FLIGHT MANUAL
MIG-29

CHANGE NOTICE

LATEST CHANGED PAGES SUPERSEDE THE SAME PAGES OF PREVIOUS DATE.
INSERT CHANGED PAGES INTO BASIC PUBLICATION.
DESTROY SUPERSEDED PAGES.

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

NOTE

Information that has been changed is indicated by a vertical line on the outer margins of the page, an asterisk, a screen, or a miniature pointing hand.

TOTAL NUMBER OF PAGES IN THIS PUBLICATION IS INCLUDING FRONT PAGES AND EMPTY PAGES IS 462, CONSISTING OF THE FOLLOWING:

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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.

Additional copies of this publication can be obtained from LwMatKdo I C 3.

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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 МИГс

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.

<u>YOUR</u> 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

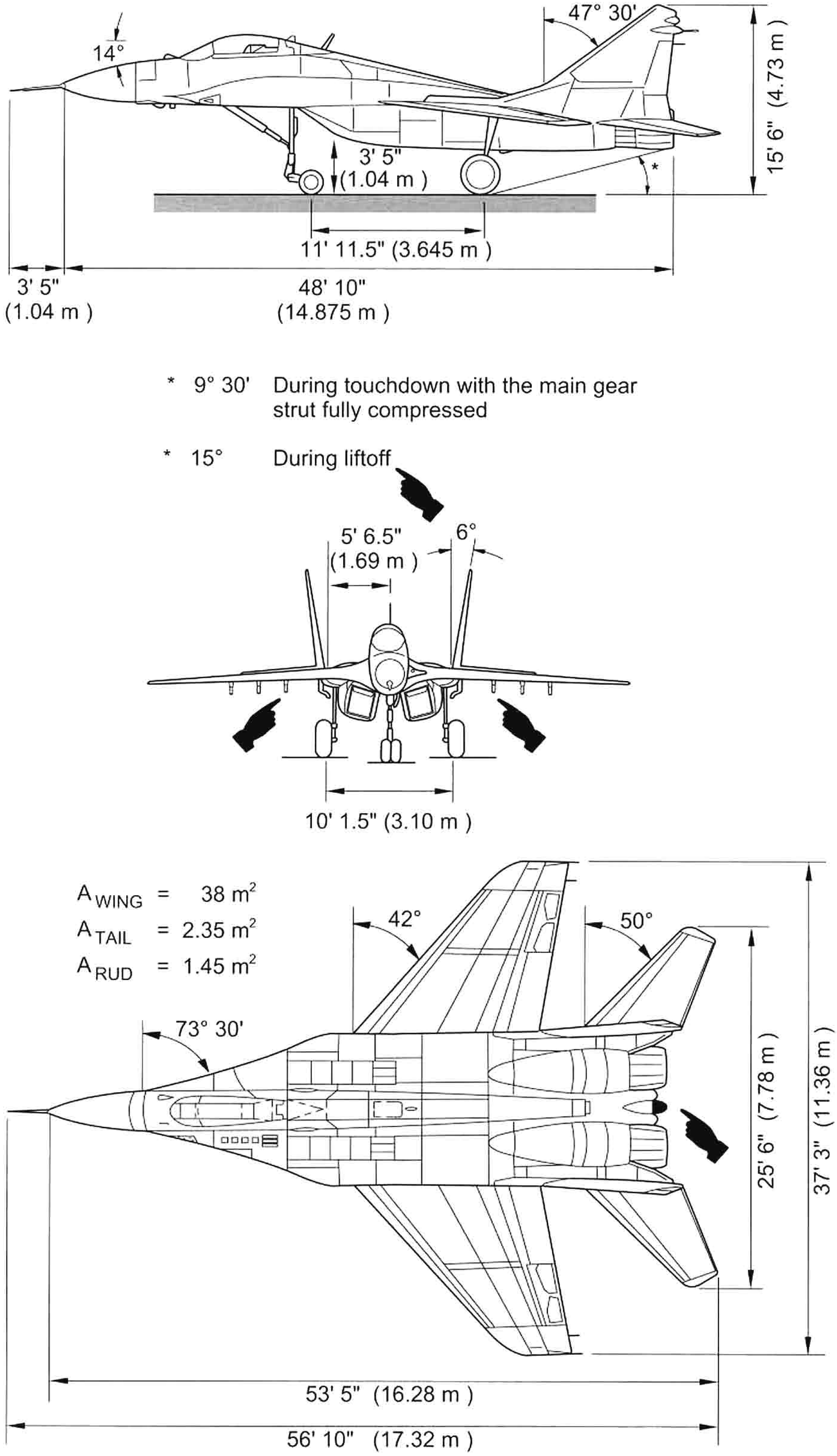


Figure 1-0

AIRCRAFT DIMENSIONS MIG-29 GT

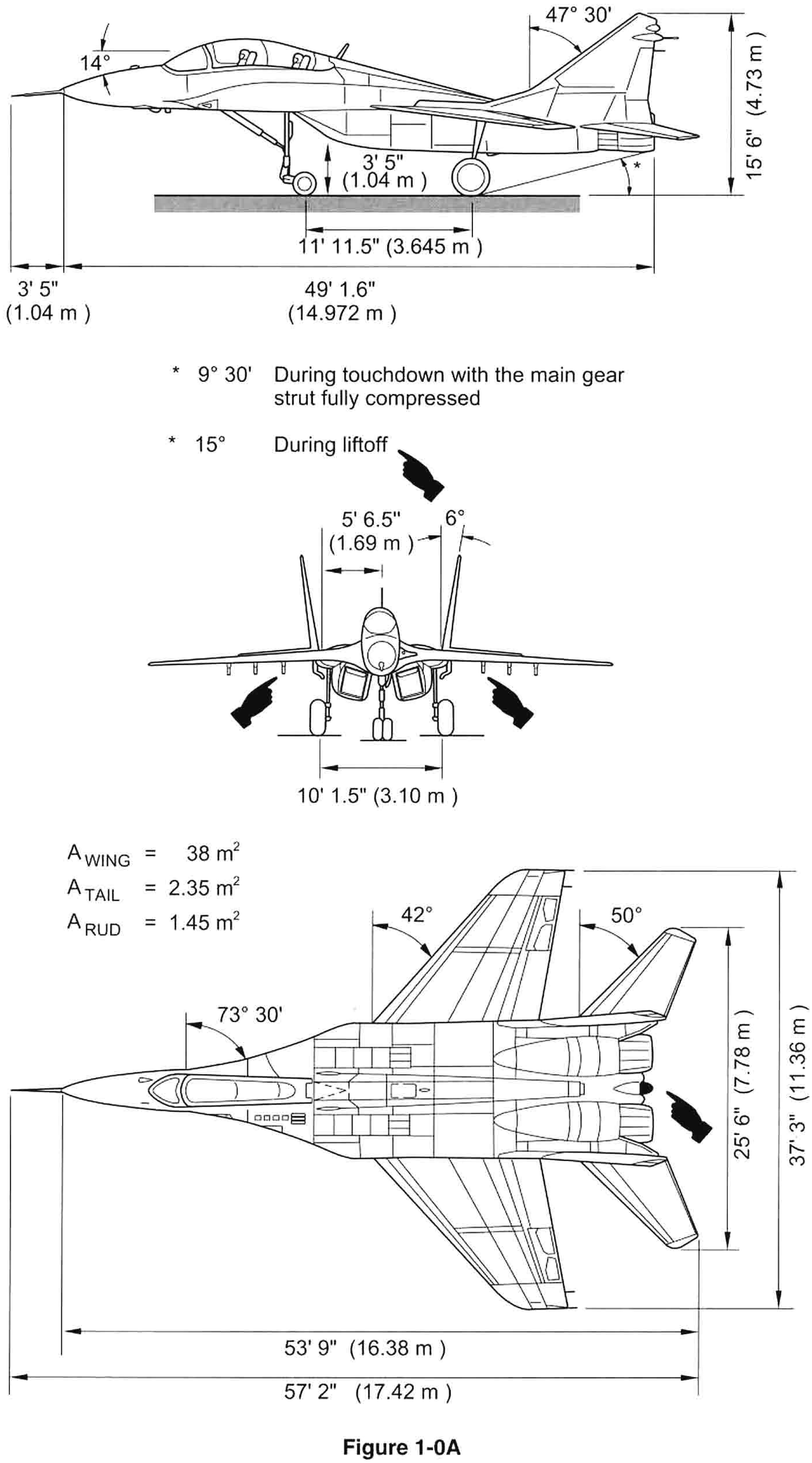


Figure 1-0A

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.

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

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.

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

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.

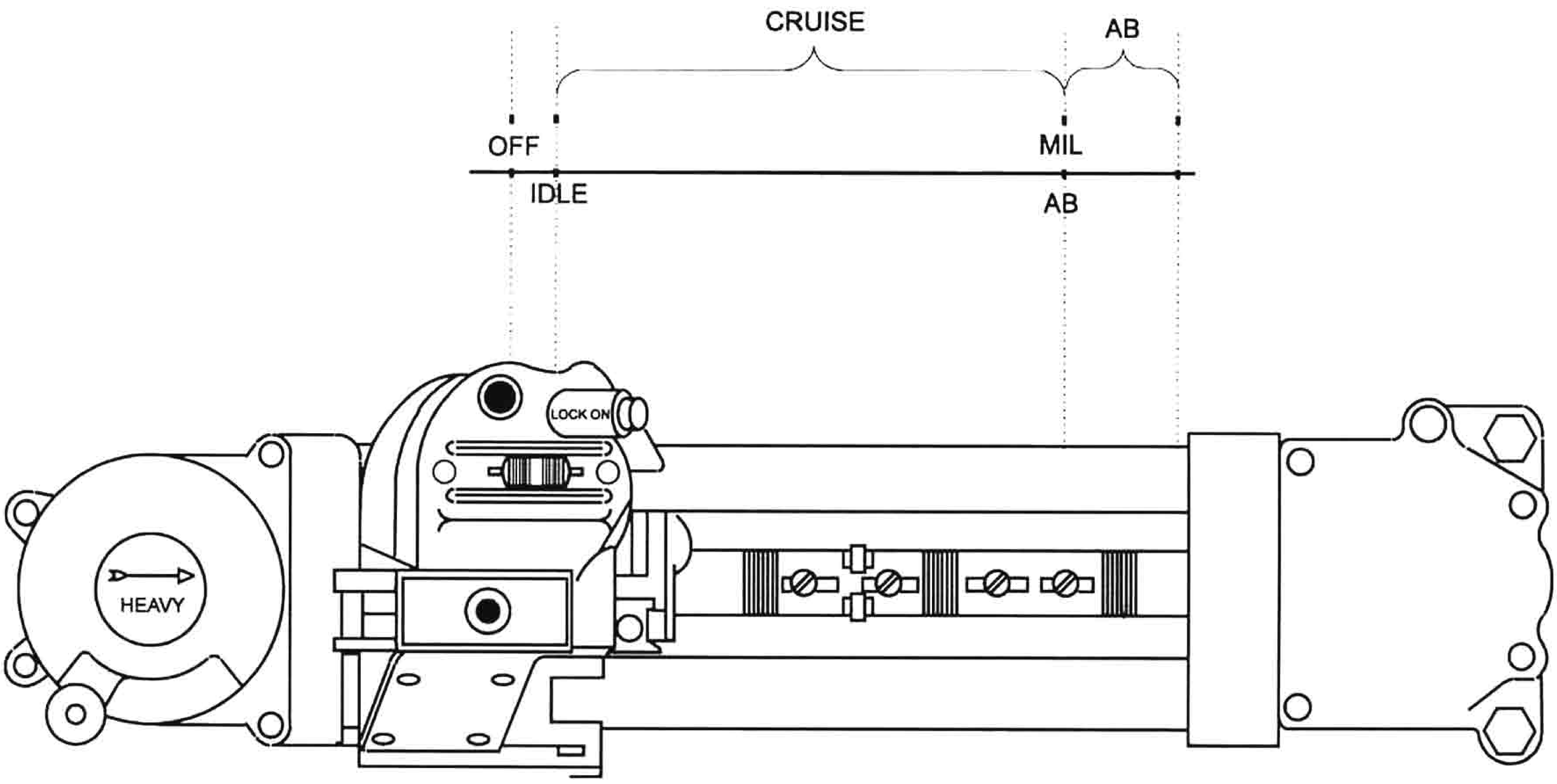
The throttle produces an engine speed demand signal which is routed mechanically to the ECP and to the AB and nozzle control unit. The ECU supplies electrical control signals to solenoids in the ECP, the AB and nozzle control unit and to the engine starter unit in order to modify engine performance within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

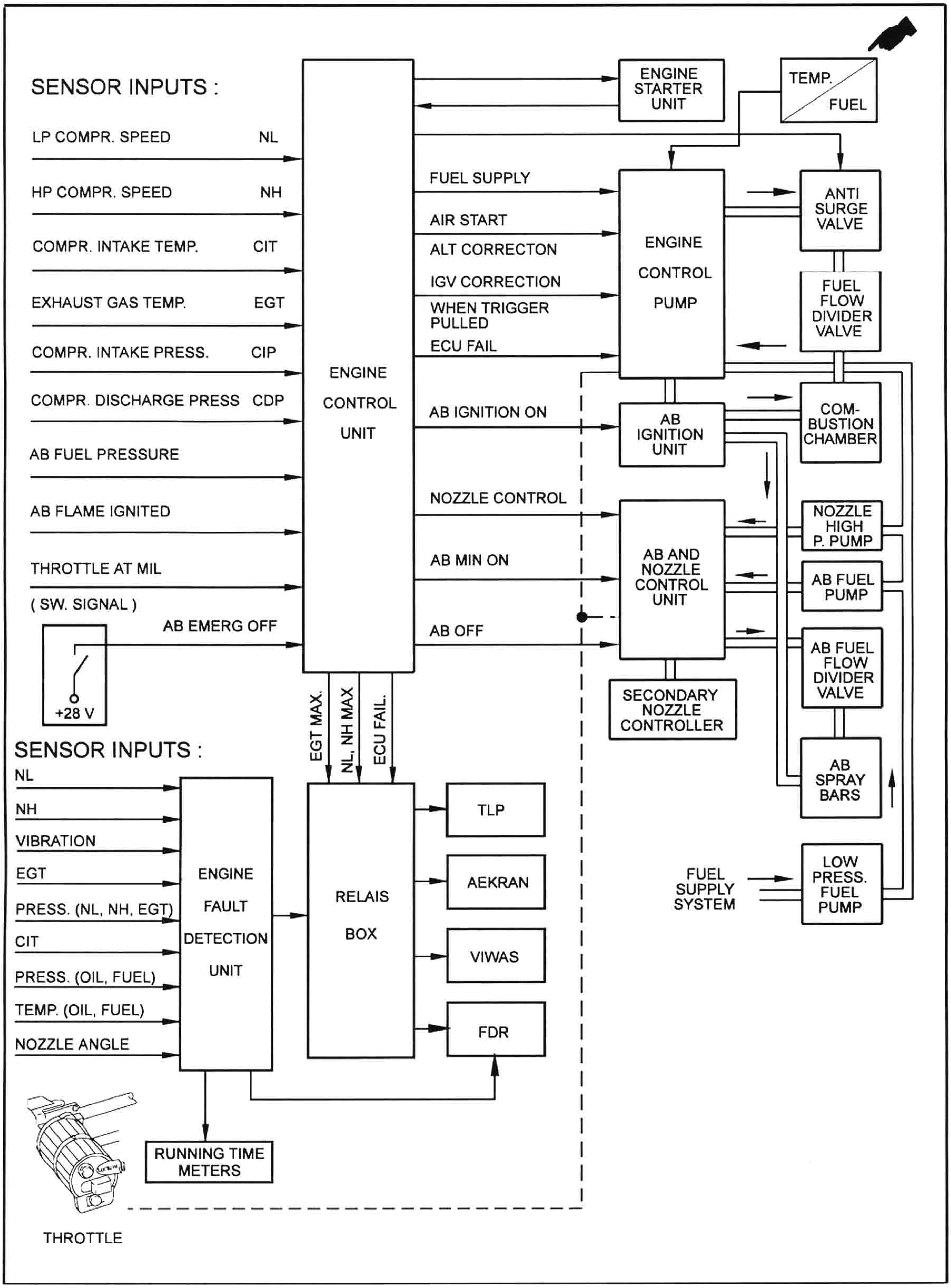


Figure 1-1A

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

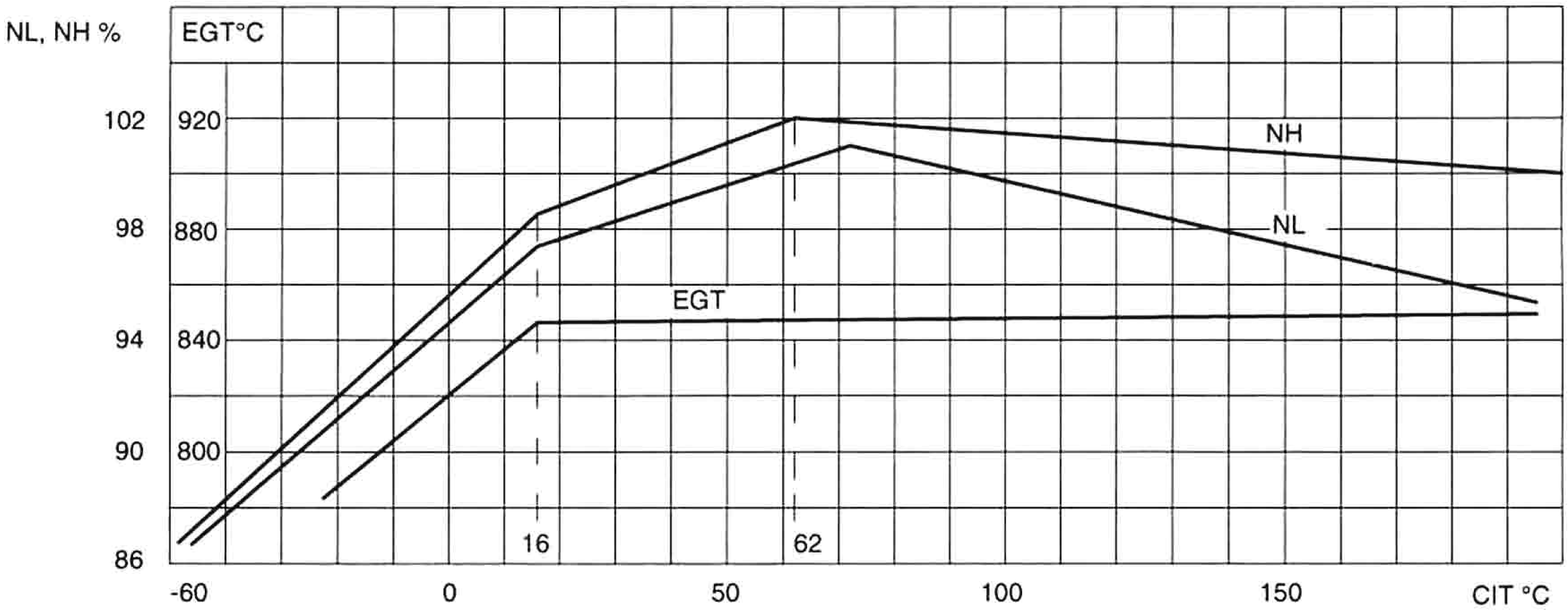


Figure 1-2

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

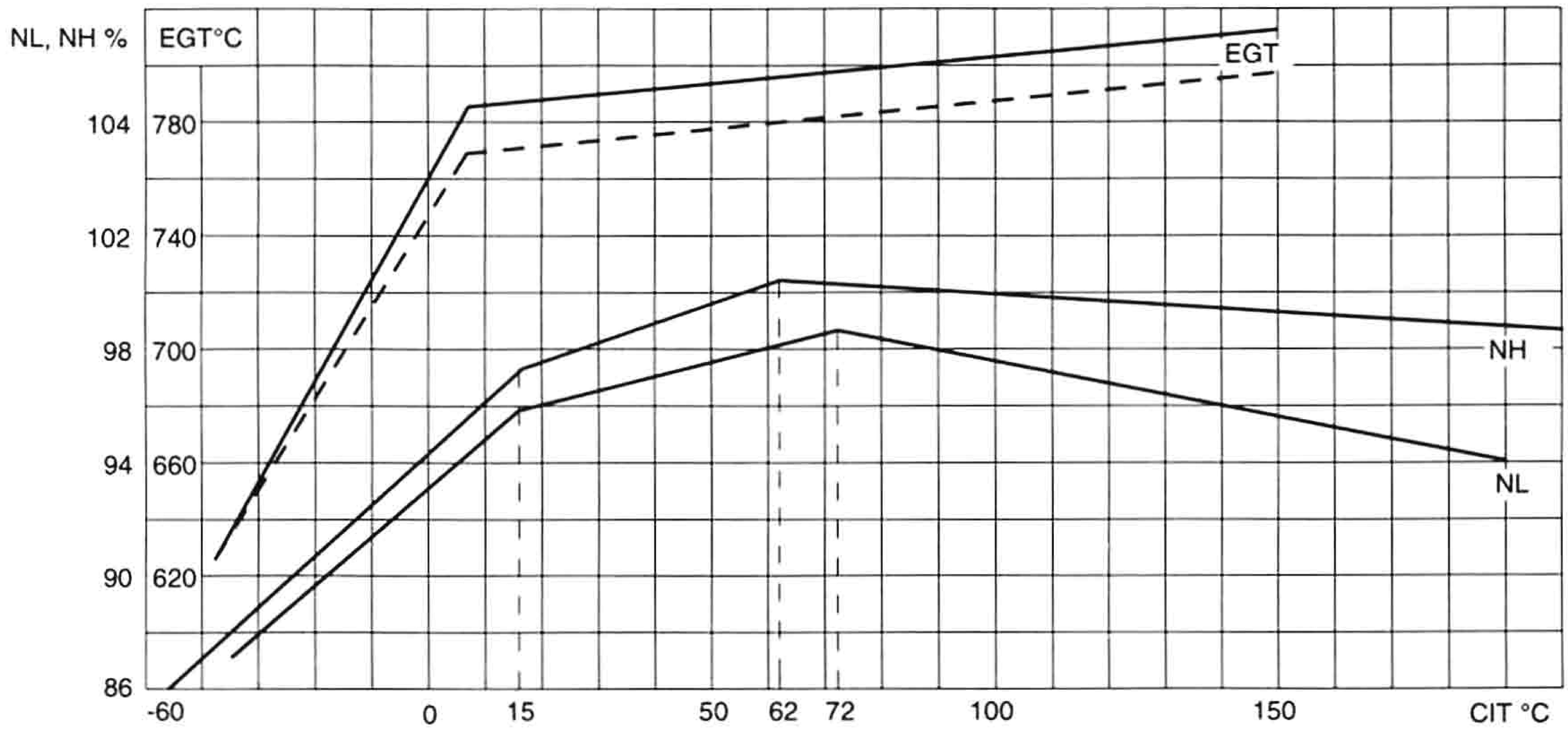


Figure 1-3

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)

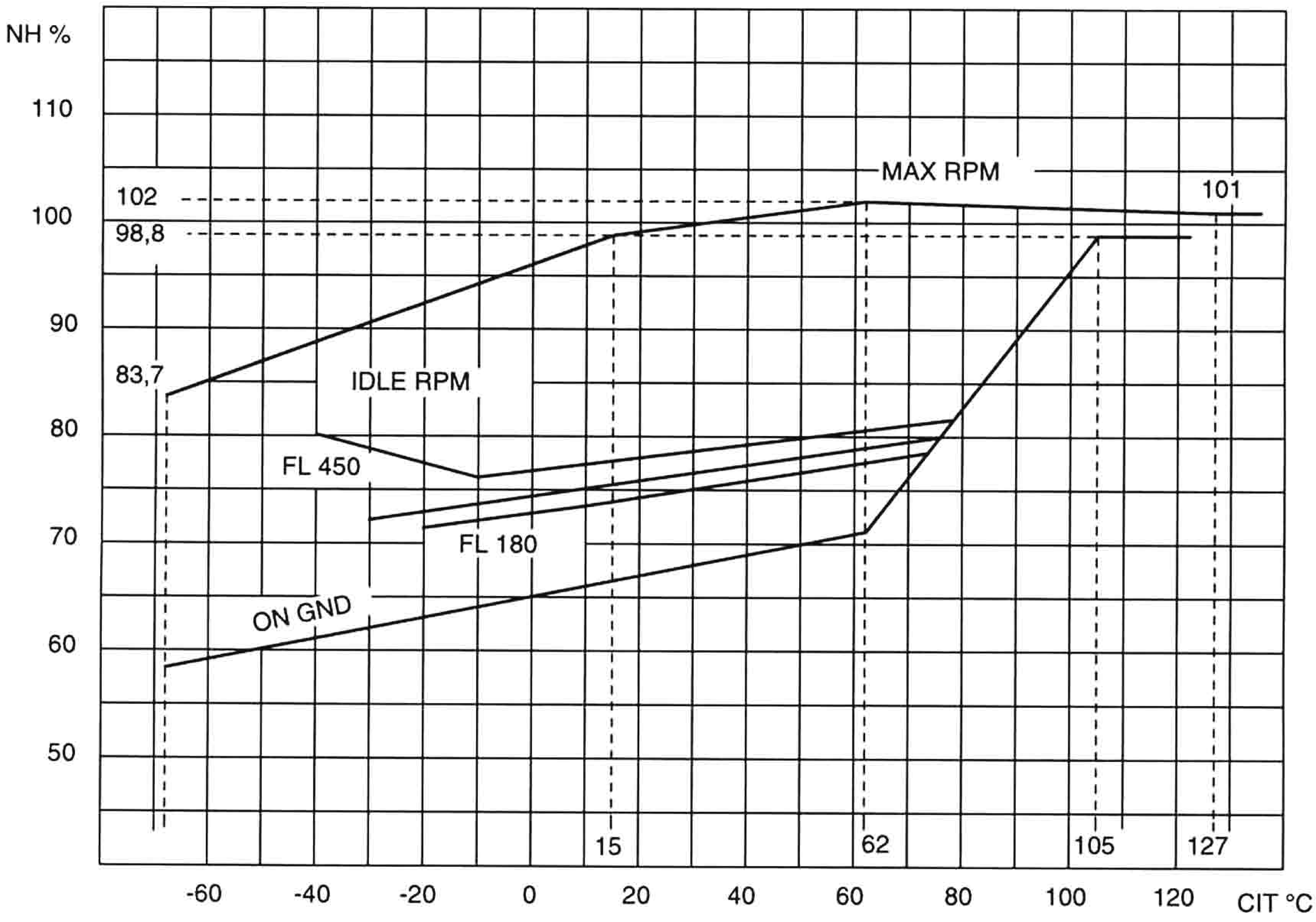


Figure 1-4

FUNCTION OF THE NH SPEED CONTROLLER
(LIMITED POWER MODE)

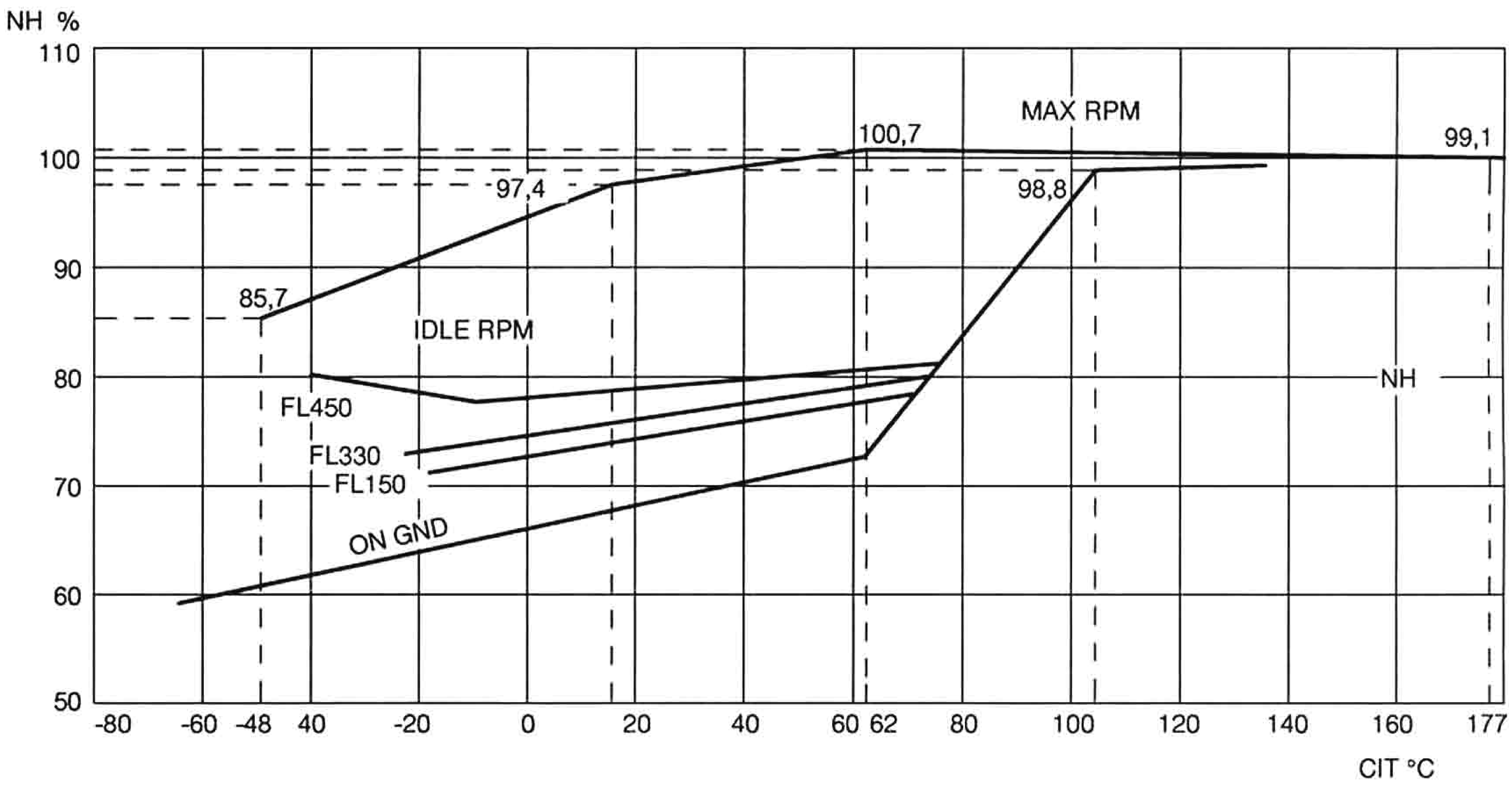


Figure 1-5

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**	Engine in surge condition and overheat left / right.
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 AB ENG**	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.

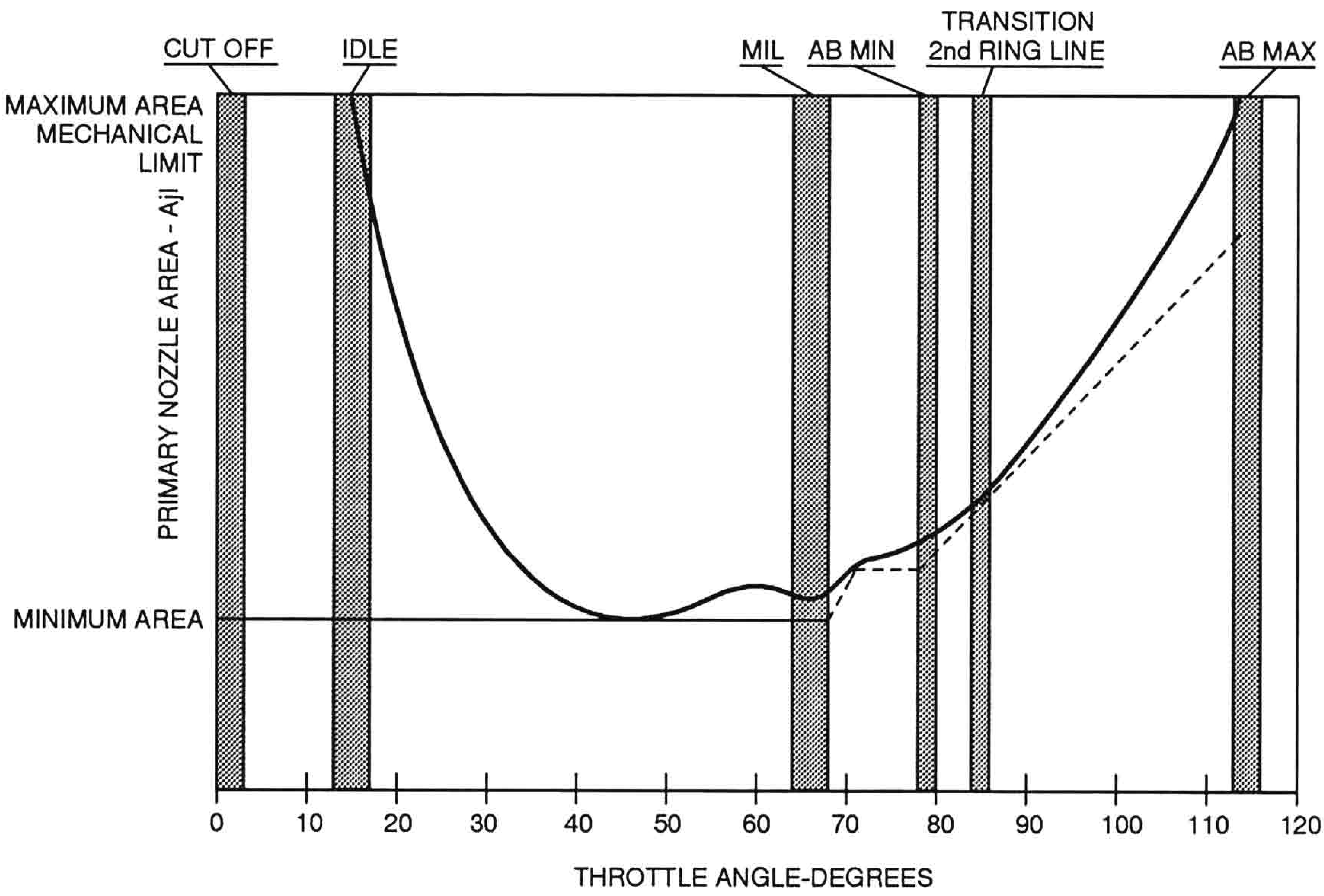
When the nozzle control lane of the ECU fails, the primary nozzle area is modulated by the AB and nozzle control unit as a function of throttle setting and CIT.

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.

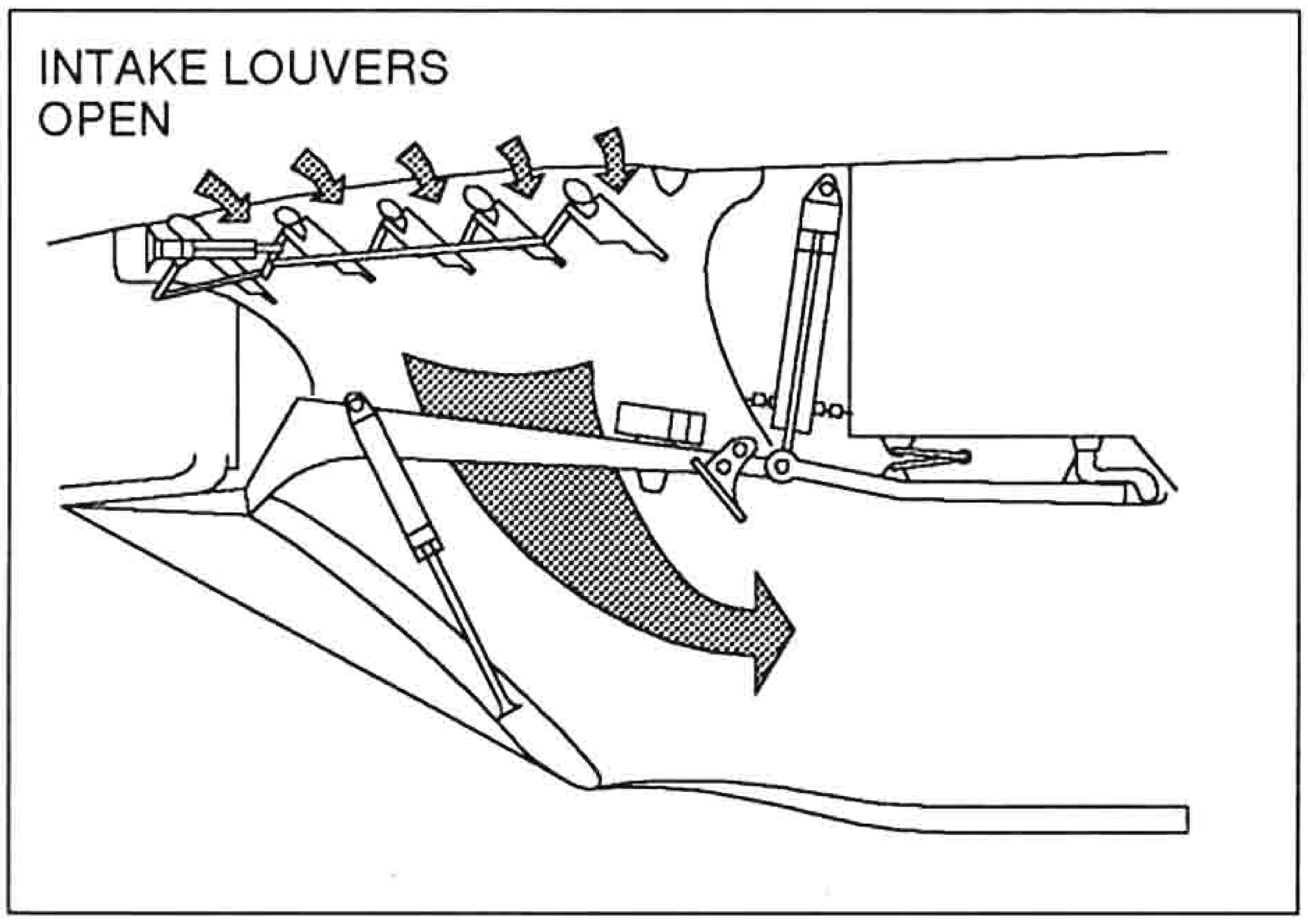
To prevent FOD during takeoff and landing, the duct ramps are closed and the air intake louvers are opened.

VARIABLE DUCT RAMP SYSTEM

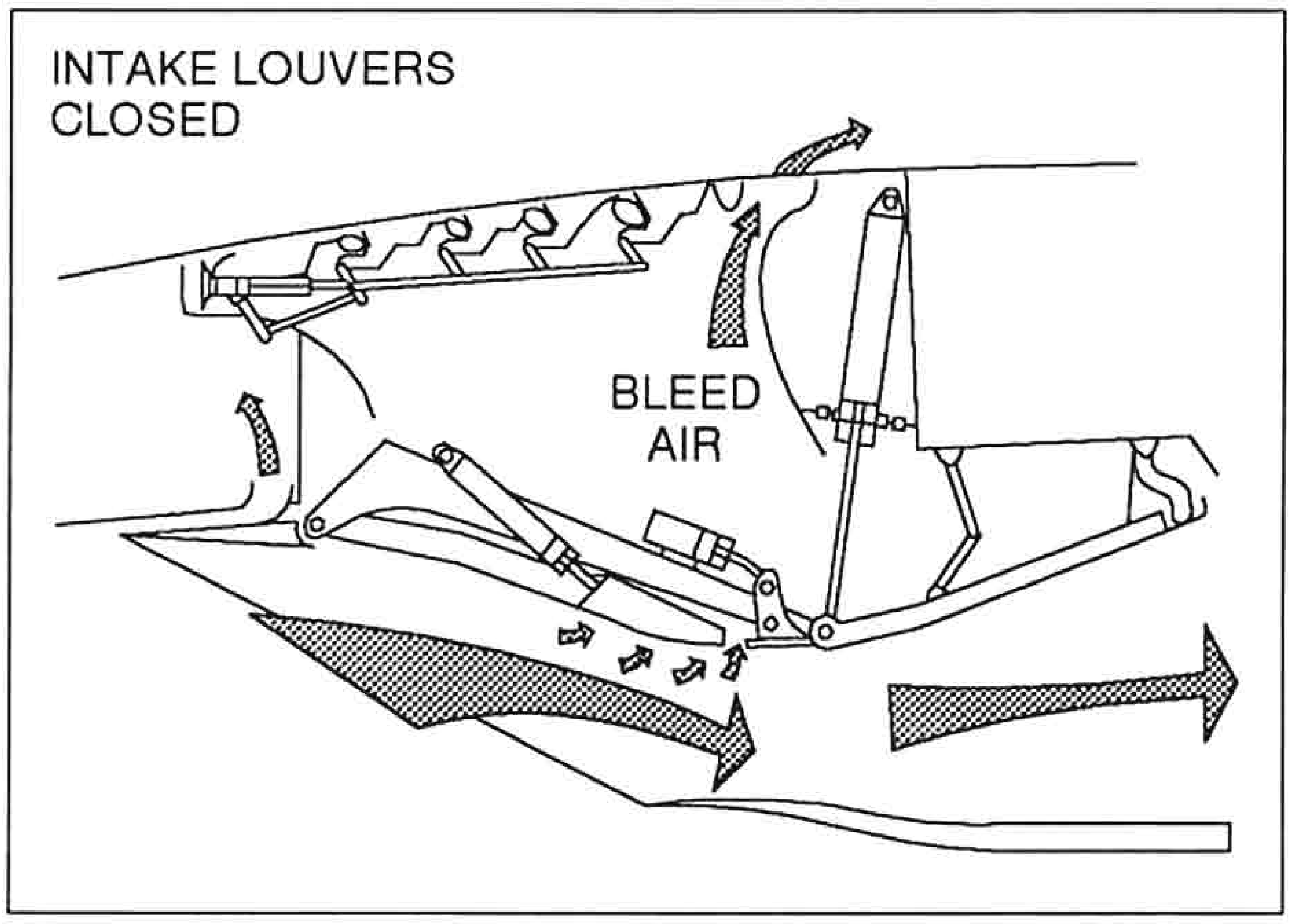
The variable forward ramp system provides engine air at optimum subsonic airflow to the low-pressure compressor face throughout a wide range of speeds. The ramp assembly consists of a variable forward ramp, a variable aft ramp, a bleed-off valve and a ramp control unit. The bleed-off valve and perforated 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.

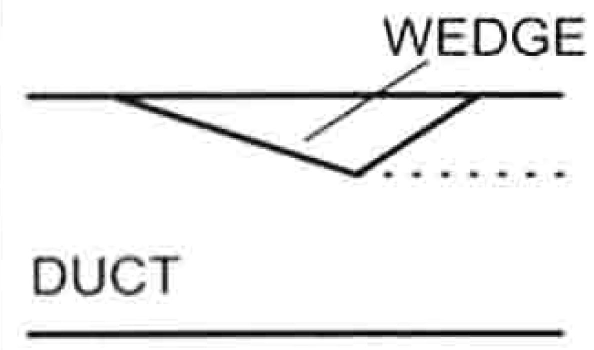
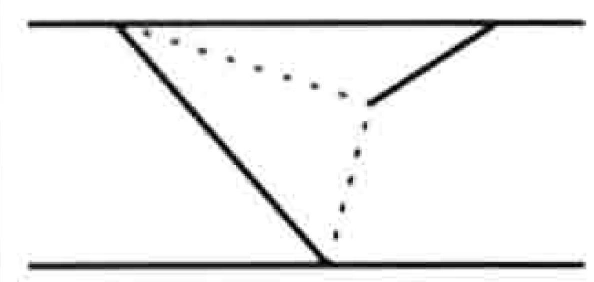
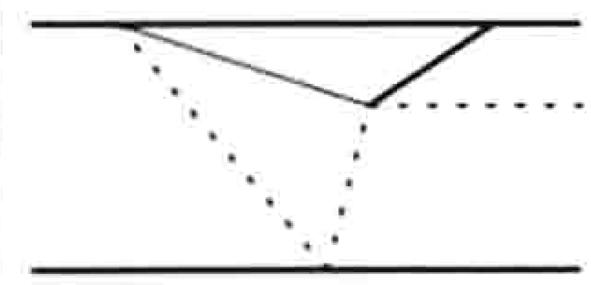
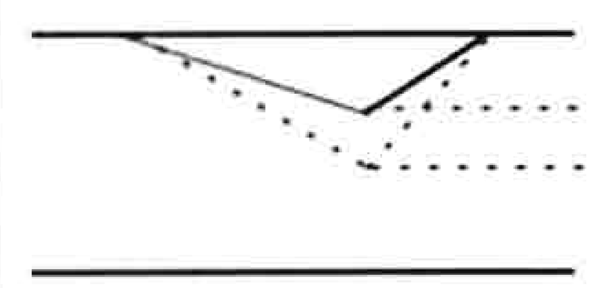
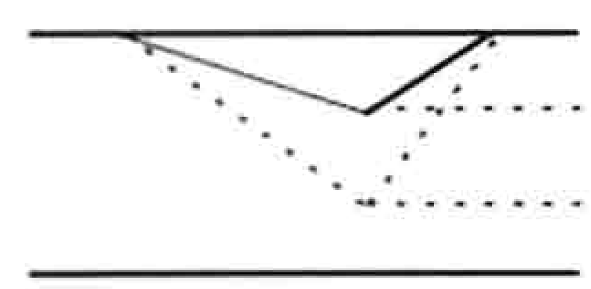
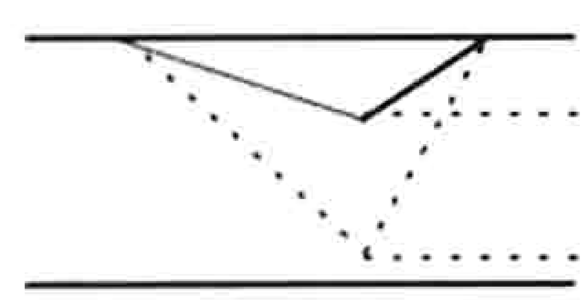
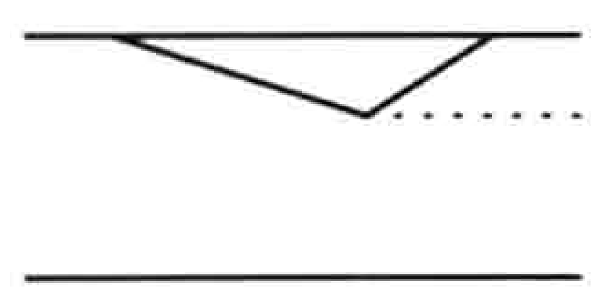
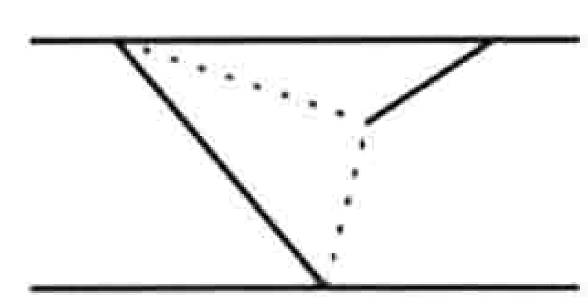
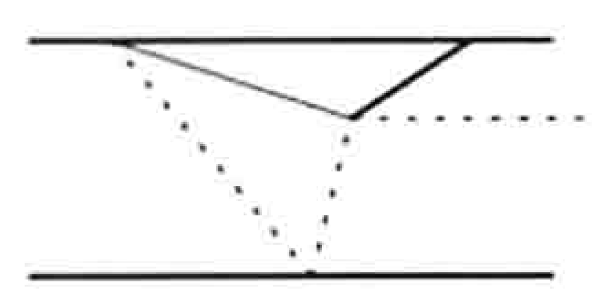
The second travel schedule controls the wedge angle between 0 % and 60 % extension.

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

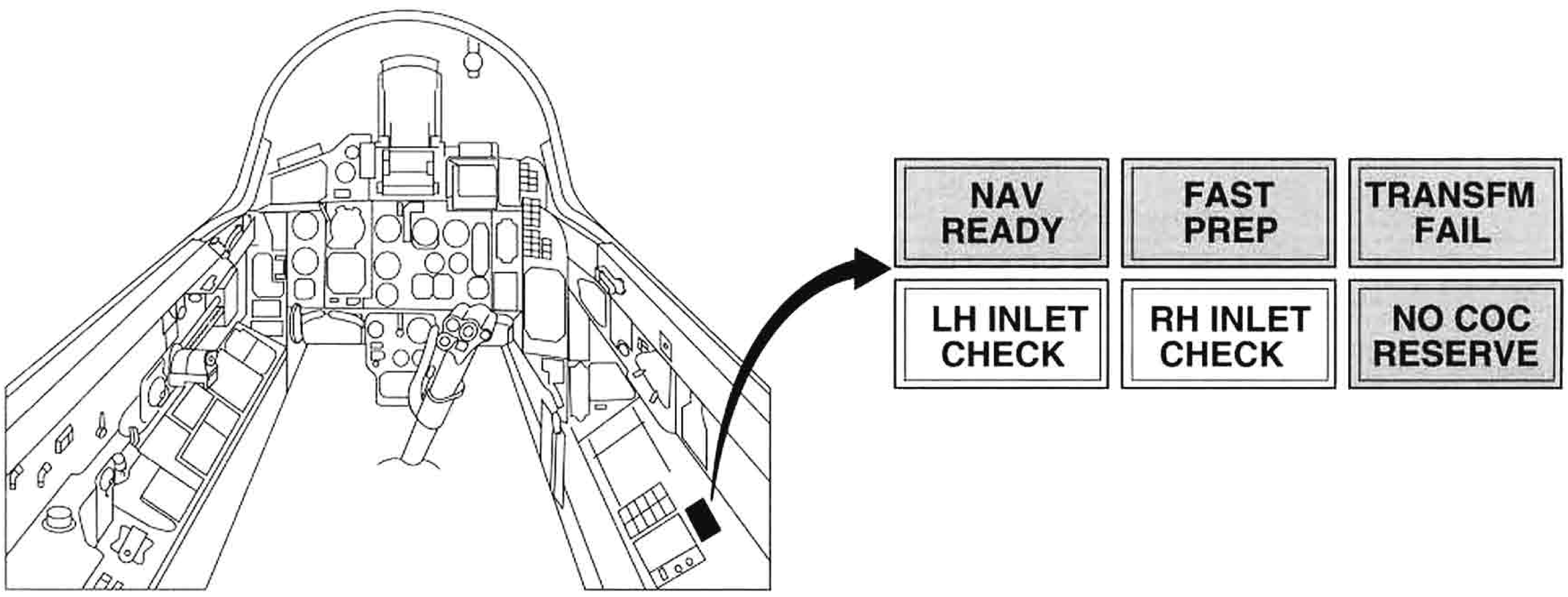


Figure 1-9

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.

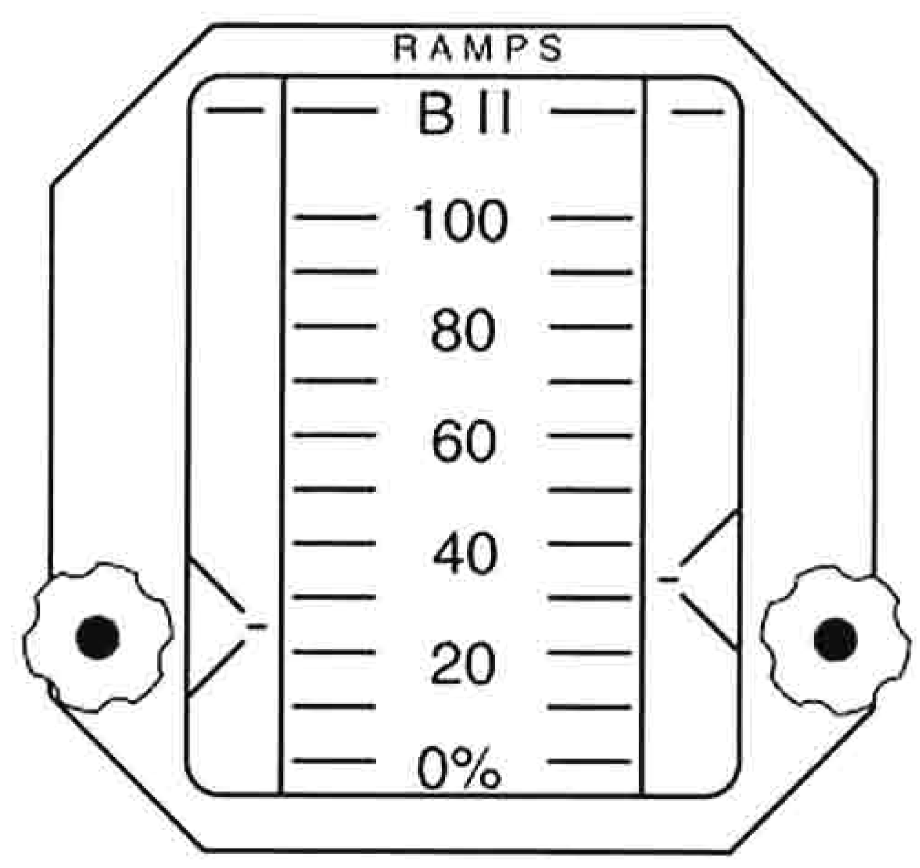


Figure 1-10

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.

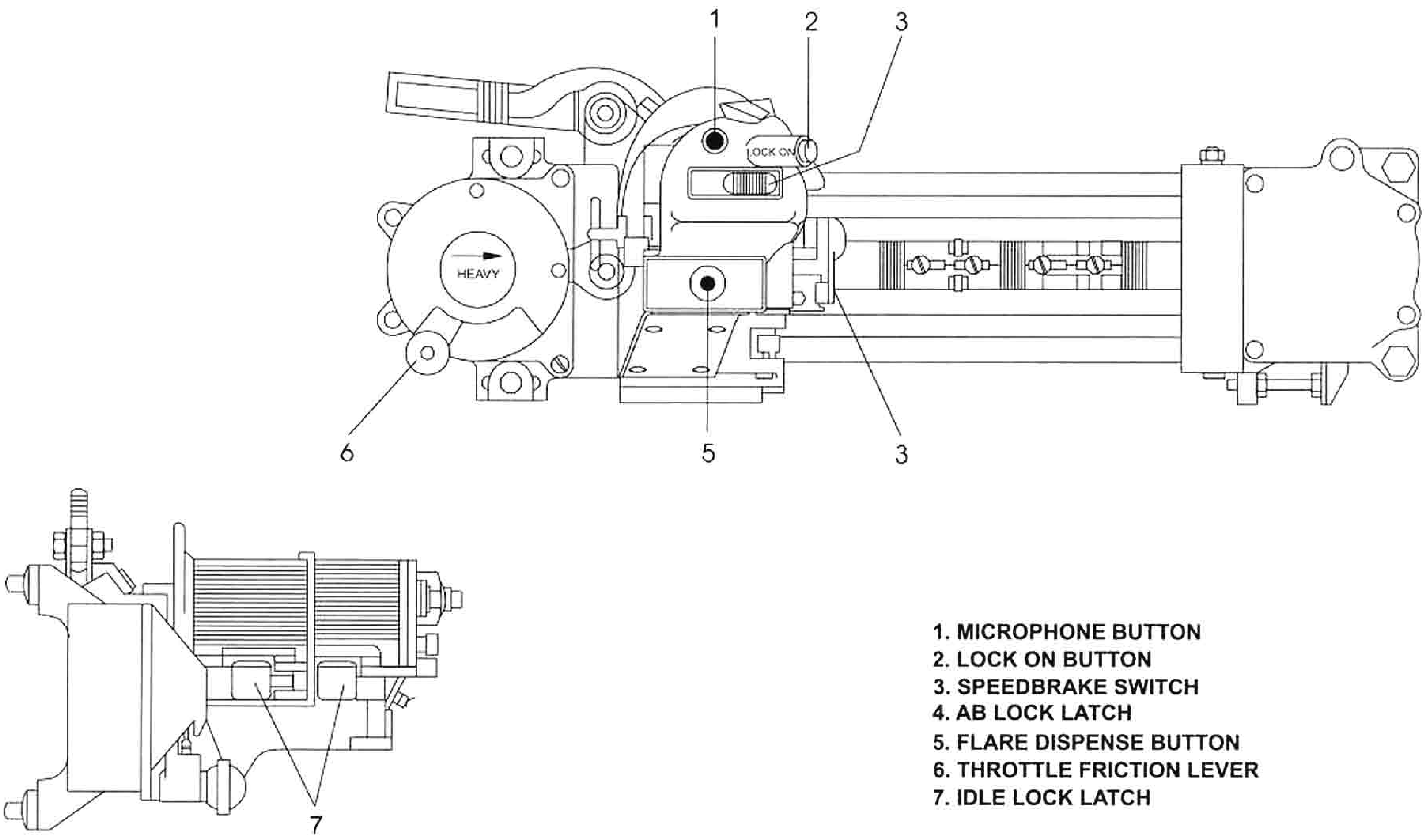
The throttles are not interconnected.

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



1. MICROPHONE BUTTON
2. LOCK ON BUTTON
3. SPEEDBRAKE SWITCH
4. AB LOCK LATCH
5. FLARE DISPENSE BUTTON
6. THROTTLE FRICTION LEVER
7. IDLE LOCK LATCH

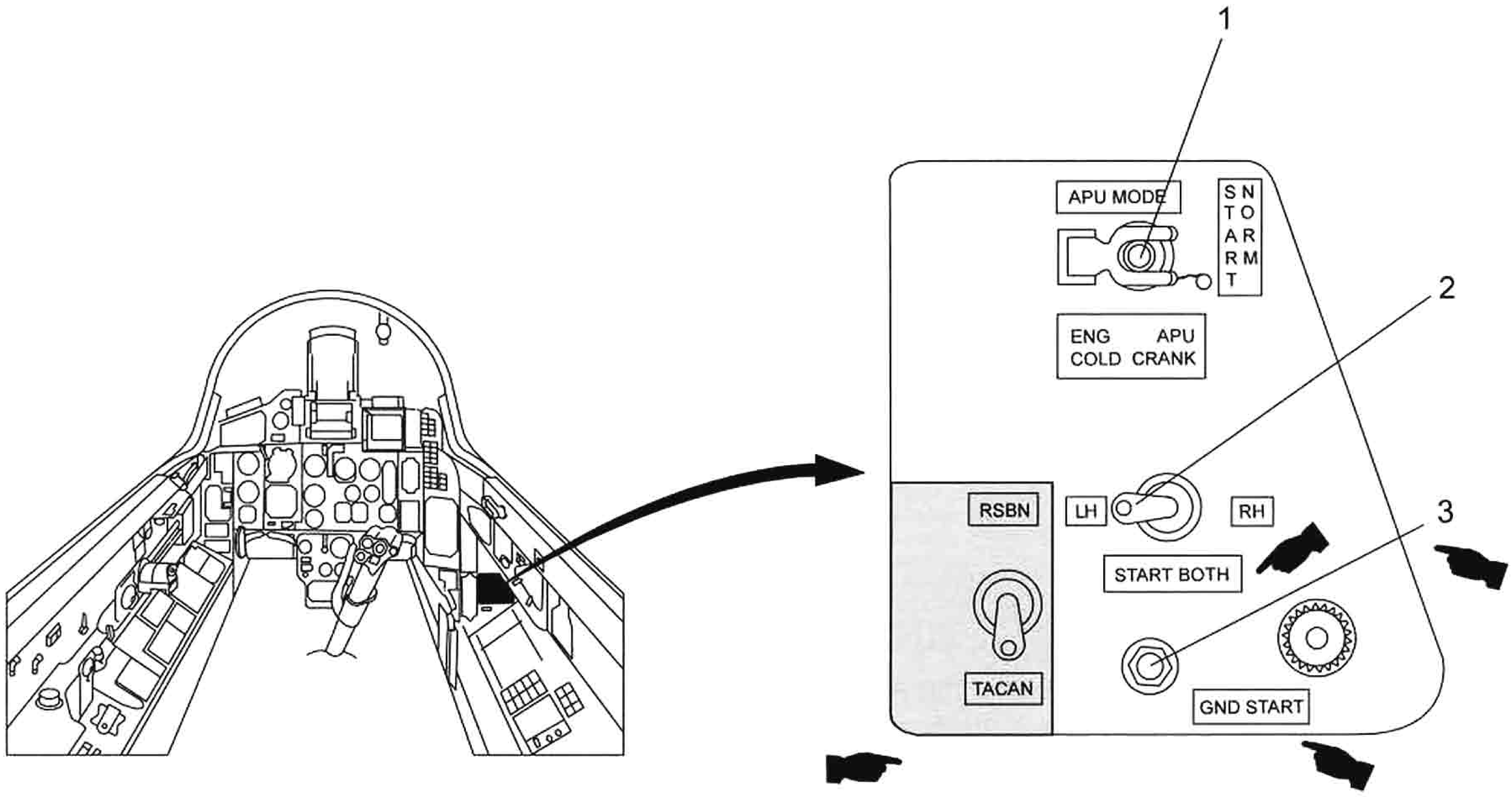
Figure 1-11

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

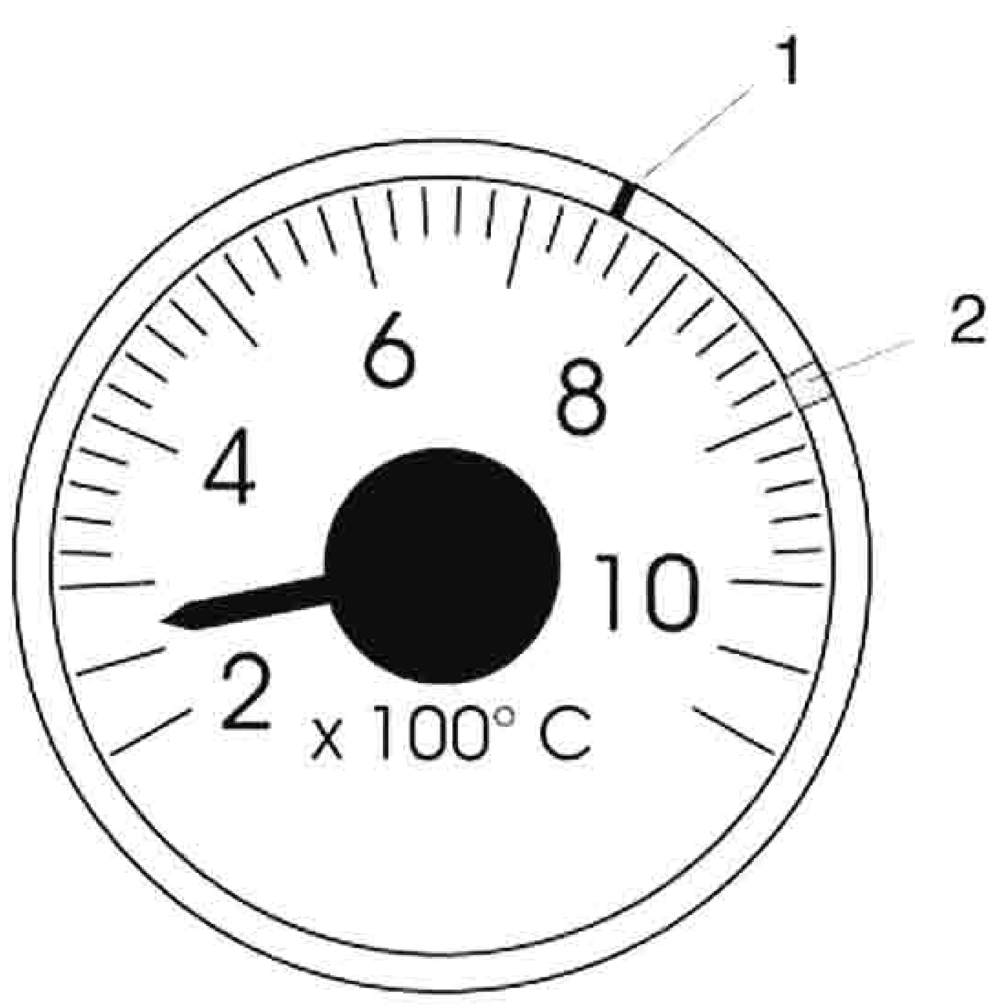
The GND START button is located at the engine start panel. Pressing the button initiates the start and ignition cycle of either the APU and the engines or the APU only depending on APU switch and start-up mode switch setting.

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.

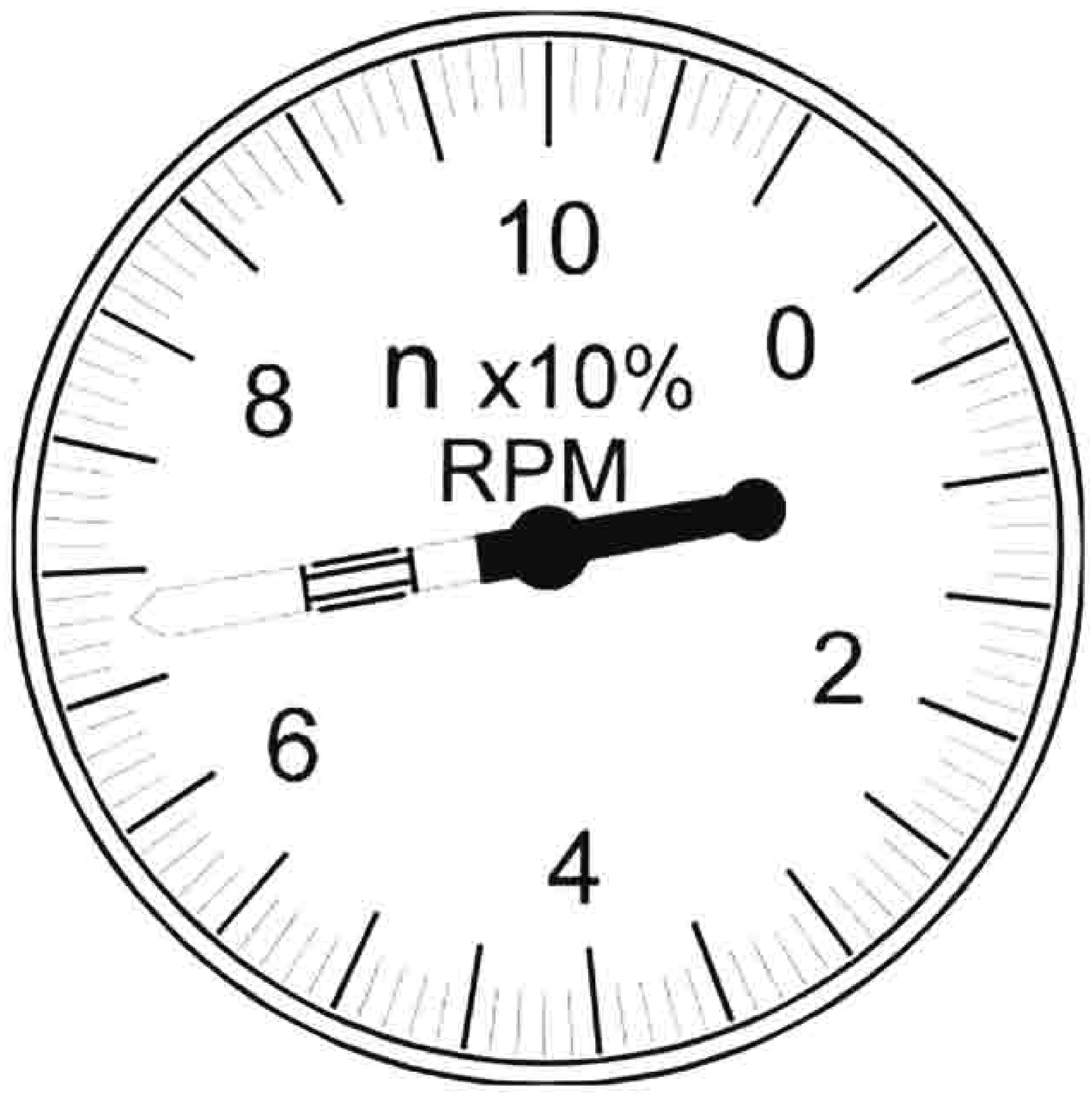


Figure 1-14

Figure 1-15, deleted

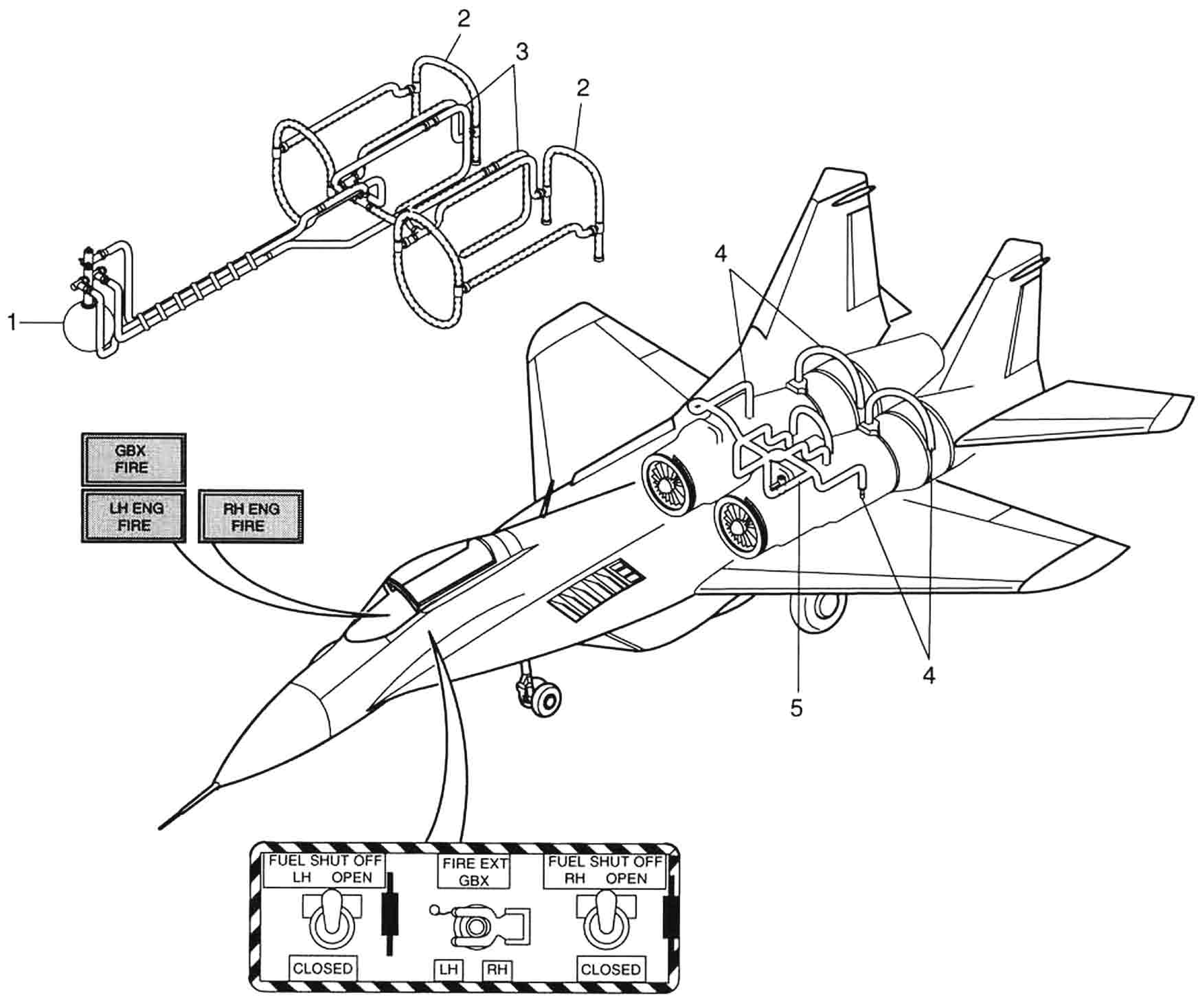
ENGINE FIRE DETECTION SYSTEM

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

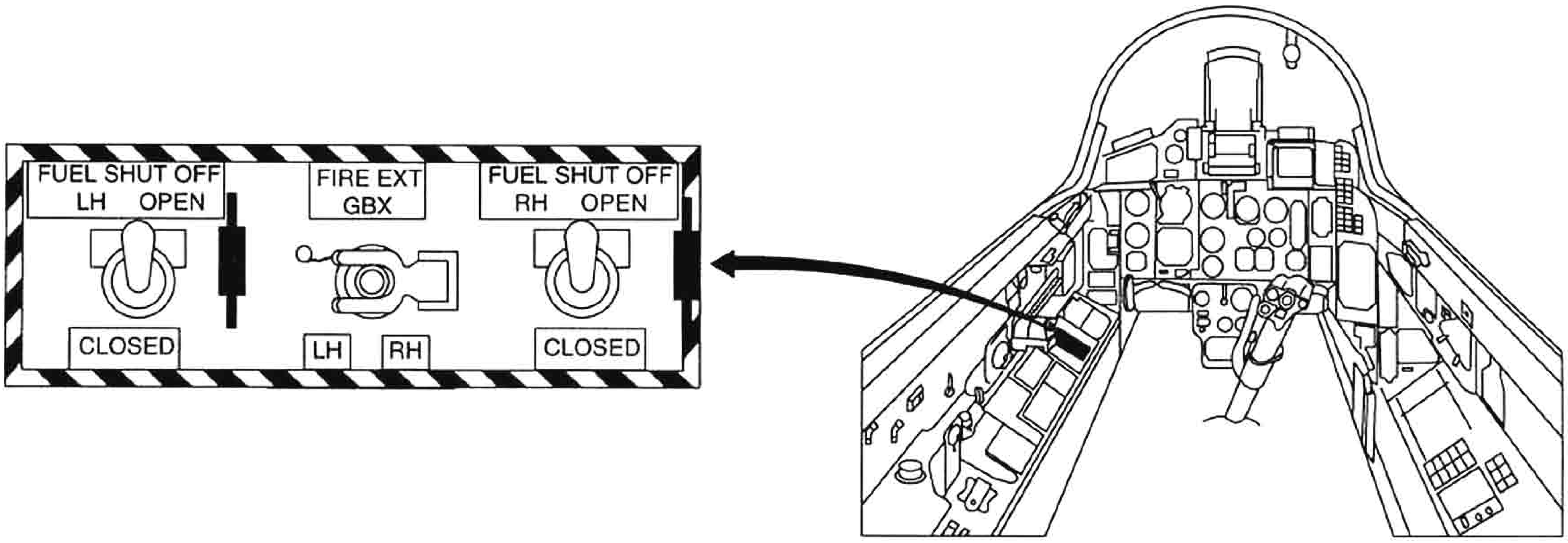


Figure 1-17

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 $M \geq 1.5$.

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

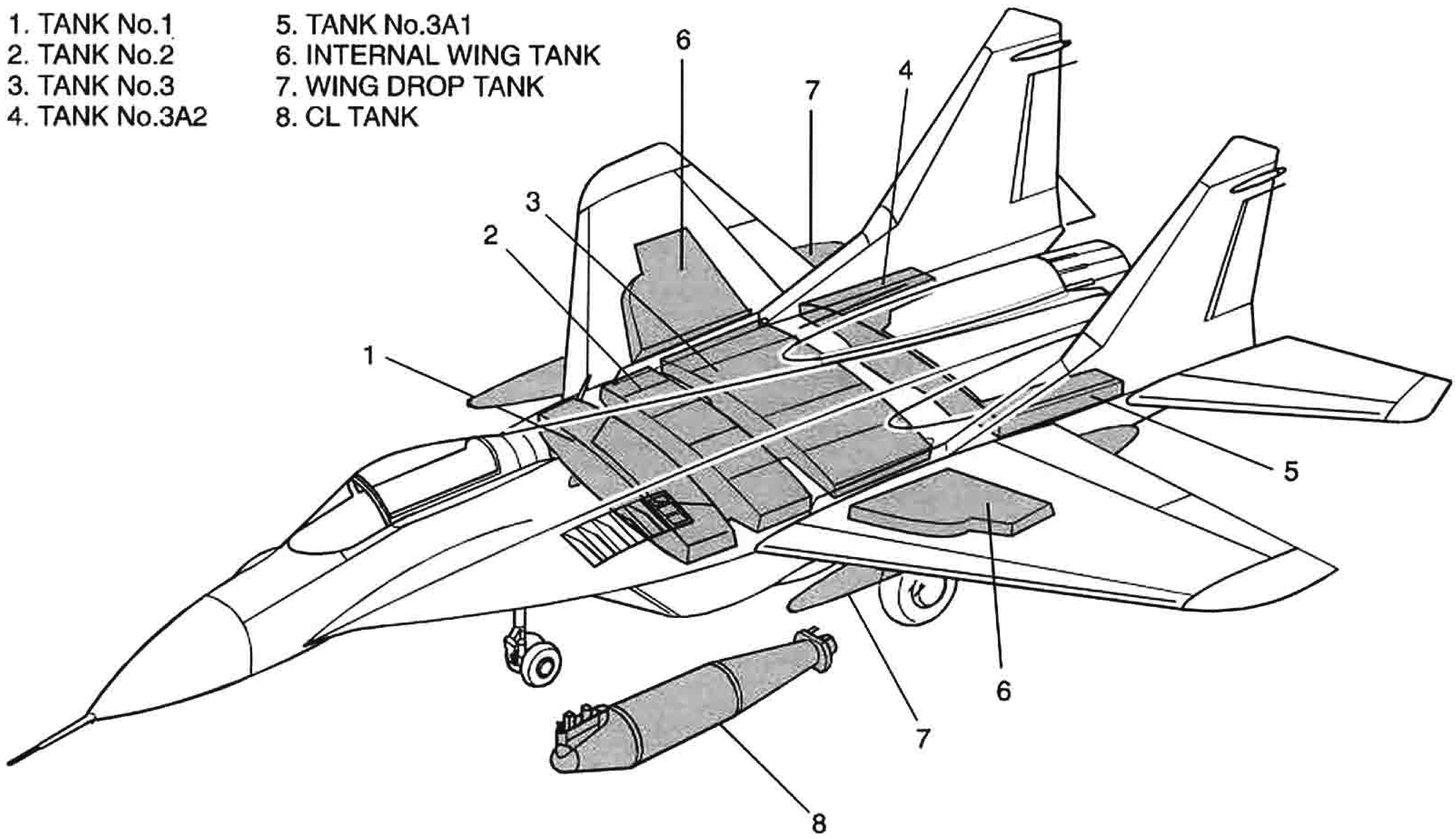


Figure 1-18

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.

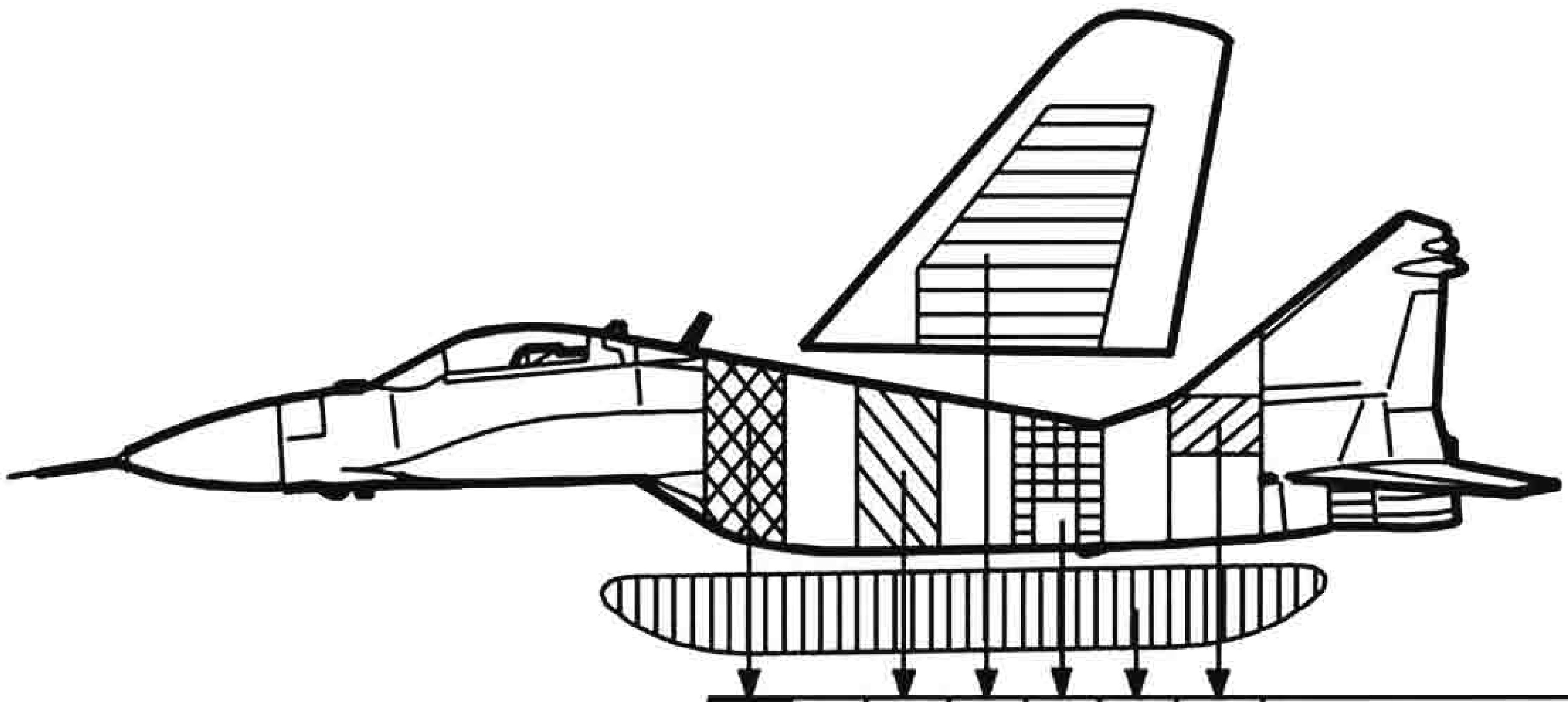
Upon depletion of approximately 50 l from tank 3, the jet-type transfer pumps in tanks 3A and the internal wing tanks start transferring fuel to tank 3. When the internal wing tanks are empty, an empty sensor signals fuel depletion to the fuel indicator.

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



FUEL TRANSFER

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

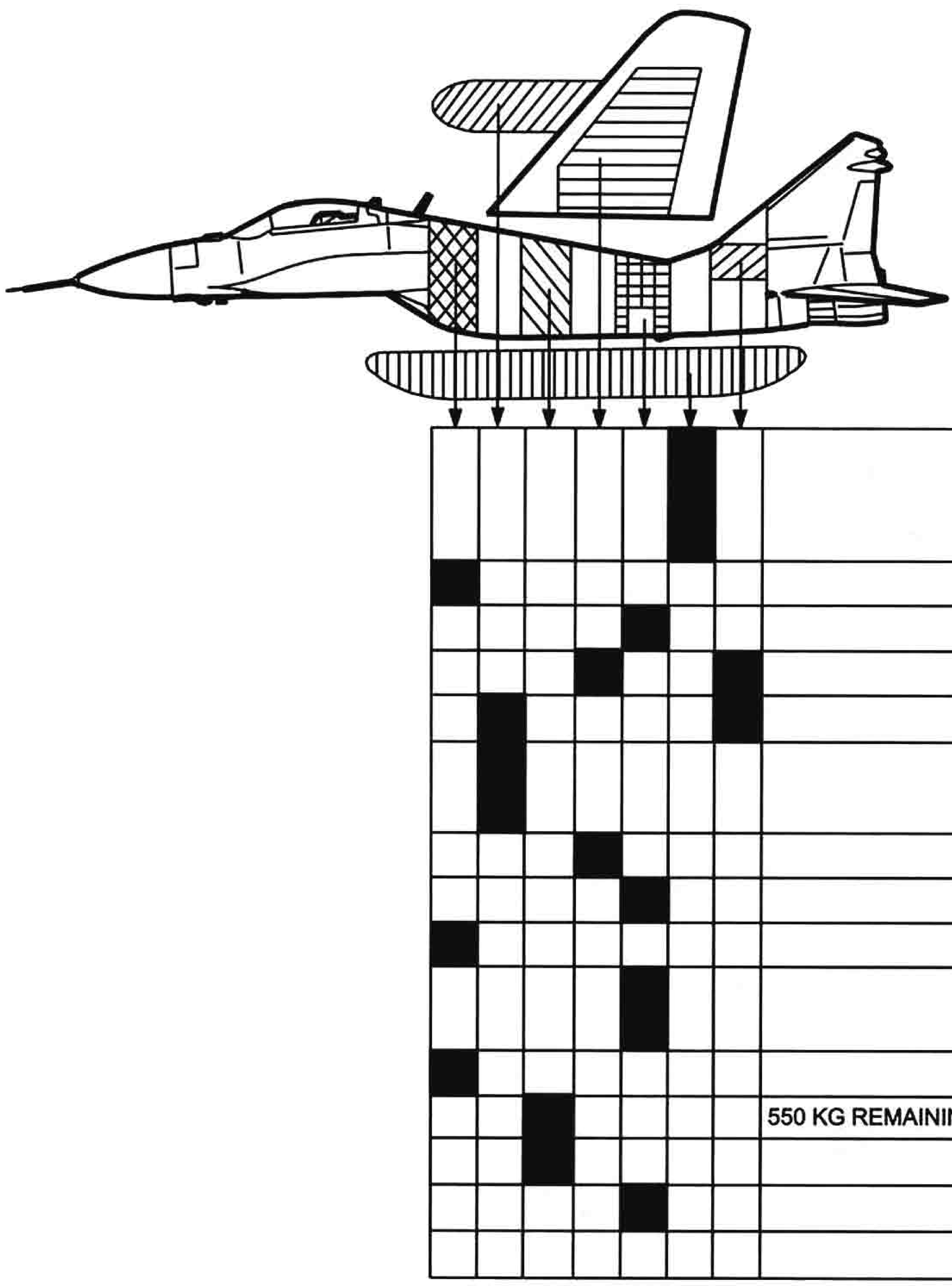
GT:

	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

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.

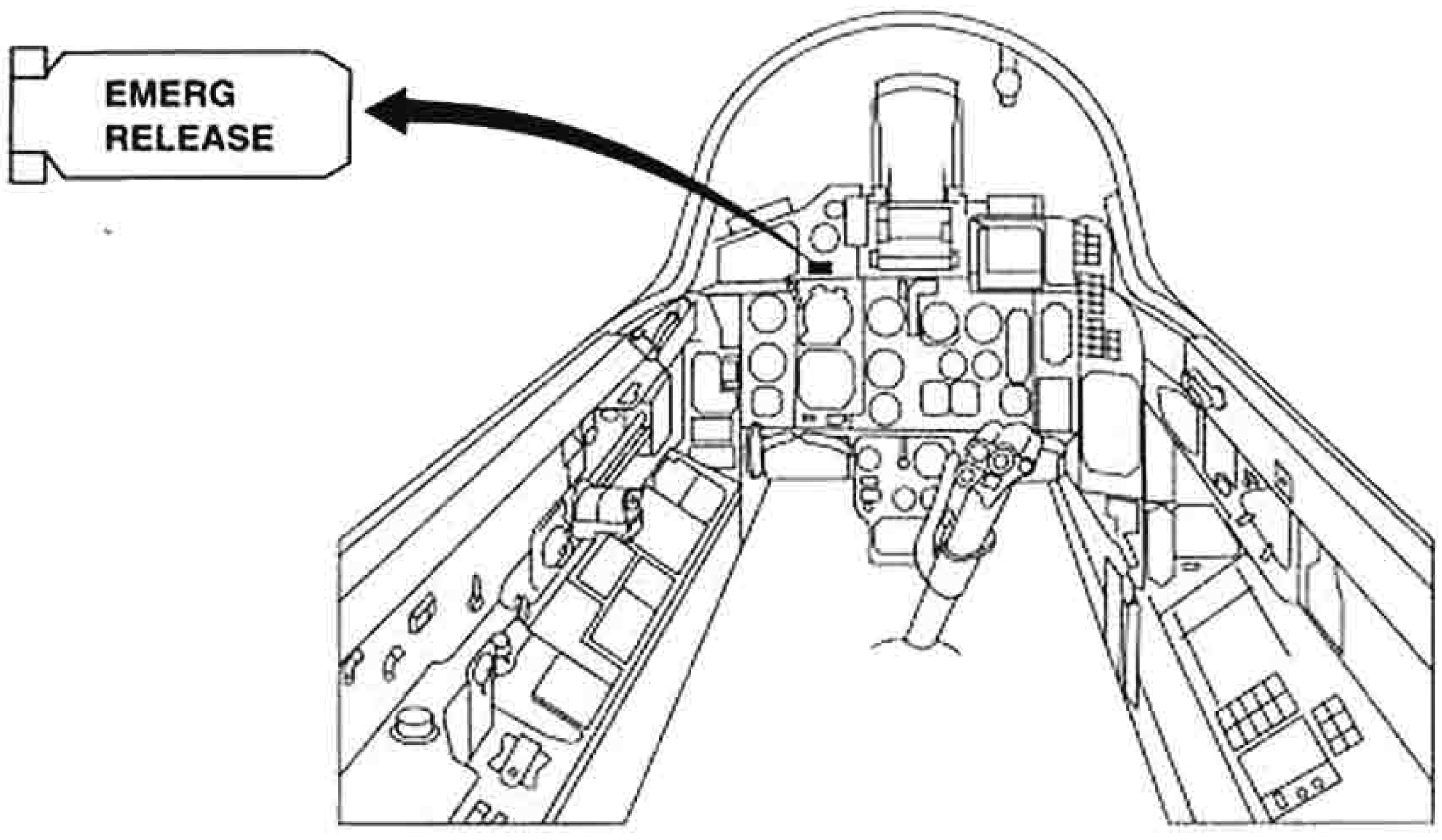


Figure 1-19B

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.

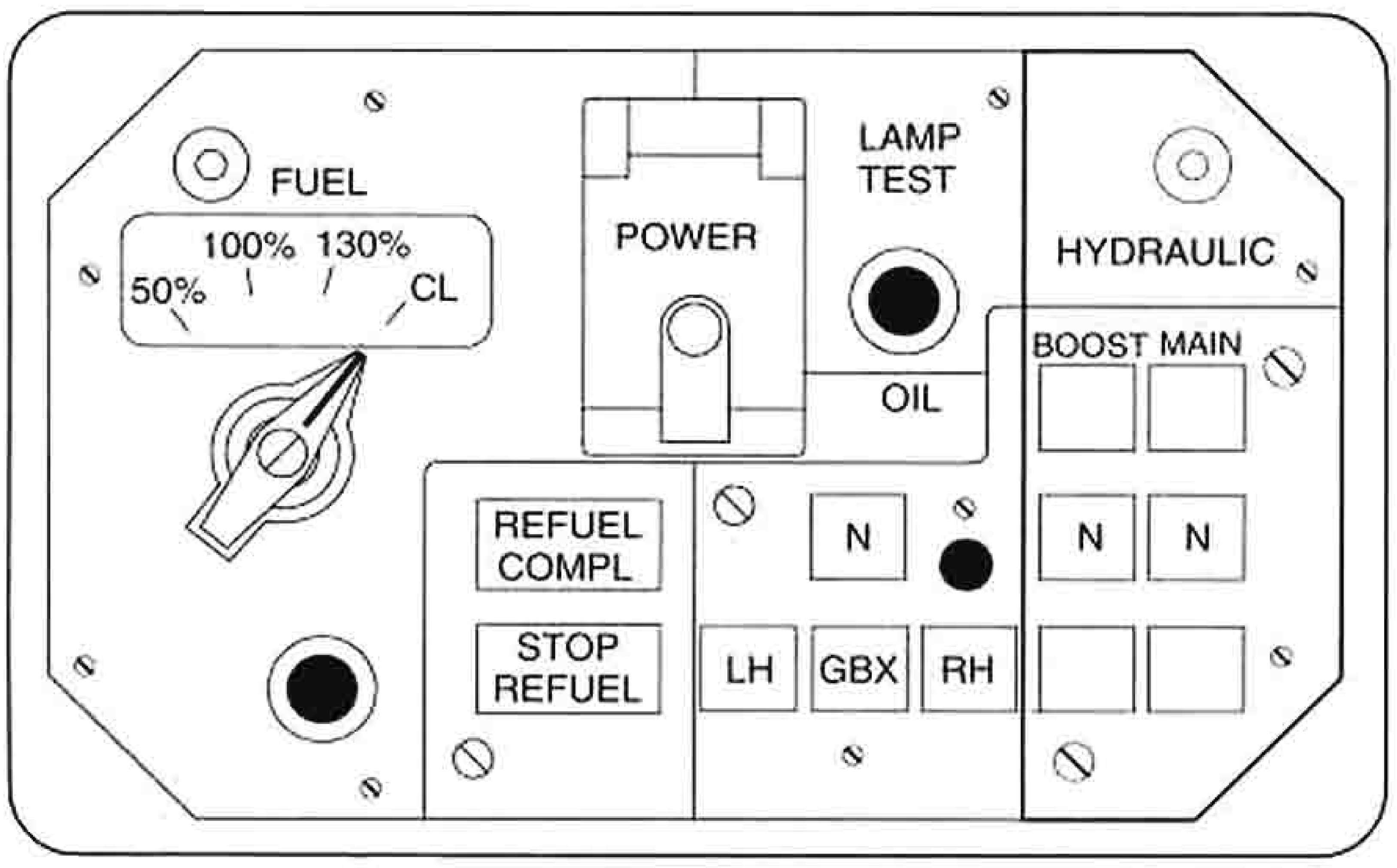


Figure 1-19D

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

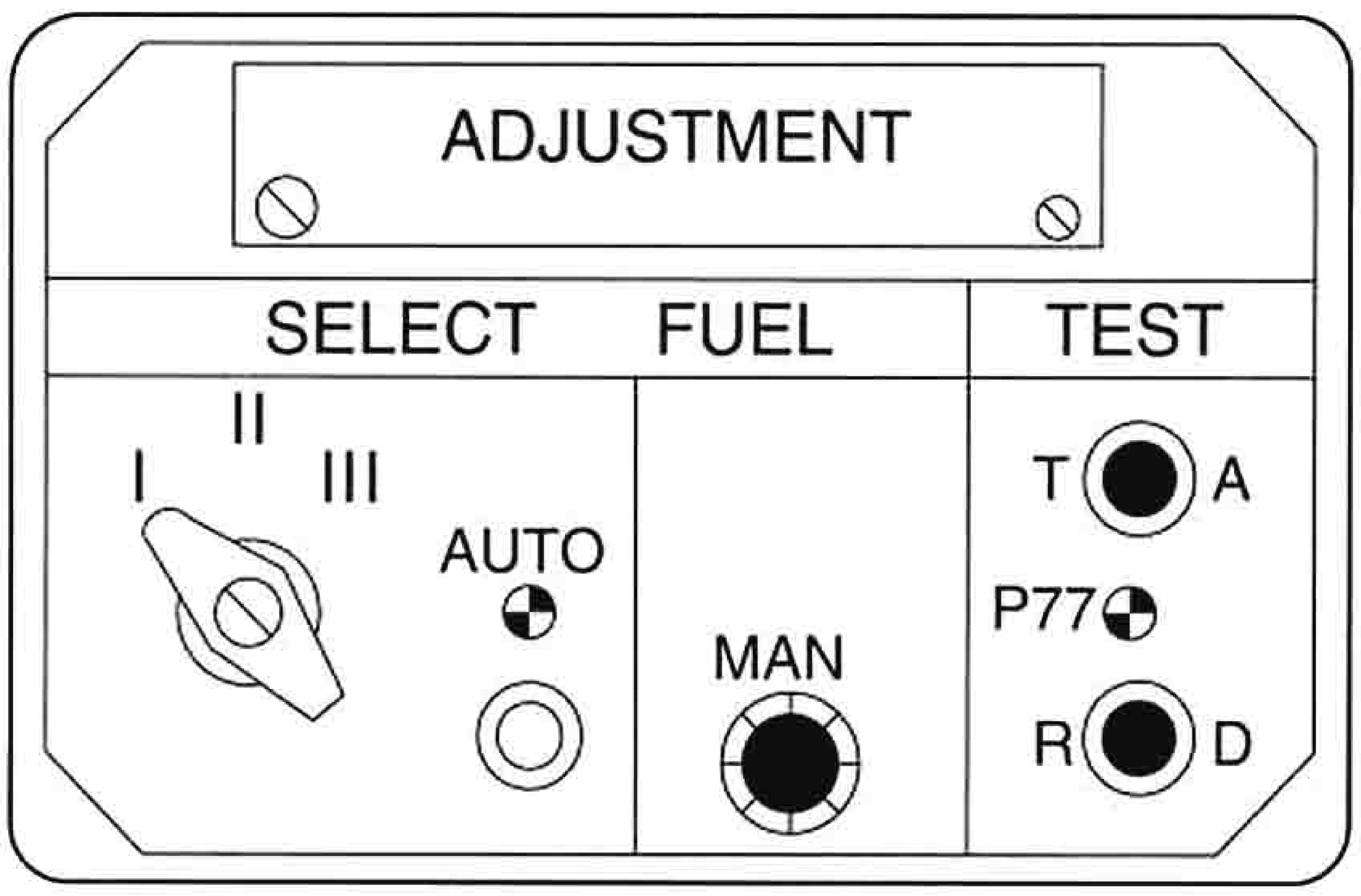


Figure 1-19E

AFTER MODIFICATION WITH WING DROP TANKS

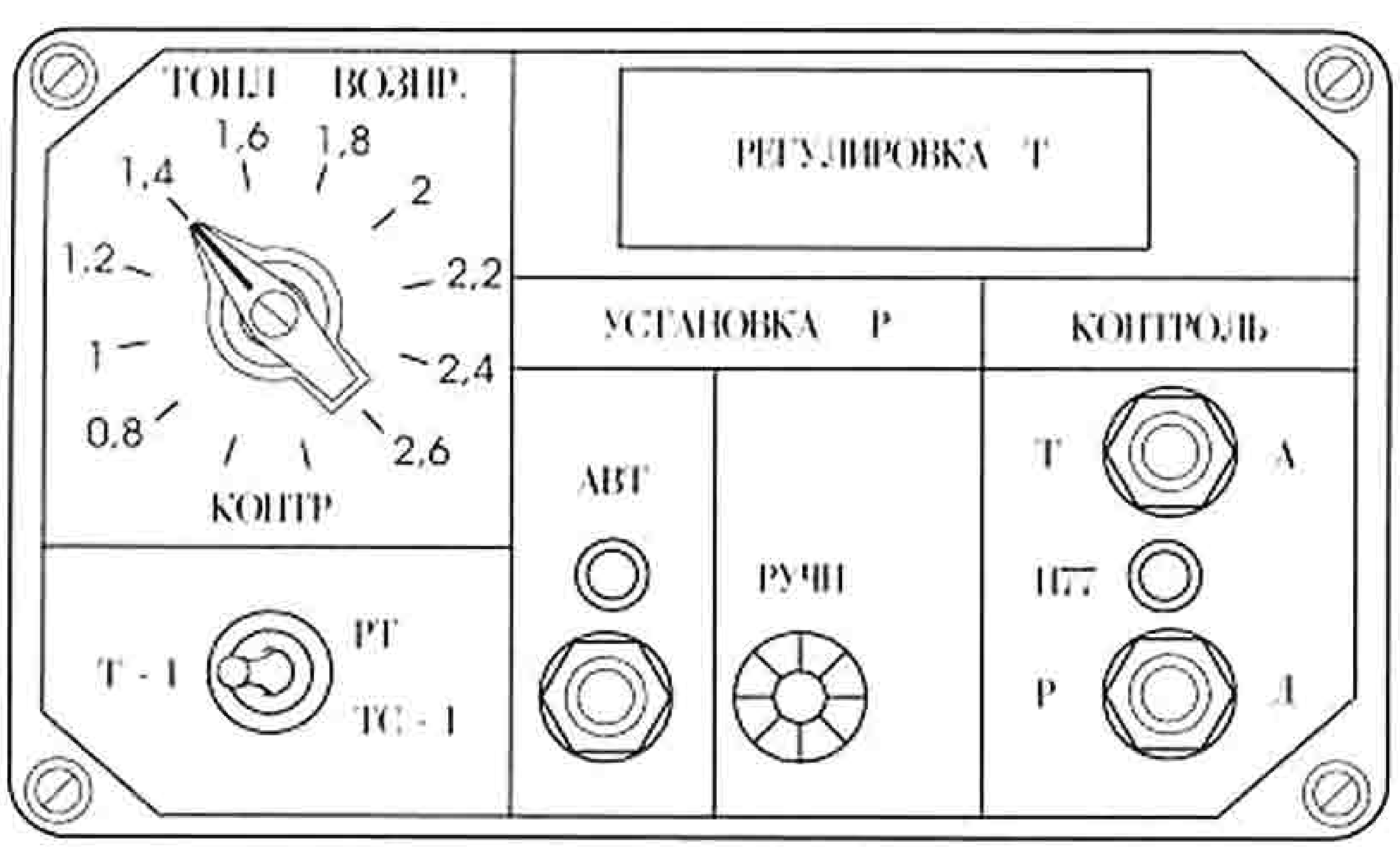


Figure 1-19F

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.

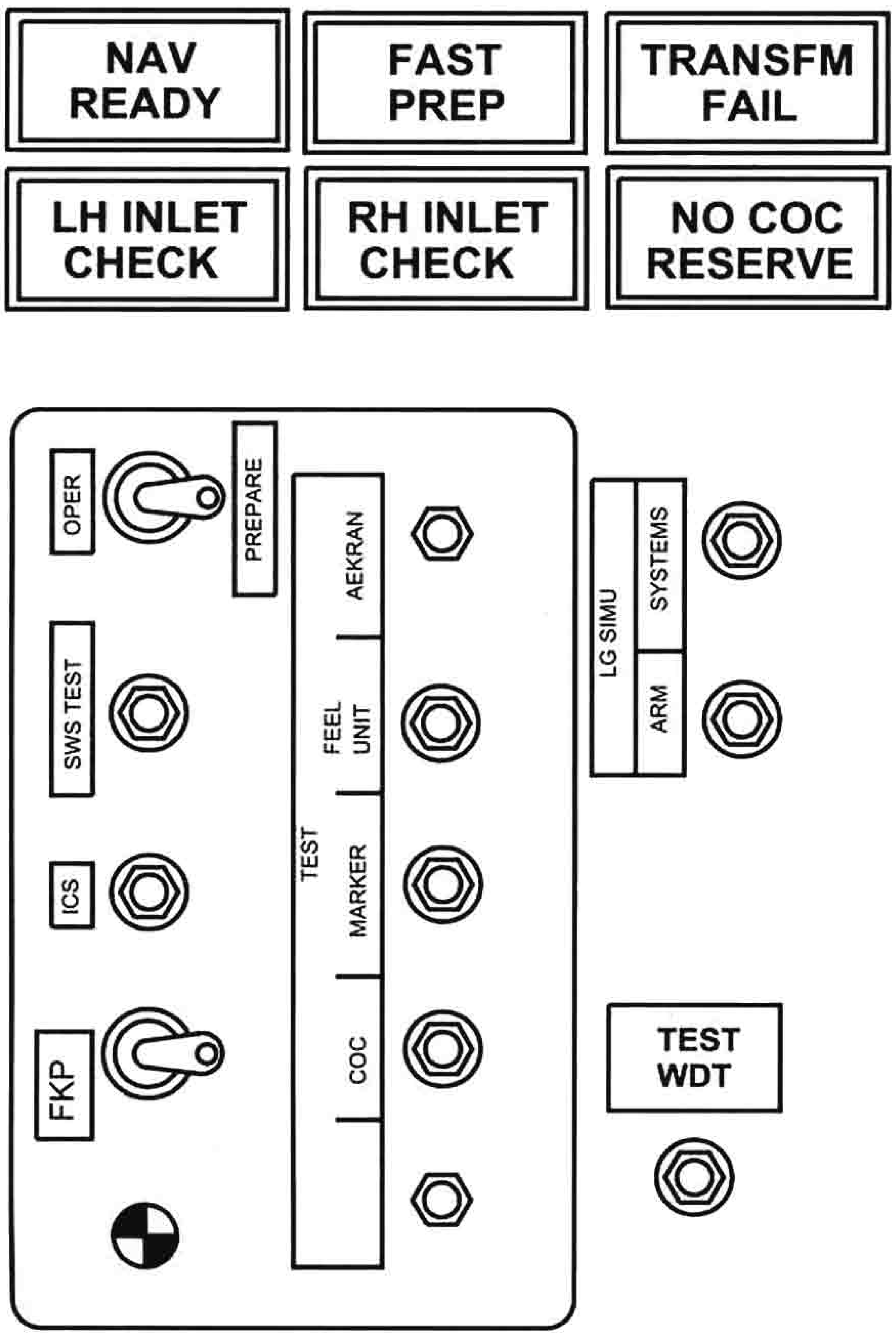


Figure 1-19G

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

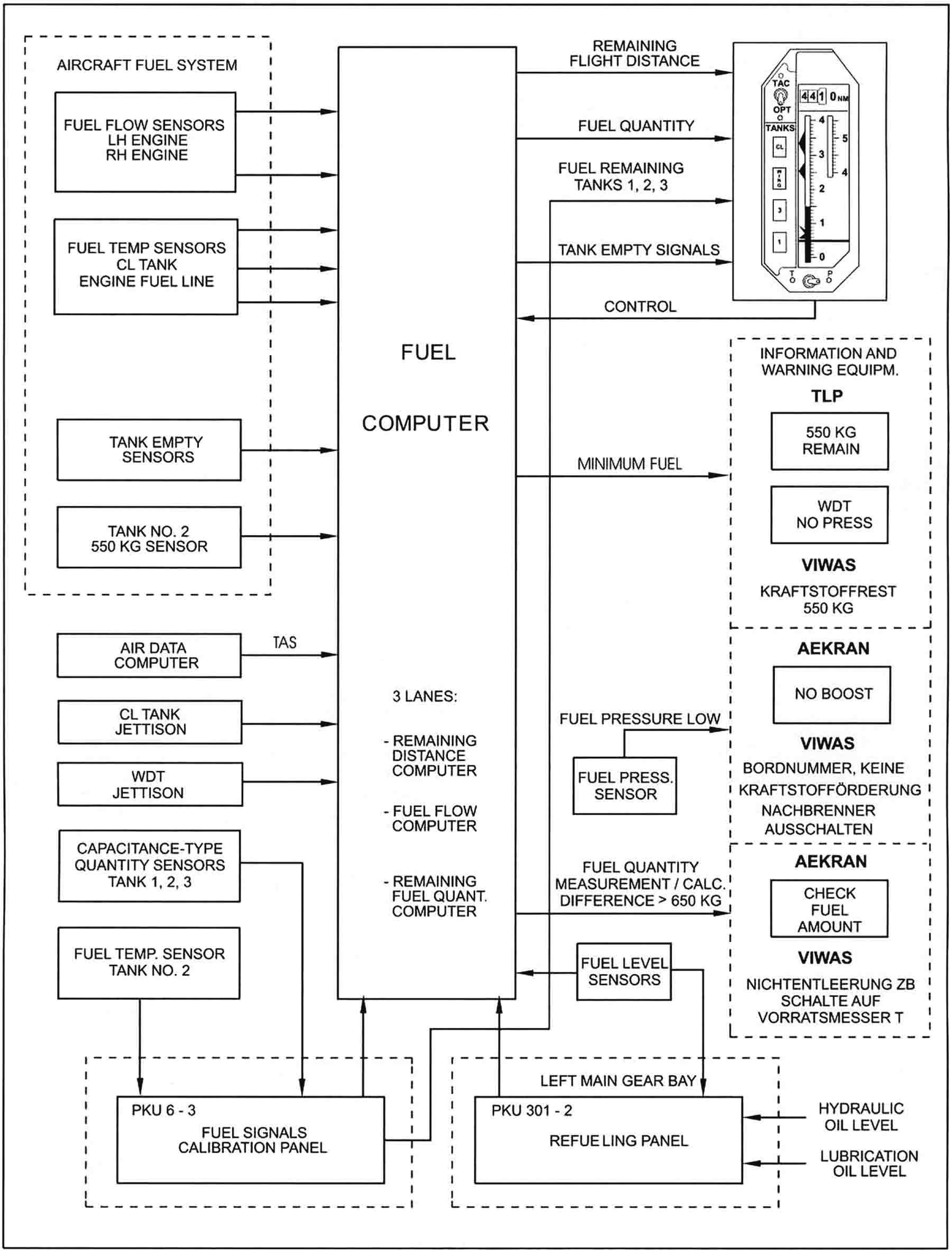


Figure 1-20

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.

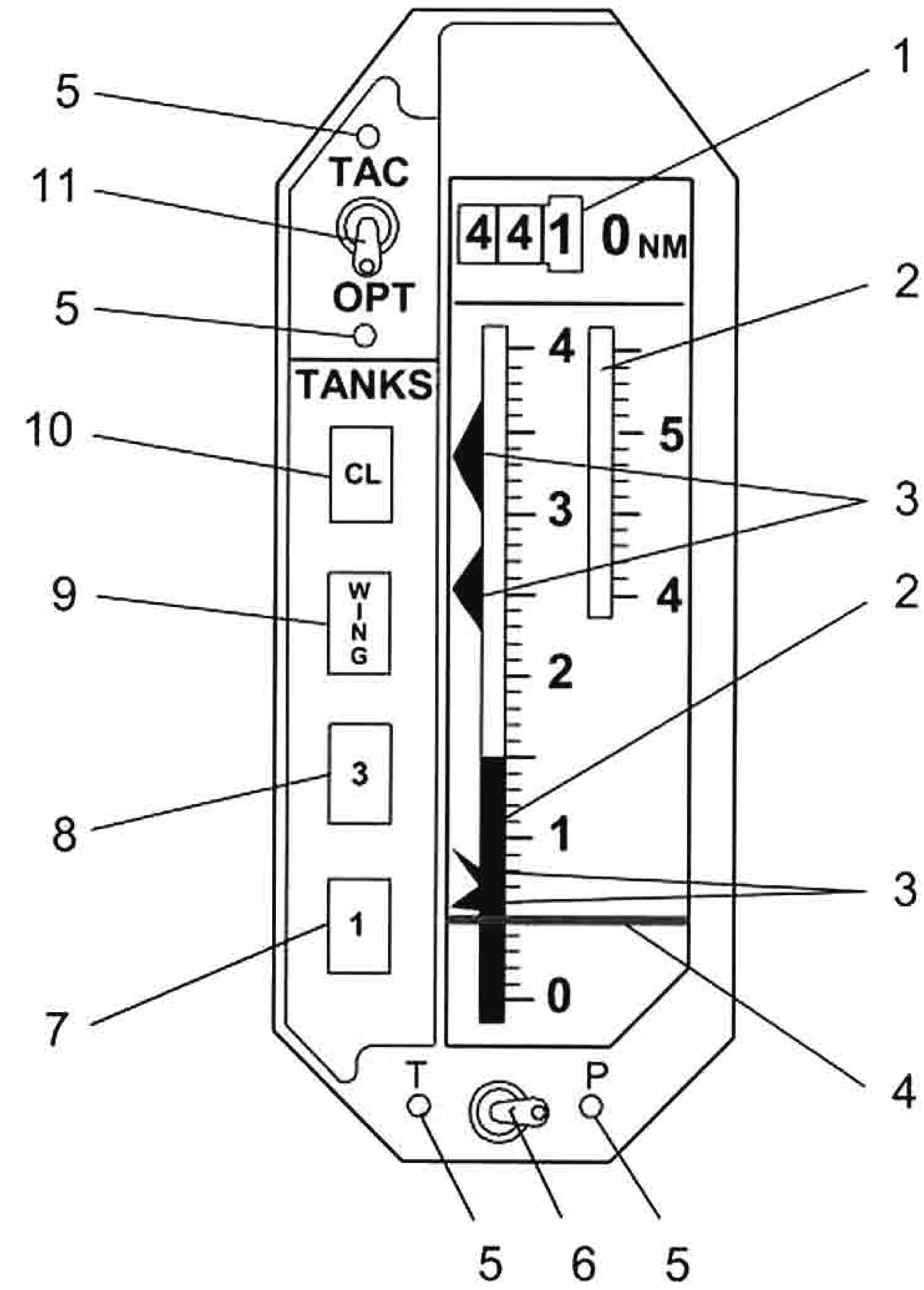
Four triangular shaped markers along the scale indicate the tolerable remaining fuel quantity when the particular tank empty caption illuminates. The markers point in the direction of the corresponding indicator caption.

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

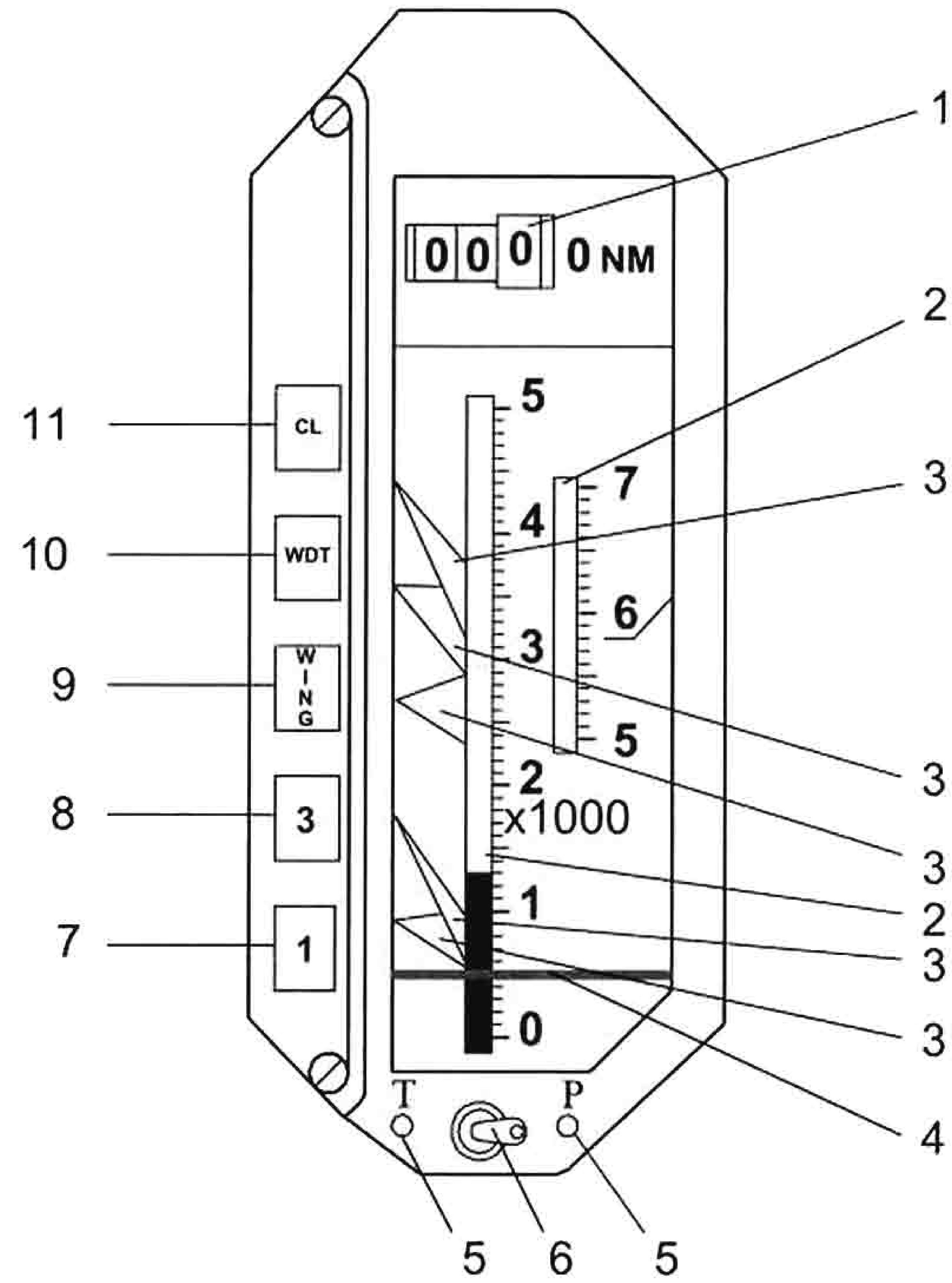
- During level, unaccelerated flight without afterburner, the tank 1 empty caption must not illuminate prior to illumination of the tank 3 empty caption.
- After all fuel from the wing drop tank has been transferred, the wing drop tank empty light caption may flash.

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.

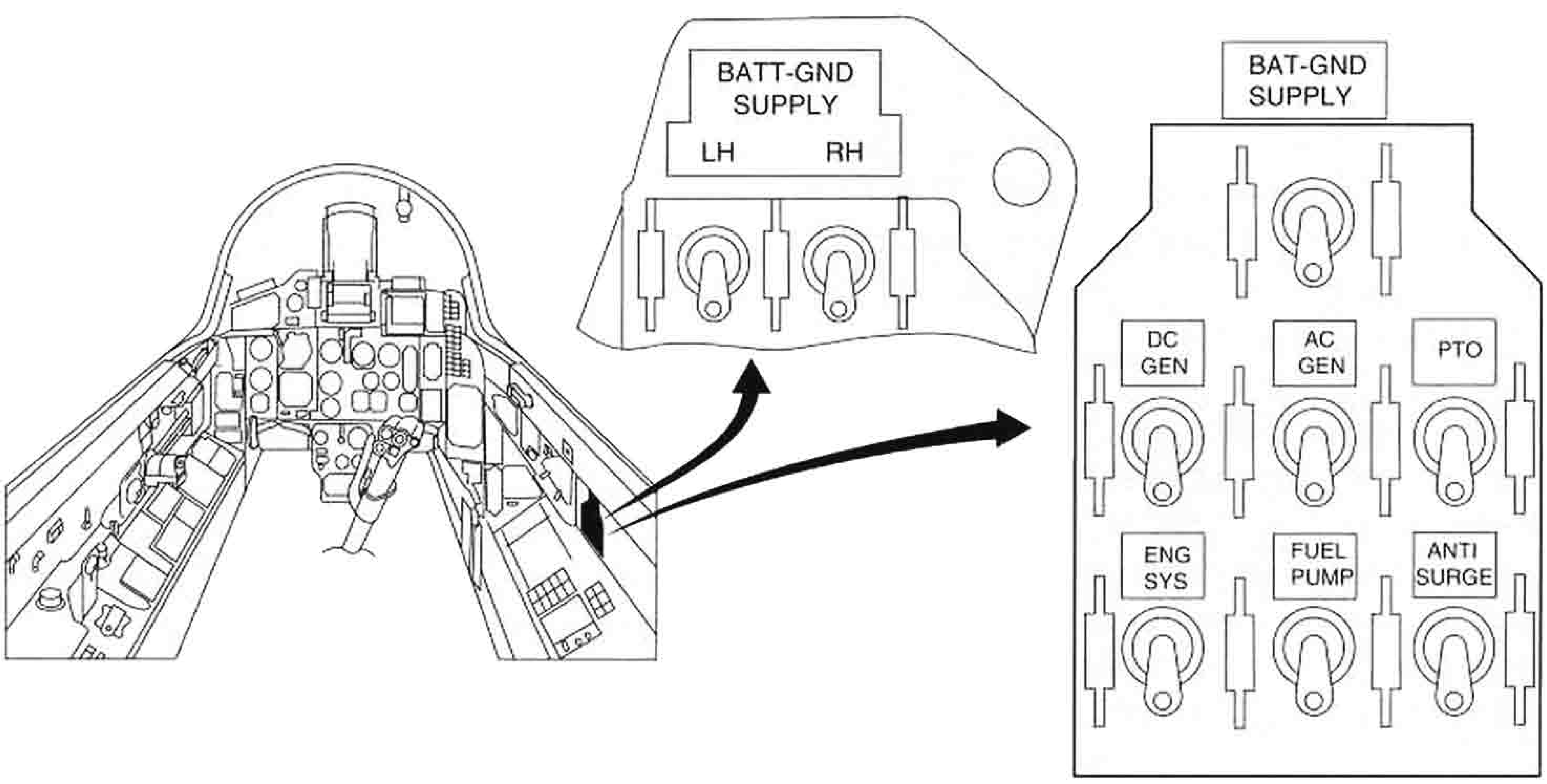


Figure 1-22

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

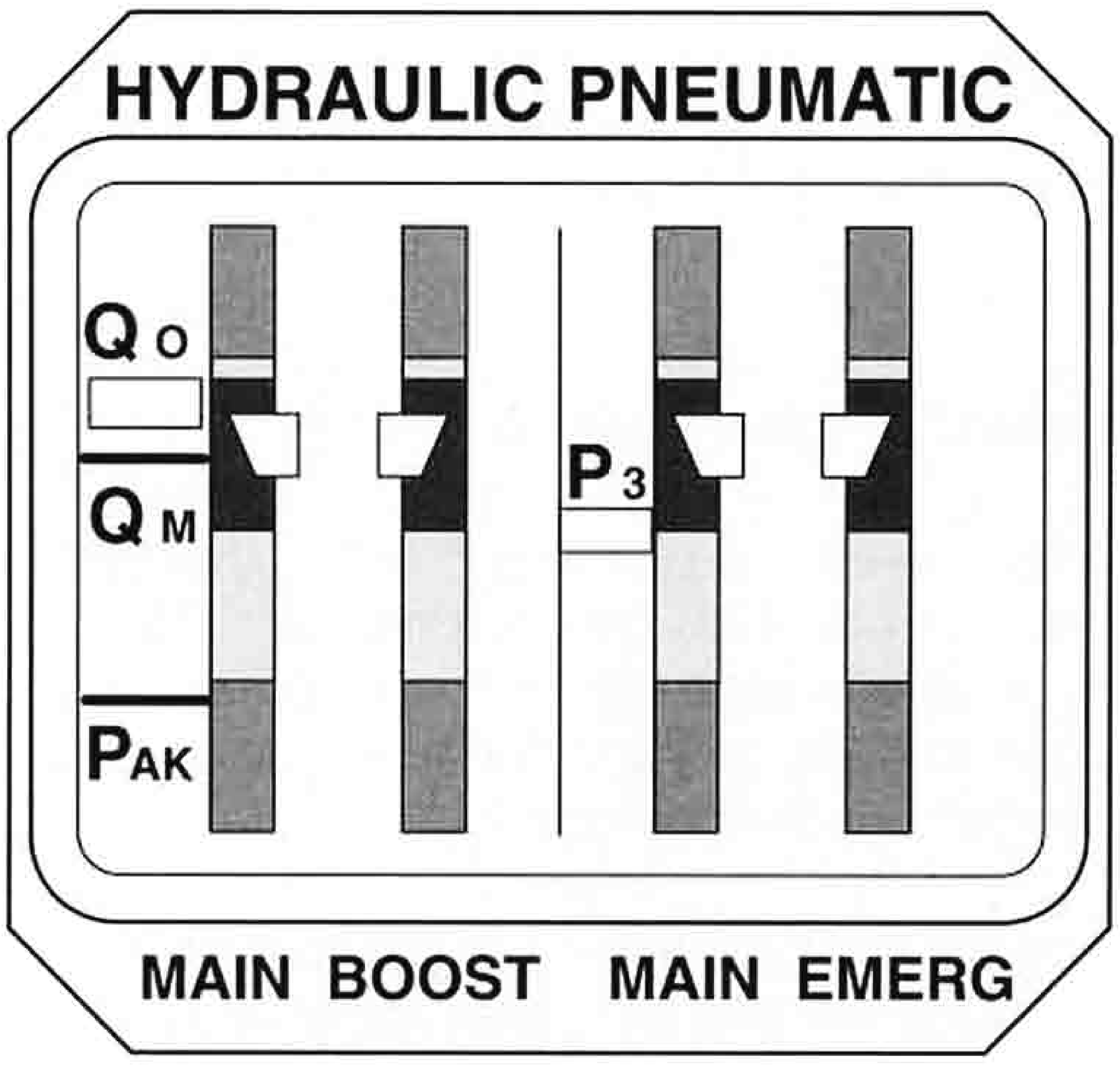


Figure 1-23

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

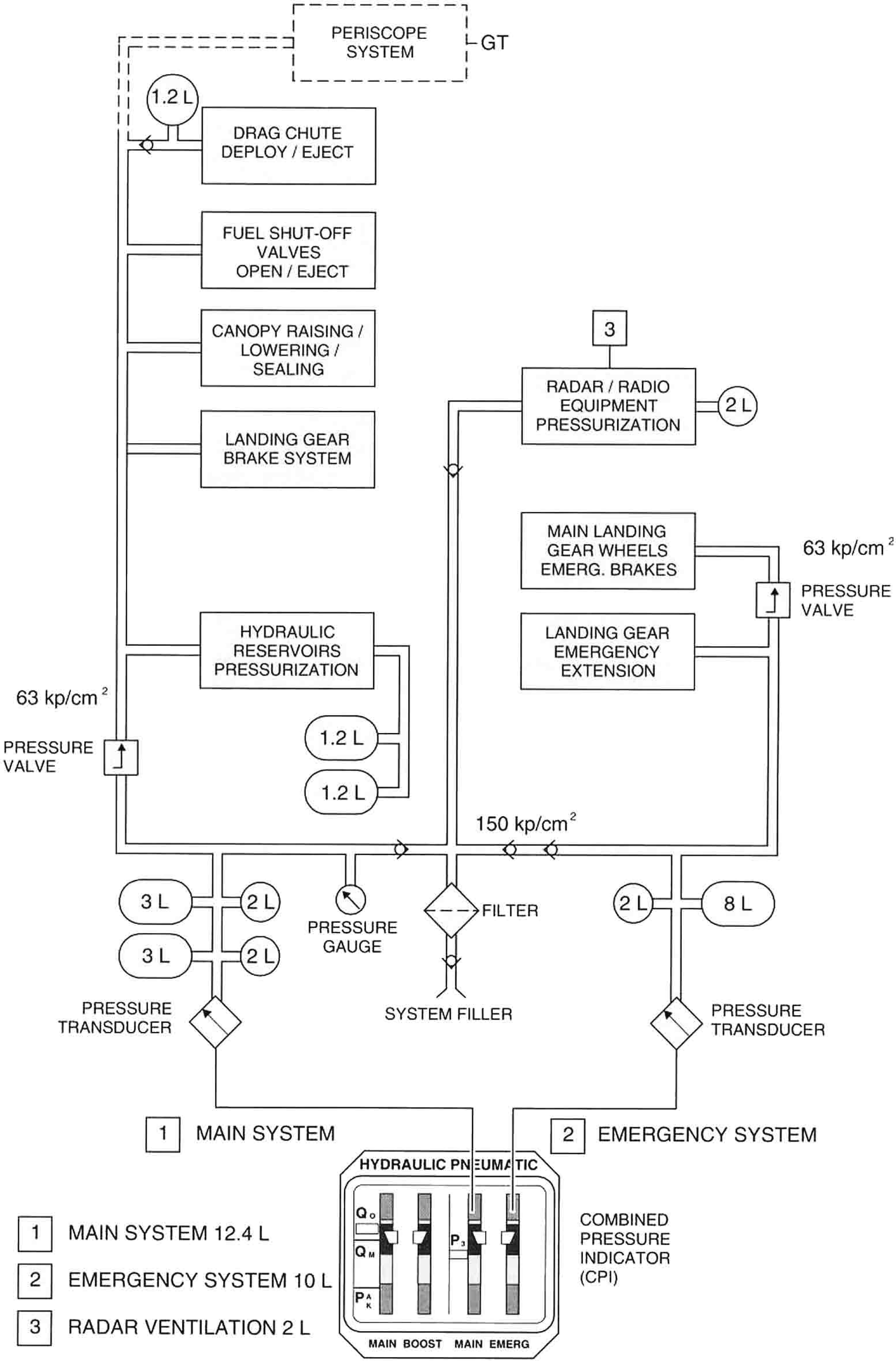


Figure 1-24