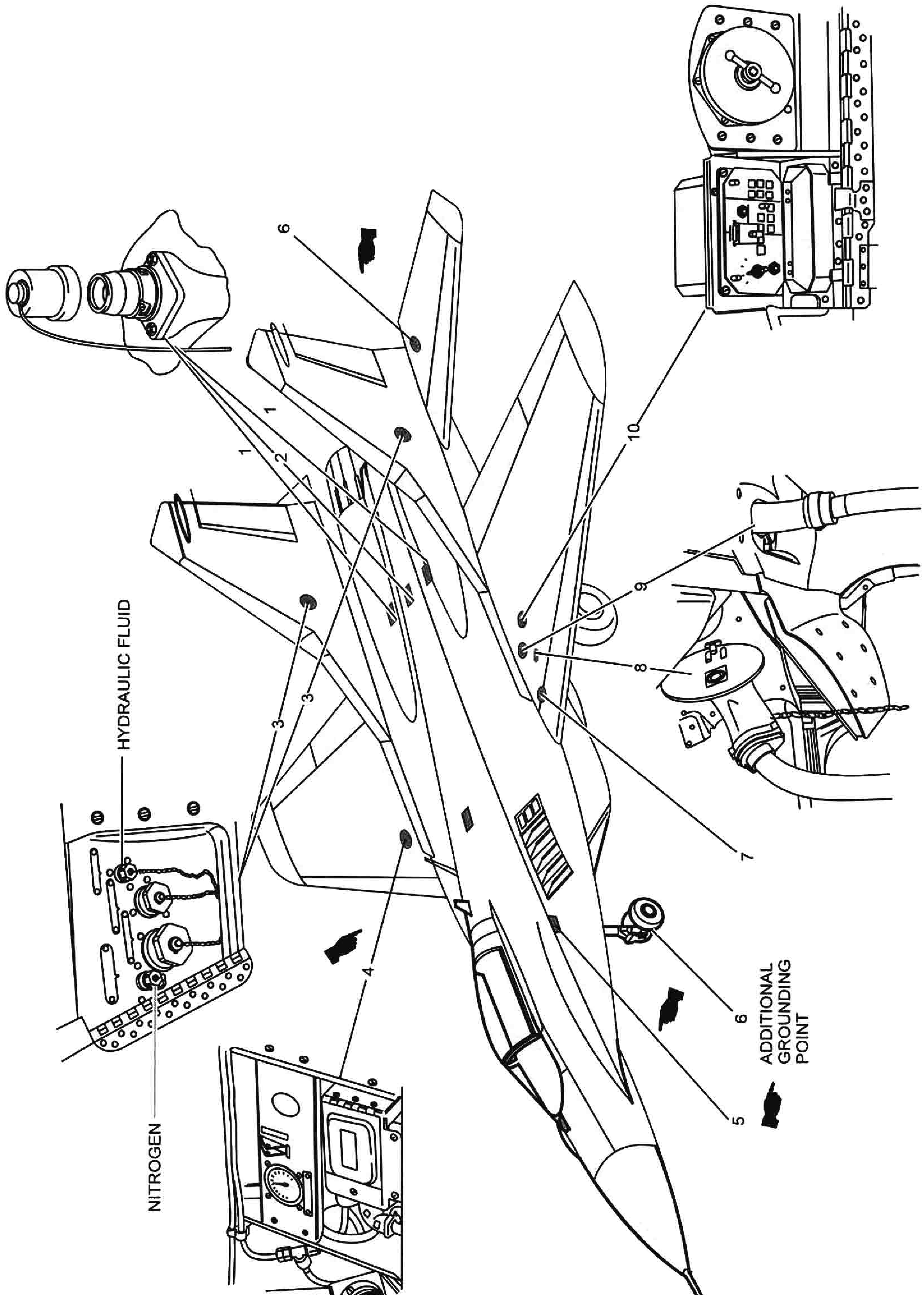


Figure 1-72

HYDRAULIC FLUID
NITROGEN
1
2
1
3
3
4
6
5
6
7
8
9
10
ADDITIONAL
GROUNDING
POINT