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ROYAL AUSTRALIAN AIR FORCE

DEFENCE INSTRUCTION (AIR FORCE)

AAP 7213.003-2-14B2

MIRAGE AIRCRAFT GENERAL AND TECHNICAL MANUAL

(BOOK 2 OF 4)

Date of Issue: 19SEP83

(S. D. EVANS)
Air Marshal
Chief of the Air Staff

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AMENDMENTS TO PUBLICATION

1. Attention is drawn to DI(AF) ADMIN 1-2, paras 17 and 18, in regard to amending this publication. In particular the member to whom this publication is on 'temporary issue' is personally responsible for:

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LIST OF EFFECTIVE PAGES

NOTE

The portion of text affected by an AL is indicated by a thick vertical line in the outer margin of the page.

Subject	Page No	AL No
PRELIMINARY PAGES		
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* The asterisk indicates pages changed by the current AL.

AMENDMENT CERTIFICATE

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

Amendment List		*Topic Affected*	** Amendment Effect*	*Amended by*	*Date*
No	*Dated*				
1					
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* **Note.** Insert brief details of page(s) amended, inserted or cancelled.

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NOTE

A comprehensive Table of Contents/List of Figures precedes each chapter.

LIST OF TABLES

NOTE

A List of Tables is contained in the Table of Contents that precedes each chapter in Books 1 and 2.

LIST OF FIGURES

NOTE

A List of Figures precedes each chapter in Books 3 and 4.

MODIFICATION STATE

1. As all RAAF Mirage aircraft had all modifications up to Modification 177 incorporated prior to delivery, it is assumed that the original General and Technical Information publications would have reflected that modification status.

2. The Modifications listed below have been identified in the original General and Technical Information publications, and have been incorporated (unchanged) in this publication.

179	Electrical — Provision of Noise Suppression Filter in Radar Wiring
180	Electrical — To Avoid UNDERCARRIAGE NOT DOWN Warning Light from Operating at LOW IAS at Altitude
197	Electrical — Provision for Future Fitment of Flasher Unit in Undercarriage Warning System
213	Air Conditioning — Provision of Filter in Radar Pressurization Ducting
237	Fuselage — Introduction of Hydraulically-Controlled Distributor in the Supply Line to the U/C and U/C Doors Hydraulic Electro Distributors
258	Electrical Reposition of Sperry Flux Valve in the Fin in lieu of the Wing
266	Fuselage — Installation of the Flasher Unit of the U/C DOWN, LOCKED FORWARD, Indicator in the Cockpit
322	Electrical — Introduction of a Connector in the Cockpit and Equipment Regulation Amplifier Circuits
329	Radio — Provision of Facilities for GPL Doppler Type AN/APN-153(V)
365	Radio — Introduction of Estimated Point Marker on Radar Scope and Provision for Drift Alignment
406	Wings — Introduction of Leading Edge Fuel Tank
614	Fuselage — Introduction of an Access Door to the Throttle Control Microswitches
635	Fuselage — Introduction of New Type Canopy Seal
826	Electrical — Introduction of a Master Armament Switch
835	Instrument — Mirage III D Aircraft — Revised Static Pressure Supply for Rear Cockpit Vertical Speed Indicator

3. The Modifications listed below have been incorporated prior to the issue of this Publication.

572	Fuselage — Nose Gear Leg — Revised Charging Pressure for the Shock Absorber
591	Fuselage — Nose Gear Leg — Rework of Truss Universal Joint Pin
725	Undercarriage — Landing Gear Hydraulic System — Installation of a Shuttle and Non-return Valve
881	Anti-G Piping — Technological Improvements of Disconnectable Elbow Coupling
919	Isolation of the AC Power Supply to the MATRA Missile Computer and Harmonization Unit
926	Electrical — Improvements to Air Weapons — Switching Selections in Mirage III O and III D Aircraft
958	Fitment of Recording Equipment to Monitor Live Missile Firing
978	Conversion of Type OM4 Ejection Seat to Type OM6
1112	Provision on Aircraft for Carrying MATRA R550 Missiles at No 2 Attachment Points
1113	Provision on Aircraft for Quick Select Gun Plus Magic Missile Firing Mode
1124	Cockpit Modification to Facilitate OM6 Ejection Seat

4. The Modifications listed below have been checked and found to have no effect on this Publication.

190	Electrical — Introduction of a Link between Cyrano Radar and MATRA 530 Harmonization Unit
227	Fuselage — Introduction of Messier Type A5 × 22 879/0 Nose Gear Leg in lieu of Type A22 879M50
240	Wings — Introduction of Messier Type A33 × 21 289M2/0 Brake Assemblies in lieu of Type A33 × 21 289M1/0
241	Wings — Introduction of Messier Type A4X 24372 M1 and A5-24372M1 Undercarriage Sequence Distributors in lieu of Types A4X 24372 and A5X 24372
242	Wings — Introduction of Messier Type A33-202.89.M2/0 Main Wheels in lieu of Type A33-202.89/0
265	Wings — Introduction of Improved Main U/C Oleo Legs Messier Type A1X 23-317/0V3 and A1X23-318/0V3 in lieu of A1X23-317/0 and A1X23-318/0
287	Wings — Introduction of Improved Main U/C Actuating Truss Jack Type Messier A1X23-177 M1/0 (LH) and A1X23-178 (RH) in lieu of Messier A1X23-177/0 (LH) and A1X23-178/0 (RH)
407	Electrical — Installation of Firing Equipment for MATRA Missile

429 Wings — Introduction of Main U/C Legs Messier A1-23317M1-0 L/H and A1-23318M1-0 R/H in lieu of Type A1X23317/0 L/H and A1X23318/0 R/H

451 Fuselage — Introduction of Nose Gear Truss Jack Type A1-22786.M1 in lieu of Type A1-22786

543 Wings — Main Landing Gear Jacks — Introduction of Improved Seals on the Sliding Shafts

544 Fuselage — L/G Doors — Introduction of Improved Unlocking Interlinking Controls

566 Equipment — MATRA Missile Launcher — Locking of Co-axial Breakaway Connectors

567 Fuselage — Introduction of Retaining Cable for the Access Door Supporting Missile Supporting Connectors

573 Fuselage — Introduction of Messier Nose Gear Leg Type A15-22879-0 in lieu of Type A5-22879-0

574 Fuselage — Introduction of Messier Nose Gear Leg Type A20-2287900 in lieu of Type A15-22879-0

636 Wings — Introduction of Main L/G Wheels Type A33-20289-M3-0 in lieu of Type A33-20289-M2-0

637 Wings — Main Wheel Assemblies — Increased Clearance between Brake Block and Wheel

649 Fuselage — Nose Wheel Gear — Improvements to Uplock

664 Main Landing Gear Jacks — Local Protection Against Corrosion

667 Fuselage No 1 Hydraulic System — Anti-surge Accumulator PN A4-23139-0 — Improved Locking of Quick Disconnect Coupling

687 Electrical — Missile Firing Circuit — Sustained Priming Function

759 Fuselage — Gate Valve Fitted in Temperature Regulating Circuits — Improvement of Sliding Conditions of Gate

805 Undercarriage — Nose Gear Truss — Improved Lubrication

840 Fuselage — Attachment of Undercarriage Emergency Hydraulic Pipe in Alternator Compartment

882 Main Landing Gear Legs — New Adjustment of the Damper

885 42 L/MIN Hydraulic Pumps

895 Main Landing Gear Jack — New Nut for the Jack Bottom

945 Main Landing Gear — Improvement of the Jack Extended Locking System

951 Hydraulic Pumps — New Compensation Spring on the Swash Plate

982 Introduction of Aeroquip Hydraulic Hoses

998 To Improve Sealing of the Nose Gear Leg Damper at the Level of the Main Piston

CHAPTER 9

POWER PLANT

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CHAPTER 9

POWER PLANT

Table 9-1 Function of Components

Index No	Description		Characteristics and Functions
REFUELLING — VAPOUR RELIEF			
5Q 6Q	LP pumps (see normal operation)		Deliver fuel to the engine feed line. Accelerate the refuelling of the rear bay tank, wing tanks, gun bay tank and inverted flight accumulator; flow rate : 2640 U.K. gal (12000 l/hr).
21Q	Cross-feed valve		This is an electrically operated valve with microswitch limit switches. In the closed position, it isolates the LH and RH parts of the fuel system. In the open position, it enables the fuel levels to be balanced in the fuselage tanks (at the end of transfer); in this position, it also permits the fuel tanks to be refuelled from one side only.
30	Non-return valve		This valve prevents the fuel from being discharged to the fuselage tanks (when the engine is supplied from the inverted flight accumulator or in case of failure of a LP pump).
22 and 28	Manual controls (75)	Refuelling valves	In the open position, these valves enable refuelling of the wing tanks, the bay tank and the gun bay tank. These valves are closed in flight.
49 and 68		Manually operated vapour relief valves	In the open position, these valves allow the vapours to escape from the fuel tanks. They are closed in flight.

FUEL SUPPLY SYSTEM		
3Q — 4Q	LP pump switches	These are the energization relays of the LP pumps. They are energized by operating the control switches 9Q and 10Q.
5Q — 6Q	LP pumps	These fuel pumps deliver 2640 U.K. gal (12000 l/hr). They supply the engine feed pipe and the inverted flight accumulator (see Refuelling — Vapour Relief).
9Q — 10Q	LP pump control switches	When placed in the ON position, these control switches are used to energize the LP pump power supply switches (3Q — 4Q).
11Q — 12Q	LP pump noise suppressors	These are used for noise suppression on the LP pumps (5Q — 6Q).
16Q 216Q	LP cock control switches 216Q — III D Only	When placed in the ON position, these switches are used to directly supply the LP cock (17Q).
17Q	LP cock	This cock controls the fuel supply to the engine.
13K 213K	Afterburner cock control switches 213K — III D Only	These switches are lock-wired in the ON position for direct control of the afterburner cock. They are opened in normal flight and closed in case of afterburner fire.
14K	Afterburner microswitch (in front cockpit power control quadrant)	This microswitch is operated as the front cockpit (or rear cockpit for III D) throttle control lever is being tilted; it is used to open the turbo-pump electro-valve as well as the afterburner ignition electro-valve.
18K	Afterburner cock	This cock is electrically controlled by switch (13K); it provides for fuel supply to the afterburner.
20K	Afterburner electro-valve	This valve is controlled by microswitch (14K); it is provided to control the P2 air supply (from 9th compressor stage) to the afterburner turbo-pump.

Index No	Description	Characteristics and Functions
	FUEL SUPPLY SYSTEM	
4.	Inverted flight accumulator	This is used to supply the engine when the pumps are no longer submerged. Under the action of the P2 reduced pressure, the accumulator diaphragm then forces the fuel out. The accumulator is automatically refilled in normal flight as soon as the pumps are submerged again. The accumulator incorporates an expansion valve calibrated at 40.6 lb/in² (2800 mb), provided to cater for any possible fuel expansion on the ground resulting from a temperature rise.
30	Non-return valve	See Refuelling — Vapour Relief.
33	Turbo-pump	This is a high-pressure fuel pump driven by a turbine using P2 compressor outlet pressure as a driving agent.
36.	Pressure reducing valve (pressurization of fuselage tanks)	This valve is used to lower the pressure of the P2 air from the engine compressor to a relative overpressure of 1.45 lb/in² (103 mb). It is provided with a built-in safety valve (39) calibrated at 2.17 to 4.35 lb/in² (152 to 305 mb) and a built-in restrictor limiting the air flow.
37 — 41	Non-return valves	These valves prevent the fuel from flowing to the engine during refuelling.
38	Air filter	This filter protects the accessories against entry of foreign matter.
40.	Pressure reducing valve (pressurization of inverted flight accumulator)	This valve drops the pressure of the P2 air from the engine down to 11.6 lb/in² (800 mb) in the inverted flight accumulator. It is provided with a built-in safety valve (42) calibrated at 15.9 lb/in² (1100 mb) and a built-in restrictor limiting the air flow.
59	Pressure relief valve	This valve protects the LH and RH fuselage fuel tank groups; it is calibrated at 2.2 lb/in2 (150 mb) and is provided with a built-in filter.
60.	Underpressure valve	This valve is provided with a built-in filter. It supplies the LH and RH fuselage fuel tank groups through the emergency system; it is calibrated at 0.38 lb/in² (26 mb).
61	Pressure relief valve	This valve protects the air supply system during inverted flight; it is calibrated at 10.73 lb/in² (750 mb).

	WING PYLON TANK TRANSFER SYSTEM	
31Q	LH pylon tank pressure switch	This pressure switch is used to close the electric circuit of the fuel transfer indicator light when the fuel pressure decreases (end of transfer).
32Q	RH pylon tank pressure switch	This pressure switch is used to close the electric circuit of the fuel transfer indicator light when the fuel pressure decreases (end of transfer).
40Q 240Q	Fuel transfer indicators 240Q — III D Only	Amber lights — (push-to-test type). These are used to indicate the end of the transfer of the pylon tanks, rear bay tank and gun bay tank.
6	242 — 374 U.K. gal (1100 — 1700 l) pylon tanks III O Only, 286 — 374 U.K. gal (1300 — 1700 l)	Containing : Two tank bottom valves (float valves) one pressure relief valve calibrated at 10.2 + 0 − 0.6 lb/in² (65) One vacuum relief valve calibrated at 0.22 lb/in² (15 mb) (65)

Index No	Description	Characteristics and Functions
WING PYLON TANK TRANSFER SYSTEM		
52	Pressure reducing valve (pressurization)	This valve is used to lower the pressure of the P2 air from the engine compressor down to 10.15 lb/in^2 (697 mb) above ambient in the pylon tanks. It is provided with a built-in restrictor limiting the air flow and a built-in safety valve (58) calibrated at 12.33 lb/in^2 (852 mb).
53 — 55	Non-return valve	This valve maintains the pressure in the pylon tanks in case of failure of the pressure reducing valve. It prevents the pressure from dropping in one tank if the other tank is leaking.
54 — 56	Air filter	This filter protects the accessories against entry of foreign matter.
57	Self-sealing coupling	This coupling is open when the tank is installed; it is closed to prevent any air loss when the tank is not installed or after it has been jettisoned.
79	Fuel filter	This filter protects the transfer system against entry of foreign matter.

FUSELAGE PYLON TANK AND GUN BAY TANK TRANSFER SYSTEM		
19Q	Transfer switch (front cockpit)	This switch electrically controls the transfer valve of the gun bay fuel tank.
22Q	Fuel transfer valve	In the open position, this valve provides for transfer of the fuel from the gun bay tank to the fuselage tanks. In the closed position, it prevents automatic transfer of the gun bay tank to permit transfer of the fuselage pylon tank.
37Q	Gun bay tank pressure switch	This pressure switch is used to close the electric circuit of the fuel transfer indicator light when the fuel pressure decreases (end of transfer).
153	Fuselage pylon tank pressure switch	This pressure switch is used to close the electric circuit of the fuel transfer indicator light when the fuel pressure decreases (end of transfer).
40Q 240Q	Fuel transfer indicator 240Q — III D Only.	See Wing Pylon Tank Transfer System.
85	Pressure reducing valve (gun bay tank pressurization)	This valve drops the pressure of the P2 air from the engine to between 8.7 lb/in^2 (600 mb) and 9.71 lb/in^2 (670 mb) in the gun bay fuel tank. It is provided with a built-in safety valve (90) calibrated from 11.13 to 12.67 lb/in^2 (780 to 880 mb).
86 — 102	Non-return valves	These valves are used to maintain the pressure in the fuel tanks in case of failure of the pressure reducing valve. They also prevent the fuel from flowing to the pressure reducing valve during refuelling.
87	Filter	This filter protects the accessories against entry of foreign matter.
89 — 91	Valve box	This valve box is provided to protect the fuel tank against excessive overpressures or underpressures Overpressure : 9.9 to 11.8 lb/in^2 (686 to 814 mb) Underpressure : -0.07 lb/in^2 (-5 mb)
100	Non-return valve	This valve is open when the fuel is being transferred from the fuselage pylon tank. It is closed during transfer of the fuel from the gun bay tank to prevent the fuel from returning to the fuselage pylon tank.

Index No	Description	Characteristics and Functions
	FUSELAGE PYLON TANK AND GUN BAY TANK TRANSFER SYSTEM	
101	Pressure reducing valve	This valve drops the pressure of the P2 air from the engine compressor to between 9.7 lb/in^2 (670 mb) and 10.6 lb/in^2 (730 mb) in the fuselage pylon tank. It is provided with a built-in safety valve (103) calibrated from 12.3 lb/in^2 (850 mb) to 13.8 lb/in^2 (950 mb).
102	Non-return valve	This valve is used to maintain the pressure in the tank in case of failure of the pressure reducing valve.
104	Valve box	This valve box is provided to protect the fuel tank against excessive overpressures or underpressures Overpressure : 10.8 lb/in^2 (750 mb) Underpressure : 0.29 lb/in^2 (20 mb)
108	Filter	This filter protects the accessories against entry of foreign matter.
	ROCKET MOTOR BAY TANK, REAR BAY TANK AND WING TANK TRANSFER SYSTEM	
35Q	LH wing : Rocket motor bay tank (rear bay tank, III D) pressure switch	This pressure switch closes the electric circuit to the fuel transfer indicator light when the fuel pressure starts decreasing (end of transfer).
36Q	RH wing : Rocket motor bay tank (rear bay tank, III D) pressure switch	See 35Q
40Q 240Q	Fuel transfer indicator panel 240Q — III D only	See External Wing Tank Transfer System)
43 — 46	Pressure reducing valves	These valves are used to lower the pressure of the P2 air from the engine compressor down to 8.7 ± 6% lb/in^2 (600 ± 6% mb) in the rocket motor bay tank (rear bay tank III D) and wing tanks. They are provided with a built-in safety valve (50 — 51) calibrated at 10.88 lb/in^2 (750 mb) and a built-in restrictor limiting the air flow.
44 — 47	Non-return valves	These valves maintain the pressure in the rocket motor bay tank (rear bay tank, III D) and wing tanks in case of failure of the pressure reducing valve. They also prevent the fuel from flowing to the pressure reducing valve during refuelling.
45 — 48	Filters	These filters protect the accessories against entry of foreign matter.
63 — 64	Valve box	This valve box is provided to protect the fuel tank against excessive overpressures or underpressures Overpressure : 9.4 to 11.4 lb/in^2 (647 to 784 mb) Underpressure : −0.007 lb/in^2 (−0.5 mb)
96 — 97	Restrictors	These restrictors limit the air flow in the fuselage tanks at the end of the wing tank fuel transfer.
	FUEL LOW LEVEL WARNING SYSTEM — CROSSFEED SYSTEM — FUEL QUANTITY INDICATING SYSTEM — INSTRUMENTS	
20Q	Crossfeed switch	Controls valve 21Q

Index No	Description	Characteristics and Functions
	FUEL LOW LEVEL WARNING SYSTEM — CROSS-FEED SYSTEM — FUEL QUANTITY INDICATING SYSTEM — INSTRUMENTS	
21Q	Crossfeed valve	See Refuelling — Vapour relief.
23Q	Relay box	This box is fed with signals from the tank units and causes the fuel low level warning light to illuminate.
26Q	RH fuel tank unit	Measures the fuel quantity in the RH fuselage tank group and feeds the fuel gauge with fuel quantity signals.
27Q	LH fuel tank unit	Measures the fuel quantity in the LH fuselage tank group and feeds the fuel gauge with fuel quantity signals.
28Q 228Q	Fuel low level warning light 228Q — III D Only	Red push-to-test light. This light is illuminated when 132 U.K. gal (600 l) fuel is remaining in the fuselage tanks.
29Q 229Q	Fuel gauge 229Q — III D Only	The fuel gauge includes two pointers marked P (port) and S (starboard). Indicates the fuel quantity in each fuselage tank group (including the inverted flight accumulator).
33Q	Push-to-test button (front cockpit)	This pushbutton enables the fuel quantity indicating system to be tested for correct operation.
38Q 39Q	RH amplifier LH amplifier	These amplifiers are supplied with 115 V power. They are used in conjunction with the fuel tank units and amplify the tank unit fuel quantity signals for the fuel gauge (29Q).
51Q 251Q	Fuel remaining indicator 251Q — III D Only	With the total fuel quantity aboard pre-set before the flight, this indicator subtracts the fuel quantities as they are consumed.
52Q	Fuel consumption transmitter (flowmeter)	This transmitter is mounted on the engine feed pipe; it feeds the fuel remaining indicator amplifier with fuel consumption information.
53Q	Fuel remaining indicator amplifier	This amplifies the signal from the transmitter and feeds it to the fuel remaining indicator (52Q) and (252Q) III D rear cockpit.

	SHOCK CONE CONTROL SYSTEM	
2C	Air data computer output multiplier	Feeds Mach number information to the slaving box (46C).
42C 242C	Shockcone position indicator 242C — III D Only	This indicates to the pilot the position of the shock-cones according to the Mach number.
43C	Control panel	Including : a. An automatic control engage pushbutton which is self-holding. b. A manual control switch with two positions marked IN-OUT with a the neutral centre position, this switch provides for power supply to the automatic control pushbutton.
46C	Slaving box	This box is fed with information from the output multiplier of the air data computer which it converts into electrical signals for automatic control of the shock-cones.
59C	Shockcone actuator	This actuator is used to move the shockcones and is fitted with limit switches.

Index No	Description	Characteristics and Functions
SHOCK CONE CONTROL SYSTEM		
60C	Potentiometer box	This determines the position of the shockcones according to the information received from the output multiplier of the air data computer and re-transmits position signals to the shockcone position indicator.

STARTING — IGNITION		
2K	Ignition contactor	This contactor, which is energized by pressing the starting button, supplies the ignition box.
3K	Ignition box	This box supplies the starting electro-valve and the igniter plugs.
4K	Starting box	This box is used to control the engine starting sequence.
6K	Starting relay	This relay is energized by pressing the engine starting button. It supplies the starting units of the engine and self-contained starter and also the by-pass electro-valve and is slaved to the speed detector.
7K	Engine starting button	Pressing this button puts the engine starting units into operation.
8K	In-flight relight switch	Aft position : Ignition contactor supplied through the starting relay (if the latter relay is energized). Forward position RELIGHT : Ignition contactor directly supplied.
9K	Ignition ventilation switch	IGN position : Starting units and ignition contactor simultaneously supplied. VENTIL position : Starting equipment supplied (only in case of a starting incident).
On engine	By-pass electro-valve	This valve is open during engine starting; it limits the pressure of the fuel supplied to the injectors during the starting sequence.
On engine	Starting electro-valve	This valve is open during engine starting and in-flight relight; it allows the fuel to flow to the starting injectors.
On engine	Starter speed detector box	This is closed during engine starting; it disconnects the self-contained starter when the engine r.p.m. is between 1800 and 2000.
On engine 43K 44k 45k 46k 47k	Self-contained starter system components : Fuel control unit Supply electro-pump Three-way electro-valve Starter Ignition coil	Refer to Engine Handbook — AAP 7111.007-2-3.

AFTERBURNER		
13K 213K	Afterburner cock control switch 213K — III D only	See Fuel Supply System
14K	Afterburner microswitch (in power control quadrant)	see Fuel Supply System

Index No	Description	Characteristics and Functions
	AFTERBURNER	
15k 215k	afterburner operating indicator light (A/B ON) 215k — III D Only	this is an amber push-to-test light; it is supplied by the ionization probe amplifier to indicate that the afterburner is operating.
16k 216k	afterburner injector indicator light (A/B INJ) 216k — III D Only	This is a red push-to-test light; it is grounded through the pressure switch and indicates that the afterburner injection system is operating.
17K	Afterburner emergency pushbutton (EMERG)	This pushbutton is used to operate the afterburner when it fails to ignite through the normal system.
18K	Afterburner cock	See Fuel Supply System.
19K	Afterburner relay	This relay is energized by the fuel control system (air data computer); it is used to cut off the power supply to the ionization probe amplifier.
20K	Afterburner electro-valve	See Fuel Supply System.
21K	Ionization probe amplifier	This is used to amplify the signals from the ionization probe and to re-transmit them to the engine fuel control system.
On engine	Ionization probe	This probe detects the ignition of the afterburner manifolds and transmits corresponding signals to the probe amplifier.
On engine	Afterburner ignition electro-valve	This valve is open to permit the afterburner manifolds to be supplied with fuel.
On engine	Thrust-corrector electro-valve	This valve is slaved to the engine fuel control system.
On engine	Afterburner pressure switch	This pressure switch is mounted in the afterburner injection pipe to indicate that the injection system is operating through the illumination of the associated indicator light.

	ENGINE OVERSPEED SYSTEM	
19K	Afterburner relay	This relay is energized by the fuel control system connected to the air data computer; it is used to open the thrust corrector electro-valve circuit as soon as the engine overspeed system is started.
51K	Time delay relay	This relay is energized through the overspeed switch; it is provided to establish the circuit to the booster electro-valve. Operating time is limited to one minute for manual operation.
52K	Overspeed switch (in power quadrant)	This switch enables the overspeed system to be manually operated on take-off with the throttle control lever in the maximum afterburner position and at a ground temperature equal to or above 27.0°C (80.6°F).
53K	Overspeed button	This button is used to pre-set the manual mode of operation of the engine overspeed system.
54K 254K	Overspeed indicator light 254K — III D Only	This is an amber light situated on the instrument panel; it is illuminated when the engine overspeed system is operating.

Index No	Description	Characteristics and Functions
EMERGENCY REGULATION — EMERGENCY OIL SYSTEM		
27K	Emergency nozzle control switch	This switch is provided with a cover guard; when turned on, it performs the following functions : Starting of the oil electro-pump. Opening of the LP 2 discharge electro-valve. Electrical power supply of the actuator control key.
28K (III O) or 40K (III D)	Actuator control key	This key enables the pilot to control the electric actuator of the emergency regulation system.
27	Electric actuator	Controlled by the actuator control key, this actuator operates the fuel metering valve of the main fuel control unit in the desired direction.
28.	Anti-overspeed spring	This spring prevents engine overspeed in case of oil failure.
30.	Emergency oil tank	This tank has a capacity of 0.77 U.K. gal (3.5 l); it is provided for lubrication of the No 1 bearing only in case of oil failure.
31.	Oil electro-pump	This pump is started by control switch (27K) to provide the oil pressure required for lubrication of the No 1 bearing.
32.	Check valve	This valve maintains the pressure of the oil to the No 1 bearing.
33.	Filter	This filter prevents foreign matter from entering the oil system.
34.	Nozzle	Mounted on the No 1 bearing, this nozzle provides for ball-bearing lubrication during operation of the emergency oil system.

ENGINE INSTRUMENTS		
1E 201E	R.P.M. indicator 201E — III D Only	This indicator is fed with current from the engine tachometer-generator and indicates the engine speed in r.p.m.
2E	Fuel LP pressure switch	This switch establishes a contact whenever the fuel pressure is less than 10.2 lb/insup2 (700 mb) (absolute pressure).
3E 203E	Jet pipe temperature indicator 203E — III D Only	This indicates the jet pipe temperature sensed by the thermocouples.
4E 204E	Adjustment box 204E — III D Only	This enables the line resistances to be balanced for adjustment.
7E	Test connections	These connections are used to test the thermocouple circuit.
8E	Thermocouples	These thermocouples are provided to sense the jet pipe temperature.
9E	Connector boxes	These are provided for connection of the thermocouple output leads.
IZ 201Z	FUEL warning light 201Z — III D Only OIL warning light (on failure warning panel)	These lights are illuminated when the corresponding circuit is closed by the associated pressure switch.

Index No	Description	Characteristics and Functions
	ENGINE INSTRUMENTS	
On engine	Oil pressure switch	This pressure switch closes the circuit in case of oil pressure drop (For calibration value refer to engine handbook — AAP 7111.007-2-3.)
On engine	Engine tachometer-generator	This is driven by the engine; it generates on a.c. voltage whose magnitude and frequency are proportional to the engine r.p.m.

	FIRE DETECTION SYSTEM	
2J	Control box	This box contains the fire detection relays slaved to the two banks of detectors.
3J	Fire detectors (qty : 9)	These detectors sense the heating in the cold zone. Calibrated at 200°C.
5J	Fire detectors (qty : 6)	These detectors sense the heating in the hot zone. Calibrated at 300°C.
6J	Fire detectors (qty : 3)	These detectors sense the heating in the hot zone. Calibrated at 300°C.
4J 204J 7J 207J	Engine fire warning light 204J — III D Only Afterburner fire warning light 207J — III D Only	Red push-to-test lights. These warning lights are supplied by the fire detection relays in control box 2J.
8J	Fire detector test pushbutton	This pushbutton enables testing of the fire detection system.
37K	III O Only Control box	This box contains the fire detection relays slaved to the rocket motor fire detection system.
38K	III O Only Rocket motor fire warning light	This is a red push-to-test light; it is supplied by the fire detection relay in the control box.

GENERAL

Description (refer to Figs 9-1 and 9-2)

901. The MIRAGE aircraft is powered by a SNECMA ATAR 9 C turbo-jet engine (1). The turbo-jet engine uses :

a. *Fuel* F-34 (DEF (AUST) 5240).

b. *Oxidizer* Ambient air.

902. The fuel is contained in the following :

a. Permanent tanks (2).

(1) Front and rear fuselage tanks

(2) Wing tanks

(3) Leading edge tanks, Post Mod 406.

b. Removable tanks (3)

(1) Gun bay tank

(2) Rocket motor bay tank (III O only)

(3) Rear bay tank (III D only).

c. External tanks (4)

(1) Under-wing tanks

(2) Under-fuselage tanks.

903. The oxidizer is directed to the engine by two symmetrical air intake ducts (5). The air flow is adjusted by means of mobile shockcones (6). In addition to propelling the aircraft, the engine provides the following power sources :

a. *Electrical* : DC power by driving a generator.
AC power by driving an alternator

b. *Hydraulic* : By driving two pumps.

c. *Pneumatic* : By tapping air from the compressor.

904. For III O Only,on certain missions, the aircraft can be equipped with a SEPR 844 booster rocket motor (9). The rocket motor uses engine fuel and Nitric acid as the oxidizer. The rocket motor is mechanically driven by the turbo-jet engine through an accessory gear box (8).

905. The accessory gear box also drives the following equipment :

a. A generator.

b. An alternator.

c. A hydraulic pump.

906. Fuel supply to the turbo-jet engine and (for III O) the rocket motor, is ensured by an installation

using the following :

a. Electrical power (pumps).

b. Pneumatic power (transfer systems).

907. The operation of the complete installation is monitored and controlled from the cockpit by means of various controls and indicators.

TURBO-JET ENGINE

General (refer to Figs 9-3 to 9-6)

908. The ATAR 9 C turbo-jet engine is in the category of thermal propulsion engines. The chemical energy in the fuel it uses is converted into thermal energy through combustion. The thermal energy thus released is used to heat the mass of air from the inlet, accelerating it to higher velocity at the outlet, and thereby developing the thrust which pushes the engine forward.

909. The MIRAGE turbo-jet engine is provided with an afterburner system. A second combustion zone is created at the exit of the turbine by injecting fuel into the gas stream (which still contains sufficient air to support further combustion). The resulting increase in temperature further accelerates the gas stream and increases the thrust. The afterburning system is of the controllable type (ie, the maximum afterburner power can be reduced in flight).

910. The engine is provided with a self-contained lubrication system. However, an emergency system (independent of the normal system) enables the pilot to control the lubrication of the No 1 bearing in case of oil failure. The engine fuel control system is automatic in normal operation; only the approach control system and the emergency fuel control system are manually selected.

911. The engine is provided with r.p.m., temperature and oil pressure monitoring systems.

Brief Description (refer to Fig 9-3)

912. The turbo-jet engine consists of the following main components :

a. A self-contained starter.

b. A nine-stage axial-flow compressor.

c. An annular combustion chamber with burners and ignition systems.

d. A two-stage turbine.

e. An afterburner nozzle.

f. A flap-type variable area exhaust nozzle.

Installation (refer to Figs 9-7 and 9-8)

913. The turbo-jet engine installation consists of :

a. The attachments to the aircraft.

b. The various connections to the aircraft systems.

c. Drains.

d. Power take-off for the accessory gear box.

e. Power take-off for the No. 1 hydraulic system pump.

914. **Attachments (refer to Fig 9-7).** The engine is attached to the fuselage at four points through the following :

a. Two trunnions (1) which are diametrically opposed at frame 27. These trunnions which are located in a horizontal plane constitute the main engine suspension. They carry most of the weight and all the thrust.Each trunnion consists of :

(1) A pin (11) with two tapered bearing surfaces. This pin is centered in a steel sleeve (13) by a bush (15). The pin (11) fits into a swivel fitting (12) mounted on the engine close to its centre of gravity. The pin is a loose fit in the swivel fittings to cater for expansion through end play.

(2) The sleeve (13) is secured to an attachment fitting (14) integral with the aircraft structure by means of screws (16).

(3) The bush (15) is clamped by a nut (17) safety-locked by a spring (18).

b. A centring rod (2) at the upper part of frame 27 connecting the engine to the aircraft structure and transmitting the side loads. This rod consists of two threaded shank fork-fittings (5 and 6) and a bushing (21) permitting the length of the rod to be adjusted. The centring rod (2) is permanently mounted on the aircraft structure. The fork-fitting (6) attaches to a fitting located on the engine. On removal of the engine the centring rod is swung back and is held in position by a fitting (4) located behind an access door.

c. A rod (3) at frame 38, which forms the engine rear suspension and is used to transmit the vertical reaction loads. The rod is fitted with a fork-fitting (7) at its upper part for permanent connection to the aircraft structure; it is fitted with an adjustable end fitting (8) at its lower part. The end fitting (8) enables the length of the rod to be adjusted; it is locked by a nut (9) provided with a lock plate. The end fitting (8) is also used for connection of the rod to the attachment fitting (10) integral with the engine. This assembly caters for expansion through the tilting motion of the rod. Upon removal of the engine, a screw, integral with the rear section of the fin, retains the rod in the raised position.

915. Sealing between the air intake duct and the engine inlet nozzle is obtained by means of an extruded seal (19). This seal is clamped between a ring (22) riveted to the rear face of frame 23 and a folded plate (20). This assembly is secured by screws (23).

916. **Connections (refer to Fig 9-8).** The following connections are made between the engine and the aircraft systems :

a. *Mechanical connections* :

(1) R — Power control. The power control is connected to the control box on the LH side of the engine

by means of a bolt and a nut fitted with a split pin.

(2) S — Accessory gear box drive on LH side of the engine. The drive consists of a toothed ball joint with internal splines, ref to para 923.

(3) T — No 1 hydraulic system pump drive (10) on RH side of the engine. The pump is driven by a splined shaft fastened by means of quick-release clamps (12). Appropriate ground ancillary equipment is provided for use between frames 25 and 26 for temporary attachment of the pump during removal operations to prevent the hydraulic system from being drained.

b. *Fuel Connections :*

(1) U — The dry engine feed system is connected by means of a toroidal coupling (on RH side).

(2) V — The afterburner feed system is connected by means of a toroidal coupling (bottom of the engine).

c. *Pneumatic Connections, air tapping pipes to engine :*

(1) N — Canopy seal inflation, cockpit air conditioning, anti-g valve, Giffard outlet, TACAN/radar : toroidal coupling.

(2) P — Equipment air conditioning, toroidal coupling. (III O Only)

(3) O — Turbo-pump air, toroidal coupling.

(4) Q — Fuel tank and hydraulic reservoir pressurization, toroidal coupling.

d. *Electrical Connections :*

(1) One connector (108 Y) on RH side of the engine

(2) One connector on ignition box (3K) at the lower part of the engine.

(3) One connector (109 Y) on the afterburner harness support on LH side of the engine.

(4) One connector (111 Y) on the starting harness on LH side of the engine

(5) One connector (112 Y) on afterburner manifold support on the LH side of the engine.

917. **Drain Systems (refer to Fig 9-8).** The engine has the following drains :

a. *Jet Pump, Pressurizing and Dump Valve and Combustion Chamber Drain System.* The air and oil from the jet pump and the fuel from the pressurizing and dump valve and the combustion chamber are drained through two flexible pipes (4) and two rigid pipes (5). Each flexible pipe is connected to the engine by a swivel coupling (6). The flexible pipes connect with the rigid pipes through Quinson quick-disconnect couplings. The two rigid pipe (5) outlets are located at the front of frame 35, one under the port fillet (drainage fuel discharge), the other under the starboard fillet (air and oil discharge).

b. *Engine Compartment Drain System.* This consists of a rigid pipe run from a pocket recess located at the lower part of frame 27 to the outside (under port fillet at frame 35).

c. *Accessory Gear Box Drain System.* The accessory gear box drain system consists of flexible and rigid pipes (13). The end of the flexible pipe is connected to a manifold (15) on the accessory gear box door.

d. *Turbo-pump Drain System.* This consists of flexible and rigid pipes (14). The rigid discharge pipe continues to the rear of the fuselage and ends at the front of frame 35 under the starboard fillet.

ACCESSORY GEAR BOX

Brief Description (refer to Fig 9-9)

918. The engine-driven accessory gear box (20) provides four power take-offs to drive the power generating accessories listed below. The various drives are protected by torque limiters. The accessories are attached by means of quick release clamps.

a. A generator (IP), on LH side of the aircraft.

b. An alternator (IV) and a hydraulic pump (21) (No 2 hydrualic system) on RH side of the aircraft.

c. A drive (26) to which the rocket motor shaft is connected at the rear of the engine. The rear drive (26) is not used on III D aircraft and is sealed off by a cap.

d. A drive, a breather (27) and a discharge coupling at the top of the engine.

e. The following parts are located at the bottom of the case :

(1) A filler plug (25).

(2) An overflow plug (24).

(3) A drain plug (23).

Attachment to Aircraft (refer to Fig 9-9)

919. The accessory gear box is attached at four points under a drop-forged structural support (2) which forms the engine compartment floor between the lower web assemblies from frame 23 to frame 26. Each attachment point includes a threaded pin (1) fitted through the gear case (3). The pin is screwed onto the support (2) with a floating nut (4) on the engine compartment side. The floating nut (4) is secured to the structure by means of a screwed clamp (6).

Connections (refer to Figs 9-10 and 9-11)

920. The engine and the accessory gear box are mechanically interconnected by means of a telescopic

drive shaft (see para 923). This shaft has a toothed and splined ball joint at each end. On III O Only, the accessory gear box also drives the rocket motor through a sliding shaft. This shaft connects to a drive (26) using the same principle as the telescopic shaft.

Drain System (refer to Fig 9-9)

921. The oil is drained through flexible pipes from the breather (27) to the manifold located on the accessory gear box door. The manifold also receives the fluid drained from the No 2 hydraulic system pump mounted on the accessory gear box.

Accessory Gear Box Drive (refer to Fig 9-10)

922. The accessory gear box is driven by a telescopic shaft (2). The rotational motion of the drive (1) from the engine is transmitted to a drive (11) on the accessory gear box. The drives (1 and 11) are fitted with toothed ball-joints (3) with internal splines. These internal splines are used for mechanical connection of the telescopic shaft. The drives are protected by a cover (5). Each drive is fitted with a grease nipple (16) and a grease overflow plug (17).

923. **Telescopic Shaft with Integral Locking Device (refer to Fig 9-10).** The telescopic shaft consists of two shafts which slide one inside the other. The driving shaft (6), on the engine side, drives the driven shaft (7), on the accessory gear box side, through its internal splines.

924. The telescopic shaft is held extended by a spring (8). It is locked in this position by a pin (9), integral with a leaf spring (14) and held in place by a moving bush (15). Two fingers (10) are provided to lock the telescopic shaft in the ball-joint (3) on the accessory gear box side. These fingers are held in position by a piston (4) with a taper end-fitting (12) fitted with a spring (13).

925. **Removal of the Telescopic Shaft.** After rotating the locking bush (15) 180 degrees, the inscription FREE appears above the spring (14). In this position, the spring faces the notch in the bush and permits withdrawal of the pin (9) from its housing. The shaft is then slid downwards, the spring is compressed and the shaft is freed from the drive (1) on the engine. The pin (9) fits into another housing and holds the shaft in the retracted position. During this movement, the piston (4) has been brought against the bottom section of the shaft, which resulted in moving the taper end-fitting and, consequently, in releasing the locking fingers (10). The telescopic shaft assembly can now be withdrawn from the drive (11) on the accessory gear box.

Rocket Motor Drive — III O Only

926. **Sliding Shaft (refer to Fig 9-11).** A sliding shaft is used to transmit the rotational motion from the accessory gear box to the rocket motor. The shaft is connected to a mechanical assembly which enables the shaft to be retracted for flights made without the rocket motor or after the rocket motor has been jettisoned in flight.

927. The sliding shaft consists of an internal driving shaft (1) and an external driven shaft (2). One end of the driving shaft (1) is fitted with a splined crown (3) connected to the accessory gear box. Guide splines (4) are located at the other end of this shaft. The driving shaft drives the driven shaft (2) through internal splines (5); a seal (6) is provided for sealing between the two shafts. The driven shaft (2) transmits the rotational motion to the rocket motor through a splined endpiece (7); it is fitted with two bearings.

928. The inner bearing retainers (8) are made integral with the driven shaft (2) by a nut (9), a safetied check-nut (10), a spacer (11) and a ring (12). The outer bearing retainers (13) take the ball-bearing cage (14) attached to the retraction device.

929. **Retraction Device.** This is a mechanical assembly which transmits an axial retraction motion to the driven shaft (2). It consists of a casing (15) bolted onto frame 27 and of two drop-forged bell-cranks (16 and 17), hinged on the casing (15) and fitted with fork-fittings. bell-crank (16) is connected to the piston (18) of the explosive actuator (19) and to the six-sided manual drive device (20). Bell-crank (17), made integral with bell-crank (18) by a link-rod (21), is connected to two springs (22) and to the ball-bearing cage (14).

930. The casing (15) is provided with a ball-and-socket joint (23) inside which the ball-bearing cage (14) is accommodated when the shaft is in the retracted position. The front flange (24) is fitted with a seal (25) fastened by a ring (26) and a clamp (27). When the rocket motor is not used, the ring (26) is replaced by a cover (28).

931. **Operation.** The shaft is held in the driving position against the tension of the springs (22) which are attached to the mechanical assembly. The mechanical assembly can be adjusted using the stop screw (29) and lock nut (30). When either the explosive actuator (19) or the manual drive device (20) is operated, the spring tension draws the ball-bearing cage (14), which fits into the ball-and-socket joint (23), forward into the retracted position.

PNEUMATIC SYSTEM SUPPLY

Description (refer to Figs 9-12 to 9-15)

932. The four P2 pressure pick-offs on the engine are used for :

a. Pressure pick-off O — top RH side; supply of the turbo-pump (system A).

b. Pressure pick-off N — top LH side; supply of the following units through a pre-cooler (3) :

   (1) Giffard valve (system B).

   (2) Cockpit air conditioning (system C).

   (3) III O Only : Radar (system D).

   (4) III D Only : TACAN and nose cone (environment) (system D).

   (5) Anti-g valve (system E).

   (6) Canopy seal inflating system through a filter (4) (system F).

c. Pressure pick-off P — III O Only, bottom LH side; supply of the following systems through a pre-cooler (1) :

   (1) Gyro-amplifier air conditioning (G).

   (2) TACAN and gyro output multiplier air conditioning (H).

   (3) BZ tail warning radar compartment air conditioning (I).

(4) MATRA missile air conditioning (J).

(5) UHF compartment air conditioning (K).

d. Pressure pick-off Q — bottom RH side; supply of the following systems through a pre-cooler (1) and a filter (2) :

(1) Fuel tank pressurization (M).

(2) Hydraulic reservoir pressurization (L).

FUEL SUPPLY SYSTEM

General (refer to Figs 9-16 to 9-18)

933. The fuel system consists of a compounded fuel, pneumatic and electrical system performing the following functions :

a. Refuelling of the tanks and vapour-relief.

b. Fuel tank pressurization.

c. Fuel transfer.

d. Engine supply.

e. Monitoring of the above functions and detection of possible failures.

934. The complete installation divides into two separate symmetrical installations not including the inverted flight accumulator (4), the gun bay fuel tank (24) and the fuselage pylon tank (109).

935. The fuel system consists of :

a. *Permanent fuel tanks :*

(1) Two fuselage fuel tanks (1 and 2), each 113 U.K. gal (515 l) capacity.

(2) Two wing fuel tanks (3), each 120 U.K. gal (545 l) capacity.

(3) Two leading-edge fuel tanks (150), each 30.8 U.K. gal (140 l) capacity (Post-Mod 406).

(4) One inverted flight accumulator (4), 13.2 U.K. gal (60 l) capacity.

(5) III D,O , one rear bay fuel tank (5), 120 U.K. gal (545 l)capacity.

b. *Removable fuel tanks :*

(1) One gun bay tank (24), 72.7 gal (330 l) capacity.

(2) III O Only, one rocket motor bay tank (5), 119.9 U.K. gal (545 l) capacity.

c. *External fuel tanks :*

(1) Under the wings (6) two supersonic tanks, each 110 U.K. gal (500 l) capacity or, III O Only, two tanks, each 286 U.K. gal (1300 l) capacity, or two tanks, each 374 U.K. gal (1700 l) capacity. III O Only, two bomb carrier tanks, each 55 U.K. gal (250 l) capacity.

(2) Under the fuselage, III O Only; One Tank, 286 U.K. gal (1300 l) capacity. III D Only, One TANK 242 U.K. gal (1100 l) Capacity.

936 The Air System performs the following functions :

a. Vapour relief during refuelling of the tanks.

b. Pressurization of the tanks in flight.

937. The Electrical System performs the following functions :

a. Driving of the LP pumps.

b. Operation of certain valves.

c. Monitoring of fuel consumption.

d. Monitoring of fuel transfer.

e. Monitoring of fuel levels in the fuselage fuel tanks.

f. Monitoring of the fuel feed pressure.

Description (Refer to Figs 9-16 to 9-18)

938. **Fuel System.** Fuel is contained in the following tanks :

a. *Fuselage Fuel Tanks.* There are four tanks (1 and 2) plus two feeder tanks (7); these tanks occupy the free space between the centre air intake duct and the fuselage skin; they extend from frames 17 to 23. Each front tank (1), between frames 17 and 20, contains a float-valve (8) controlling the fuel transfer from the pylon tanks. Each rear tank (2), between frames 20 and 23, contains a float-valve (10) controlling the fuel transfer from the gun bay tank and the fuselage pylon tank and a float-valve (9) controlling the fuel transfer from the wing tanks and the rocket motor bay tank (III O) or rear bay tank (III D). These valves control the upper, intermediate and lower levels by stopping or restricting the corresponding flows. Each feeder tank (7), between frames 20 and 22, contains a LP pump (5Q and

Table 9-2 Fuel Capacities

A/C Type	Condition	Tank Type and Location	Capacity in U.K. gal (litres) Pre-Mod 406	Capacity in U.K. gal (litres) Post-Mod 406
III O	Permanent tanks :	Fuselage 2 × 113.5 Wings 2 × 119.9 Inverted flight accumulator	227 (1030) 239.8 (1090) 13.2 (60)	227 (1030) 301.8 (1370) 13.2 (60)
	Removable tanks :	Gun bay Rocket motor bay	72.7 (330) 119.9 (545)	72.7 (330) 119.9 (545)
	Total capacity of internal fuel tanks (the fuel in the gun bay tank can be used for either the rocket motor or the engine).			
	a. Without bay tanks b. With gun bay tank c. With rocket motor bay tank. d. With both gun and rocket motor bay tanks		480 (2180) 553 (2510) 600 (2725) 673 (3055)	542 (2460) 615 (2790) 662 (3005) 735 (3335)
III D	Permanent tanks :	Fuselage 2 × 113.5 Wing 2 × 119.9 Leading edge 2 × 31 Inverted flight accumulator Rear bay		227 (1030) 239.8 (1090) 62 (280) 13.2 (60) 119.9 (545)
	Removable tank :	Gun bay		72.7 (330)
	Total capacity of internal tanks a. Without gun bay tank b. With gun bay tank			662 (3005) 735 (3335)

Table 9-3 Fuselage Fuel Tank Capacities at Valve Level

A/C Type	Valve	Capacity in U.K. gal (litres)
III O	Upper (8) — External wing transfer	119 (905)
	Intermediate (10) — Gun bay tank and fuselage pylon tank transfer	179 (815)
	Lower (9) — Wing tank and rocket motor bay tank transfer	158 (720)
	Contactor level	132 (600)
III D	Upper Intermediate — As for III O	
	Lower (9) — Wing tank and rear bay tank transfer	158 (720)
	Contactor level	121 (550)

NOTE These capacities do not include the inverted flight accumulator.
All values are less than the fuel gauge readings by 13.2 U.K. gal (60 l).

6Q) and a tank gauge unit (26Q and 27Q).

b. *Wing Tanks (3).* There are two wing tanks; these structural fuel tanks occupy the complete main wing box volume. The fuel flows from each wing tank through a strainer (12) located at the rear of each tank (bottom point in nose-up attitude).

c. *Leading Edge Tanks (150) — Post-Mod 406.* There are two leading edge tanks; these tanks constitute the inboard leading edge section from the wing root to the leading edge slot. Each tank is interconnected to the corresponding wing tank by a pipe (151) and has a pressurization pipe (152).

d. *Pylon Tanks (6).* The various pylon tanks are described in paras 9134. to 9143. Each pylon tank contains a sump pipe tipped by two bottom valves (13).

e. *External Tanks (6).* The various external tanks are described in paras 9134 to 9143. Each tank contains a sump pipe tipped by two tank bottom valves (13).

f. *Gun Bay Fuel Tank (24).* This is a 72.7 U.K. gal (330 l) tank mounted in place of the gun-pack; it is used to supply either the rocket motor (III Only) or the engine. The sump pipe is fitted with a tank bottom valve (88).

g. *Rocket Motor Bay Fuel Tank (5) (III O) or Rear Bay Tank (III D).* This is a 120 U.K. gal (545 l) tank consisting of two independent compartments.

III O Only, it is mounted in place of the rocket motor and is used to supply the engine via the fuselage tanks.

939. **Fuel Tank Interconnections.** The following fuel tank interconnections are installed :

a. *Interconnection Between Tanks on Each Side of the Aircraft.* The front and rear fuselage tanks are connected by tubular interconnectors. The feeder tank is made integral with the rear tank through a large bolted interconnecting flange. These tanks for operational purposes constitute a single capacity. The leading edge tank (150) (Post-Mod 406) is connected to the wing tank (3) by a pipe (151) and to the pressurizing system by pipe (152). The wing tank in turn is connected to the rocket motor tank (III O) or rear bay tank (5) (III D) by a rigid pipe (17). The pipe (17) is used for refuelling and transfer from the wing tank and the leading edge tank as well as for pressurization of the rocket motor tank (III O) or rear bay fuel tank (III D). The rocket motor tank or rear bay tank is connected to the rear fuselage tank (2) (lower valve (9)) by two rigid fuelling and transfer pipes (18). The wing pylon tank (6) is connected to the fuselage tank (upper valve (8)) by a rigid transfer pipe (19) equipped with a filter (79) accommodated in the pylon.

b. *Between LH and RH Sides of the Aircraft.* The two feeder tanks are interconnected by a pipe (23) controlled by an electrically operated full-flow valve (21Q). The two pipes (18) are interconnected by two filler pipe sections (21), from the distribution manifold, which are controlled by mechanically operated full-flow valves (22).

c. *Gun Bay Tank to Fuselage Tank Connections.* The gun bay tank (24) is connected to the fuselage tanks (2) by transfer pipes (25 and 26) connected to the intermediate float valves (10). The filler pipe (27) is provided with a cock (28) which is connected to the wing tank filler pipe (17). The valve (28) is closed in flight and is manually opened for fuelling.

For III O Only; the fuelling pipe (27) is fitted with the rocket motor supply self-sealing coupling.

940. **Fuel Tank to Engine Connections.** The two LP pumps (5Q and 6Q) on the feeder tanks and the inverted flight accumulator (4) are connected by rigid pipes and a four-way coupling (29). The pipe from each pump is provided with a non-return valve (30). The centre conduit from the four-way coupling extends into the distribution manifold, the electrically controlled fuel shut-off valve (main LP cock) (17Q) and the fuel consumption transmitter (52Q). The centre conduit divides then into two branches leading to the dry engine fuel system through a rigid pipe (31) and a flexible pipe (32) and to the afterburner turbo-pump (33) through a pipe controlled by an electrically operated valve (18K).

941. A heat-exchanger (95) for cooling of the hydraulic fluid is mounted in the dry engine fuel system. The fuel is used as a cooling medium for the hydraulic fluid.

942. **Air System (refer to Figs 9-19 to 9-24).** This system performs the following functions :

a. Vapour relief during refuelling.

b. Pressurization of the fuel tanks in flight to provide for fuel transfer and to back-up the LP pumps.

943. The air bled from the engine compressor (pressure pick-off Q) is supplied via a filter and a non-return valve (35) to :

a. A bank of five pressure reducing valves in parallel corresponding to the various tanks.

b. The gun bay tank reducing valve (85 and 90) (through pipe 34)

c. The fuselage pylon tank reducing valve (101 and 103) (through coupling 106).

Table 9-4 Fuel Pressurization System Components

Pressurization System	Pipe	Reducing Valve	Safety Valve	Restrictor	Non-return Valve	Filter	Valve Box
External wing tanks (6)	34	52	58		53 and 55	54 and 56	65 (in pylon)
Wing and rear bay tanks (3 and 5) (port side)		46	51	97	47	48	63
Wing and rear bay tanks (3, 5 and 150) (starboard side)		43	50	96	44	45	64
Fuselage tanks (1 and 2)		36	39		37	38	59 and 60
Inverted flight accu. (4)		40	42		41		
Gun bay tank (24)		85	90		86	87	89 and 91
Fuselage pylon tank (109)		101	103		102	108	104
Index numbers relate to Items shown in Figs 9-19 to 9-21							

Table 9-5 Fuel System Operational Equipment

Controlled Unit	Index No	Control Unit	Index No	
			Cockpit	Rear Cockpit III D Only
LH LP pump	5Q	Switch	9Q	
RH LP pump	6Q	Switch	10Q	
Fuel shut-off cock (LP cock)	17Q	Switch	16Q	216Q
Cross-feed cock	21Q	Switch	20Q	
Gun bay tank fuel transfer valve	22Q	Switch	19Q	
Afterburner cock	18K	Switch	13K	213K
Afterburner electro-valve	20K	Afterburner microswitch (throttle lever)	14K	

Table 9-6 Fuel System Monitoring Equipment

Sensing Unit	Index No	Indicating Unit	Index No	
			Cockpit	Rear Cockpit III D Only
LH fuel tank unit	27Q	Fuel gauge indicator	29Q	229Q
RH fuel tank unit	26Q	(Test button)	33Q	
LH fuel tank unit	27Q	130 gal (600 l) warning light	28Q	228Q
RH fuel tank unit	26Q			
Fuel remaining indicator transmitter	52Q	Fuel remaining indicator	51Q	251Q
LH wing tank pressure switch	35Q			
RH wing tank pressure switch	36Q			
LH pylon tank pressure switch	31Q			
Fuselage pylon tank pressure switch	153	Fuel transfer indicator	40Q	240Q
RH pylon tank pressure switch	32Q			
Gun bay tank pressure switch	37Q			
Fuel LP pressure switch	2E	Failure warning panel	1Z	201Z

944. All pressure reducing valves incorporate a safety valve and a static-pressure tube.

945. The external tank system is provided with self-sealing couplings (57 and 106) in case the external tanks are not installed.

946. A pipe fitted with a pressure relief valve (61) connects the pressurization system of the inverted flight accumulator to the fuselage tanks. A pipe fitted with a pressure relief valve (59) and an underpressure valve (60) provided with a built-in filter connects the fuselage tanks to the static pressure system. The static pressure inlet is located under the LH wing fillet at frame 35. The static pressure inlet for the gun bay tank system is located under the RH wing fillet at frame 35.

947. The vapours in the pylon tanks escape through the tank filler ports. The vapours in the other tanks escape through the fuselage tanks.

948. A number of connections are provided for testing on the ground with the engine stopped (refer to Fig 9-21) :

a. Pressurization connections (70 and 93)

b. Pressure test connections (69, 71, 72, 73 and 92) for each system.

c. The pylon tank pressure is checked by connecting a pressure gauge in place of the filler plugs (74).

949. **Mechanical Valve Controls (refer to Fig 9-22).** These controls permit refuelling of the wing tanks (3 and 150) and gun bay tank (24). Each control consists of a handle (75) provided to operate the following valves through a flexible control :

a. LH handle (through a single control); the refuelling valve (28) and the vapour relief valve (68) of the gun bay tank.

b. RH handle (through two controls); the refuelling valves (22) and the vapour relief valves (49) of the wing tanks.

950. **Electrical System (refer to Fig 9-17).** In addition to driving the LP pumps and operating the valves, the electrical system is used to give the following information to the pilots :

a. Fuel consumption.

b. Fuel transfer.

c. Fuel levels in the fuselage tanks.

d. Fuel feed pressure.

951. **Draining and Bleeding Systems (refer to Fig 9-25).** The following draining and bleeding systems are installed :

a. *Draining of the Fuselage Tanks.* The fuselage tank draining system consists of a valve (1) screwed onto the distribution manifold (detail D). Sealing is ensured by an O-ring (2). The valve contains a clapper (3) fitted with a seal (4). The clapper is held closed by a spring (5) seated in a perforated cage (6). The cage is attached to the valve by means of screws (7). Draining is accomplished by means of a drain adapter (8). As the drain adapter is being installed, its point lifts the clapper (3) which compresses the spring (5). The fluid is then allowed to flow through the ports (9) of the cage (6). On removal of the drain adapter, the spring pushes the clapper (3) back and forces the seal (4) onto its seat (10).

b. *Draining of the Gun Bay, Rocket Motor Bay Tanks (III O), Rear Bay Tank (III D) and the Wing Tanks.* This operation is accomplished by means of a device including a body (12) with a valve (11) screwed into it. Sealing is achieved by compressing an O-ring (14). A spring (13) prevents the valve (11) from loosening. A brass ring (15) mounted in the perforated cage (16) holds a cap (17) which supports the spring (13). The assembly is locked in position by a plate (18) attached by two screws. After the plate (18) has been removed the body (12) can be removed using a peg spanner fitted into the two cylindrical holes (19).

c. *Bleeding.* All bleed valves and screws are located at the bottom points of the pipe lines and enable the water which may have collected in each section of the system to be drained. The following bleeds are provided :

(1) On the wing and bay tanks, screw in the valve (11). This lifts the valve off its seat, compresses the spring and releases the O-ring (14). The fluid is then allowed to flow through the perforations of the cage (16) and the two holes drilled in the valve (11).

(2) One on the accumulator (detail A)

(3) One on each LP pump (detail B)

(4) One on the distribution manifold (detail C).

(5) There is a bleed screw on each manifold LH and RH (detail E).

952. **Description of Bleed Valves (refer to Fig 9-25).** A needle valve (21), maintained in position by a spring (22), is screwed into a body (23). This hollow needle valve is connected to the hollow screw (24) used to attach the banjo union. Unscrewing the needle valve (21) allows the fluid to flow through the bores.

953. **Description of Bleed Screw (refer to Fig 9-25).** The bleed system consists of a threaded hollow screw (31). The bleed screw, screwed into the base-plate (32) integral with the manifold (33), is locked by means of lockwire (34). The assembly is sealed by a seal (35). Unscrewing the bleed screw (31) allows the fluid to flow through the centre bore of the screw. The external tanks are drained by pumping the fuel directly out through the fuelling ports.

954. **Draining of the Leading-edge Tanks — Post-Mod 406 (refer to Fig 9-25).** This operation is accomplished by means of a device including a body (41) with a valve (40) screwed into it. Sealing is achieved by compressing an O-ring seal (42). A spring (43) prevents the valve (40) loosening and is held in position by an end-fitting (44) screwed into the body (41). The assembly is attached to the structure by means of three screws (45) sealed by a nylon cap (46) and gasket (47).

Operation

955. **Fuelling.** The tanks are refuelled through two filler ports located at the top of the rear fuselage tanks. From the rear fuselage tanks, the fuel is directed as follows :

a. By gravity, to front fuselage tanks.

b. By the LP pumps, to the wing tanks, rocket motor bay tank (III O) or rear bay tank (III D), gun bay tank and inverted flight accumulator.

956. The external tanks are fuelled through a port provided on the tank.

957. The engine is supplied with fuel as follows :

a. By the LP pumps (in the feeder tanks) in normal flight.

b. By the inverted flight accumulator during inverted flight. This accumulator automatically fills up in flight.

958. III O Only,the rocket motor is supplied with fuel from the gun bay tank.

959. Fuel transfer from the external tanks, gun bay tank, rocket motor bay tank (III O) or rear bay tank (III D) and wing tanks to the fuselage tanks is fully automatic. The energy required for fuel transfer is provided by the pressurization air system. The flow is regulated by the level control valve mounted in the fuselage tanks.

960. **Crossfeed.** If the fuel is transferred more rapidly from one side than the other side, the fuel tanks can be interconnected by opening the crossfeed cock (21Q). This allows the fuel levels to balance out and ensure the tanks empty evenly.

961. **Refuelling — Vapour Relief (refer to Figs 9-23 and 9-24).** For details of refuelling vapour relief, refer to Table 9-7.

Table 9-7 Refuelling and Vapour Relief Procedure

Action	Effect	Refuelling and Vapour Relief
PRELIMINARY Circuit breakers 7Q, 8Q and 15Q closed		The tanks are refuelled through ports (11) located on the rear fuselage tanks. Venting takes place through the free space around the refuelling nozzle.
Switch 16Q — OFF	Main LP cock 17Q closed	
Switch 20Q — ON	Crossfeed valve 21Q open	
Opening of full-flow valves (22 and 49) and (28 and 68) through manual controls (75) on LH and RH sides.		
REFUELLING — VAPOUR RELIEF Switches 9Q and 10Q — ON (only one pump is necessary).	The LP pumps (5Q and 6Q) are supplied by the relays (3Q and 4Q) through the noise suppressors (11Q and 12Q).	The fuel is forced into the inverted flight accumulator (4) by the pumps through the non-return valves (30). The air contained in the accumulator is discharged into the rear fuselage tank through the relief valve (61). The fuel is also forced into the rocket motor bay tank (5) (III O) or rear bay tank (III D) by the pumps through the fuelling valves (22). The air contained in the rocket motor bay tank or rear bay tank flows into the wing tanks (3) through pipe (17) and to the leading edge tanks, (Post-Mod 406) through the pipe (151); it is then directed to the fuselage tanks through valves (49). When the rocket motor bay tank (III O) or rear bay tank (III D) is full, the fuel is conveyed through pipe (17) to the wing tanks (3) and gun bay tank (24) (through valve 28). Post-Mod 406, the leading-edge tanks (150) are refuelled after the wing tanks through pipe (151). The air is discharged through pipe (152) to the fuselage tanks. The air contained in the gun bay tank is discharged into the fuselage tanks through valve (68).
AFTER REFUELLING		LP pumps 5Q and 6Q stop
	Switches 9Q and 10Q — OFF	
	Switch 20Q — OFF	Crossfeed valve (21Q) closes.
	Manual closing of full-flow valves.	

NOTE

A new accumulator requires filling and draining several times to remove all the air before it can be completely filled.

A safety device prevents the valve control access doors being closed as long as the valves are in the OPEN position.

962. The external tanks are refuelled through a port (77) located on each tank. Venting takes place through the free space around the refuelling nozzle.

963. **Dry Engine Normal Fuel Supply System (refer to Figs 9-26 and 9-27).** With the pumps (5Q and 6Q) operating and the main LP cock (17Q) open, the fuselage fuel tanks are pressurized through the pressure reducing valve (36), the non-return valve (37) and the filter (38). The LP pumps deliver the fuel through non-return valves (30) to the engine supply system through the main LP cock (17Q). Excessive pressures are discharged by the pressure relief valve box (59). An underpressure valve (60) is also mounted in the tank pressurization system. The valves (30) prevent :

a. The fuel flowing from one tank to the other should the delivery pressures of the two pumps not be equal.

b. The inverted flight accumulator fuel being forced into the fuselage fuel tanks.

c. One of the tanks being drained when an operation is to be performed upstream of the non-return valve of the other tank.

964. **Inverted Flight Supply System (refer to Figs 9-28 and 9-29).** During inverted flight the pumps (5Q and 6Q) are no longer immersed in fuel. The air under pressure flows through the pressure reducing valve (40) and the non-return valve (41) and exerts a pressure on the diaphragm (78) of the inverted flight accumulator (4) forcing the fuel out into the engine fuel feed pipes. The pressure is also used to close the two non-return valves (30) located in the pump outlet systems. Excess pressures are discharged by the pressure relief valve (61).

965. **Afterburner System (refer to Figs 9-30 and 9-31).** The electrical power from the afterburner fuse breaker (12K) passes through the afterburner cock control switch (13K) locked in the ON position (this switch can be turned to OFF after pulling on the knurled button). The afterburner cock (18K) is open. Electrical power is maintained at the afterburner microswitch (14K) on the throttle control lever.

966. Tilting the handgrip upwards closes the afterburner control microswitch (14K) and causes the afterburner electro-valve (20K) (which opens) as well as the afterburner ignition electro-valve to be energized. The P2 air (from pick-off O on the compressor) available at the electro-valve (20K) supplies the turbo-pump (33) and is discharged on the RH side between frames 25 and 26.

967. One LP pump is sufficient to meet the dry engine demand. With the afterburner selected, the two LP pumps must be operating to meet the engine demand. In case of failure of one pump, the fuel pressure is no longer sufficient and the afterburner must be shut down.

968. **External Wing Tank Automatic Transfer System (refer to Figs 9-32 and 9-33).** The tanks are pressurized by air bled from the engine (tapping Q) through a pressure reducing valve (52), two non-return valves (55) and two filters (56). The function of the non-return valves (55) is to protect the system should an air leakage occur on the tank line between the tank and the valve. Valve boxes (65) protect the tank against any possible overpressure or underpressure condition.

969. At high angles of attack and if there is fuel left in the tank, one of the tank bottom valves (STOP AIR) (13) is uncovered and the valve is closed by the action of its float. Transfer continues through the valve still submerged. The fuel starts being transferred only when the upper level valves (8) in the fuselage tanks are uncovered. The fuel from the external tanks is then forced to the fuselage tanks through the tank bottom valves (13) and the filters (79).

970. In each system, there is a pressure switch (31Q and 32Q) which remains open as long as the fuel pressure is acting on the capsule of the pressure switch. When the fuel pressure decreases and the pressure switch is no longer submerged, the switch closes the circuit and causes the corresponding light to illuminate on the fuel transfer indicator (40Q) located on the instrument panel.

971. **Gun Bay Tank Automatic Transfer System (refer to Figs 9-36 and 9-37).** The gun bay fuel tank can be used to supply two systems as follows :

a. The engine fuel supply system.

b. For III O Only, the rocket motor fuel supply system.

NOTE

Paras 972 and 973 are applicable to III D and III O when the rocket motor is NOT fitted. Para 974 is applicable to III O Only and applies when the rocket motor is fitted.

972. The gun bay tank is pressurized by air bled from the engine (tapping Q) through a non-return valve (35), a pressure reducing valve (85), a non-return valve (86) and a filter (87). A valve box is provided for protection against any possible overpressure or underpressure condition.

973. With the GB TANK TRANSFER switch (19Q) in the ON position and the valve (22Q) open, the fuel level drops in the fuselage tanks and valves (10) open when no longer submerged. The fuel is then forced to valves (10) through the tank bottom valve (88). The pressure switch (37Q) in the delivery system remains open as long as the fuel is exerting a pressure. When the pressure starts dropping at the end of the transfer sequence, the switch closes the circuit and the corresponding light illuminates on the fuel transfer indicator (40Q) and for III D (240Q).

974. Since the gun bay tank fuel is partially used by the rocket motor, fuel transfer will take place only after rocket motor operation. To transfer the fuel, place the GB TANK TRANSFER switch (19Q) in the ON position to open valve (22Q). The fuel remaining in the tank will be transferred as described above.

975. **Gun Bay Tank and Fuselage Pylon Tank Automatic Transfer System (refer to Figs 9-34 to 9-37).** Both these tanks can be installed at the same time; they use the same transfer system :

a. The gun bay fuel tank is pressurized by air bled from the engine (pick-off O) through non-return valve (35), pressure reducing valve (85), non-return valve (86) and filter (87).

b. The fuselage pylon tank is also pressurized by air bled from the engine through non-return valve (35), pressure reducing valve (101), a non-return valve and filter (108).

976. Fuel is transferred firstly from the fuselage pylon tank and then the gun bay tank as follows :

a. With the gun bay tank transfer switch (19Q) in the OFF position; the indicator light on the fuel transfer indicator (40Q) and for III D (240Q) corresponding to the fuselage pylon tanks is extinguished.

b. When valves (10) are uncovered in the fuselage tanks, the fuel is then forced to valves (10) through the tank bottom valves (13), the filter (79) and the non-return valve (100).

c. The pressure switch (37Q) in the fuel transfer system remains open as long as the fuel is exerting a pressure. When the pressure starts dropping at the end of the transfer sequence, the switch closes the circuit to the corresponding light on the fuel transfer indicator and the light illuminates.

d. After the fuselage pylon tank is empty and valves (10) are uncovered, the gun bay tank transfer switch (19Q) is placed in the ON position and valve (22Q) opens. Fuel is then transferred as described in para 973.

977. **Wing Tank Automatic Transfer System with Rocket Motor Fitted — III O Only (refer to Figs 9-38 and 9-40).** The wing tanks are pressurized by air bled from the engine (pick-off O) through two pressure reducing valves (43 and 46), two non-return valves (44 and 47), two restrictors (96 and 97) and two filters (45 and 48). The fuel starts being transferred from the wing tanks only when the lower float valves (9) are uncovered in the fuselage tanks. The fuel from the wing tanks (3) is then forced to the fuselage fuel tanks (2) through a strainer (12), a pipe and a manifold (80).

978. A pressure-switch (35Q and 36Q) in each system is kept open as long as the fuel pressure is acting on the capsule of the switch. When the fuel pressure starts decreasing and the pressure switch is no longer immersed in the fuel, the switch operates the corresponding light on the fuel transfer indicator (40Q) and for III D (240 Q).

979. **Wing Tank, Rocket Motor Bay Tank (III O) and Rear Bay Tank (III D) Automatic Transfer System — Rocket Motor Not Fitted (refer to Figs 9-39 and 9-40).** The fuel is transferred as in the case of the wing tanks but it is first passed from the wing tanks (3) into the rocket motor bay tank (5) or rear bay tank before being forced into the fuselage tanks (2).

980. With the aircraft in a nose up or nose down attitude when the wing and bay tank fuel has been transferred, the pressure reducing valve (46) delivers air into the fuselage tanks through valve (9). A loop (81) enables any excessive pressure to be discharged through valve (59) and prevents any siphoning at this valve.

Fuel System Instruments

981. **Fuel Quantity Indicating System (Refer to Fig 9-42).** The fuel quantity indicating system provides information on the fuel quantity remaining in the fuselage fuel tanks. The system consists of a tank unit (27Q, left hand) and (26Q, right hand) mounted in each fuselage fuel tank group.

982. Through its dielectric constant (which is about 2), the fuel level acts upon a self-balancing bridge. The unbalance voltage resulting from the fuel level change is amplified (amplifiers 38Q and 39Q) and is then applied to a motor and potentiometer assembly forming the two-pointer fuel gauge (29Q) and for III D (229 Q).

983. Each pointer indicates the quantity of fuel remaining in each fuselage fuel tank group. When the fuel gauge pointers come to a stop on the 0.3 reading, 13.2 U.K. gal (60 l) of fuel is remaining which corresponds to the capacity of the inverted flight accumulator (4).

984. A pushbutton (33Q) enables the system to be tested for correct operation. When the pushbutton is depressed, the fuel gauge pointers are slowly moved to the 0.3 reading. When the pushbutton is released, the two pointers should return to their original position.

985. **Fuel Remaining Indicator (refer to Fig 9-42).** The fuel remaining indicator provides information on the total quantity of fuel remaining in the aircraft. A fuel volume meter (52Q) feeds the indicator (51Q) (III O) and for III D (251 Q) with engine fuel consumption information through an amplifier (53Q). The consumed fuel quantities are continuously subtracted from the total quantity preset before flight.

986. **Fuel Transfer Indicating System (refer to Fig 9-42).** Six pressure switches are electrically connected to the fuel transfer indicator (40Q) and for III D (240Q). Two pressure switches (31Q and 32Q) are located in the LH and RH external wing tank fuel transfer pipes.

987. Two pressure switches (35Q and 36Q) are located in the wing leading edge tank, wing tank and rocket motor bay tank (III O) or rear bay tank (III D) fuel transfer pipes. One pressure switch (37Q) is located in the gun bay tank fuel transfer pipe and one pressure switch (153) in the ventral tank fuel transfer pipe.

988. **Fuel Feed Low Pressure Indicating System.** A pressure switch (2E) located in the dry engine fuel feed pipe is connected to the FUEL warning light on the failure warning panel (1Z) and for III D (201Z) and the FAIL light (3Z).

989. **Fuel Low Level Warning System (refer to Fig 9-42).** Once all the transfers are completed, the fuel level starts dropping in the fuselage tanks. When it reaches one detector on the LH or RH tank unit (26Q or 27Q), it causes the 130 G warning light (28Q) and for III D (228 Q) to illuminate through the relay box (23Q). Switch (20Q) can then be used to open the crossfeed valve (21Q) to equalize the levels in the fuselage fuel tanks.

Fuselage Fuel Tanks (refer to Figs 9-43 and 9-44)

990. **General.** The fuselage fuel tanks are made of Superflexit flexible rubber. They form two symmetrical groups which each include a front tank (1) and a rear tank (2) with a feeder tank (7). The tanks are

interconnected by tubular interconnectors (3). The feeder tank (7) is attached to the rear tank (2) through a bolted interconnecting flange (4).

991. The front and rear fuel tanks are secured to the aircraft structure by means of anchor-studs (5) for the outer faces, the centre faces and the bottom and by means of spring clips (6) for the inner faces and the ends of the tanks (at frames 17, 20 and 23). The feeder tank is also attached by three clips (6) at frames 20 and 20N and on its outer face.

992. **Front Fuselage Tank (Refer to Figs 9-43 and 9-44).** The front fuselage tank (1) contains the following units :

a. The external tank transfer float valve (8) the support (12) of which is screwed onto the mounting flange (13) of the access door (14).

b. A lower interconnecting coupling (3), between the front tank and rear tank.

c. An upper interconnector (15), between front tank and rear tank.

993. On the outer face is a door (14) for access to the float-valve (8). At the bottom of the tank is a large mounting flange (16) for a circular door (17) provided for access to the inside of the tank for installation of the internal units; access to this mounting flange is gained through the gun bay. A mounting flange (18) is provided for connection of the external tank transfer pipe (19) and a coupling (20) is also provided for connection of the vent pipe.

994. **Rear Fuselage Tank (refer to Figs 9-59 and 9-60).** The rear fuselage tank (2) contains the following units :

a. The wing and rocket motor bay tank (III O) or rear bay tank (III D) transfer float-valve (9) the support (21) of which is screwed onto the mounting flange (22) of the access door (23) on the outer face.

b. The fuel tank units (26Q and 27Q).

c. The gun bay tank transfer float-valve (10) the supports (24 and 25) of which are screwed onto the mounting flange (27) of the upper access door (28).

d. The flange (30) of the wing tank transfer pipe and a circular door (31), for access to the inside of the tank for installation of the internal components, are screwed onto a large mounting flange (29). Access to the latter flange, which is located at the bottom of the tank, is gained through the LH or RH wheel well.

e. The filler port consisting of a flange (32) and a screwed counterflange (33) is blanked by a notched plug (11).

f. A door (34) retained by a spring and resting on a seal provides for sealing in case of incorrect locking of the plug.

g. The counterflange (33) which supports a sheet metal neck (35) tipped by a filter.

h. A port (4) provided for connection to the feed tank.

995. **Feeder Tank (refer to Fig 9-43).** The feeder tank (7) contains the supply pumps (5Q and 6Q) and the tank units (26Q and 27Q) attached at the bottom of the feeder tank to a mounting flange (36).

996. The rubber material of the tank (7) is clamped between the flange (36) and the counterflange (37). Mounted to the inner face are :

a. A flange (38) for interconnection of the LH and RH fuel tank groups.

b. A flange (44) for connection of the gun bay tank transfer pipe (26).

997. **Sealing (refer to Figs 9-43 and 9-44).** The following components are sealed by clamping the rubber edge around each port between a flange and studded counterflange :

a. Door mounting flanges (16 and 17).

b. Pump and fuel tank unit mounting flanges (36) (lower part of tanks and feeder tanks).

c. Fuel transfer pipe connecting flange.

d. Flange (18) for connection of external tanks.

998. A gasket (39) is used to seal the circular doors (17 and 31) (lower part of front and rear tanks) and is fastened by means of nuts fitted to the flange studs. A gasket (40) is used for sealing the wing tank transfer pipe connecting flange (30) which is attached by screws. A gasket (41) is used for the mounting flange of the pumps (5Q and 6Q) and fuel gauges (26Q and 27Q) and is attached with nuts to the studs.

999. Sealing is also provided by :

a. Sealed doors (14,23 and 28); the projecting parts of the tanks are clamped by nuts to the studs of counterflanges (13, 22 and 27).

b. Tangent screw clamps that clamp interconnector (15) and tubular interconnecting couplings (3).

c. The rubber edge of the tank (2) being clamped between counterflange (32) and flange (33) of the plug by means of screws to seal the fuelling point.

d. A rubber edge around the port is clamped between spacer (42), provided with studs integral with the aircraft structure, and flanges (43) located on either side to seal the interconnector (4) between the rear tank (2) and feeder tank (7).

Inverted Flight Accumulator (refer to Fig 9-45)

9100. **General.** The light alloy inverted flight accumulator (4), 13.8 U.K. gal (63 l) in capacity is made up of two parts (1 and 2). A rubber diaphragm (78) separates the fuel from the pressurizing air (P2).

9101. The chambers forming the upper part (1) and the lower part (2) of the inverted flight accumulator (4) include a welded double walled perforated bottom (3) which supports the diaphragm (78). The upper chamber (1) incorporates an expansion valve (62) connected to the vent pipe of the fuselage tanks and a pipe coupling (5) used for both refuelling and fuel supply.

The lower chamber (2) incorporates a welded flange (6) for connection of the pressurization system and bleed system.

9102. The edges of the rubber diaphragm, clamped between the flange of each chamber, are used as seals. The chambers are assembled by means of screws and Nylstop nuts (7).

9103. **Installation.** The inverted flight accumulator (4) is vertically installed between frames 15 and 17 between a lower support (8) and a stirrup (9) attached to the aircraft structure by screws (10). The lower support (8), with lightening flanged holes, is designed to suit the contour of the accumulator which is kept in position by the stirrup. The support and the stirrup are lined with rubber strips.

9104. The expansion valve (62), which is located at the upper part of the accumulator, is screwed into a sleeve (11) welded to the accumulator. Sealing is ensured by a copper-asbestos gasket (12). This valve consists of a fixed part including a body (13) and a guide (14) tipped by a stop (15) and a sliding part including a piston (16). This piston is held against the O-ring seal holder (17) by a calibrated spring (18). At the centre part of the piston, a valve (19) is held against the piston port by a centre spring (20).

9105. **Operation.** The overpressure compresses the spring (18) by pushing the piston (16) back. The valve (19), which comes into contact with the stop (15), opens and compresses the centre spring (20) and the overpressure is discharged through the chamber (21) and the holes (22) provided in the guide (14) and the valve body.

Crossfeed Unit (refer to Fig 9-46)

9106. **Description.** This unit consists of a full-flow valve controlled by an electric actuator and two connections to the LH and RH fuel tanks. The valve consists of a body formed by two flanges (1 and 2) containing the closing flap (3) and its link-rod (4).

9107. Flange (1) carries the control block (5) and the angle drive (6) which is connected to the electric actuator (7). The flange also carries the reducing coupling (8) onto which is screwed the connection (9) for the RH tank.

9108. Flange (2) takes a reducing coupling (10) for the connection to the LH tank which is installed by means of studs and nuts (11). This connection consists of a straight section (12) carrying a threaded plug and a section tipped by a toroidal end piece and a nut (13).

9109. The assembly is sealed by O-rings (14), two of which are fitted at the connecting ports on either side of the flap to perfectly seal the valve in the closed position.

9110. **Operation.** The actuator (7) operates the closing flap (3). This motion is transmitted through an angle-drive (6) and a splined shaft (15) to which the link-rod (4), connected to the flap (3) by a shouldered screw (16), is attached.

9111. **Installation.** The crossfeed unit attaches through a plate to a support, between frames 20 and 21 on the RH side, in the lower bay. The connection to the LH feeder tank is made by a toroidal coupling and the connection to the RH feeder tank by a Wig-O-Flex coupling.

Coupling Block (refer to Fig 9-47)

9112. **Description.** The coupling block consists of a full-flow valve, controlled by an electric actuator (7) and two connections one to the turbo-pump and one to the fuel inlet. The valve consists of a body joined by two flanges (1 and 2) housing the closing flap (3) and its link-rod (4). Flange (1) carries the control block (5) connected to the electric actuator (7) and a straight union (6) onto which is screwed the fuel inlet connection (8). This connection incorporates two ports fitted with Wig-O-Flex couplings (9). Flange (2) takes a reducing union (10) onto which is screwed the turbo-pump connection (11). This connection is fitted with an endpiece and a nut (12).

9113. The assembly is sealed by O-rings (14), two of which are fitted at the connecting ports on either side of the flap (3) to perfectly seal the valve in the closed position.

9114. **Operation.** The actuator operates the closing flap (3). The motion is transmitted through a splined shaft (13) to which the link-rod (4), connected to the flap (3) by a shouldered screw (15), is attached.

Pre-Cooler (refer to Fig 9-48)

9115. **Description.** The pre-cooler is an air-to-air heat exchanger intended for lowering the temperature of the P2 air bled from the engine for cockpit air conditioning. The temperature of the air, which is 300°C at the entry of the pre-coolers is lowered by approximately 100°C through thermal exchange with ram air from the outside of the aircraft.

9116. The pre-cooler consists of a welded plate rectangular casing stiffened by four walls (1) and three flanged webs (2). The two cambered faces are reinforced by top-hat sections (3) inserted in each wall. The inlet coupling (4) and outlet coupling (5) are connected to lateral ducts (6 and 7) which direct the air to the compartments (9) between the walls. On the inside of the casing is a tube cluster (10) silver-brazed to the two outer walls.

9117. **Operation.** The cooling air from a ram air-scoop flows across the pre-cooler through the tube cluster (10). It is then discharged overboard by a pipe between frames 25 and 26 on the RH side of the fuselage. The conditioning or pressurization air conveyed by the air inlet duct (6) is distributed in the compartments (9) of the pre-cooler. It flows around the tubes of the tube cluster (10) and is directed to the various systems by the outlet duct (7).

Dual Pre-Cooler (refer to Fig 9-49).

9118. **Description.** The dual pre-cooler is intended to lower the temperature of the air used for pressurization of the fuel tanks and hydraulic reservoirs. The cooler fitted is designed for connection of two air systems; however, only one system is used. The dual pre-cooler operation is identical to that of the pre-cooler described in para 9117.

9119. The front section (16), provided with an air inlet (17) and an air outlet (18), is attached to the rear section by means of screws (19). The air inlet and outlet are connected by a tube cluster (20). The assembly is stiffened by two walls (21).

Filters (refer to Fig 9-50).

9120. **Description.** The following filters are fitted in the tank pressurization systems :

a. Wing and external tank pressurization.

b. Fuselage tank pressurization.

c. Gun bay tank pressurization.

d. Tank pressurization.

9121. Each filter consists of two bodies (1 and 2), screwed one into the other, and a retainer ring (3) and contains the following components :

a. A filter element (4) which has two faces (5 and 6) made of light alloy. The edge of the face (6) of the element is clamped between the two bodies to form a seal.

b. A baffle plate (7).

9122. The filters are connected to the pipes by means of Wig-O-Flex couplings (8) or Arsaero couplings (9). The direction of flow is indicated on the filter body by an engraved arrow.

9123. **Operation.** The hot air is deflected by the baffle plate (7) between the inner walls of the body (1) and the filter element (4); it then flows out through the centre section of the filter element.

Wing Tank Pylon (refer to Fig 9-51)

9124. **Description.** The wing tank pylon is designed to carry one of the following :

a. A 286 U.K. gal (1300 l) tank.

b. A 374 U.K. gal (1700 l) tank.

9125. The pylon consists of two sections as follows :

a. A front section including :

(1) Two ribs, one (1) upper and one (2) lower attached to the front web (3) and the rear centring fitting support (4). This assembly is stiffened by five intermediate webs (5).

(2) A leading edge skin (6) stiffened by two ribs (7). An inner skin (8) provided with a door (9) for access to the internal air-fuel units. An outer skin (10) also provided with four access doors :

(a) Door (11) for access to the adjustment point of the air-fuel coupling (pylon-to-wing) and front centring swivel fitting (23).

(b) Door (12) for access to the adjustment point of the air-fuel coupling (pylon-to-tank).

(c) Door (13) for access to the ejector squibs.

(d) Door (14) for access to the rear centring fitting.

(3) A tank bearing plate (15) bolted to the lower rib (2).

b. A rear section including :

(1) An upper rib (16) and a lower rib (17) attached to a finger-plate (18) and a ledge (19).

(2) Two webs (20) used to stiffen the assembly.

(3) A two-piece skin (21). The rear skin piece is screwed onto the rear centring fitting support (4).

9126. The pylon is fitted with the following :

a. At the upper part (pylon-to-wing connection) :

(1) A front centring swivel fitting (23) which can be adjusted vertically.

(2) A pressurization coupling (24) carrying an opening finger.

(3) A fuel coupling (25).

(4) The upper flange (26) for the ejector squibs.

(5) An electric connector (88Y RH side and 87Y LH side).

(6) A retainer cable (55) preventing the electric connectors from being damaged on removal of the pylons.

(7) The anchorage spindle (27).

(8) A rear centring swivel fitting (28).

b. At the lower part (pylon-to-tank connection) :

(1) A quick-release connector (29) for connection to combined fuel tank/ rocket launcher.

(2) A pressurization coupling (30) with a sealing valve.

(3) A fuel coupling (31).

(4) The valve box coupling (32).

(5) A seat (33) for the front centring swivel fitting.

(6) The lower flange (34) for the ejector squibs.

(7) A microswitch (35). The purpose of this microswitch is to prevent firing of the emergency initiator on operation of the wing jettison button if the wing fuel tanks have previously been jettisoned by operation of the tank jettison button (normal initiator).

(8) The mounting flange (36) of the anchorage spindle.

(9) A seal (37) for the rear centring swivel fitting.

(10). A quick-release connector (38) for the fuel dump valve.

c. Internal :

(1) Couplings (24 and 30) connected by a rigid pressurization pipe (39).

(2) Couplings (25 and 31) connected by a fuel pipe (40) provided with a filter (41).

(3) Coupling (32) which carries the overpressure-underpressure valve box (42). The couplings are fitted with a seal and can be adjusted vertically.

(4) The electric cable from the connector (87Y or 88Y) runs to resistor box (43) and is connected to the two initiators (44) of the ejector squibs (34) and to connectors (29) and (38).

NOTE

The ejector squib connection is fitted with a plain blanking plug (61) used with the pylon tanks.

9127. **Installation (refer to Fig 9-51).** The pylon is stiffened laterally by locking the spindle (27) into the corresponding fitting (46) in the wing by means of a threaded pin (45). It is stiffened longitudinally by two centring swivel fittings (23 and 28) which can be adjusted vertically for correct bearing in the seats (47) provided in the wing.

9128. The tank is set two degrees nose-down relative to the level reference line. This setting is obtained by slightly rotating the sleeve (48) inside the wing fitting (46) the bore of which is off-centred. The fuel coupling (25) can be adjusted vertically by the nut (49). This connection is sealed when the seal (51) is slightly compressed by the coupling (50). The pressurization coupling (24) is fitted with a finger (52) enabling the wing valve (53) to be opened and can be adjusted vertically by the nut (49).

Fuselage Tank Pylon (refer to Figs 9-52 and 9-53)

9129. **Description.** The pylon is the junction component between the fuselage and the pylon tank. It is designed to carry a 286 U.K. gal (1300 l) or 242 U.K. gal (1100 l) tank. The pylon consists of three sections which are assembled by means of screws. The loads in the pylon assembly are distributed by the skin panels and the upper and lower plates. The pylon assembly consists of :

a. A front section including :

(1) An upper plate (1) and a lower plate (2) assembled by two frames (3). This assembly is reinforced by stiffeners.

(2) A RH skin panel (4) which can be removed for access to the internal components.

(3) A LH riveted skin panel (5) forming the leading edge.

b. A centre section including :

(1) An upper plate (7) and a lower plate (8) assembled by four frames (9) and the two anchorage fittings (10). This assembly forms the main structure of the pylon.

(2) A RH skin panel (11) provided with access doors.

(3) A door (13) for access to equipment.

(4) A LH skin panel (14) provided with access doors.

(5) A door (15) for access to the ejector initiators.

(6) A door (16) for access to the fuelling port.

(7) A door (6) for access to the pylon-to-fuselage couplings.

c. A rear section including :

(1) An upper plate (17) and a lower plate (18) assembled by two frames.

(2) A one-piece skin panel (19) used to stiffen the assembly.

9130. The pylon is fitted with the following :

a. At the upper part (pylon-to-fuselage connection) :

(1) A front fitting (20) with an anchorage system and a centring spigot (21).

(2) A pressurization coupling (22).

(3) A fuel coupling (23).

(4) An electric connector (122Y).

(5) A rear fitting (25) with an anchorage system and a centring swivel fitting (26).

b. At the lower part (pylon-to-tank connection) :

(1) A pressurization coupling (27).

(2) A fuel coupling (28).

(3) A blanking plug (29).

(4) A front swivel fitting support (30).

(5) The lower flange of the ejector and the tank support plate (31)

(6) A rear swivel fitting support (32).

c. The pressurization couplings (22 and 27) are connected by a system including :

(1) A pressure reducing valve (34).

(2) A non-return valve (33).

(3) A filter (35).

(4) A valve box (36).

(5) A self-sealing valve built into the coupling (27).

d. The fuel couplings (23 and 28) are connected by a system including :

(1) A filter (38).

(2) A pressure switch (39) connected to the valve box.

e. An electric cable is provided to supply the ejector initiators (40) and the pressure switch through a resistor box (41).

9131. **Installation.** The pylon is centred by engaging the front spigot (21) and the rear swivel fitting (26) into the corresponding fittings provided in the fuselage. Ensure that the locking devices are in the right position. The plunger (43) is moved to the down position and the balls (44) lock the system. Correct positioning of the plunger is checked by means of a feeler (45) which fits into a slot of the plunger and locks it. Tightening the screw (46) brings the contact points (47) of the anchorage fittings (frames 21 and 26) against the fuselage attachment fittings which stiffens the assembly. The rear upper part is connected to the aircraft by a fairing (53) attached to the pylon by means of screws.

9132. After installation, the pressurization coupling (22) can be adjusted vertically by means of a nut (48). The coupling is sealed by the cylindrical seal (49) being applied on the fuselage coupling; the finger (50) is used to open the fuselage system valve.

9133. The fuel coupling (23) can be adjusted in a like manner. This connection is sealed by the cylindrical seal (51) being compressed on the fuselage side and by an O-ring between the fixed and mobile parts of the coupling. On the fuselage side, the coupling is fitted with a self-sealing valve (52).

Pylon Tanks — 242 and 286 U.K. gal (refer to Fig 9-54)

9134. The description given below covers the 242 U.K. gal (1100 l) and 286 U.K. gal (1300 l) tanks.

9135. **Description.** The tank is a welded plate streamlined body with fins at the rear. Frames (1) and anti-surge baffles (2) stiffen the assembly. The tank is provided with two bolted doors (3) for access to the anchorage device. The tank includes the following :

a. A filler plug (4).

b. An earthing connector.

c. A pressurization coupling (5).

d. A fuel coupling (6) connected to an internal pipe fitted with two float valves (13).

e. An overpressure-underpressure air coupling (7).

f. A front centring swivel fitting (8).

g. An anchorage fitting (9) in a recess.

h. A rear centring swivel fitting (10).

i. An electric connector connected to the fuel dump valve (12).

9136. **Installation on Pylon.** When the clamping nut (14) of the tulip fitting (15) is loosened, the catches (16) fit into the groove of the tulip fitting and are locked by the ejectable piece (17). Tightening the nut (14) brings the pylon support plate into contact with the tank flat and the swivel fittings (8 and 10) into the cups (19) provided in the pylon. The tulip fitting (15) is prevented from rotating by two pins (20) and is fitted with a set of dished washers (21) permitting the assembly to be clamped unstressed.

9137. The couplings (5, 6 and 7) are fixed couplings. The connection at the couplings (22) on the pylon is sealed by compressing the seals (23). Correct compression is obtained by adjusting the couplings (22) vertically by means of nuts (24). Coupling (5) is provided with a finger (25) used to open the valve (26) on the pylon.

Pylon Tank — 374 U.K. gal

9138. For information on the 374 U.K. gal (1700 l) tank refer to AAP 7271.035-3M.

Supersonic Tank (refer to Fig 9-55)

9139. **Description.** The tank is a welded plate streamlined body stiffened by frames (1) some of which are fitted with anti-surge baffles. The tank has a capacity of 110 U.K. gal (500 l) and is installed directly under the wings; a small fairing is used to maintain the profile. Two doors (2) permit access to the internal components, one door is located in the side and the other on the lower surface. The internal equipment consists of a pipe assembly (12) connected to the fuel coupling (6) and provided with a filter (13) and valve (14). The tank includes the following :

a. At the upper part :

(1) A filler plug (3).

(2) A front centring swivel fitting (4).

(3) A pressurization coupling (5).

(4) A fuel coupling (6).

(5) A pressure relief valve (7).

(6) An anchorage fitting (8) with a sealing plug in a mounting recess.

(7) A rear centring swivel fitting (9).

b. At the lower part :

(1) A plug (10) for access to the anchorage system.

(2) Two bleed plugs (11), one at the front, one at the rear.

9140. **Installation.** A spindle (17) and a support plate (15) are mounted at the wing attachment point prior to installing the tank. Tightening the nut (16) brings the support plate (15) on to the tank bearing surface and the centring swivel fittings (4 and 9) into the corresponding seats (18) provided in the wing. The front centring swivel fitting (4) is preset vertically for correct clamping of the tank and for the tank to be set at the correct angle of incidence. The spindle (17) is prevented from rotating by two pins (21) and is fitted with a set of dished washers (22) which permits the tank to be clamped unstressed.

9141. The air and fuel systems are connected by means of couplings (5 and 6) which can be adjusted vertically. The connections are sealed by the seals (20) being compressed on the wing couplings (21). These seals are mounted on swivelling supports (19) enabling the seals to correctly bear on the wing couplings. The swivelling support (19) is fitted with a finger (23) used to open the valve in the wing coupling.

Gun Bay Fuel Tank (refer to Figs 9-57 and 9-58)

9142. **Description.** A fuel tank (1) can be installed in the gun bay located at the bottom of the fuselage between frames 17 and 20. The tank is a welded plate tank internally stiffened by webs and stiffeners with flanged holes (anti-surge). The front section includes a housing (2) for the pressurization equipment; this housing is closed by welded and riveted cut-out plates. The tank includes the following :

a. *Inside the Tank.*

(1) An engine fuel suction pipe (3) fitted with a valve (4).

(2) A drain plug (5).

(3) A refuelling and suction pipe (21); this pipe is free to swivel.

NOTE

III O Only. Pipe (21) is used to supply the rocket motor.

(4) At the top point, two pipes (6) used for venting, vapour relief and pressurization.

b. *Outside the Tank.*

(1) In the housing, a pressure reducing valve (12) which receives the compressed air from the aircraft system through a pipe (13) fitted with a charging connection (14). This pressure reducing valve is provided with a reference pressure inlet and delivers a differential pressure of :

$7.25 \begin{matrix} +0.43 \text{ lb/in}^2 \\ -0.29 \text{ lb/in}^2 \end{matrix} (0.5 \begin{matrix} +0.03 \\ -0.02 \end{matrix} \text{ bar})$.

It is fitted with a safety valve connected to the vent system; this valve is calibrated at 8.7 ± 0.72 lb/in² (0.6 ±0.05 bar). It is connected to a valve box (16) through a non-return valve (17). The overpressure valve is calibrated at 9.9 to 11.8 lb/in² (0.686 to 0.814 bar) and the underpressure valve at 0.07 lb/in² (−0.005 bar). The valve box is provided with a pressure test connection (18) for pressure checking on the ground.

(2) From the valve box an overpressure/underpressure relief pipe is connected to the vent system and a pressurization pipe connected to the internal pipes through a tee-union and a filter (19).

(3) A vapour relief pipe and a bleed plug (20) connected to the tee-union.

(4) Four anchorage fittings (22) and two guide tracks (23).

(5) Access to the equipment housing is gained through two doors located at the bottom of the tank; one (24) of these doors is attached by Dzus fasteners, the other (25) by screws.

(6) A door at the rear of the tank for access to couplings (38 and 43).

(7) Access to the internal components is gained through two doors (7 and 8) sealed by gaskets (9) and attached by means of bolts (10).

(8) On the rear door is mounted the attaching flange (11) of the suction pipe (3) fitted with a Wig-O-Flex coupling. The internal front door (8) is located in the equipment housing.

9143. **Installation.** The gun bay fuel tank is attached and connected to the aircraft systems as follows :

a. *Attachment.* After engaging the guide-tracks (23) the tank is attached by tightening the four captive screws (26) on the attachment fittings (22). On the front attachment fitting, the captive screw (26) fits into the internally threaded sleeve (27) and is then prevented from rotating by a pin (28). As the screw is being tightened, the sleeve (27) is moved down inside the bushes (29) until it is abutting. The screw is finally prevented from rotating by the lockplate (30) fitting into one of the notches of the screw head. On the rear attachment fitting, the captive screw (31) fits into the aircraft attachment fitting (22). It is prevented from rotating by a lockplate (32).

b. *Connections.*

(1) The tank is connected to the pressurization (B) and vent (C) systems by means of one swivel Quinson coupling (33) and one straight Quinson coupling (34). These couplings are fitted with a crimped shouldered ring (35) which seals the connection together with the seals (37) as the nut is tightened.

(2) The removable suction pipe (38) is connected to the aircraft system and to the tank by means of Wig-O-Flex couplings.

(3) The vapour relief connection (46) is made by a coupling (47). Sealing is achieved by compressing O-ring (48) on the pipe by tightening the nut.

(4) The removable refuelling and rocket motor fuel suction pipe (43) is connected to the aircraft system and to the tank by toroidal couplings. Tightening the nut (44) seals the connection.

(5) When the gun bay tank is not installed, the aircraft pressurization, venting and suction systems are plugged. The vapour relief system is blanked off by a door at frame 17.

Rocket Motor Bay Fuel Tank (III O) or Rear Bay Fuel Tank (III D) (refer to Fig 9-59).

9144. **Description.** The fuel tank consists of two sections, one section (1) accommodated between frames 27 and 33 and the other section (3) which forms a fin providing for lateral stability; at the end of this fin is a photographic camera housing (4).

9145. The tank is made of welded plates forming two sealed compartments (5 and 6) separated by a centre web (7) and not interconnected. It is stiffened internally by a lattice structure of webs and stiffeners cut out at their lower and upper parts for interconnection; this structure is used as anti-surge baffles. The tank includes the following :

a. *Inside the Tank in Each Sealed Compartment.*

(1) A refuelling and suction pipe (8), connected to a flange located on the front face and routed through the webs and stiffeners down to the bottom point of the tank.

(2) A vapour relief, pressurization and fuel transfer pipe (9), connected to a flange located on the front face and routed up to the top point of the tank.

b. *Outside the Tank.*

(1) One centring stud (10) on the front

face.

(2) Two inlet couplings (11) and two outlet couplings (12) in recesses in the front face.

(3) Six ball-joint cages (13) bolted to welded plate supports (14) on the front section (1).

(4) Two ball-joint cages (16) bolted to welded plate supports (17) on the fin section (4). These ball-joint cages are fitted with captive pins (18).

(5) Two drain plugs (15) at the bottom point of each compartment

9146. **Installation (refer to Fig 9-59).** The tank is attached and connected to the aircraft systems as follows :

a. *Attachment.* There are six attachment points on the front section (1) and two attachment points on the fin section (4). At the front section, the ball-joints (13) fit into the aircraft attachment fittings (19) and are axially locked by self-locking nuts (20). At the fin section, the ball-joints (16) bear on the aircraft attachment fittings (17) and are axially locked by captive pins (18).

NOTE

The tank cannot be jettisoned.

b. *Connection.* There are two symmetrical connection points consisting of :

(1) A clamp (22) with two ports; this clamp is attached to the manifold (23) by means of captive screws (24) and sealed by O-rings (25) fitted to the centring bushes (26).

(2) Two fixed couplings (27 and 28) connected to the clamp by Wig-O-Flex couplings (29). These couplings are attached in the bay by means of screws and nuts (30).

(3) Two mobile couplings (11 and 12) which are part of the tank.

(4) Coupling (12) is sealed by the sliding motion of the ball-joint (31) resulting from the action of the spring (32); however, the travel of the ball-joint is limited by the shouldered plunger (33). As it is rotated, the nut (34) pushes the piston (35) and plunger (33) assembly upwards which results in compressing the seal (36) through the ball-joint (31) and opening the sealing valve (37).

(5) Coupling (11) is similar to coupling (12) but is not fitted with a sealing valve.

NOTE

III O Only. In case of flights without the rocket motor bay fuel tank (rocket motor configuration), the manifold (23) is blanked by a double coupling (38). This coupling is fastened by means of the captive screw (24). The sealing method is similar to that of clamp (22) in the tank.

Bomb Carrier Tank

9147. For information on the bomb carrier tank, refer to AAP 7271.299-2M and AAP 7271.299-3.

AIR SUPPLY SYSTEM AND COOLING

General (refer to Figs 9-61 to 9-65)

9148. The engine uses outside air for combustion and cooling which is supplied via two air intake ducts located on either side of the fuselage. The air flow is regulated by a moving shock cone (also known as Mice) in each air intake. The position of the shock cones is slaved to the Mach number.

9149. The air intake ducts are also provided with auxiliary air intakes intended to increase the air flow according to the engine demand (low airspeed). The cooling air from the boundary layer bleeds (19) is distributed in the engine compartment. Various air inlets and ducts are also provided to cool the accessories.

Engine Combustion Air Supply (refer to Fig 9-61)

9150. The air is conveyed to the engine by the LH and RH air intake ducts. The semi-circular cross-section of the duct at the air intake (20) is variable up to the convergence point of the two air ducts at frame 20. A vertical dividing wall (22) maintains the air flows separated up to the engine inlet at frame 23 in order to protect the engine compressor against any possible variance in the air supply. The auxiliary air intake ducts (21) merge into the main air ducts.

9151. The door (23) of each air duct is shaped to match the outer fuselage contour and is calibrated to open at 4.35 lb/in^2 (300 mb) by a spring (24). It is opened by the differential pressure, ie external pressure being higher than internal pressure. The continuity of the air flow is maintained by a deflector plate (25). In the open position, the door fits into a dimple provided in the deflector plate. The door travel is limited by a stop (26).

Shock cones (refer to Figs 9-66 to 9-69)

9152. Each air intake is provided with a variable position shock cone. The shock cone is required at the high supersonic speed range where disturbances are encountered which could seriously affect the engine air supply.

9153. For a given speed, an oblique shock-wave is created, the apex of which is situated at the apex of the shock cone. Across the shock-wave, the speed of the molecules decreases. There is another shock-wave (straight wave) the location of which varies with the air flow to the engine as follows :

a. *High Speed Air Flow.* The straight wave enters the air intake duct : supercritical conditions.

b. *Low Speed Air Flow.* The straight wave moves towards the tip of the shock cone : subcritical conditions.

9154. The straight shock-wave, which also reduces the speed of the molecules which are passing through it, intercepts the conical shock-wave. The speeds downstream of both of these waves being different, a pulsatory phenomenon known as Buzz is encountered at the interception point. When the conical wave and the straight wave intersect beyond the air intake, the Buzz is encountered on the outside of the air duct.

9155. Other flight conditions can move the intersection point of the two waves closer (subcritical conditions) and the Buzz is encountered in the air intake duct, which disturbs the air supply to the engine compressor.

9156. As the shock cone is moved forward, the conical shock-wave is displaced with the normal shock wave practically remaining at the same point (its position varies with the air flow to the engine). This moves the intersection point of the waves outside the entry zone of the air intake, thereby avoiding disturbances in the air supply to the engine compressor.

9157. The above phenomenon requires the shock cone to be slaved to the flight envelope to prevent any inadvertent occurrence of the Buzz which would result in a sudden drop in engine efficiency.

9158. **Mechanical Units (refer to Figs 9-68 and 9-69).** A reduction gear (16) is driven by a motor (59C). The transmission between the reduction gear and the actuating jack (2) consists of shafts (1) (and (4) for III D) with universal joints (17) articulated at the reduction gear jack connections. Each output shaft (18) is connected to the wheel of the reduction gear train and has a universal joint fork (19) secured by a nut (20) and a lock-plate (21).

9159. For III D, rotational movement of shaft (1) is transmitted through an angle drive (30) to shaft (4).

9160. Each jack is secured to a support (3) integral with the aircraft structure and consists of a gear box with a threaded shaft. The end of the threaded shaft (5) is connected to the shock cone structure by means of a fork-end (6) and is prevented from rotating. Any rotational motion of the gear box causes the threaded shaft to move and move the shock cone.

9161. The yoke (22) and the fork (19) of the universal joint are assembled with the sleeve (23) by means of plain pins (24) and pinned nuts. The universal joint shaft (1) has a splined shaft (25) which meshes with the internal splines of the forked sleeve (23) and takes up the side play.

9162. **Slaving Units (refer to Fig 9-67).** The automatic shock cone slaving system consists of :

a. A slaving box (46C) (shock cone amplifier).

b. A potentiometer box (60C) situated on the shock cone actuator and including two potentiometers; one potentiometer is used for slaving purposes, the other is operated in connection with the shock cone position indicating system.

c. Three contactor relays (56C, 57C and 58C) in the shock cone box (this box and the slaving box (46C) are located between frames 16 and 17 at the bottom RH side).

d. A MICE pushbutton on panel (43C).

e. The slaving box is connected to d.c. and a.c. power systems and is fed with Mach number information from the air data computer output multiplier (2C).

9163. The manual shock cone slaving system consists of :

a. Two relays (55C — extension) (54C — retraction) in the shock cone box.

b. Control and position indicating units :

(1) An IN-OUT switch on panel (43C) (manual control).

(2) A position indicator (42C) and for III D (242C).

9164. The position of the shock cones is indicated by a shock cone position indicator (42C) and for III D (242C), fed with information from a potentiometer (60C) controlled by the shock cone actuator. The indicator is calibrated in terms of Mach number and shows the position in which the shock cones should be, according to the indicated Mach number.

9165. **Automatic Operation (refer to Figs 9-66 to 9-69).** The automatic mode of operation is selected by depressing the MICE pushbutton on the control panel (43C). Depressing this self-held button energizes the automatic system relay (56C) which closes. The current is kept available at the terminals of the Auto-Extension contactor relay (57C) and Auto-Retraction contactor relay (58C).

9166. Starting from a Mach number of 1.2, and for increasing airspeeds of the aircraft, the slaving box (46C) determines control signals according to Mach number information supplied by the output multiplier (2C) of the air data computer. These control signals are delivered in the form of energization currents which are applied to the electro-magnet of the Auto-Extension contactor relay (57C), which closes.

9167. The power at the terminals of 57C and 58C supplies the electric actuator (59C) of the shock cones. The actuator also drives the potentiometers (60C) which monitor the shock cone displacement until the control signal from the slaving box (46C) is balanced. The energization current to the electro-magnet of the contactor relay (57C) is then cut-off by the slaving box.

9168. As the aircraft is being decelerated, the output multiplier of the air data computer transmits the opposite information through the slaving box (46C). The electro-magnet of the Auto-Retraction contactor relay (58C) is energized, the relay closes and supplies power to the shock cone actuator (59C) until the control signal is balanced by the potentiometer (60C). The supply to the electro-magnet of the contactor relay (58C) is then cut-off.

9169. **Manual Operation (refer to Fig 9-67).** The shock cones can be directly controlled by the IN-OUT shock cone control switch. Moving the control switch to the IN or OUT position cuts-off the supply to the relay of the control panel (43C) which opens and cuts-off the energization of relay (56C) which also opens, thus isolating the Auto relays (57C and 58C).

9170. Placing the switch in the OUT position energizes the electro-magnet of relay (55C) which closes. Placing the switch in the IN position energizes the electro-magnet of relay (54C) which closes. The electric actuator (59C) is supplied only while the control switch is being operated.

9171. **Installation (refer to Figs 9-68 and 9-69).** The shock cone components are installed as follows :

a. *Attachment of Shock cone Reducing Gear.* The reducing gear (16) is attached between frames 14 and 15 under the nose undercarriage well ceiling as follows :

(1) At the front, to an attachment fitting (7) by means of a pin (12) and Nylstop nuts; this fitting is secured to the slanting frame structure by means of screws, nuts and washers.

(2) Laterally, by two brackets (8 and 9)

secured to the reducing gear and the support (15) integral with the structure by means of screws (14).

b. *Shock cone Actuating Jack.* The shock cone displacement takes place in three different planes. To allow for angular displacements of the actuating jack (2), the latter is hinged on to a fitting (3) attached to the aircraft structure through a universal joint system (26). The lower and upper parts of the universal joint yoke are attached to the forks of the fitting (3) (fixed point). The LH and RH side parts are attached to the forks of the actuating jack (2). A plain pin, a washer and a nut (27) are used at each attachment point.

Engine Cooling System (refer to Fig 9-61)

9172. The air is conveyed by two ducts (19) into a centre duct (27), then through an upper compartment (28) and a lower compartment (29) to the distributing ducts (31 and 32). Each compartment includes a cut-out deflector plate (30) providing for correct air flow. The upper duct (31) and the lower duct (32) lead into the engine compartment (33). The air flows in this compartment and cools the internal and external surfaces of the engine anti-heat radiation blanket (34).

Rear Fuselage Structure Box Cooling (refer to Figs 9-61 and 9-63)

9173. The rear fuselage structure box is cooled by air from air scoops (13) (LH and RH fillets). The air flows through the frames between the engine compart-

Table 9-8 Cooling — External Air Ducts

Index No	Description — Function	Location
1	Cockpit fresh-air system air scoop	Bottom — Between frames 1 and 2
2	Air conditioning system evaporator steam outlet	Bottom; Nose cone frame VII (III D) Bottom — frame 10 (III O)
3	Air conditioning system heat exchanger air inlet	Centre section of boundary layer bleed (LH) (III O) Nose cone front section (III D)
4	Air conditioning system heat-exchanger air outlet	Nose cone lower section (Frame VI) (III D) Bottom — frame 14 (III O)
5	gun barrel top plate cooling air inlet	Wing lower surface LH and RH sides
6	gun barrel top plate cooling air outlet	Bottom — LH and RH sides between frames 15 and 16
7	Equipment bay cooling air outlet	Bottom — LH and RH sides between frames 15 and 16
8	Gun pack cooling air inlet	Bottom — On gun pack
9	Gun pack cooling air outlet	Side doors of gun pack
10	Generator cooling air inlet and outlet	Generator door — Bottom — LH side between frames 24 and 26
11	Accessory gear box cooling air outlet	Accessory gear box door — Bottom between frames 24 and 26
12	Alternator cooling air inlet and outlet	Alternator door — Bottom — RH side between frames 24 and 26
13	Rear fuselage structure box cooling air inlet	Lower surface fillet
14	Rear fuselage structure box cooling air outlet	Top — LH and RH sides between frames 37 and 38
16	Pre-cooler air inlets	Top — LH and RH sides between frames 23 and 24
17	Pre-cooler air outlets	Top — LH and RH sides between frames 25 and 26
18	Rear lateral bay air outlets	Bottom — LH and RH sides between 33 and 35
19	Engine cooling air inlets	Engine Air intakes LH and RH
21	Ballasted nose cone cooling air inlet	In nose cone (III O)
22	Ballasted nose cone cooling air outlet	In nose cone (III O)
23	Upper box cooling air inlets	On fairing between frames 17 and 23 (III D)
24	Upper box cooling air outlets	On fairing between frames 17 and 23 (III D)
25	Rear side bay cooling air inlets	Bottom; LH and RH sides between frames 33 and 34 (III D)
26	Turbo-pump air outlet	RH side between frames 25 and 26 (III D)
Index numbers refer to items shown in Figs 9-62 and 9-66		

ment sheets and the skin. The air is then discharged to the outside through louvres (14) located on either side of the fin.

Accessory Cooling System (refer to Figs 9-62 to 9-65)

9174. **Gun barrel Housing Top Plate Cooling.** The air used to cool the LH and RH gun barrel housing top plates is obtained from an air scoop (5) located at the lower surface of the leading edge of each wing. The air is conveyed by a tube leading to the rear section of the top plate (frame 17) and is discharged through louvres (6) within the depression zone at the front of the top plate (rear of frame 15).

9175. **Gun Housing Cooling.** The air is collected by a scoop (8) located beneath the front section of the gun pack. The air is conveyed by two ducts routed in the gun pack; it is directed into the LH and RH gun housings and is discharged through louvres (9) located at the bottom rear end of the gun pack.

9176. **Pre-coolers.** The air used for the pre-coolers is collected by a scoop (16) and is discharged through outlet (17).

9177. **Air-To-Air Heat Exchangers — III 0 Only (refer to Fig 9-62).** The following air-to-air heat exchangers are fitted :

a. *Cockpit Air Conditioning Heat Exchanger.* The cooling air is collected by an inlet (3) (centre section of LH boundary layer bleed) and is discharged through outlet (4) at frame 14.

b. *Equipment Air Conditioning Heat Exchanger.* Mounted on the RH side, the cooling system of this exchanger is symmetrical with the previous one.

9178. **Air-To-Air Heat Exchangers — III D Only (refer to Fig 9-63)** The air supplying the heat-exchanger of the cockpit and nose cone (TACAN) air conditioning system is obtained from an air inlet (3) located at the front of the nose cone. It is discharged to the outside through an outlet (4) situated at the bottom of the nose cone (frame VI).

9179. **Accessory Gear Box, Alternator and Generator Cooling (refer to Figs 9-64 and 9-65).** The door (35) which closes the accessory gear box housing between frames 24 and 26 is provided with two louvres (36) for cooling of the accessory gear box.

9180. The alternator (1V) is completely enclosed in a casing. Connected to this casing is a cooling duct (37) which is closed in its centre section and forms both a cooling air inlet and outlet. This duct is located on the alternator housing door, between frames 24 and 26, on the RH side.

9181. The generator (1P) uses the same cooling principle as the alternator. Its casing is connected to two air inlet and exhaust scoops (40) located on the generator housing door (41), between frames 24 and 26, on the LH side.

ENGINE CONTROLS AND INSTRUMENTS

Power Control (refer to Figs 9-70 to 9-74)

9182. **Front Cockpit.** The power control consists of :

a. A power control quadrant (1) located on the LH console.

b. A flexible control shaft (2) connecting the power control quadrant (1) to the single control box (3) on the engine. The flexible control shaft (2) is fitted with fork-ends (4), nuts (5) and an eye-bolt (6) on the engine side and a swivel fitting (7) on the power control quadrant side. The flexible control shaft is sealed by a vinyl sheath (8) attached by a clamp (9) and its passage through the cockpit bulkhead is sealed by means of a mounting flange (10) with a threaded cap (11) together with a seal (12) fitted in between a shim (13) and a washer (14). The swivel fitting (7) is clamped to the aircraft structure by a nut and a checknut (15) on a gusset (16). The fork-ends are assembled by means of threaded pins, nuts, washers and split-pins (17).

9183. **Front Cockpit Power Control Quadrant.** The power control quadrant consists of :

a. A two-piece casing (18).

b. A lever (19) including :

(1) A control lever (20)

(2) A hand grip (21)

9184. The following items are located at the front of the casing :

a. The rocker (22) to override the dry engine idle stop (23).

b. The in-flight RELIGHT switch (24).

c. The APPR. CONT. (approach control) manual control lever (25).

d. The throttle control lever friction lock.

9185. The following items are located inside the casing :

a. The adjustable maximum afterburner stop (27)

b. The dry engine idle stop (23).

c. The adjustable OFF lock (28).

d. The dry engine (29) and afterburner (30) double quadrant.

e. The contactor box (31) including the U/C not down, double brake pressure, afterburner ignition, and engine overspeed microswitches.

f. The approach control contactor (26K).

g. The control lever (20) attached to the casing (18) by a shouldered pin (33).

9186. The control lever carries the following items :

a. The hand grip (21) hinged about a pin (34).

b. A hook (35) maintained by a spring (36) (approach control automatic cut-off system).

c. A rocker (37) fitted with a roller (38).

d. A set of link-rods (39) connecting the shaft of the hand grip to the rocker and to a notched bell-crank (40). A link-rod

(41) controlling the contactor box (31) is connected to the notched bell-crank (40).

e. A ball and spring assembly (42) stiffening the tilting motion of the hand grip.

f. A ball-joint (51) in the centre of the lever (for connection to rear cockpit control).

g. A ball-joint (43) at the lower part of the lever (engine control).

9187. The hand grip carries the following items :

a. The airbrake control switch (36C)

b. The microphone press-to-talk button (10R).

c. The sight gyro caging button (27A).

9188. The airbrake rocker (44) operates the contactor (46) through the notched lever (47). This lever is fitted with a locking finger (48). The finger is maintained in the locked position by a spring and a ball (50) on the lever.

9189. **Front Cockpit Adjustments.** There are two adjustment points as follows :

CAUTION

The fork-ends should not be unscrewed beyond the adjustment hole (53) (refer to Fig 9-71).

a. *Adjustment of the Flexible Control Shaft Travel.* The flexible control shaft travel is adjusted by turning the fork-ends (4) as required on the shaft (2).

b. *Adjustment of Throttle Control Lever Friction.* Throttle control lever (19) friction is increased by turning the friction lock (26) on the pin (33). This compresses the friction plates (51) located between the flanges (52) on either side of the lever (20).

9190. **Rear Cockpit (refer to Fig 9-71).** The power control consists of a power control quadrant (51) located on the LH console. This quadrant is connected to the front cockpit power control quadrant levers by rods (52) fitted with fork-ends that are connected by means of threaded pins, nuts, washers and split-pins.

9191. **Rear Cockpit Power Control Quadrant.** The rear cockpit power control quadrant is mechanically linked to the front cockpit quadrant; it permits the dry engine thrust and ignition and the afterburner to be controlled from the rear cockpit. It consists of a casing (1) and a lever including a control lever (2) and a hand grip (3). The lever is fitted at its lower part with a ball-joint (4) for connection to the front cockpit power control quadrant.

9192. The tilting motion of the hand grip (3) is transmitted by a link-rod (5) to a lever (6) the pivot pin of which is concentric with that of the lever (2) tipped by a fork-end (for connection to the front cockpit power control quadrant). The two levers (2 and 6) are hinged about a shouldered pin (7) clamped by a nut (8) between the flanges (9 and 10) of the casing. A washer (11), adjusted on assembly, is provided to stiffen the joint.

9193. The front and rear cockpit power control quadrants are interconnected by means of rods. These rods have their adjustable fork-ends connected to ball-joints on the throttle control levers. The fork-ends are connected to the ball-joints by means of pinned pins. The two cockpit controls are synchronized through adjustment of the fork-ends. The hand grip is fitted with an airbrake control switch (236C).

9194. **Power Controls — Operation (refer to Fig 9-74).** For operation of the power controls, refer to Table 9-9

Self-Contained Starter (refer to Figs 9-75 and 9-76)

9195. The self-contained starter fitted to the ATAR 9C engine is a gas generator which uses the aircraft power sources, ie, fuel and electric power. It is used for driving the engine up to a speed high enough to allow ignition.

9196. The self-contained starter consists of a casing containing the following main components :

a. An electric starting motor (46K) and its step-up gear (1).

b. A centrifugal compressor (2) coupled to the step-up gear.

c. An annular combustion chamber (3).

d. A first turbine (4) integral with the compressor shaft.

e. A second turbine (5) driving the reduction gear (6).

f. A mechanical coupling system (7) connected to the engine rotor.

9197. The main items of equipment required for operation of the self-contained starter are as follows :

a. Starting box (4K).

b. Fuel electro-valve (45K).

c. Electro-pump (44K).

d. Igniter (47K).

e. Fuel control unit (43K) (including pressure switches (43K1) and (43K2).

9198. **Principle of Operation (refer to Fig 9-76).** The compressor (2) draws in air at ambient pressure P1 and compresses this air up to pressure P2. The P2 air is directed to the combustion chamber (3) where fuel is injected and ignited. This causes the gas temperature to rise and the gases to expand and drive the first turbine (4). The developed power :

a. Drives the first turbine (4) integral with the compressor shaft, which provides for continuity of the cycle.

b. Drives the second turbine (5); this arrangement constitutes a pneumatic coupling. The second turbine then drives the engine rotor through the reduction gear (6) and the mechanical coupling system (7).

2199. Starting the turbo-starter requires the following :

a. A generator starting device (electric motor) (46K).

b. A fuel sypply system.

c. An air-fuel mixture ignition system.

d. A starting box (4K) that controls the various fuel and electric power supply sequence. It contains the 7-second and 14-second delay relays (4K4) and (4K3) controlling the sequences of operations. These relays also act as safety devices during the different starting phases.

Starting the Turbo-Starter

9200. The turbo-starter can be operate from the front cockpit only.

9201. **Phase 1.** On starting, the position of the controls is as follows :

a. Throttle lever — OFF.

b. RELIGHT switch (8K) in the normal position.

c. IGN-VENTIL switch (9K) in IGN position.

9202. Depressing the starting pushbutton (7K) energizes the self-holding relay (6K); this energizes relay

Table 9-9 Power Control Operation

Action	Fig 9-74 Detail No	Detailed Operation
The engine rpm is selected by moving the power lever (19).	1	Once the engine starting operations have been completed, the throttle control lever (19) is placed in the dry engine IDLE position. To place the lever in the IDLE position, the lever is pushed forward to be unlocked from the OFF position, and then further forward to the IDLE stop (23).
	2	The power is increased by pushing the throttle control lever forward, which drives the flexible control shaft (2) connected to the single control box on the engine. As it is moved over the DRY ENGINE power quadrant (29), the throttle lever operates various microswitches : the U/C not down microswitch (31G) at the mid-travel position approximately, the double brake pressure microswitch (30G) at the 3/4 travel position approximately. The DRY ENGINE power quadrant is limited (full power position) by the roller (38) contacting the AFTERBURNER power quadrant (30).
	3	The second power quadrant (30) is used in connection with the afterburner (ignition and afterburner power changes). The lever is shifted from the first to the second power quadrant by tilting the hand grip upwards. This tilting motion results in moving the roller (38) and operating the afterburner ignition microswitch (14K). At this point, the lever of the afterburner fuel control unit is in the MINIMUM AFTERBURNER position.
	4	As it is moved over the AFTERBURNER power quadrant (30), the throttle lever operates the lever of the afterburner fuel control unit up to the MAXIMUM AFTERBURNER stop (27) through the flexible control shaft (2). At the 9/10 travel position, the lever operates the engine overspeed microswitch (52K).
	5	The throttle control lever is returned into the OFF position by : a. Retarding the throttle lever to the MINIMUM AFTERBURNER position indicated by the roller (38) contacting the DRY ENGINE power quadrant (29). b. Tilting the hand grip down to cancel the AFTERBURNER. c. Retarding the lever again until reaching the DRY ENGINE IDLE stop (23). d. Depressing the rocker (22) to override the IDLE stop. e. Pulling the lever fully aft to engage the OFF lock (28). The approach speed control system lever (25), which is in the ON position (direction of the arrow on the power control quadrant), is returned to the OFF position by the hook (35) attached to the throttle control lever when the lever is passing through the full dry power position. If desired, the approach speed control system lever may be returned to the ON position.

(4K1) in starting box (4K) and provides power to the following units :

a. Fuel electro-valve (45K).

b. Electro-pump (44K).

c. Energization circuit of the 14-second time-delay relay (4K3).

9203. **Phase 2.** The electro-pump then raises the fuel pressure up to 21.3 lb/in^2 (1470 mb) which results in :

a. Reversal of the fuel pressure switch (43K1).

b. Energization of relay (4K2) through pressure switch (43K2) connected in series in the circuit.

c. De-energization of the 14-second time-delay relay (4K3).

9204. The energized relay (4K2) simultaneously supplies the electric starter motor (46K) and the igniter firing system (ignition unit 47K).

9205. The compressor is driven as well as the turbine (4) mounted on the same shaft. The fuel (whose flow is metered by the control unit 43K) is injected into the combustion chamber and ignites upon contact with the igniter plugs. The thermodynamic cycle is thus initiated and the turbo-starter speed increases up to self-sustaining speed.

9206. **Phase 3.** The P2 pressure increases and reaches 4.3 lb/in^2 (294 mb) which results in reversal of pressure switch (43K2) and de-energization of relay (4K2) causing :

a. De-energization of the electric starter motor (46K).

b. De-energization of the igniter plug firing system (ignition unit 47K).

c. De-energization of the 7-second time-delay relay (4K4).

d. Energization of the relay (4K5).

9207. The closing of relay (4K5) permits power being fed to :

a. The starting by-pass electro-valve.

b. Energization circuit of the 14-second time-delay relay (4K3).

c. The starting electro-valve and igniter (3K) following energization of relay (2K).

9208. **Phase 4.** The increased turbo-starter power drives the second turbine (5) which then drives the engine rotor. The engine rotor speed increases gradually; when it reaches 600 r.p.m., the throttle lever is advanced to the IDLE position. When the engine speed reaches 1900 $\pm$100 r.p.m., the speed detector cuts-in to de-energize relay (4K1), thereby interrupting the feed to :

a. The fuel electro-valve (45K) which closes.

b. The fuel electro-pump (44K) which stops.

c. The self-holding circuit of relay (6K).

The turbo-starter is then disconnected.

9209. In case of a hot start, a thermostat closes the circuit of the fuel pressure switch (43K1) and energizes the P2 pressure switch (43K2). Power is then fed to :

a. Electro-valve (45K).

b. Electro-pump (44K).

c. Electric starter motor (46K).

d. Igniter plug firing system (47K).

Starting Incidents — Action of Time-delay Relays (refer to Fig 9-75)

9210. If the turbo-starter fails to attain self-sustaining speed after starting, the 7-second delay relay (4K4) cuts-out the following units :

a. Electro-valve (45K).

b. Electro-pump (44K).

c. Electric motor (46K).

d. Igniter plug firing system (47K).

The turbo-starter stops driving the engine.

9211. If the turbo-starter fails to become self-sustained and reach 2000 r.p.m. The 14-second time-delay relay (4K3) cuts out the above-mentioned units and the turbo-starter stops.

Ventilation (refer to Fig 9-75)

9212. After a false start, it may be necessary to perform a dry ventilation. The procedure is identical with that for a normal start, except for the following :

a. The IGN-VENTIL switch (9K) is placed in the VENTIL position; this prevents fuel being fed to the injectors and the igniter plugs being energized.

b. The throttle lever is placed in the OFF position.

c. The engine rotational speed must be held between 1200 and 1500 r.p.m. Ventilation will be stopped by time-delay relay (4K3) after 14 seconds

In-Flight Relight (refer to Fig 9-75)

9213. In flight, the engine is relit by placing the RELIGHT switch (8K) in the FLIGHT position. The starting electro-valve and the igniter plugs are then permanently energized and there is no need to depress the starting pushbutton (7K). Once the engine relights, the RELIGHT switch (8K) is returned to the NORMAL position.

Afterburner (refer to Fig 9-77)

9214. The following afterburner controls and indicators are fitted :

a. A/B INJ indicator lights (16K) and for III D (216K).

b. A/B ON indicator lights (15K) and for III D (215K).

c. Ionisation probe amplifier (21K)

d. A/B control switch (13K) and for III D (213K) that controls the electrically operated afterburner cock (18K). These switches are lockwired in the ON position.

e. Microswitch (14K).

f. Afterburner relay (19K) slaved to the air data computer.

9215. As the throttle control lever is tilted upward (minimum afterburner position), the microswitch (14K) closes providing power to the following units :

a. Afterburner electro-valve (20K) which opens and allows the air supply to the turbo-pump. The afterburner fuel system is then supplied.

b. Afterburner ignition electro-valve through the ionization probe amplifier (21K). Fuel is injected.

9216. The A/B INJ indicator lights (16K), and for III D (216K), illuminate due to the circuit being closed by the pressure switch.

9217. Subjected to the afterburner flame, the ionization probe starts feeding the ionization probe amplifier (21K) with information. The amplifier then causes :

a. The afterburner ignition electro-valve circuit to cut-off the afterburner injection sequence.

b. The A/B ON indicator lights (15K), and for III D (215K) to illuminate.

c. The thrust corrector electro-valve to open.

9218. **Afterburner Emergency Ignition System.** In case of failure of the normal system, the current available at the terminals of the A/B EMERG injector push button (17K) enables the afterburner to be ignited by energizing the afterburner ignition electro-valve.

NOTE

The afterburner emergency ignition system can be only operated from the front cockpit.

Normal Fuel Control System (refer to Fig 9-78)

9219. The engine fuel control system enables the engine capabilities to be fully utilized by means of a single lever under all flight conditions. The engine fuel control system uses two parameters; engine rotational speed and gas temperature upstream of the turbine. The engine is provided with a main fuel control unit (dry engine) and an afterburner fuel control unit.

9220. **Main Fuel Control Unit.** The main FCU includes :

a. *An Engine Speed Governor.* The purpose of the speed governor (1) is to hold the engine speed constant in keeping with the setting of the throttle lever. The speed governor unit consists of a flyweight-type governor (3), an oil distributor (slide valve) (4) and a fuel metering piston (5). The slide-valve (4) is positioned at the balance point between two forces; the spring (6) force, which is determined by the position of the throttle lever, and the flyweight (3) force. When one force becomes greater than the other, the slide-valve (4) moves away from its balanced position and its displacement determines the direction of control of the metering piston (5). The metering piston alters the rate of fuel flow to the engine to maintain the desired r.p.m. as determined by the throttle lever setting.

b. *Temperature Regulator.* The purpose of the temperature regulator 91) is to maintain the optimum engine operating temperature corresponding to the selected r.p.m. It varies the jet pipe exhaust area by acting on the nozzle flaps (10) through oil actuated jacks (9). A capsule (7) is subjected to the P2 pressure (which varies as a function of the temperature) from the engine. The movements of this capsule are transmitted to a distributing slide-valve (8). The position of the slide-valve controls the oil slaving system operating the actuators (9) of the nozzle flaps (10). Depending on the slide-valve position, the nozzle flaps open or close.

c. *Connection Between Speed Governor and Temperature Regulator.* The P2 pressure variations resulting from the changes in the position of the nozzle flaps result in r.p.m. changes. To maintain a correct fuel flow, taking these changes into account, it is necessary that a correction be made at the fuel metering piston (35) of the main fuel control unit. This correction is obtained through a lever (14) which provides a mechanical link between the temperature regulator (2) and the speed governor (1).

9221. **Afterburner Fuel Control Unit (refer to Fig 9-78).** The afterburner FCU consists of a capsule (11), a fuel metering piston (12) and a differential linkage (13). The throttle lever movements change the position of the metering piston (12), thereby determining the fuel flow to the afterburner manifolds. When P2 varies, the capsule (11) moves the metering piston linkage (13) and thus varies the fuel flow. The function of the differential linkage (13) is to establish a balanced condition between the metering unit and the capsule for all A/B positions of the throttle lever.

Engine Overspeed System (refer to Figs 9-78, 9-79 and 9-81)

9222. This system can be controlled either automatically or manually from the front cockpit only. It permits the engine speed to be increased by 250 r.p.m. by increasing the calibration value of the speed governor either :

a. Manually on take-off to shorten the take-off roll (when the outside temperature is more than or equal to 27°C

b. Automatically when the altitude is more than 30,000 ft and the total temperature is more than or equal to 27°C.

9223. **Description.** The engine overspeed system consists of :

a. An OVERSPEED presetting button (53K) under a lockwired guard cover on the RH side of the cockpit instrument panel in the front cockpit only.

b. An overspeed contact (52K) in the power control quadrant.

c. An OVERSPEED amber indicator light

(54K) and for III D (254K) on the instrument panel.

d. An overspeed electro-valve on the engine.

e. A hydraulic actuator (26) on the speed governor.

9224. **Operation.** The system operates as follows

a. *Manual Operation* When the outside air temperature on the ground is more than or equal to 27°C, a switch located in the air data computer (1C) permits power at the terminals of the pushbutton (53K); depressing this button applies power to contactor (52K). When the throttle control lever reaches the maximum afterburner position, contactor (52K) is closed. The time-delay relay (51K) is energized and allows the over speed electro-valve to be supplied; this supply is limited to one minute. The servo-control oil then moves the small piston of the actuator (26) which acts on the spring (6) of the speed governor, thereby increasing the calibration of the latter and increasing the engine speed up to the overspeed limit (8710 ±50 r.p.m.). A valve (25) mechanically actuated by the throttle lever cuts-off the oil supply to the piston when the throttle lever is retarded below 8150 r.p.m.

b. *Automatic Operation.* The requirements for this mode of operation to directly supply the overspeed electro-valve are as follows :

(1) Altitude higher than or equal to 30000 ft; contactor closed in output multiplier (2C).

(2) Temperature more than or equal to 27°C; contactor closed in air data computer (1C).

9225. Regardless of the operating mode, the overspeed indicator light (54K) and for III D (254K), remains on as long as the overspeed system is operated. On starting, the relay (19K) is open and the thrust corrector electro-valve is no longer supplied.

Emergency Fuel Control System and Emergency Oil System (refer to Fig 9-80)

9226. The previously described control systems use hydraulic power only for command signal generation, transmission and actuation. An oil system failure will therefore be a serious failure which is partially compensated for by the emergency fuel control system and the emergency oil system.

9227. **Purpose.** The purpose of these systems is to provide enough time to reach an alternate airfield and to prevent destruction of the engine due to lack of No 1 bearing lubrication and/or engine fuel control failure resulting in over speeding or thrust loss.

9228. The installation consists of two systems — an electrically-operated emergency fuel control system and an emergency oil system for separate lubrication of the No 1 bearing (lost oil).

9229. **Emergency Fuel Control System** The system electrically controls the fuel metering piston in the event of failure of the hydraulic servo system. The emergency fuel control system includes the following :

a. In the cockpit :

(1) An EMERG NOZZLE control switch (27K) (with a guard) on the LH console.

(2) An EMERGENCY THROTTLE control (28K for III O) (40K for III D) for control of the fuel metering piston. This control is on the LH side console in the front cockpit only.

b. On the main fuel control unit :

(1) A mechanical fuel metering piston actuator controlled by an electric motor (27).

(2) An anti-overspeed spring (28) used to prevent engine overspeed in case of oil system failure and to return the metering piston to the minimum flow position.

9230. **Operation of the Emergency Fuel Control System.** Closing the EMERG NOZZLE control switch (27K) results in opening of the P2 discharge electro-valve (29) and energization of the EMERGENCY THROTTLE actuator control key (28K for III O) (40K for III D). The opening of the discharge solenoid valve (29) results in decreasing the exhaust nozzle flap servo pressure. This allows the exhaust nozzle flaps to fully open, thereby preventing thrust surges.

9231. Actuating the EMERGENCY THROTTLE control forward or aft operates the electric actuator (27) acting on the fuel metering piston (35). Depending on the actuating direction, the fuel flow is thus increased or decreased and enables engine power to be controlled until the aircraft has been landed.

NOTE

Maximum engine r.p.m. not to exceed 9000.

9232. **Emergency Oil System.** The fully independent emergency oil system provides lubrication for the No 1 engine bearing. Lubrication is ensured until the emergency oil tank is exhausted. The maximum operating time is 20 minutes. The emergency oil system includes :

a. In the cockpit, an EMERG NOZZLE control switch (27K) fitted with a guard.

b. On the engine :

(1) An oil tank (30), 0.77 U.K. gal (3.5 l) capacity.

(2) An electro-pump (31).

(3) A check valve (32).

(4) A filter (33).

(5) A No 1 bearing oil jet (34).

9233. **Operation of the Emergency Oil System.** Closing the EMERG NOZZLE control switch (27K) starts the electro-pump (31) which sucks oil from the emergency oil tank (30) for lubrication of the engine No 1 bearing. A check valve (32) and a filter (33) are mounted in the electro-pump discharge system. The oil reaches the No 1 bearing through the jet (34).

NOTE

Depending on nature of the failure, the throttle control lever must be placed in the IDLE position for oil pressure failure or the full dry power position for other failures. In no case shall the throttle control lever be placed in the STOP (III O) or OFF (III D) position : such an action would completely shut off the engine fuel supply.

Approach Speed Control System (refer to Fig 9-79)

9234. The approach speed control system controls the flying speed during particular landing conditions (eg GCA). For any change in the aircraft angle of incidence, this system automatically corrects the thrust so that a given flying speed is maintained. The approach control system holds the indicated airspeed (IAS) constant by varying the engine thrust through varying the exhaust nozzle cross-sectional area (flaps) while the engine speed remains constant. The system includes :

a. On the power quadrant (front cockpit only) :

(1) An approach control lever (15).

(2) A microswitch (26K) operated by the lever (15).

b. On the fuselage :

(1) A total pressure inlet (curved pitot-tube (16) between frames 1 and 2).

(2) A static pressure inlet (17) at frame 38.

c. On the engine :

(1) A three-way solenoid valve (18).

(2) An approach control unit (19).

9235. **Operation** With the throttle lever in a position corresponding to an engine speed between 7300 and 8100 r.p.m., lifting the APPR. CONT. lever (15) causes microswitch (26K) to close and energise the three-way solenoid valve (18). As a result, a P2 pressure (depending on IAS), modulated by the approach control unit (19), is substituted for the normal P2 pressure. This modulation is obtained by means of an orifice (20) whose area is varied by a needle (21) integral with a diaphragm (22). The diaphragm is subjected to static pressure on one side and to total pressure on the other side, therefore, its displacements are governed by IAS. The variations in P2 pressure are transmitted to the capsules of the temperature regulator (7); these capsules control either the opening or the closing of the exhaust nozzle flaps (10) to maintain a thrust depending on the IAS.

NOTE

The pilot can recover full control over the engine speed at any time during the approach by advancing the throttle past 8100 r.p.m; the approach control system lever will then return into its OFF position and microswitch (26K) will open.

Engine Instruments (refer to Fig 9-82)

9236. **Engine Speed Indicating System.** The engine r.p.m. is detected by a tachometer-generator delivering a current proportional to the engine r.p.m. This current is applied to the r.p.m. indicator (1E) and for III D (201E).

9237. **Fuel Low Pressure Warning System.** A pressure switch (2E) is subjected to the engine fuel inlet pressure. Whenever the fuel pressure drops below the calibration value of the pressure switch, the switch closes the circuit to the FUEL warning light which illuminates on the failure warning panel (1Z) and for III D (201Z).

9238. **Oil Pressure Warning System.** A pressure switch is subjected to the engine oil pump delivery pressure. This pressure switch is connected to the OIL warning light on the failure warning panel (1Z) and for III D (201Z) and to the audio warning horn (7Z).

9239. **Jet-pipe Temperature Indicating System.** The exhaust gas temperature is detected by four dual thermocouples (8E). Each thermocouple is connected to two connecting boxes (7E) and (9E), the latter box being used for checking purposes. From 7E the thermocouples are connected to a terminal board (5E) and for III D (205E) by special leads and to the jet-pipe (T5) temperature indicator (3E) and for III D (203 E) through an adjustment box (4E) and for III D (204 E) provided to balance the line resistance.

Thermocouples (refer to Figs 9-82 and 9-83)

9240. The jet-pipe temperature thermocouples (1) are of the chromel-alumel type. They are designed to be used with temperature indicators capable of reading temperatures as high as 800°C. The conductors are insulated with asbestos and glass-silk and then painted and are permanently secured to each thermocouple. The mechanical protection consists of a flexible metal tube (4) attached to the thermocouple. Each conductor may be identified by a coloured sleeve at the free end of :

a. The red wire — Chromel, positive (2)

b. The blue wire — Alumel, negative (3).

9241. **Installation.** There are four dual thermocouples attached to threaded bushings (6) made integral with the engine by anchor nuts (7) on the probes. Sealing is ensured by tightening the conical parts (5). The thermocouples are connected to the aircraft system by connecting the leads to the connecting boxes (9E and 7E) by means of the terminal lugs fitted to the leads. The terminal board (5E) and for III D (205E) and the adjustment box (4E) and for III D (204E) are mounted by means of screws in the lower lateral bay, on the RH side at frame 35.

FIRE DETECTION SYSTEM

General (refer to Figs 9-84 to 9-86)

9242. The following system is provided only for fire detection; no fire extinguishing system is provided. The fire detection system covers the turbo-jet engine and in III O, the system also covers the rocket motor.

9243. There are two fire detection sections on the engine and for III O, one on the rocket motor, with fire detectors distributed as follows :

a. *Cold Zone.* Nine fire detectors calibrated at 200°C

b. *Hot Zone.* Nine fire detectors calibrated at 300°C.

c. *For III O Only.* Two fire detectors calibrated at 300°C are located on the rocket motor.

9244. Two red warning lights are provided on the instrument panel to indicate abnormal overheating. A third red warning light for the rocket motor fire detection system is provided on the instrument panel in III O. All the warning lights have provision for testing.

Description

9245. There are two fire detection sections on the engine.

a. *Cold Zone.* Within this zone there are nine fire detectors (3J) calibrated at 200°C to detect any fire hazard at the fuel pumps and connections. Four detectors are located at the lower part of frame 25 and two are located at the upper part. Two detectors are located between frames 24 and 25 and one between frame 27 and frame 28.

b. *Hot Zone.* Afterburner jet pipe. There are two fire detector rings within this zone. One is located at frame 34 and includes six fire detectors (5J) calibrated at 300°C. The other is located at frame 38 (tail pipe fairing) and includes three fire detectors (6J) also calibrated at 300°C.

9246. The fire detectors are connected to the warning lights through a control box (2J) as follows :

a. Front cockpit :

(1) ENG FIRE warning light (4J)

(2) A/B FIRE warning light (7J)

(3) III O Only, FIRE ROCK MOTOR warning light (38K)

b. Rear cockpit.

(1) ENG FIRE warning light (204J)

(2) A/B FIRE warning light (207J).

Operation (refer to Figs 9-84 to 9-86)

9247. As soon as electrical power is available, the control box (2J) is energised through circuit breaker (1J). Relays (2J1) and (2J2) are energised, the earth return being via the series connected fire detectors of each zone. When energised, relays (2J1) and (2J2) open-circuit the fire warning lights.

9248. A fire or overheat condition is sensed by one or more of the fire detectors which opens and breaks the energisation circuit to the corresponding relay thus causing the appropriate warning light to illuminate.

9249. Each set of detectors can be tested by depressing the corresponding fire warning light. Correct operation causes the appropriate FIRE light to illuminate.

9250. For III O Only, the rocket motor detector circuit is independent and has no earth return unless the rocket motor is installed. The warning light (38K) is inoperative if the rocket motor is not installed.

CHAPTER 10

HYDRAULIC INSTALLATION

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Figures for this chapter are contained inDI(AF) AAP 7213.003-2-14, Book 4

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CHAPTER 10

HYDRUALIC INSTALLATION

Table 10-1 Function of Components

Index No	System	Description	Characteristics and Functions
4C	2	Pressure transmitter	Detects the pressure in the system downstream of the servo-control accumulator and electrically feeds the dual hydraulic pressure indicator (9C) with pressure information.
5C	1	Pressure transmitter	See 4C.
8C 208C	1 and 2	Selector switch 208C — III D Only	Enables pressure selection on the dual hydraulic pressure gauge (9C).
9C 209C	1 — 2 and Emerg.	Dual hydraulic pressure gauge 209C — III D Only	This gauge includes two scales graduated from 0 to 5000 lb/in² (0 to 300 bars). It provides the following information with the selector (8C) switch on : *SERVO* LH — No. 1 system pressure RH — No. 2 system pressure *U/C BRAKE* LH — Undercarriage and brake system pressure RH — Emergency brake system pressure.
11C	1	Control box	Causes the electro-valve (13C) to close when it is energized by the capacitor (12C).
12C	1	Capacitor	Electrically supplied through circuit breaker 10C, this capacitor generates an electrical signal when the capacity usable by the No. 1 system has dropped to 2.4 U.K. gal (10.9 l) in the reservoir (12).
13C	1	Electrically operated valve	Enables the air brake, undercarriage and wheel brake systems to be isolated when controlled by 11C.
14C	Emerg.	Electro-pump circuit breaker	Supplies power to electro-pump (18C) via pressure switch (15C).
15C	Emerg.	Pressure switch	This switch controls energization of the electro-pump power relay (16C). *Energization pressure* :1778 ± 43 lb/in² (122.5 ± 3 bars) *De-energization pressure* :2133 ± 70 lb/in² (147 ± 5 bars)
16C	Emerg.	Electric pump relay	This is controlled by the pressure switch (15C); it controls the electric power supply to the electric pump (18C).
18C	Emerg.	Electric pump	Delivers a rated pressure of 2175 lb/in² (150 bars) in the system with a maximum output of 1.32 U.K. gal/min (6 l/min).
27C	1	Pressure switch	Causes the warning light come on when the pressure is less than 1991 ± 43 lb/in² (137 ± 3 bars) and to go out when the pressure is more than 2347 ± 70 lb/in² (161.7 ± 5 bars).
28C	2	Pressure switch	See 27C.
29C	Emerg.	Pressure switch	Causes the warning light to come on when the pressure is less than 1276 ± 43 lb/in² (88 ± 3 bars) and to go out when the pressure is more than 1636 ± 70 lb/in² (112.7 ± 5bars).
on1Z 201Z	Emerg	EMG. HYD warning light (red) 201Z — III D only	Operated by pressure switch (29C)
	1	HYD. 1 warning light (red)	Operated in connection with pressure switch (27C).
	2	HYD. 2 warning light (red)	Operated in connection with pressure switch (28C).
1	2	Reservoir	This reservoir provides a 2.86 U.K. gal (13 litres) FHS hydraulic fluid reserve. It is pressurized and is fitted with a drain cock (29).
2	2	Low-pressure filter	Filters the hydraulic fluid sucked by the pump. Filtering capacity : 6 to 15 microns.
3	2	Ground connection with self-sealing valve.	Suction port for test-bench (ground testing).

Index No.	System	Description	Characteristics and Function
4	2	Self-regulating pump	Delivers a rated pressure of 3046 lb/in² (210 bars) in the system; maximum output : type A2-5400-11 : 9.85 U.K. gal/min (45 l/min) or Type A1-24700-0 : 9.46 U.K. gal/min (43 l/min).
5	2	Heat-exchanger	This is used to cool the hydraulic fluid returned from the pump regulation system; engine fuel is used as a cooling agent.
6	2	Restrictor	Protects the pressure-switch against rapid pressure variations.
7	2	Ground connection with self-sealing valve.	Pressure port for test-bench (ground testing).
8	2	Non-return valve	This valve isolate the hydraulic power generation system upstream of the servo-control accumulator (9) in case of failure of the system (ruptured pipe or pump drive shaft).
9	2	Servo-control accumulator	Charged with nitrogen to 1421 lb/in² (98 bars). Because of its 0.48 U.K. gal (2.2 l) capacity, it dampens the pressure variations and constitutes a power reserve only for the flight controls.
10	2	Pressure relief valve	This valve operates when the pressure is equal to or more than 3568 lb/in² (246 bars).
11	2	Servo-control filter	Filtering capacity : 10 microns.
12	1 and Emerg.	Reservoir	Capacity : 3.96 U.K. gal (18 l); this reservoir is pressurized by the same system as the No. 2 system reservoir (1). It includes a compartment providing a 0.88 U.K. gal (4 l) fluid reserve for the emergency system. It is fitted with a capacitor (12C) and two drain cocks (30 and 31).
13	1	See 2	
14	1	See 3	
15	1	see 4	
16	1	See 5	
17	1	See 6	
18	1	See 7	
19	1	See 8	
20	1	See 9	
21	1	See 10	
22	1	See 11	
23	Emerg.	See 2	
25	Emerg.	See 8	
26	Emerg.	Buffer accumulator	Charged with nitrogen at 1065.75 ± 43 lb/in² (73.5 ± 3 bars), this accumulator, due to its 0.15 U.K. gal (0.7 l) capacity, dampens the pump surge and maintains the emergency system under pressure during normal operation.
27	Emerg.	Pressure relief valve	This valve operates and returns the fluid to the reservoir when the pressure is equal to or more than 2560 lb/in² (176.5 bars).
28	Emerg.	Non-return valve	Prevents the No. 1 system fluid from flowing into the emergency system.
29	2	Drain cock	Enables the No. 2 system reservoir (1) to be drained.
30	1	Drain cock	Enables the Normal compartment of the No. 1 system reservoir (12) to be drained.

Index No.	System	Description	Characteristics and Functions
31	Emerg.	Drain cock	Enables the Emergency compartment of the No. 1 system reservoir (12) to be drained.
32	Pressurization	Non-return valve	Isolate the power generation system to permit ground pressurization tests to be performed with the engine stopped.
33	Pressurization	Pressure reducing valve	Regulates the pressure of the reservoir pressurizing air at 11.31 ± 0.58 1.01 lb/in² (0.780 ± 0.04 0.07 bar).
34	Pressurization	Test connection	Enables the reservoir pressurization tests to be performed on the ground.
35	1 and 2	Non-return valve	Prevents hydraulic fluid leaks in case of rupture of a return pipe from the following utility systems: Airbrake system, undercarriage system, wheel brake system (No. 1 system) and undercarriage emergency system (No. 2 system). Enables the accessories to be removed without draining the No. 1 system reservoir.
36	2	Anti-vibration accumulator	Charged with nitrogen at 1421 lb/in² (98 bars), this accumulator dampens the fluid surges created by the pump. Capacity :0.017 U.K. gal (0.078 l).
37	1	See 36	
38	1	Mechanically operated shut-off valve	OPEN : Normal operation. CLOSED : Allows only the undercarriage, wheel brake and airbrake utility systems to be supplied. Permits the undercarriage locking pressures to be read on the ground. Located in generator housing between frames 25 and 26 on the LH side.
39	1	Filter	Filtering capacity : 66 microns. Protects the utility system.
40	2	Undercarriage emergency system filter	Prevents contamination of the returned No. 1 system fluid in case of emergency undercarriage extension. Filtering capacity : 10 microns.
52 53	1	Filter	Protects the inboard control surface servo-controls
57	2	Filter	Protects the undercarriage, wheel brake and airbrake systems. Filtering capacity : 15 microns.

GENERAL

1001. This chapter only deals with the hydraulic power generating system. The hydraulic installation can be broken down into three separate systems as follows :

a. *No. 1 System.* This system is supplied by an engine driven pump which feeds a rated pressure of 2988 lb/in² (206 bars) to :

(1) Pre-servos on pitch and roll linkages.

(2) Elevon servo-controls.

(3) Rudder servo-controls.

(4) Undercarriage jacks.

(5) Wheel brakes.

(6) Airbrake jacks.

b. *No. 2 System.* This system is supplied by a pump driven by the accessory gear box; this pump feeds a rated pressure of 2988 lb/in² (206 bars) to :

(1) Pre-servos on pitch and roll linkages.

(2) Elevon servo-controls.

(3) Inboard flap servo-controls.

(4) Rudder servo-control.

(5) Undercarriage jacks (emergency lowering).

c. *Emergency System.* The No. 1 reservoir includes a reserve of fluid on which the emergency system electric pump operates. This pump will deliver a pressure rated at 2175 lb/in² (150 bars) in the event of failure of the No. 1 system. In an emergency, the pump provides for operation of :

(1) Pre-servos on pitch and roll linkages.

(2) Elevon servo-controls.

(3) Rudder servo-control.

The electric pump is controlled by a pressure switch and automatically comes into operation when the pressure in the No. 1 system drops below 1778 ± 43 lb/in² (122.5 ± 3 bars) and stops delivering into the system when the pressure rises again to above 2133 ± 70 lb/in^2 (147 ± 5 bars).

Description (refer to Figs 10-2 and 10-3)

1002. The No. 1 and No. 2 systems each include the following units :

a. Low-pressure filters (2 and 13).

b. Two suction (3 and 14) and delivery (7 and 18) connections with self-sealing valves, for connecting a ground rig with the pump by-passed.

c. Fixed-plate self-regulating pumps (4 and 15).

d. Nitrogen inflated anti-vibration accumulators (36 and 37).

e. Nitrogen inflated buffer accumulators (9 and 20). Capacity : 0.45 U.K. gal (2.04 l).

f. Non-return valves (8 and 19) ensuring that fluid is retained in the accumulator for use in the event of failure of the generating system.

g. Pressure relief valves (10 and 21).

h. Porous bronze filters (11 and 22) for protecting the servo-control circuits.

i. In the No. 1 system, a shut-off valve (38) located in the LH main wheel well that enables undercarriage tests to be performed.

j. A signalling system including transmitters (4C and 5C) and pressure switches (28C and 27C) protected by restrictors (6 and 17).

NOTE

When the engine is removed, the pipes of pump (15) are not disconnected. This pump is uncoupled from the engine and stowed on a rest plate integral with the aircraft structure as shown in Fig 10-2.

1003. The following units are common to the No. 1 and No. 2 systems :

a. The dual heat-exchanger which includes two independent sections (5 and 16). The hydraulic fluid returned from the pump regulating system is cooled through the heat exchange with the engine fuel.

b. The pressurization system of the reservoirs (1 and 12). The air bled from the engine (tapping Q) is passed through a non-return valve (32) and a pressure reducing valve (33) calibrated at 11.31 +0.58 −0.01 lb/in^2 (0.780 +0.04 −0.07 bars).

c. A pressurization connection (34) enables the reservoirs to be pressurized on the ground with the engine stopped.

d. The failure warning panel (1Z) and for III D (201Z), includes the HYD.1, HYD.2 and EMG.HYD. failure warning lights.

1004. **No. 1 System Reservoir (refer to Fig 10-11).** The No. 1 system reservoir (12) includes a normal compartment and an emergency compartment. The return fluid from the ancillaries (undercarriage NORMAL and EMERGENCY control systems, airbrakes and wheel brakes) flows into the emergency compartment through a non-return valve (35). Cocks (30 and 31) permit the draining of the normal and emergency compartments respectively. The reservoir incorporates a capacitor (12C).

1005. **Electrically Operated Valve (refer to Fig 10-2).** The electrically operated valve (13C), which is controlled by the capacitor (12C), isolates the airbrake system, undercarriage system and wheel brake system whenever the fluid quantity falls below 2.4 U.K. gal (10.9 l) to enable priority to be given to the servo-control system.

1006. **No. 2 System Reservoir (refer to Fig 10-12).** The No. 2 system reservoir (1) contains the fluid required to operate the components of the No. 2 hydraulic system. The reservoir can be drained by means of cock (29).

1007. **Filters (refer to Fig 10-13).** The undercarriage, wheel brake and airbrake systems (No. 1 hydraulic system) are protected by a common filter (57). The No. 2 hydraulic system is provided with a filter (40) intended to prevent contamination of the No. 1 system in case of emergency undercarriage operation.

1008. **Emergency System (refer to Figs 10-2 and 10-3).** Emergency system consists of the following units :

a. A low-pressure filter (23).

b. An electrically driven pump (18C).

c. An accumulator (26) charged with nitrogen.

d. A pressure relief valve (27).

e. A non-return valve (28) preventing fluid flow from the No. 1 system into the emergency system.

f. A pressure switch (15C) controlling the electrical power supply to the pump (18C) through a relay (16C).

g. A non-return valve (25) ensuring that fluid is retained in the accumulator for use in the event of failure of the emergency generating system.

h. A pressure switch (29C) used with the pressure indicating system.

Operation

1009. **No. 1 And No. 2 Systems (refer to Figs 10-3 and 10-4).** The following description holds for both hydraulic systems. The engine driven pumps (15 and 4) draw fluid from the reservoirs (12 and 1) through filters (13 and 2). The excess fluid from the pump regulating system is returned to the reservoir through the heat exchangers (16 and 5). The fluid under pressure is then directed to the ancillary systems.

1010. The ground connections (14 and 3) and (18 and 7) with self-sealing valves, enable a ground test rig to be connected up for pressurizing the systems with the engine stopped. The ground rig pump draws fluid through (14 and 3) and delivers through (18 and 7). A manually operated valve (38) enables the undercarriage system to be isolated from the servo-control system for the purpose of measuring the undercarriage locking and unlocking pressures. In the event of failure of the regulating system, the pressure relief valves (21 and 10) allow fluid flow into the return lines if the pressure exceeds 3568 lb/in^2 (246 bars).

1011. **Emergency System (refer to Fig 10-4).** With the emergency system accumulator discharged, the electrically driven pump (18C) is energized as soon as power from the GPU is fed through the external power receptacle. This pump draws fluid from the emergency compartment of reservoir (12) and delivers it into the system. When the pressure in the system reaches 2133 ± 70 lb/in^2 (147 ± 5 bars), pressure switch (15C) cuts off the power supply to the pump through relay (16C). The pump is energized again at 1778 ± 43 lb/in^2 (122.5 ± 3 bars).

NOTES

- In case of failure of the pressure switch (15C), the pressure relief valve (27) allows fluid flow into the return line if the pressure exceeds 2560 lb/in^2 (176.5 bars).
- On ground, with the battery switch ON the electrically driven pump circuit-breaker (EMG.HYD.) must not be closed.

TYPICAL GENERATING SYSTEM FAILURES

No. 1 System Failure (refer to Fig 10-5)

1012. In case of a pressure drop in the No. 1 system :

a. The pressure switch (15C) starts the electric pump (18C).

b. The emergency system substitutes for the No. 1 system in supplying hydraulic power to the following :

(1) Pitch and roll pre-servos.

(2) Elevon servo-controls.

(3) Rudder servo-control.

c. The No. 1 system pressure gauge indicates the emergency system pressure.

d. The pressure switch (27C) causes the HYD 1 warning light to illuminate on the failure warning panel (1Z) and for III D (201Z). The audio warning device does not operate.

1013. With a No. 1 system failure the following conditions exist :

a. Systems still 100% operative :

(1) Emergency undercarriage control system.

(2) Inboard control surfaces.

b. Systems affected operative at 85% power :

(1) Elevons.

(2) Rudder.

c. Systems inoperative :

(1) Airbrakes.

(2) Normal undercarriage system.

(3) Normal wheel brake system.

NOTE

The aircraft can be braked by the PARK EMG BRAKE handle and the undercarriage extended by the EMERG. U/C control handle.

No. 2 System Failure (refer to Fig 10-6)

1014. In case of pressure drop in the No. 2 system, only the No. 1 system continues to deliver hydraulic power. The pressure switch (28C) causes the HYD. 2 warning light to illuminate on the failure warning panel (1Z) and for III D (201Z) and the audible warning device to operate.

1015. With a No. 2 system failure the following conditions exist :

a. Systems still 100% operative :

(1) Airbrakes.

(2) Normal and emergency wheel brake systems.

(3) Pitch and roll pre-servos.

(4) Oscar device.

b. Systems affected — operative at 50% power :

(1) Elevons.

(2) Rudder.

c. Systems inoperative :

(1) Inboard control surfaces.

(2) Emergency undercarriage control system.

Emergency System Failure (refer to Fig 10-7)

1016. In case of a pressure drop in the emergency system, the pressure switch (29C) causes the EMG.HYD warning light to illuminate on the failure warning panel (1Z) and for III D (201Z) and the audio warning horn to operate. No systems are affected; the purpose of the failure warning system is to warn the pilot that he can no longer use the emergency system.

Level Drop In No. 1 System Reservoir (refer to Fig 10-8)

1017. Because the aircraft cannot be flown with the servo-controls inoperative, it is most essential that these be given prefential treatment in supply of fluid. Therefore, if the amount of fluid in the No. 1 system reservoir (12) falls below 2.4 U.K. gal (10.9 l), the capacitor (12C), which is a level switch, causes the electrically operated valve (13C) to close through the control box (11C) and remove hydraulic power from the following systems :

a. Undercarriage.

b. Wheel brakes.

c. Airbrakes.

NOTE

The aircraft can be braked by the PARK EMG BRAKE handle and the undercarriage extended by the EMERG. U/C control handle.

Simultaneous Failure Of No. 1 and Emergency Systems (refer to Fig 10-9)

1018. Such a failure combination may result from either the complete deterioration of the No. 1 and emergency system reservoir (12) or the rupture of a pipe located after the servo-control accumulator. If this occurs :

a. The pressure switches (27C and 29C) cause the HYD. 1 and EMG HYD. warning lights to illuminate on the failure warning panels.

b. The pressure switch (29C) also causes the horn to operate.

c. The No. 1 system pressure gauge reads zero.

1019. With a simultaneous failure of the No. 1 and emergency systems, the following conditions exist :

a. Systems still 100% operative :

(1) Inboard control surface.

(2) Emergency undercarriage control system.

(3) Emergency brake system (accumulator).

b. Systems affected — operative at 50% power :

(1) Elevons.

(2) Rudder.

c. Systems inoperative :

(1) Undercarriage.

(2) Normal wheel brake system.

(3) Airbrakes.

NOTE

When extended in an emergency, the undercarriage uses fluid from the No. 2 system reservoir (1); this fluid is then returned to the emergency compartment of the No. 1 system reservoir (12). The quantity of fluid delivered during the lowering of the undercarriage may be sufficient to resupply the emergency system for a short period of time.

Simultaneous Failure Of No. 1 And No. 2 Systems (refer to Fig 10-10)

1020. Such a failure combination may result from seizure of the engine and result in :

a. The pressure switches (27C and 28C) causing the HYD.1 and HYD.2 warning lights to illuminate.

b. The pressure switch (28C) also operates the warning horn.

c. The No. 1 system pressure gauge reading the emergency system pressure; the No. 2 system hydraulic pressure gauge reading zero.

1021. With a simultaneous failure of the No. 1 and No. 2 systems there is no system fully operative and the following conditions exist :

a. Systems affected :

(1) Oscar device.

(2) Pre-servos.

b. Systems operative at 35% power :

(1) Elevons.

(2) Rudder.

c. Systems inoperative :

(1) Inboard control surfaces.

(2) Undercarriage.

(3) Emergency undercarriage control system.

(4) Normal wheel brake system.

(5) Airbrakes.

NOTE

It is still possible to brake the aircraft through the emergency brake system. If the failure is due to a seizure of the engine, the alternator and the generator are no longer driven and the time that hydraulic power is available depends on the aircraft battery rundown time, which is very short.

PRESSURE INDICATING SYSTEM (refer to Fig 10-8)

Description

1022. The pressure indicating system consists of two separate systems :

a. Pressure indicating system.

b. Pressure drop warning system.

1023. The pressures can be read on the dual hydraulic pressure gauge (9C) and for III D (209C), located on the instrument panel. The pressure gauge is energized through a dual switch (8C) and for III D (208C) which enables the combinations shown in Table 10-2 to be selected.

Pressure Drop Warning System (refer to Figs 10-9 and 10-10)

1024. The HYD.1, HYD.2 and EMG.HYD. warning lights respectively illuminate on the failure warning panel when the corresponding pressure switches (27C, 28C and 29C) in the No. 1, No. 2 and emergency systems detect a minimum pressure of :

a. 1991 ± 43 lb/in^2 (137 ± 3 bars) for the No. 1 and No. 2 systems.

b. 1276 ± 43 lb/in^2 (88 ± 3 bars) for the emergency system.

1025. In addition to illuminating the associated warning lights, the pressure switches (28C and 29C) of the No. 2 and emergency systems cause the horn to operate. As soon as the pressure in the faulty system rises again to above a given value, the corresponding warning light goes out. This value is :

a. 2347 ± 70 lb/in^2 (161.7 ± 5 bars) for the No. 1 and No. 2 systems.

b. 1636 ± 70 lb/in^2 (112.7 ± 5 bars) for the emergency system.

Table 10-2 Hydraulic Pressure Gauge Readings

Switch Position	Associated Pressure Transmitter	Pressure Gauge Scale	
		LH	RH
Up	4C and 5C	No. 1 system (Servo-controls)	No. 2 system (Servo-controls)
Down	6C and 7C	Undercarriage wheel brakes airbrakes	Emergency brake

Table 10-3 Hydraulic Power Distribution

Generation	Distribution	Supplied Units (*through filters)	Return
No. 1 System		Pre-Servos (through dual-supply valve 111C)*	
		Oscar Valve* Elevon Servo-Controls Rudder Servo-Controls	
		Undercarriage* Airbrakes* Filter (57) Wheel Brakes* Emerg. and Park. Selector Valve	
No. 2 System		Pre-Servos (through dual-supply valve 111C)*	
		Elevon Servo-Controls Inb'd Flap Servo-Controls* Rudder Servo-Controls*	
		Emerg. Selector Valve (U/C)* Filter (40)	
Emerg. System		Pre-Servos (through dual-supply valve 111C)* Oscar Valve* Elevon Servo-Controls Rudder Servo-Controls*	

NOTE

Under overload conditions (simultaneous hydraulic power demands of several ancillary systems) one or more HYD warning lights may come on for a few seconds without implying failure of the system.

1026. In case of failure of the No. 1 system, the pressure gauge reads the pressure in the emergency systems.

HYDRAULIC POWER DISTRIBUTION

Description (refer to Fig 10-15)

1027. As a safety measure, the hydraulic systems are split up into sections each provided with its own accumulator. The units supplied by each system are shown in Table 10-3.

No. 1 And Emergency System Hydraulic Reservoir (refer to Fig 10-11)

1028. The reservoir contains the quantity of fluid required to operate the units and systems which depend on the No. 1 hydraulic system and the emergency hydraulic system. A pneumatic system is used to pressurize the reservoir so that the pumps are always supplied with fluid under pressure and good pump performance maintained at high altitude.

1029. **Description.** The body (28) is divided by two walls; one wall (26) forms the emergency compartment (30) and the other wall (34) divides the reservoir into two chambers (32) and (36).

1030. Located on the outside of the reservoir are :

a. A quick-release filler cap (2) blanking the filler port; this is fitted with a metal filter (4) at the bottom of which is a white mark (1) for visually checking that the reservoir is properly filled.

b. The control wheel (8) of a needle valve the opening of which automatically determines the fluid level (accumulators empty) when filling the reservoir (ports 6 and F).

c. The control wheel (9) of another needle valve used for bleeding the air from compartment (32) while the reservoir is being filled.

d. The tapped endpiece (A) of a valve box (14) to which the pneumatic system is connected. This valve box incorporates an inlet valve (12) which opens for an inlet pressure of 0.29 lb/in² (0.020 bar) and a pressure relief valve (10) which opens for a pressure of 17.11 +2.55 −0.084 lb/in² (1.186 +0.176 −0.058 bar). These two valves regulate the pressure inside the reservoir. They enable the reservoir to breathe during fluid volume fluctuations caused by the operation of the various systems.

e. The five-pin electrical connector (20) of a capacitor (12C).

f. Eight tapped connection pieces (B, C, D, E, F, J, H and K) to which the various hydraulic lines connect, ref to legend in Fig 10-11.

g. A tapped port (G) blanked by a plug enabling chamber (36) to be drained.

1031. Inside the reservoir, the chambers (30, 32 and 36) are interconnected by an anti-siphon tube (24) and a ball valve (16). The ball valve automatically closes during inverted flight to prevent the fluid in chambers (30 and 32), where the suction ports are, from flowing into chamber (36).

1032. The three return ports (C, D and H) are fitted with an anti-emulsion nozzle (22). The various internal components are so designed and arranged as to permit satisfactory pump suction whatever the aircraft configuration. The reservoir operating details are stamped on an identification plate welded to the side of the reservoir.

1033. **Installation (refer to Fig 10-11).** Four lugs (38) three of which are provided with anchor nuts, are used to attach the reservoir in position.

No. 2 System Hydraulic Reservoir (refer to Fig 10-12)

1034. The reservoir contains the quantity of fluid required to operate the units and systems which depend on the No. 2 hydraulic system; it is pressurized by the same pneumatic system as is used to pressurize the No. 1 system reservoir.

1035 **Description.** The body (28) is divided into two communicating chambers (32 and 36) by a wall (34). On the outside of the reservoir are :

a. A filler cap (2).

b. The control wheels (8 and 9) of two needle valves.

c. The tapped endpiece (A) of a removable valve box (14).

NOTE

These components are identical with and serve the same purpose as those described in para 1030 for the No. 1 system reservoir.

1036. There are five tapped connection pieces (B, C, D, E and F) for connecting the various hydraulic lines, ref to legend on Fig 10-12. A tapped connection piece (G) with a blanking plug enables chamber (36) to be drained. Inside the reservoir, the chambers (32 and 36) are interconnected by an anti-siphon tube (24) and a ball valve (16) whose function is similar to that of the ball valve of the No. 1 system reservoir. The two return ports (C and D) are fitted with an anti-emulsion nozzle (22). The various internal components are designed and arranged to permit satisfactory pump suction whatever the aircraft configuration. The reservoir operating details are stamped on an identification plate welded to the side of the reservoir.

1037. **Installation.** Four lugs (38) three of which are provided with anchor nuts are used to attach the reservoir in position.

Low Pressure Filters (refer to Fig 10-13)

1038. Low pressure filters are fitted to the No. 1, No. 2 and emergency systems and consist of a filtering cartridge (1) housed in a case (2) provided with a filter head (3) retained by a butterfly screw (4). The cylindrical metal case is fitted with the following parts :

a. A pin (5) fitted through the base to which it is welded on the outside.

b. A circular flange (6) welded on the upper periphery.

1039. The case contains the following moving components :

a. A spring (7).

b. A cartridge support (8).

c. A filtering cartridge (1).

d. A flange (9).

1040. **Filter Head.** The filter head is a cast part through which the attaching butterfly screw (4) is fitted. It includes :

a. The fluid inlet and outlet ports.

b. A mounting lug (10) with two holes for attachment to the aircraft structure.

NOTE

The No. 2 system filter is attached by means of studs (11) screwed on a mounting flange.

1041. **Installation of the Filtering Cartridge.** The filtering cartridge (1) is installed in the filter case between the cartridge support and the flange. Felt washers (12) are placed in between the cartridge, the support and the flange. The spring (7), which bears on the bottom of the case and is guided by the pin (5), compresses this assembly to ensure sealing.Each filter should show no sign of leakage when tested at a pressure of 21.33 lb/in² (1.47 bars).

1042. **Operation.** The fluid under pressure enters the filter through the filter head then flows through the filtering cartridge from the outside to the inside and upward through the centre passage up to the outlet port. The flow direction in the filter accumulates the impurities on the outside of the filtering cartridge which makes cleaning easier. The filters have a filtering capacity of 6 to 15 microns.

Filters (refer to Fig 10-13).

1043. The following filters are fitted :

a. Servo-controls (11 and 22).

b. Rudder servo-controls (51).

c. Pre-servos.

d. Oscar valve.

e. Inboard control surface servo-controls (52 and 53).

1044. These filters each consist of a drilled cylindrical body (1) inside which a conical filter element (2) is mounted. The filter element (2) is retained in the body (1) by a cap (3) and a lock nut (4). The assembly is sealed by seals (5 and 6).

Heat Exchanger (refer to Fig 10-14)

1045. The heat exchanger is designed to cool hydraulic fluid through thermal exchange with the aircraft fuel supply. The heat exchanger consists of :

a. A cylindrical body (1) provided with two inlet ports and two outlet ports (hydraulic fluid).

b. An inlet connection (2) and an outlet connection (3) (fuel) which are sealed by means of 0-rings (4) and flexible screw-clamps (6).

1046. There are two identical chambers inside the body (5 and 16) which are connected each to one hydraulic system; these chambers incorporate :

a. Two flanges (8) carrying an 0-ring (9) and a bundle of pipes (10).

b. A perforated wall (11) forming two staggered compartments and through which the pipes are run.

c. Two flanged, roll-formed sheets (12) ducting the hydraulic fluid.

d. A spacing washer (28) which separates the two chambers.

1047. **Operation.** The engine fuel (which is the coolant) flows across the heat exchanger through the pipes (10). The hydraulic fluid enters the heat exchanger, flows from one compartment into the other and has its heat passed into the fuel as it passes between the pipes.

1048. **Installation.** The heat exchanger is attached to a gusset (26) integral with the structure by means of a flexible screw-clamp (27).

1049. The fuel outlet is connected to the aircraft system by a Wig-0-Flex flexible coupling (13). Tightening the nut (14) seals the assembly by compressing the two 0-ring seals (15) between a spacer (7) and two washers (17).

1050. The fuel inlet is connected to the aircraft system through a Wig-0-Flex coupling (22). Tightening the nut (23) seals the connection by compressing the 0-ring seal (24) between two washers (25).

1051. The hydraulic fluid inlet and outlet are connected to the aircraft system through a screwed end-fitting (18) and an Arsaero coupling. The connections are sealed by a copper gasket (19) and by compression of a biconic ring (20) by tightening nut (21).

CHAPTER 11

ELECTRICAL INSTALLATION

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LIST OF FIGURES

Figures for this chapter are contained in DI(AF) AAP 7213.003-2-14, Book 4

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CHAPTER 11

ELECTRICAL INSTALLATION

Table 11-1 Function of Components

Index No	Description	Characteristics and Functions
16F	Inverter (standby gyro-horizon)	This inverter provides 115 V 400 Hz three-phase a.c. power (1 phase grounded) for the standby gyro-horizon; maximum output power 10 VA.
1P	Generator	This is an engine-driven generator which supplies the aircraft system with a 28.5 V regulated d.c. voltage. Maximum continuous power :9 kW (with cooling requirements taken into account).
2P	DC generator reverse current relay	Isolates the generator from or connects it to the main d.c. busbar. It makes when the generator voltage is higher by 0.3 V than the battery voltage. It breaks for a reverse current above 20 A.
3P	Overvoltage protection	This causes the generator reverse current relay (2P) to open in case of overvoltage conditions or closing of the generator switch thereby interrupting the energization current to the generator.
4P	Voltage regulator	This regulator keeps the generator voltage constant at 28.5 ± 0.5 V.
10P	Battery	This is a cadmium-nickel alkaline type battery; rated voltage : 24 V, rated capacity : 40 AH. This battery performs two functions, it provides an emergency source of d.c. power and operates as a buffer battery.
35P	Magnetic polarised relay	This relay causes the generator failure warning light (on 1Z and for III D 201Z) to illuminate and the audio warning horn (7Z) to operate whenever the generator output is insufficient.
40P	III O Only. Preheating switch	This switch has two positions : OFF (locked) : Preheating bus de-energised. PREHEAT : Preheating bus supplied by external power units.
1V	Vario-alternator	This alternator provides 208 V, 400 Hz three-phase power for the aircraft a.c. system; maximum continuous power : 6 kVA under appropriate cooling conditions.
2V	Energization box	This box provides the required energization current for the inductors and the speed changer of the alternator (1V) according to the control signals from the detector box (3V) so as to maintain 208 V ± 2% and 400 Hz ± 1%.
3V	Detector box	This box detects the voltage and frequency variations in the a.c. system and feeds the energization box (2V) with corresponding corrective signals.
4V	Protection panel	This is used to detect the overvoltage, overfrequency, undervoltage, underfrequency and overheat limits and to control the opening of the line contactor (6V).
13V	Overheat detector	This is used to cause, through the protection panel (4V), the opening of the line contactor (6V) in case of alternator overheat.
15V	Inverter	This inverter is supplied by the d.c. system and provides 200 V, 400 Hz three-phase power for an a.c. subsystem; maximum output power : 150 VA.
26V	III O Only. Computing voltage filter	This filter transforms and filters the 200 V, 400 Hz power into three 26 V voltages, two of which are filtered and fed to certain items of equipment.

Index No	Description	Characteristics and Functions
30V	Phase sequence relay	This relay permits the aircraft system to be supplied through the external power receptacle (11V) only if the phase sequence is correct.
11Z	Transformer-rectifier	This is supplied by the a.c. system through the transfer switch (9Z) and provides d.c. power from the main bus when it is supplied. Maximum output power available for the d.c. system : 4k W.

GENERAL

Description (refer to Figs 11-1 to 11-4)

1101. There are two main electrical power distribution systems in the aircraft :

a. *DC System.* Voltage-28.5 V, this system is powered by :

(1) A generator (1P).

(2) A battery (10P).

(3) A transformer-rectifier (11Z) supplied by the a.c. system in case of failure of the generator.

(4) A ground power unit connected through an external power receptacle (17P).

b. *AC System.* Rated three-phase voltage — 119/206 V ± 2% Frequency — 400 Hz ± 15%. This system is powered by :

(1) An alternator (1V).

(2) A ground power unit connected though an external power receptacle (11V).

1102. The d.c. system supplies an inverter (15 V) which provides, 115/200 V, 400 Hz three-phase a.c. power for a sub-system. The generator and the alternator are driven by the engine through the accessory gear box. The installation of these units is described in Chapter 9.

1103. The power supply main contactors and protection systems are contained in boxes the schematic composition of which is shown in Fig 11-28 for the d.c. system and Figs 11-29 and 11-30 for the a.c. system (alternator and inverter).

DC POWER GENERATION

Description — (refer to Figs 11-1 and 11-5)

1104. The wiring diagrams for the d.c. power generation systems are contained in AAP 7213.001-2-7 for III O and AAP 7213.002-2-8 for III D. The system consists of :

a. The following power generation circuits :

(1) Generator circuit.

(2) Battery circuit.

(3) External power receptacle circuit.

b. The following busbars :

(1) Main busbar.

(2) Automatic load-shedding busbar.

(3) Controlled load-shedding busbar.

(4) Battery busbar.

(5) III O Only. Pre-heating busbar.

c. The following interconnecting boxes are shown on Fig 11-1 :

(1) Power supply box (A).

(2) AC box (B).

(3) Battery box (C).

(4) Distribution box (D).

(5) Circuit Breaker box (E).

(6) Load-shedding box (F).

(7) Contactor box (G).

(8) Central box.

d. Control units.

e. Interrupting units incorporated or not incorporated in the circuits.

Generator Circuit (refer to Figs 11-5 and 11-6)

1105. The generator circuit consists of the following units :

a. An engine-driven generator (1P) mounted on the LH side of the accessory gear box.

b. A voltage regulator (4P) located between frames 16 and 17 on the LH side; this carbon pile type regulator is used to maintain a constant voltage of 28.5 ± 0.5 V.

c. A generator reverse current relay (2P) located in the power supply box.

d. An overvoltage protection box (3P) located at frame 14 on the LH side; this box causes the generator reverse current relay (2P) to open in case of overvoltage conditions.

e. A polarized magnetic relay (35P) located in the power supply box.

f. A voltage test connector (37P) located between frames 16 and 17 on the LH side.

Battery Circuit

1106. The battery circuit consists of a battery (10P) directly supplying the battery bus and a battery reverse current relay (12P).

External Power Receptacle Circuit (refer to Fig 11-6)

1107. The external power receptacle circuit includes :

a. An external power receptacle (17P) with three pins; one supply terminal, one terminal to (2P) and one ground terminal.

b. A circuit breaker (20P).

c. A contactor (18P).

d. III O Only, a preheating switch (40P).

e. III O Only, a preheating relay (39P).

f. An isolating relay (21P).

Transformer-Rectifier Circuit

1108. The transformer-rectifier circuit includes a transformer-rectifier (11Z) and a safety contactor (27P).

Protective Systems — Brief Description of Operation

1109. **Generator Circuit (refer to Figs 11-3, 11-4 and 11-6).** The generator circuit contains the following protective circuits :

a. *Regulation.* The generator inductor current is passed through a carbon pile (13) the resistance of which is governed by the electro-magnet (14) either compressing or relaxing the carbon pile according to the generator output voltage.

b. *Cutting-in.* As soon as the generator voltage is more than the battery voltage, the solenoid (1), which is polarized, closes the contact (2). This results in closing the relay (3) (provided that the external power receptacle is not supplied and the contacts (5) in (3P) are closed).

c. *Reverse Current.* The field in the solenoid (4) causes the contact (2) to open. This results in opening the relay (3) (generator cutting-out). The generator automatically cuts in when the generator voltage becomes greater than the battery voltage.

d. *Overvoltage.* The relay (6) closes, supplying the solenoid (7) which causes the contacts (5) to open. This results in interrupting the energization current to the generator and opening the relay (3); the generator cuts out. The generator is manually reset by depressing the GEN reset pushbutton located on the failure warning panel (1Z). Depressing the reset pushbutton re-energizes the generator (by applying d.c. current to the inductor) and closes the contacts (5) through the solenoid (8). Releasing the pushbutton allows the generator to cut in again through 2P. If the overvoltage occurs again while the pushbutton is being depressed, solenoid (7), supplied by relay (6), opens contacts (5) as the action of solenoid (8) is nullified by relay (13). In this case, or if the generator cuts out again, it is to be considered unserviceable and no further attempt to reset is to be made.

e. *Undervoltage.* The magnetic field in the polarized relay (35P) becomes too weak to maintain its contact; the relay contact opens and the GEN warning light on the failure warning panel (1Z) and for III D (201Z), and the warning light (3Z) are illuminated and the audio warning horn (7Z) sounds.

f. *Turning the Generator Off.* Closing the GENERATOR switch on the failure warning panel (1Z) supplies the solenoid (7) which results in opening contacts (5) and interrupts the energization current to the generator; the generator cuts out. The opening of the generator reverse current relay (2P) in flight results in :

(1) Opening the automatic load-shedding contactor (23P).

(2) Closing the contactor (9Z).

(3) Closing the contactor (27P).

(4) Illuminating the GEN warning light on the failure warning panel (1Z) and for III D (201Z).

(5) Illuminating the FAIL warning light (3Z).

(6) Operating the audio warning horn (7Z).

1110. **Battery Circuit (refer to Fig 11-6).** The battery circuit contains the following protective circuits :

a. *Switching the Battery On.* Closing the BATT switch on 1Z supplies the relay (9) of the reverse current cut-out (provided the external power receptacle is not supplied), through contact (11) in the a-b position.

b. *Battery Cut-off.* In the event of a charge current higher than 300 amps, the field in the coil (10) brings the contact (11) into the a-c position, this contact is maintained by the solenoid (12).

c. *Manual Resetting.* Depressing the BATT reset pushbutton on the failure warning panel (1Z) cuts off the power supply to the solenoid (12). The contact (11) is returned to the a-b position and relay (9) closes.

d. *Automatic Resetting.* In case of generator overvoltage conditions, the contacts (5) are opened and the ground return of the solenoid (12) is interrupted. The contact (11) is returned to the a-b position and relay (9) closes.

1111. **External Power Receptacle Circuit (refer to Fig 11-6).** When the external power receptacle (17P) is supplied, contact (A) of the isolating relay (21P) is closed provided the circuit breaker (20P) is closed; contacts B and C are opened. The opening of contacts (B and C) may also result in the opening of the battery reverse current cut-out (12P) if the generator and battery switches on 1Z are ON and if the generator is delivering current. When the external power receptacle is supplied, relay 3 (in 2P) is no longer energized and the generator cuts out.

Table 11-2 DC Control Units

Index	Description	Open		Closed	
		Position	Effects	Position	Effects
24P	Load-shedding switch	LOAD SHED (down)	Manual and auto load-shed busbars are disconnected from the d.c. power system.	Up	Manual and auto load-shedding busbars normally supplied.
26P	Load-shedding prohibition switch	Aircraft in flight	Auto load-shedding system ready to function	Aircraft on the ground	Auto load-shedding system prevented from operating
40P III O Only	Preheating switch	PREHEAT	Connection : preheat busbar — missile preheat (direct); preheat busbar; ground telephone — gyro centre (through 39P). Isolation : missile control — radar supply.	OFF	Ground telephone and gyro-centre preheat cut-off. Supply through 39P : Ground telephone : from main busbar. Gyro-centre : from manual load-shedding busbar. Missile control circuit : from auto load-shedding busbar. Radar : from battery busbar.
on 1Z	Battery switch	Down	Battery off	Up	Battery on
	Battery reset pushbutton	Depressed	Battery ready to reset	Released	Battery reset after it has been disconnected due to reverse current conditions and if these conditions no longer exist.
	Generator switch	Up	Generator ready to cut in	Down	Generator cut-out.
	Generator reset pushbutton	Released	No effect	Depressed	Generator RCR disabled; over-voltage protector reset.

Table 11-3 DC Switching Units

Index	Description	Operating Function	Rest Position When :	Working Position When :
2P	Generator reverse current relay	Connection between generator and main d.c. bus	GEN switch (to 1Z) closed; external power receptacle supplied; reverse current to generator; generator voltage insufficient	Generator circuit normally operating
12P	Battery reverse current cut-out	Connection between battery and main d.c. bus	BATT switch (to 1Z) open; external power receptacle (17P) supplied; charge current to battery over 300 amps; battery voltage insufficient.	Battery circuit normally operating
18P	External power receptacle contactor	Connection between external power receptacle and main d.c. bus	External power receptacle (17P) not supplied and/or circuit breaker (20P) open.	External power receptacle (17P) supplied; circuit breaker (20P) closed.
21P	Isolating relay	Isolates the generator and the battery from the main d.c. bus	External power receptacle not supplied; circuit breaker (20P) open.	External power receptacle (17P) supplied; circuit breaker (20P) closed.
23P	Automatic load-shedding contactor	Connection between auto load-shedding bus and main d.c. bus	Switch (24P) to LOAD SHED. Reverse current relay (2P) open in flight. Auto load-shed bus not supplied.	Switch (24P) closed. Normal operation of reverse current relay (2P). Auto load-shed bus supplied from the main bus.
25P	Manual load-shedding contactor	Connection between manual load-shedding bus and main d.c. bus	Switch (24P) to LOAD SHED. Manual load-shed bus not supplied.	Switch (24P) closed. Manual load-shed bus supplied from the main bus

Index	Description	Operating Function	Rest Position When :	Working Position When :
39P III O Only	Preheating contactor	Connection between preheating bus and and preheated equipment.	Switch (40P) to OFF. Contactor (39P) de-energized.	Switch (40P) on PREHEAT. Contactor (39P) energized

AC POWER GENERATION

Description (refer to Figs 11-3 to 11-7)

1112. Wiring diagrams for the a.c. power generation systems are contained in AAP 7213.001-2-7 for III O and AAP 7213.002-2-8 for III D. The a.c. system includes :

a. The alternator circuit (1V) and the external power receptacle circuit (11V).

b. The following busbars :

(1) Alternator busbar.

(2) Auto load-shedding a.c. busbar.

(3) III O Only. Radar preheating busbar.

(4) III O Only. Gyro centre preheating busbar.

c. The a.c. box containing the main contactors and the protective systems.

Alternator Circuit

1113. The alternator circuit includes :

a. An alternator (1V) which mounts on the RH side of the accessory gear box; it is driven by the engine.

b. An excitation box (2V) located at frame 17, on the RH side.

c. A detector box (3V) located on the rear face of frame 10.

d. A protective panel (4V) located on the RH side of frame 17.

e. A line contactor (6V) in the a.c. box.

f. A filter (17V) mounted on frame 17 on the RH side.

External Power Receptacle Circuit (refer to Fig 11-7)

1114. The external power receptacle circuit includes :

a. An external power receptacle contactor (12V).

b. An external power receptacle (11V) with six pins :

(1) Three power pins.

(2) One ground pin.

(3) One pin connected to the battery.

(4) III O Only. One pin connected to preheating switch (40P).

c. For III O, these last two pins are shunted when (11V) is connected. They permit either contactor (12V) or preheating relays (4S) and (3F) to be supplied, according to the position of switch (40P). Contactor (12V) is supplied through a phase sequence safety relay (30V).

Inverter Circuit (refer to Figs 11-3 and 11-4)

1115. The inverter circuit includes :

a. An inverter (15V) located under the control pedestal in the cockpit. It is supplied by the manual load-shedding busbar through switch (16V) and provides 200 V, 400 Hz, three-phase current for a sub-system.

b. An inverter busbar which supplies the sub-system.

Protective Systems — Brief Description of Operation

1116. **Regulation System (refer to Fig 11-8).** The alternator is voltage and frequency regulated. There are two regulation phases, detection of the error in the detection box (3V) and correction of the error in the energization box (2V) and the alternator (1V).

1117. The voltage is regulated by varying the induction field in the alternator and the frequency is regulated by varying the rotational speed of the alternator which is achieved by driving the inductor (4) by a variable speed drive (E).

1118. The regulation system comprises the following sub-systems :

a. *Voltage Regulation.* Part of the system electrical power is transformed (5), rectified (6) and used to supply the inductor (4). To cause the inductor supply voltage to vary, the magnetic circuit of the transformer (5) is saturated to a greater or lesser extent through the winding (7). If the system voltage increases, the output voltage of the detector potentiometer (8) accordingly increases and the saturation degree of the transformer (5) is increased. The secondary voltage decreases causing the induction field and therefore the alternator output voltage to decrease. On starting, the alternator is excited by supplying the inductor (4) with current from the main d.c. bus. The purpose of the rectifier (20) is to prevent reverse current flows.

b. *Frequency Regulation.* The rotor (4) of the alternator is driven at a constant speed (8000 r.p.m.) by a speed changer (19) consisting of an electromagnetic brake (22) and a bell-shaped frame (21). As this speed changer is not capable of exceeding a speed ratio of 1 to 2, it is driven by a mechanical assembly operated as a two speed range gear box (2500 — 4500 — 8000 r.p.m.). A speed signal (23), taken at the input of the assembly, enables the regulator to electrically control the speed range changes, through the electro-magnetic brake.

c. *Drive — First Range.* A differential assembly (28), the planet gear of which acts as a step-up gear when the satellite carrier is locked, drives the electro-brake (22) carried by a ratchet wheel assembly (27). A second differential assembly (29), the planet gear of which drives the alternator (4), is accelerated by the bell-shaped frame (21) of the speed changer (acceleration is gradually decreasing as the input speed increases).

d. *Drive Speed Range Change.* When the input speed (E) reaches a given value (between 5150 and 5400 r.p.m.), the clutch (2) is de-energized and frees the step-up gear. The satellite gears of the first differential assembly (28) are prevented from driving the electro-brake (22) in the opposite direction by the ratchet wheel assembly (27). The satellite gears (or planetary gears) of the second differential assembly (29) are then slowed down by the bell-shaped frame (21) of the speed changer, the electro-brake (22) of which is stopped. In the opposite direction, when the input speed (E) decreases, the clutch (2) is energized and the ratchet wheel assembly is inoperative.

e. *Regulation.* The electro-brake (22) of the speed changer is energized by current from the transformer (9) passed through the rectifier (10). On either side of 400 Hz supply, filters (11) are used to vary the saturation value of the transformer. The energization current is then regulated in such a way that the instantaneous torque (A) transmitted by the speed changer results in the alternator delivering the required power at 400 Hz. The drift between the electro-brake (22) and the bell-shaped frame (21) creates eddy currents in the iron body of the frame; the property of these eddy currents is to transmit the instantaneous torque (A). Power resulting from the speed differences is converted to heat. The accuracy of such a regulation method is better than 1%.

1119. **Protection (refer to Fig 11-8).** The line contactor (6V), controlled by the potection panel (4V), opens in the following cases :

a. Undervoltage.

b. Underfrequency.

c. Overvoltage.

d. Overfrequency.

e. Alternator overheating.

1120. In the event of undervoltage or underfrequency conditions, the relay (18) opens and interrupts the power supply to line contactor (6V). It closes automatically again if normal values are restored.

1121. In the event of overvoltage or overfrequency conditions, the solenoid (13) is energized and opens the contact (14) which results in interrupting the energization current to the alternator through the relay (17) and opening the line contactor (6V).

1122. The opening of the line contactor (6V) is delayed :

a. By three seconds for undervoltage, underfrequency or overfrequency conditions.

b. By a time period varying inversely as the overvoltage value for overvoltage conditions.

1123. The opening of the line contactor (6V) causes the ALT warning light to illuminate on the failure warning panel (1Z) and for III D (201Z) and the audio warning horn (7Z) to operate.

1124. Line contactor (6V) may be reset by depressing the reset pushbutton (on 1Z) which causes the solenoid (16) to return the contact (14) to the closed position; this results in closing the relay (17) and the line contactor (6V).

1125. The alternator switch (9V) enables the alternator energization current to be cut off (relay 17).

Table 11-4 AC Control Units

Index	Description	Open		Closed	
		Position	Effects	Position	Effects
on 1Z	Alternator reset pushbutton	Released	No effect	Depressed	Alternator reset after it has cut out due to an overvoltage, overfrequency or overload condition, and if such condition no longer exists
on 1Z	Alternator switch	Down	Alternator excitation cut-off	Up	Alternator on
40P III O Only.	Preheating switch	OFF (locked)	Preheat busbar de-energised	PREHEAT	Preheating busbar supplied from external power receptacles
16V	Inverter switch	Down	Inverter off	Up	Inverter on

Table 11-5 AC Switching Units

Index	Description	Operating Function	Rest Position When :	*Working Position When :
27P	Safety contactor	Connection between main d.c. bus and transformer rectifier	Generator normally operating	Automatic load-shedding contactor (23P) open. Reversing contactor (9Z) on alternator circuit
6V	Line contactor	Connection between alternator and alternator bus	External power receptacle contactor (12V) closed. ALT switch (on 1Z) open. Alternator failure or overheating. Main d.c. bus not supplied.	Alternator circuit normally operating
12V	External power receptacle contactor	Connection between external power receptacle and a.c. system	External power receptacle (11V) not connected. Battery not supplied	External power receptacle connected. III O Only. Switch (40P) in normal position
9Z	TRU input contactor	Connection between transformer-rectifier and a.c. system	Generator normally operating	Generator cut-out Automatic load-shedding contactor (23P) open. Contactor (27P) closed

*Working position corresponds to electro-magnets energized.

VOLTAGE COMPUTER 400 Hz — III O ONLY

Description

1126. The voltage computer 400 Hz transforms and filters the 200V, 400 Hz three-phase power from the alternator to provide three 26V, 400 Hz voltages for various units of the weapons and navigation systems. These three voltages are :

a. A reference voltage (R) (not filtered).

b. A computing voltage (A).

c. A computing voltage (B) (filtered).

1127. The voltage computer supplies the following units :

a. Gyro computer (25F).

b. Roll stabilizer.

c. Gyro output multiplier (27F).

d. Spherical flight indicator (28F).

e. Flux valve (8F).

1128. The spherical flight indicator and flux valve are supplied through the output multiplier (27F).

1129. The voltage computer is mounted on the equipment bay door. It is supplied with 200 V power, as soon as the aircraft is under load, either from the alternator bus (FB 1F), or from the a.c. preheating bus (FB 2F), depending on the position of the PREHEAT switch (40P). The equipments are supplied as soon as the gyro centre switch (48F) is set to ON.

LOAD-SHEDDING

Automatic Load-Shedding (refer to Fig 11-9)

1130. In the event of the generator cutting out in flight, the automatic load-shedding contactor (23P) opens and isolates the systems not vital for flight performance. The opening of this contactor results in operating the isolating contactor (27P) which operates the transfer switch (9Z) which results in :

a. The d.c. system being supplied by the a.c. system (through the transformer-rectifier 11Z).

b. The a.c. system being relieved of certain system loads.

1131. The load-shedding override switch (26P), installed on the RH undercarriage shock-absorber, prevents the contactor (23P) from opening as long as the shock-absorber is compressed (aircraft on the ground).

Manual Load-Shedding (refer to Fig 11-9)

1132. In case of d.c. generator and alternator failure, the only electrical power source is the battery, therefore it is most important to disconnect the non-essential systems. Load-shedding is achieved by manually operating the load-shedding switch (24P) which opens load-shedding contactors (25P) and (23P).

PREHEATING — III O ONLY (refer to Figs 11-14 and 11-15)

Description

1133. Under operational conditions, certain systems are energized while the aircraft is on standby to ensure that these systems are serviceable immediately on take-off.

1134. The preheating system uses power from d.c. and a.c. field units. The d.c. and a.c. preheating busbars are supplied as soon as the external power sources are connected (d.c. to 17P, a.c. to 11V). The external power receptacle (11V) includes two shunted terminals to permit the supply of power to relays (4S) and (3F) with (40P) set to PREHEAT or to (12V) with (40P) set to OFF. Missile preheating is available at both positions of switch (40P).

1135. The following conditions apply with switch (40P) set to PREHEAT :

a. *DC System.* External power receptacle (17P) isolated from the main d.c. busbar (relay 18P not supplied). Relay (39P) is energized and breaks the missile control circuit and the battery busbar supply to the radar. In this condition, the ground telephone and gyro centre are supplied from the preheating busbar.

b. *AC System.* The gyro centre and the radar are supplied from the preheating busbar via relays (3F) and (4S) respectively. The external power relays (18P and 12V) remain open, isolating all other ancillary systems.

ELECTRICAL POWER DISTRIBUTION

General

1136. Wiring diagrams for the electrical power distribution systems are contained in AAP 7213.001-2-7 for III O and AAP 7213.002-2-8 for III D.

DC Power (refer to Figs 11-28 and 11-29)

1137. The main contactors and protection systems are contained in the boxes listed below. The schematic composition of each box is shown in the referenced figure :

a. Power supply box (A).

b. Battery box (C).

c. Distribution box (D).

d. Circuit breaker box (E).

e. Load-shedding box (F).

f. Contactor box (G).

1138. The central box contains the following main ancillary system relays :

a. *For III O :* 38C, 8D, 9D, 10D, 12E, 46G, 6K, 19K, 31K, 51K, 16M, 39P, 30S, 31S, 38S.

b. *For III D:* 38C, 10D, 22F, 46G, 6K, 24L.

1139. The following boxes contain the X terminals :

a. Armament box located in the equipment bay — 1X terminals.

b. Central box — 2X terminals.

c. Armament box located in front of the pedestal — 3X terminals.

d. Servo-control box — 5X terminals.

AC Power (refer to Figs 11-30 to 11-32)

1140. The main contactors and protection systems of the a.c. power supply systems are contained in the AC power supply box (B). The schematic composition of this box is shown in Fig 11-29 (for III O) and Fig 11-30 (for III D) for the alternator system and Fig 11-31 for the inverter system.

EXTERNAL LIGHTS

Landing Lights (refer to Figs 11-33 and 11-34)

1141. The aircraft is equipped with two 450 W landing lights (26 Lg and 26 Ld) used either for landing or for taxiing. The lights are mounted on the nose undercarriage leg and are controlled by means of a three position switch (27L) marked TAXI, OFF and LDG.

1142. In the LDG position, two relays (24L and 33L) are simultaneously energized and both lights are supplied. In the TAXI position, only one relay (24L) is energized and a single light (26 Lg) is supplied. A microswitch (34L) mounted on the front shield door prevents the landing lights being energized when the undercarriage is retracted.

1143. **Installation.** Each landing light incorporates a Y-shaped support fitting, and is attached to the nose undercarriage lugs by means of screws.

Navigation Lights (refer to Figs 11-33 and 11-34)

1144. The aircraft is fitted with three navigation lights :

a. Port wing tip light (15L) :RED.

b. Starboard wing tip light (16L) : GREEN.

c. Tail light (13L) with two lamps : located on the fin, above the rudder.

1145. **Description and Installation.** Each wing tip light consists of a plexiglass shell (53) attached to the wing structure through a bolted flange (56). The bulb (57) is accessible after removing the shell.

1146. The tail light (13L) consists of two shells (58) of special Fresnel glass tinted white and yellow, which protrude from the contour. The yellow tinted light section is located at the upper part. The shells are held in position by two springs (59) and two covers (60) bolted to the fin structure. A bracket (61) carries the lamp holders (62).

1147. The flasher mechanism (12L) and the resistor box (17L) are bolted to a folded sheet metal support (63) located at the lower part of the LH air intake.

1148. **Control.** The navigation lights are controlled by means of a switch (11L — front cockpit only) situated on the LH side of the cockpit.

Formation Lights (refer to Fig 11-35)

1149. The aircraft is provided with two formation lights, (20L) starboard and (21L) port, mounted on the upper surface near the wing tips.

1150. **Purpose.** The purpose of the formation lights is to enable the aircraft to be seen by accompanying aircraft during close formation flying at night. The lights are controlled by a switch (19L — front cockpit only) located on the LH side.

1151. In the FULL position, the lights are directly supplied and brightly illuminated. In the DIM position, the lights are supplied through a resistor contained in the resistor box (17L).

1152. **Description.** Each light consists of a dome (64) enclosing the lamp and an electrical supply connector. The dome is riveted to a flange (65) bolted to the wing structure (66). The assembly is covered with a Fresnel optical glass secured by means of a bolted flange.

ELECTRICAL COMPONENTS

1153. Table 11-9 is a cross reference of electrical component identification numbers to drawing numbers. By using the drawing number referenced for an item, and referring to the index contained in front of each section of the Wiring Diagram Manual (AAP 7213.001-2-7 for III O and AAP 7213.002-2-8 for III D) the circuit containing the particular item can be readily identified.

NOTE

Drawings prefixed DA are RAAF drawings and are applicable upon the introduction of Mods 1112/1113.

Table 11-6 Circuit Operation — III O Only

	POSITION	ENGINE	EXTERNAL POWER RECEPTACLES	CONTACTS: Closed or Operated	CONTACTS: Open	POWER SUPPLY: Battery bus	Main DC bus	Auto load-shed bus	Manual load-shed bus	D.c. preheat. bus	A.c. preheat. bus	Alternator bus	A.c. load-shed bus	Invertor bus	INDICATING SYSTEMS: Warning Lights on 1Z	Warning Lights 3Z	Warning Horn	REMARKS
A/C on the ground	Rest Fig 11-10	Stopped		26P	All	X												
A/C on the ground	Battery ON Fig 11-12	Stopped		Batt on 1Z 12P-24P 25P-23P-26P		X	X	X	X						GEN ALT	yes	yes	Inverter supplied if 16V closed.
A/C on the ground	Preheat. Figs 11-14 and 11-15	Stopped	17P 11V	20P-21P 40P on P 39P-4S-3F		X				X	X				NO	no	no	Ref only: GRD. CONN. light (18M) on.
A/C on the ground	Tests Fig 11-16	Stopped	17P 11V	20P-21P 16V-26P 40P on O 18P-12V	Batt on 1Z 39P-4S-3F	X	X	X	X	X	X	X	X	X	BATT GEN ALT	yes	yes	Ref only: GRD. CONN. light (18M) on.
A/C on the ground	Tests	Running		Batt on 1Z 12P-2P-6V	21P 12V	X	X	X	X	X	X	X	X	X	All off	no	no	Auto load-shedding prohibited.
A/C in flight	Flight Fig 11-18	Running		Batt on 1Z 12P-2P 16V-6V	21P-26P 12V	X	X	X	X			X	X	X	All off	no	no	Auto load shedding ready to operate.
A/C in flight	Auto load-shed Figs 11-9 and 11-22	Running		24P-25P-27P-9Z	26P-2P 23P	X	X		X			X	X	X	GEN	yes	yes	Main d.c. bus supplied by 11Z through 27P.
A/C in flight	Manual load-shed Figs 11-9 and 11-22	Running		24P-25P 23P-26P		X	X					X			All off except in case of failures	no	no	

NOTE: The GEN and ALT switches are safety-wired in the ON position.

Table 11-7 Circuit Operation — III D Only

	POSITION	ENGINE	EXTERNAL POWER RECEPTACLES	CONTACTS: Closed or operated	CONTACTS: Open	POWER SUPPLY: Battery bus	Main d.c. bus	Auto load-shed DC bus	Manual load-shed bus	Alternator bus	Auto load-shed alt.bus	Inverter bus	INDICATING SYSTEMS: Warning lights On 1Z and 201Z	Warning lights 3Z	Warning horn	REMARKS
A/C on the ground	Rest Fig 11-11	Stopped		26P	All	X										
A/C on the ground	Battery ON Fig 11-13	Stopped		Batt on 1Z 12P-24P 25P-23P 26P		X	X	X	X				GEN ALT	Yes	Yes	Inverter supplied if 16V closed
A/C on the ground	Ground tests Fig 11-17	Stopped	17P 11V	20P-21P-16V-26P-18P-12V	Batt on 1Z 12P-2P 6V	X	X	X	X	X	X	X	BATT GEN ALT	Yes	Yes	Ref only: GRD. CONN. light (18M) on
A/C on the ground	Tests	Running		Batt on 1Z 12P-2P-6V	21P-12V	X	X	X	X	X	X	X	All off	No	No	Auto load-shedding system prevented from operating
A/C in flight	Flight Fig 11-19	Running		Batt on 1Z 12P-2P-16V-6V	21P-26P 12V	X	X	X	X	X	X	X	All off	No	No	Auto load-shedding system not prevented from operating
A/C in flight	Auto load-shedding Figs 11-9 and 11-23	Running		42P-25P-23P		X	X		X	X		X	GEN	Yes	Yes	Main bus supplied by by 11Z through 27P
A/C in flight	Controlled load-shedding Figs 11-9 and 11-27	Running		24P-25P-23P		X	X			X			All off except in case of failures	No	No	

NOTE: The GEN and ALT switches are safety-wired in the ON position.

Table 11-8 Circuit Failures

FAILURES	CONTACTS		POWER SUPPLY							INDICATING SYSTEMS			REMARKS
										Warning lights			
	Closed or supplied	Open	Battery bus	Main d.c. bus	Auto load-shed d.c. bus	Manual load-shed bus	Alternator bus	Auto load-shed alternator bus	Inverter bus	On 1Z and 201Z (IIID only)	3Z	Warning horn	
Battery Figs 11-20 and 11-21		12P		X	X	X	X	X	X	BATT	Yes	No	
Generator Figs 11-22 and 11-23	27P	2P 23P	X	X		X	X		X	GEN	Yes	Yes	Main d.c. bus supplied by 11Z (Transfer switch 9Z). Auto load-shedding.
Alternator Figs 11-24 and 11-25		6V	X	X	X	X			X	ALT	Yes	Yes	
Generator and alternator Figs 11-26 and 11-27		6V 2P-23P (Possibly) 24P-25P	X	X		Possibly			Possibly	GEN ALT	Yes	Yes	Battery supply only. Auto load-shedding. Battery run down time very short-Controlled load-shedding required.
Battery and alternator		All									No	No	Complete failure. Circuit no longer supplied by the alternator (6V open).
Battery and alternator	2P	12P-6V		X	X	X			X	BATT ALT	Yes	Yes	Total power failure.

Table 11-9 Electrical Components

Ident	Description	Drawing No
1A	Gun circuit breaker	660.001
2A	Armament safety panel	660.001 — 660.002 — 660.003/3 — DA83233001 — DA83233014 — 660.006 — 660.010
3A	LH gun firing relay	660.001
4A	RH gun firing relay	660.001 — 660.012/1/3
5A	LH gun connector	660.001
6A	RH gun connector	660.001
7A	LH gun junction box	660.001
8A	RH gun junction box	660.001
9A	LH gun	660.001
10A	RH gun	660.001
11A	Gun/missile fire button	660.001
14A	Camera pushbutton	660.015
15A	Camera	660.015
16A	Photo-cell	660.015
17A	Control panel	660.015
19A	Sight circuit breaker	660.012/1/3
20A	Missile fuse	600.011 — 660.006
21A	Missile control circuit breaker	600.011 — 660.006
22A	III O Only. Missile preheat circuit breaker	600.011
23A	Sight circuit breaker	660.012/1/3
24A	Sight circuit breaker	660.012/1/3
25A	Sight relay box	660.012/1/3 — 660.009 — 660.013/3 — 660.027/3 — DA83233001 — DA83233012
26A	Voltage regulator	660.012/1/3
27A	Gyro caging button	660.012/1/3
29A	Sight head	660.013/3
30A	III O Only. Closing setting switch	660.012/1/3
41A	Missile three-phase circuit breaker	660.006
42A	III O Only. Missile on contactor	660.006
43A	III O Only. MATRA/NA switch	660.006
44A	Preparation relay	660.006
45A	Missile three-phase relay	660.006
46A	III O Only. Battery heater relay	660.006
47A	Missile ready signal relay	660.006
48A	III O Only. Automatic fire relay	660.006
49A	Armament safety relay	660.006 — DA83233006 — DA83233015
50A	Firing relay	660.006
51A	Firing repetition relay	660.006
52A	Firing non-return cell	660.006
53A	Trigger non-return cell	660.006
55A	III O Only. Harmonization box	660.008
56A	III O Only. Computer	660.009

Ident	Description	Drawing No
57A	Coder power supply unit NOT INSTALLED	660.009
58A	Modulator transmitter NOT INSTALLED	660.009
59A	Missile antenna filter NOT INSTALLED	660.009
60A	Missile guidance antenna NOT INSTALLED	660.009
61A	Missile control stick NOT INSTALLED	660.009
62A	III O Only. Coaxial bulkhead connector	660.028
63A	III O Only. Coaxial bulkhead connector	660.028
65A	LH Sidewinder/MATRA R550 heater relay	660.010 — DA83233003 — DA83233005 — DA83233017 — DA83233019 — DA83233020
66A	RH Sidewinder/MATRA R550 heater relay	660.010 — DA83233003 — DA83233005 — DA83233017 — DA83223019 — DA83233020
67A	Sidewinder/MATRA R550 safety relay	660.010 — DA83233003 — DA83233016 — DA83233019
68A	Sidewinder/MATRA R550 firing relay	660.010 — 660.015 — DA83233004 — DA83233006 — DA83233017 — DA83233021
69A	LH Sidewinder/MATRA R550 firing relay	660.010 — DA83233004 — DA83233005 — DA83233017 — DA83233019
70A	RH Sidewinder/MATRA R550 firing relay	660.010 — DA83233005 — DA83233017 — DA83233018
71A	Sidewinder/MATRA R550 single firing relay	660.010 — DA83233004 — DA83233016
72A	Sidewinder/MATRA R550 salvo firing relay	660.010 — DA83233004 — DA83233005 — DA83233017 — DA83233018
73A	Sidewinder/MATRA R550 audio potentiometer relay	660.010 — DA83233004 — DA83233017
74A	Missile audio potentiometer	660.010
75A	LH Sidewinder/MATRA R550	660.011
76A	RH Sidewinder/MATRA R550	660.011
81A	NA missile fuse	660.006
82A	SW NA missile fuse	660.006
83A	III O Only. MATRA R530 signal fuse	660.006
84A	III O Only. Missile pylon fuse	660.006
85A	III O Only. Missile pylon fuse	660.006
86A	III O Only. Missile pylon fuse	660.006
87A	III O Only. Missile pylon fuse	660.006
90A	III O Only. Information relay	660.006
92A	III O Only. Missile unlocked light	660.006
93A	III O Only. Firing safety relay	660.006
94A	III O Only. Fire safety cell	660.006
113A	LH Missile lock on light	DA83233002 — DA83233015
114A	RH Missile lock on light	DA83233002 — DA83233015
115A	Rapid gun/MISSILE light	DA83233002 — DA83233015
120A	III D Only. Coaxial connector	660.009
121A	III D Only. Coaxial connector	660.009
123A	Transfer box	DA83233010 — DA83233023
1B	Rocket circuit breaker	660.002
2B	Rocket-and-bomb release button	660.002 — 660.003/3

Ident	Description	Drawing No
3B	Rocket-and-bomb release relay	660.002 — 660.003/3
4B	Armament panel	660.003/3 — 660.015 — 660.006 — 660.012/1/3 — 660.019/3 — 660.002 — 660.010 — DA83233000 — DA83233013
5B	Bomb circuit breaker	660.003/3
6B	Bomb emergency relay	660.003/3
7B	III O Only. AC power supply circuit breaker	660.004
8B	III O Only. DC power supply fuse	660.004
9B	III O Only. Test relay	660.004
11B	III O Only. Control box	660.004
12B	Wing store jettison button	660.005/1
13B	Wing store jettison system circuit breaker	660.005/1
14B	Heading error switch	660.014/3 — 660.012
15B	III O Only. LABS switch	660.014/3
16B	III O Only. Timer (not installed)	660.014/3
212B	III D Only. Wing store jettison button	660.005/1
1C	Air data computer	600.027 — 680.008 — 680.010 — 600.059 — 600.029
2C	Air data computer output multiplier	680.009/3 — 680.010/3 — 680.023 — 680.026 — 600.029 — 600.056 — 680.021 — 660.012 — 600.013
3C	Hydraulic pressure indicator circuit breaker	600.051/3
4C	No 2 system transmitter	600.051/3
5C	No 1 system transmitter	600.051/3
6C	Undercarriage transmitter	600.051/3
7C	Wheel brake pressure transmitter	600.051/3
8C	Hydraulic pressure selector switch	600.051/3
9C	Hydraulic pressure indicator	600.051/3
10C	Servo-control isolating system circuit breaker	600.052
11C	Control box	600.052
12C	Hydraulic reservoir capacitor	600.052
13C	Isolating valve	600.052
14C	Electric pump circuit breaker	600.052
15C	Pressure switch	600.052
16C	Electric pump contactor	600.052
17C	Electric pump fuse	600.052
18C	Electric pump	600.052
20C	Roll neutral resetting jack circuit breaker	600.058
21C	Roll trim control	600.058
22C	Roll trim actuator	600.058
23C	Roll trim indicator light	600.058
24C	III D Only. Rudder trim control relay	600.054
25C	III D Only. H 30000 ft relay	600.029 — 600.066
26C	Trim jack box	600.054/3
27C	No 1 hydraulic system pressure switch	600.053
28C	No 2 hydraulic system pressure switch	600.053

Ident	Description	Drawing No
29C	Emergency hydraulic system pressure switch	600.053
30C	Rudder neutral resetting jack circuit breaker	600.054/3
31C	Rudder trim control	600.054/3
32C	Rudder trim actuator	600.054/3
33C	Rudder trim indicator light	600.054/3
34C	III D Only. Airbrake override switch	600.055
35C	Airbrake circuit breaker	600.055
36C	Airbrake switch	600.055
37C	Airbrake indicator light	600.055
38C	Airbrake relay	600.055
39C	Airbrake electro-distributing valve	600.055
40C	RH jack switch	600.055
41C	LH jack switch	600.055
42C	Shockcone position indicator	600.057/3
43C	Auto flight function control panel	600.056/3 — 680.020 — 680.022 — 680.026 — 680.021
44C	Shockcone slaving system circuit breaker	600.056/3 — 600.057/3
45C	Shockcone slaving system circuit breaker	600.056/3
46C	Slaving box	600.056/3
48C	Shockcone system fuse	600.056/3
49C	Manual system fuse	600.056/3
50C	Automatic system fuse	600.056/3
51C	Control circuit breaker	600.056/3
54C	Manual retraction switch	600.056/3
55C	Manual extension switch	600.056/3
56C	Automatic system switch	600.056/3
57C	Automatic extension switch	600.056/3
58C	Automatic retraction switch	600.056/3
59C	Shockcone actuator	600.056/3
60C	Potentiometer box	600.056/3 — 600.057/3
61C	Temperature sensing unit	680.008
62C	Air data computer power circuit breaker	680.008
63C	Air data computer power circuit breaker	680.008
66C	Accelerometer circuit breaker	600.059/3
67C	Accelerometer sensing unit	600.059/3
68C	Accelerometer	600.059/3
70C	Servo-control circuit breaker	680.020
71C	Auto-command circuit breaker	680.020
72C	Roll auto-command circuit breaker	680.020
73C	Roll auto-command circuit breaker	680.020
74C	Servo-control circuit breaker	680.020
75C	Servo-control circuit breaker	680.020
76C	Rate gyro transformer	680.020
77C	Rudder servo-control circuit breaker	680.020

Ident	Description	Drawing No
78C	Inboard control surface servo-control circuit breaker	680.020
79C	28 V safety relay	680.020
80C	Rudder system failure indicating relay	680.020
81C	Pitch system failure indicating relay	680.020
82C	Servo-control amplifier	680.020 — 680-021 — 680.022 — 680.023 — 680.024 — 680.027
83C	Roll auto-command relay	680.020
85C	Servo-control disengage circuit breaker	680.022
86C	Damper disengage button	680.022
89C	Rate gyro	680.021/3
90C	Rudder system transformer	680.021/3
91C	Rudder servo-control	680.021/3
92C	Input inductive position pick-off	680.021/3
93C	Output inductive position pick-off	680.021/3
94C	Electro-valve	680.021/3
95C	Safety microswitch box	680.021/3
96C	Pitch rate-gyro	680.023
97C	Pitch system transformer	680.022/3
98C	Dual input inductive position pick-off	680.022
99C	LH output inductive position pick-off	680.022
100C	RH output inductive position pick-off	680.022
101C	LH inboard control surface servo-control	680.022
102C	RH inboard control surface servo-control	680.022
103C	Unlocking valve electro-valve	680.022
105C	Pre-flight test connection	680.027
107C	Auxiliary servo-control	680.023/3
108C	Inductive position pick-off	680.023/3
109C	Auxiliary servo-control transformer	680.023/3
110C	Pitch stop microswitch	680.023/3
111C	Dual feed valve	680.026/3
113C	Elevon servo-control stop microswitch	680.023/3
114C	Elevon servo-control stop microswitch	680.023/3
115C	Roll stabilizer engage button	680.024
116C	Roll pre-servo	680.024
117C	Inductive position pick-off	680.024
120C	Auto-command amplifier	680.025 — 680.026/3 — 680.027 — 680.024
121C	Rate gyro	680.025 — 680.024
122C	Linkage dynamometer	680.025 — 680.024
123C	Control stick dynamometer	680.025
124C	Pitch trim system circuit breaker	680.025
125C	Nose down trim contactor switch	680.025
126C	Nose up trim contactor switch	680.025
127C	Pitch trim relay	680.025

Ident	Description	Drawing No
128C	Nose up trim system circuit breaker	680.025
129C	Nose down trim system circuit breaker	680.025
130C	Pitch trim jack	680.025
131C	Pitch trim indicator light	680.025
132C	Auto-command failure relay	680.025
133C	Auto-command emergency circuit breaker	680.025 — 680.024
134C	Emergency disengage switch (safe tied in off position)	680.025 — 680.024
135C	Emergency disengage electro-valve	680.025 — 680.024
137C	III O Only. Servo-control secondary circuit breaker	680.020
138C	III O Only. Auto-command secondary circuit breaker	680.020 — 680.025
139C	Auto-command gain switch	680.025
141C	Pitch auto trim circuit breaker	680.025
142C	III D Only. Nose down trim, override relay	680.025
143C	III D Only. Nose up trim, override relay	680.025
146C	III D Only. Roll trim relay	600.058
147C	III D Only. Roll trim relay	600.058
150C	Protection box	680.026/3
151C	Auto-command protection relay	680.026/3
208C	III D Only. Hydraulic pressure selector switch	600.051
209C	III D Only. Hydraulic pressure gauge	600.051
221C	III D Only. Roll trim control	600.058
223C	III D Only. Roll trim indicator light	600.058
231C	III D Only. Rudder trim control	600.054
233C	III D Only. Rudder trim indicator light	600.054
236C	III D Only. Airbrake control switch	600.055
237C	III D Only. Airbrake indicator light	600.055
242C	III D Only. Shockcone position indicator	600.057
286C	III D Only. Auto-command disengage button	680.022
325C	III D Only. Nose down trim contactor switch	680.025
326C	III D Only. Nose up trim contactor switch	680.025
331C	III D Only. Pitch trim indicator light	680.025
334C	III D Only. Emergency disengage control	680.024
1D	Demisting system circuit breaker	600.069
2D	Demisting control	600.069
3D	Demisting resistor	600.069
4D	Emergency demisting system circuit breaker	600.069
5D1	Helmet visor resistor (front cockpit)	600.069
5D2	Helmet visor resistor (rear cockpit)	600.069
6D	Pitot heater circuit breaker	600.070
7D	Pitot heater circuit breaker	600.070
8D	Total pressure head relay	600.070
9D	Pitot heater relay	600.070
10D	Current relay	600.070

Ident	Description	Drawing No
11D	Pitot heater/Incidence indicator control	600.070 — 680.001 — 680.002
12D	Flight instrument total pressure head	600.070
13D	Total pressure head	600.070
14D	Pitot-static pressure head	600.070 — 894.002
202D	III D Only. Visor de-misting switch	600.069
203D	III D Only. De-misting resistor	600.069
1E	R.P.M. indicator	600.025
2E	Fuel LP pressure switch	600.025
3E	T5 temperature indicator	600.025
4E	Adjustment box	600.025
5E	Terminal board	600.025
6E	Compensating leads	600.025
7E	Connector boxes	600.025
8E	Dual thermocouples	600.025
9E	Connector boxes	600.025
10E	R.P.M. indicator circuit breaker (front cockpit)	600.025
11E	Aircraft limit indicator	600.027
12E	III O Only. Aircraft limit relay	600.027
201E	III D Only. R.P.M. indicator — Rear cockpit	600.025
204E	III D Only. Jet pipe temperature indicator adjustment box	600.025
203E	III D Only. Jet pipe temperature indicator — rear cockpit	600.025
205E	III D Only. Terminal board	600.025
210E	III D Only. R.P.M. indicator fuse-breaker — rear cockpit	600.025
211E	III D Only. Aircraft limit warning light	600.027
1F	Gyro normal a.c. circuit breaker	600.012 — 680.003
2F	Gyro preheating a.c. circuit breaker	600.012 — 680.003
3F	Gyro normal relay	680.003/1/3
4F	Gyro normal circuit breaker	600.011 — 680.003
5F	III O Only. Gyro preheating circuit breaker	600.011
7F	Compass circuit breaker	680.001
8F	Flux valve	680.004/3
9F	Incidence indicator circuit breaker	680.001 — 680.002
10F	Incidence indicator circuit breaker	680.001 — 680.002
11F	Incidence indicator amplifier	680.001 — 680.002
12F	Incidence probe	680.001 — 680.002
13F	Incidence indicator	680.001 — 680.002
14F	Push-to-test button	680.001 — 680.002
15F	Standby gyro-horizon circuit breaker	680.001
16F	Standby gyro-horizon inverter	680.001
17F	Standby gyro-horizon	680.001
20F	Gyro-centre circuit breaker	680.003/1/3 — 680.004
22F	208V safety relay	680.003/1/3

Ident	Description	Drawing No
23F	DC relay	680.003/1/3
24F	Control panel	680.003/1/3 — 680.004
25F	Gyro computer	680.005/3 — 680.004
26F	Gyro reference unit	680.005/3 — 680.013 — 680.004
27F	Gyro centre output multiplier box	660.030/3 — 660.034/3 — 680.003/1/3 — 660.034/3 — 680.003/1/3 — 660.013 — 680.004
28F	Spherical flight indicator	680.006/3 — 680.004
29F	III D Only. Rectifier box	680.003
35F	PHI circuit breaker	680.011/3
36F	PHI circuit breaker	680.011/3
37F	PHI computer	680.011/3 — 680.012/3 —680.010
38F	Navigation indicator	680.011/3 — 680.004 —680.012 — 680.013 — 660.013
39F	Station selector box	680.012/3
47F	III O Only. Potentiometer	660.021/3 — 660.031/3
48F	Gyro centre switch (with 2 locked positions)	680.003/1/3
49F	Gyro centre failure light	680.003/1/3
51F	III O Only. Doppler a.c. circuit breaker	680.014/3
52F	III O Only. Three-phase/single-phase transformer	680.014/3
53F	III O Only. Doppler transmitter-receiver	680.014/3 — 680.015/3
54F	III O Only. Doppler antenna	680.014/3 — 680.015/3
55F	Doppler control panel	680.015/3 — 680.016/3
56F	PHI — radar coupler	680.017/3 — 680.011
57F	Range setting unit	660.013
59F	III O Only. Doppler memory indicator	680.015/3
213F	III D Only. Incidence indicator	680.002
214F	III D Only. Incidence test pushbutton	680.002
217F	III D Only. Standby gyro-horizon	680.001
228F	III D Only. Ball attitude indicator	680.004
238F	III D Only. PHI indicator — rear cockpit	680.011 — 680.012 — 680.013
1G	Undercarriage control circuitbreaker	600.042
2G	Undercarriage retraction override microswitch	600.042
3G	Undercarriage control switch	600.042 — 600.044
4G	Undercarriage electro-distributing valve	600.042
5G	Undercarriage door electro-distributing valve	600.042
6G	Door open microswitch	600.042
7G	Door open microswitch	600.042
8G	Door open microswitch	600.042
10G	Undercarriage up microswitch	600.042
11G	Undercarriage up microswitch	600.042
12G	Undercarriage up microswitch	600.042
13G	III O Only. Electro-valve	600.042
14G	Undercarriage down microswitch	600.042 — 600.043
15G	Undercarriage down microswitch	600.042 — 600.043

Ident	Description	Drawing No
16G	Undercarriage down microswitch	600.042 — 600.043
18G1	Door closed microswitch	600.043
18G2	Door closed microswitch	600.042
19G1	Door closed microswitch	600.042
19G2	Door closed microswitch	600.043
20G1	Door closed microswitch	600.042
20G2	Door closed microswitch	600.043
22G	Undercarriage retraction override system circuit breaker	600.042 — 600.044
23G	Undercarriage retraction override switch	600.042 — 600.044
24G	Door manual unlocking system microswitch	600.042
25G	Door manual unlocking system microswitch	600.042
26G	Door manual unlocking system microswitch	600.042
27G	Undercarriage not down light	600.043
28G	Undercarriage position indicator circuit breaker	600.043
29G	Undercarriage position indicator	600.043
30G	Brake microswitch	600.044
31G	Throttle lever retarded microswitch	600.043
32G	Flasher	600.043
33G	Ministop system circuit breaker	600.044
34G	Ministop system control switch	600.044
35G	Ministop electro-distributing valve	600.044
36G	Ministop electro-distributing valve	600.044
39G	Accelerometer	600.044
40G	Accelerometer	600.044
41G	Brake electro-distributing valve	600.044
43G	Control stick circuit breaker	600.044 — 660.001
44G	Ground firing connector	600.044 — 660.001
45G	Undercarriage slaving microswitch	600.044 — 660.001
46G	Ground firing override relay	600.044 — 660.001
47G	Longitudinal jack microswitch	600.042 — 600.043
48G	Longitudinal jack microswitch	600.042 — 600.043
49G	Longitudinal jack microswitch	600.042 — 600.044
50G	Longitudinal jack microswitch	600.042 — 600.043
52G	Airspeed capsule	600.043
53G	Dual flasher	600.043
54G	III O Only. Audio checking pushbutton	650.001
54G	III D Only. Undercarriage downlocked audio checking system relay	650.001
55G	III O Only. Audio checking relay	650.001
55G	III D Only. Undercarriage downlocked audio checking tone button	650.001
203G	III D Only. Undercarriage control switch	600.042
227G	III D Only. Undercarriage not down warning light	600.043

Ident	Description	Drawing No
229G	III D Only. Undercarriage position indicator	600.043
234G	III D Only. Ministop installation control switch	600.044
1H	Cabin temperature control circuit breaker	600.066/1/3 — 600.068
2H	Control panel	600.066/1/3
3H	Temperature control circuit breaker	600.066/1/3
4H	Master cock	600.066/1/3
5H	Servo-amplifier	600.066/1/3
6H	Temperature sensing unit	600.066/1/3
7H	Cabin temperature sensing unit	600.066/1/3
8H	Valve	600.066/1/3
9H	Boot-strap valve	600.066/1/3
10H	III O Only. Equipment temperature control circuit breaker	600.067/3
11H	III O Only. Valve control (locked in ON position)	600.067/3
12H	III O Only. Emergency cold control switch (locked in ON position)	600.067/3
13H	III O Only. Equipment temperature valve	600.067/3
14H	III O Only. Equipment temperature amplifier	600.067/3
15H	III O Only. Duct temperature sensing unit	600.067/3
16H	III O Only. Equipment temperature control valve	600.067/3
17H	III O Only. Limiting amplifier	600.066/1
18H	Oxygen system circuit breaker	600.068
19H	Remote control panel, gauge and flasher	600.068
20H	Altitude switch	600.068
21H	Normal oxygen regulator	600.068
22H	Electronic failure detector	600.068
23H	Cabin pressure capsule	600.068 — 600.067
24H	Altitude capsule	600.068 — 600.067
25H	Overpressure-underpressure capsule	600.067 — 600.068
40H	Canopy not locked microswitch	600.067 — 600.068
202H	III D Only. Rear cockpit air conditioning control panel	600.066
218H	III D Only. Oxygen system circuit breaker	600.068
219H	III D Only. Oxygen control panel	600.068
220H	III D Only. Pressure switch	600.068
221H	III D Only. Oxygen regulator assembly	600.068
222H	III D Only. Electronic oxygen failure detector	600.068
1J	Fire detection system circuit breaker	600.026
2J	Control box	600.026
3J	Fire detectors	600.026
4J	Engine fire warning light	600.026
5J	Fire detectors	600.026
6J	Fire detectors	600.026
7J	Afterburner fire warning light	600.026

Ident	Description	Drawing No
204J	III D Only. Engine fire warning light	600.026
207J	III D Only. Afterburner fire warning light	600.026
1K	Ignition system fuse	600.021
2K	Ignition contactor	600.021
3K	Ignition box	600.021
4K	Starting box	600.021
5K	Starting system circuit breaker	600.021
6K	Starting relay	600.021
7K	Engine starting button	600.013 — 600.021
8K	In-flight relight switch	600.021
9K	Ignition — ventilation switch	600.021
10K	Starting system fuse	600.002 — 600.021
11K	Ionization probe circuit breaker	600.022/3
12K	Afterburner d.c. fuse breaker	600.022/3
13K	Afterburner valve control switch	600.022/3
14K	Afterburner microswitch	600.022/3
15K	Afterburner operating indicator light	600.022/3
16K	Afterburner injector indicator light	600.022/3
17K	Afterburner emergency injector button	600.022/3
18K	Afterburner valve	600.022/3
19K	Afterburner relay	600.022/3 — 600.029/3
20K	Afterburner electro-valve	600.022/3
21K	Ionization probe amplifier	600.022/3
22K	III O Only. Rocket motor relay control circuit breaker	600.023/3
23K	III O Only. Rocket motor power circuit breaker	600.023/3 — 600.027
24K	III O Only. Rocket motor control circuit breaker	600.023/3 — 600.027
25K	Regulation system circuit breaker	600.028
26K	Approach speed control system lever	600.028
27K	Emergency FCU control	600.028
28K	III O Only. Emergency FCU actuator control key	600.028
29K	III O Only. Rocket motor safety switch	600.023/3
30K	III O Only. Rocket motor switch	600.023
31K	III O Only. Rocket motor relay	600.023/3
32K	III O Only. Timer	600.023/3
33K	III O Only. Ground cut-out connector	600.023/3
34K	III O Only. Rocket motor engage hold indicator light	600.023/3
35K	III O Only. Thrust indicator light	600.023/3
36K	III O Only. Rocket motor fire warning light and slow drain valve circuit breaker	600.024/1/3
37K	III O Only. Detector box	600.024/1/3
38K	III O Only. Rocket motor fire warning light	600.024/1/3
39K	III O Only. Slow drain valve control	600.024/1/3
40K	III O Only. Slow drain indicator light	600.024/1/3
40K	III D Only. Emergency FCU actuator control key	600.028

Ident	Description	Drawing No
42K	Starting harness	600.021
43K	Fuel control unit	600.021
44K	Supply electro-pump	600.021
45K	Three-way electro-valve	600.021
46K	Starter	600.021
47K	Ignition coil	600.021
49K	Overspeed system circuit breaker	600.029/3
50K	Cell	600.021
51K	Time-delay relay	600.029/3
52K	Overspeed contact	600.029/3
53K	Overspeed pushbutton	600.029/3
54K	Overspeed indicator light	600.029/3
213K	III D Only. Afterburner cock control switch	600.022
215K	III D Only. Afterburner on indicator light	600.022
216K	III D Only. Afterburner indicator light	600.022
254K	III D Only. Overspeed indicator light	600.029
1L	Dial lighting system circuit breaker	600.061/1/3 — 600.060
2L	Rheostat box	600.061/1/3 — 600.060
3L	White light circuit breaker	600.061/1/3 — 600.060
4L	UV light circuit breaker	600.061/1/3 — 600.060
5L	Map light circuit breaker	600.061/1/3 — 600.060
6L	Resistor box	600.061/1/3 — 600.060
7L	III O Only. Lighting assembly	600.061/1/3
8L	III O Only. Lighting assembly	600.061/1/3
10L	Navigation light circuit breaker	600.062/3
11L	Navigation light switch	600.062/3
12L	Flasher	600.062/3
13L	Tail light	600.062/3
14L	Map light microswitch	600.061/1/3 — 600.060
15L	Navigation light	600.062/3
16L	Navigation light	600.062/3
17L	Light resistor box	600.062/3
18L	Formation light circuit breaker	600.062/3
19L	Formation light switch	600.062/3
20L	Formation light	600.062/3
21L	Formation light	600.062/3
22L	Winding reel	600.061/1/3 — 600.060
23L	Landing light circuit breaker	600.062/3
24L	Landing light contactor	600.062/3
25L	Map light	600.061/1/3 — 600.062
26Lg	Landing light	600.062/3
26Ld	Landing light	600.062/3
27L	Landing light switch	600.062/3

Ident	Description	Drawing No
28L	Standby magnetic compass/Clock light rheostat	600.061/1/3 — 600.060
29L	Clock lighting resistor	600.061/1/3 — 600.060
30L	Clock light	600.061/1/3 — 600.060
31L	PHI lighting resistor	600.061/1
32L	Landing light control circuit breaker	600.062/3
33L	Landing light contactor	600.062/3
34L	Landing light microswitch	600.062/3
37L	Dimmer cap, lamp	600.061/1/3 — 600.060
39L	Dimmer cap, lamp	600.061/1/3 — 600.060
40L	PHI lighting circuit breaker	600.061/1/3 — 600.060
41L	Light rheostat — front cockpit	600.060
42L	PHI lighting transformer	600.060
44L	Lighting unit — RH panel	600.060 — 600.061
45L	III D Only. LH console light assembly — front cockpit	600.060
47L	III D Only. LH console light assembly — front cockpit	600.060
48L)		
49L)	III D Only RH console light assembly — front cockpit	600.060
50L		
51L	III D Only PHI lighting transformer resistor	600.060
52L	III D Only PHI lighting transformer resistor	600.060
53L	III D Only Standby gyro-horizon light	600.060
54L	III D Only RH console light assembly — front cockpit	600.060
55L	III D Only LH console light assembly — front cockpit	600.060
56L	III D Only Instrument panel light — rear cockpit	600.061
57L	III D Only Instrument panel light — rear cockpit	600.061
58L)	III D Only. Canopy light assembly	600.061
59L)		
60L)		
61L)	III D Only. LH console light assembly — rear cockpit	600.061
62L)		
63L)	III D Only. RH console light assembly — rear cockpit	600.061
64L)		
65L	III D Only. RH console light assembly — front cockpit	600.060
66L	III D Only. Flight crew interphone resistor	600.060
67L	III D Only. LH console light assembly — front cockpit	600.060
68L	III D Only. Ground telephone resistor	600.061
69L	III D Only. Instrument panel light rheostat	600.061
70L	III D Only. Instrument panel light — rear cockpit	600.061
71L	III D Only. Light resistor	600.060 — 600.061
171L	III D Only. LH console light assembly — rear cockpit	600.061
202L	III D Only. Light rheostat box — rear cockpit	600.061
206L	III D Only. Light resistor box	600.061
214L	III D Only. Map light microswitch — rear cockpit	600.061
222L	III D Only. Map light winding reel — rear cockpit	600.061

Ident	Description	Drawing No
225L	III D Only. Map light — rear cockpit	600.061
241L	III D Only. Light rheostat — rear cockpit	600.061
242L	III D Only. PHI lighting transformer — rear cockpit	600.061
251L	III D Only. PHI lighting transformer resistor	600.061
1M	Fuselage store jettison system circuitbreaker	660.005/1
2M	III O Only. Fuselage store jettison button	660.005/1
3M	III O Only. Jettison system resistor	600.024/1/3
4M	III O Only. Jettison system resistor	600.024/1/3
5M	III O Only. Retraction squib	600.024/1/3
7M	III O Only. Arm retracted microswitch	600.024/1/3
8M	III O Only. Jettison squib	600.024/1/3
9M	III O Only. Jettison squib	600.024/1/3
10M	Seat adjustment system circuit breaker	600.069
11M	III O Only. Seat adjustment system	600.069
11M1	III D Only. Seat adjustment system — front cockpit	600.069
11M2	III D Only. Seat adjustment system — rear cockpit	600.069
12M	III O Only. Door open actuator circuit breaker	600.013
13M	III O Only. Control box	600.013
14M	III O Only. Door actuator	600.013
15M	III O Only. Door closed actuator circuit breaker	600.013
16M	III O Only. Door actuator relay	600.013
17M	Ground receptacle door microswitch	600.013
18M	Ground receptacle door indicator light	600.013
20M	Recorder circuit breaker	990.001
21M	Recorder control fuse	990.001
22M	Recorder control switch	990.001
23M	Recorder starting button	990.001
24M	Unwinding reel indicator	990.001
25M	Recorder	990.001
26M	Fuel recorder	990.001
27M	Temperature measurement box	990.001
28M	Temperature sensing unit	990.001
29M	Roll rate gyro	990.001
30M	Yaw rate gyro	990.001
31M	Pitch rate gyro	990.001
32M	Accelerometer	990.001
33M	Test system circuit breaker	990.001
35M	III D Only. Canopy control fuse breaker	600.071
36M	III D Only. Canopy fuse breaker	600.071
37M	III D Only. Canopy opening button	600.071
38M	III D Only. Canopy closing button	600.071
39M	III D Only. External canopy control switch	600.071
40M	III D Only. Canopy control relay	600.071

Ident	Description	Drawing No
41M	III D Only. Canopy control relay	600.071
42M	III D Only. Canopy actuator	600.071
43M	III D Only. Canopy opening limit switch	600.071
44M	III D Only. Canopy closing limit switch	600.071
45M	III D Only. Microswitch	600.071
50M	III D Only. Cell	600.072
51M	III D Only. Release microswitch box	600.072
52M	III D Only. LH initiator (cell)	600.072
53M	III D Only. RH initiator (cell)	600.072
54M	III D Only. LH initiator (battery)	600.072
55M	III D Only. RH initiator (battery)	600.072
56M	III D Only. Canopy jettison system fusebreaker	600.072
57M	III D Only. Resistor	600.072
50M	III O Only. Power supply fuse	894.001 (Photo Nose Cone A)
51M	III O Only. Power supply fuse	894.001 (Photo Nose Cone A)
52M	III O Only. Power supply fuse	894.001 (Photo Nose Cone A)
53M	III O Only. Power supply fuse	894.001 (Photo Nose Cone A)
54M	III O Only. Power supply fuse	894.001 (Photo Nose Cone A)
55M	III O Only. Power supply fuse	894.001 (Photo Nose Cone A)
56M	III O Only. Heater relay	894.001 (Photo Nose Cone A)
57M	III O Only. Valve control relay	894.001 (Photo Nose Cone A)
58M	III O Only. Heater	894.001 (Photo Nose Cone A)
59M	III O Only. Fan	894.001 (Photo Nose Cone A)
60M	III O Only. Thermostat	894.001 (Photo Nose Cone A)
61M	III O Only. Thermostat	894.001 (Photo Nose Cone A)
62M	III O Only. Thermostat	894.001 (Photo Nose Cone A)
63M	III O Only. Thermostat	894.001 (Photo Nose Cone A)
64M	III O Only. By-pass valve	894.001 (Photo Nose Cone A)
65M	III O Only. FHS fluid pump	894.001
66M	III O Only. Temperature sensor	894.001 (Photo Nose Cone A)
67M	III O Only. Capacitor	894.001 (Photo Nose Cone A)
68M	III O Only. Capacitor	894.001 (Photo Nose Cone A)
69M	III O Only. Capacitor	894.001 (Photo Nose Cone A)
70M	III O Only. Capacitor	894.001 (Photo Nose Cone A)
71M	III O Only. Capacitor	894.001 (Photo Nose Cone A)
72M	III O Only. Camera control unit	660.033/3
73M	III O Only. Nose cone overheat warning light	660.033/3
74M	III O Only. Temperature indicator	660.033/3
75M	III O Only. Camera	894.002
76M	III O Only. Magazine	894.002
77M	III O Only. Junction box	894.002
78M	III O Only. DCU-8 unit	894.003
79M	III O Only. DAR 1 unit	894.003

Ident	Description	Drawing No
202M	III D Only. Fuselage door jettison button	660.004
210M	III D Only. Seat adjustment circuit breaker	600.069
1P	Generator	600.002
2P	Generator make-and-break switch	600.002
3P	Overvoltage protective system	600.002
4P	Voltage regulator	600.002
5P	Generator resetting system circuit breaker	600.002
8P	Generator control circuit breaker	600.002
10P	Battery	600.002
11P	Battery fuse	600.002
12P	Battery reverse current relay	600.002
13P	Battery circuit breaker	600.002
17P	External power receptacle	600.002 — 600.011
18P	External power receptacle contactor	600.002 — 600.011
19P	External power receptacle fuse	600.002 — 600.011
20P	Circuit breaker	600.002 — 600.011
21P	Isolating relay	600.002 — 600.011
22P	Load-shedding system circuit breaker	600.002
23P	Automatic load-shedding contactor	600.002 — 600.031
24P	Load-shedding switch	600.002
25P	Manual load-shedding contactor	600.002
26P	Load-shedding override switch	600.002 — 600.044
27P	Transformer rectifier safety contactor	600.002
28P	Power supply system fuse	600.002
29P	Power supply system fuse	600.002
30P	Power supply system fuse	600.002
31P	Automatic load-shedding system fuse	600.002
32P	Manual load-shedding system fuse	600.002 — 600.011
33P	Transformer rectifier fuse	600.002
34P	External power unit fuse	600.002 — 600.011
35P	Polarized magnetic relay	600.002
36P	Voltage test connector circuit breaker	600.002
37P	Voltage test connector	600.002
38P	III O Only. Preheating system circuit breaker	600.002 — 600.011
39P	III O Only. Preheating system relay	600.011 — 650.005 — 680.003/1/3
40P	III O Only. Preheating switch (locked in OFF position)	600.011 — 600.012
1Q	LH low pressure pump fuse	600.031
2Q	RH low pressure pump fuse	600.031
3Q	LH pump switch	600.031
4Q	RH pump switch	600.031
5Q	LH low pressure pump	600.031
6Q	RH low pressure pump	600.031
7Q	LH pump control circuit breaker	600.031

Ident	Description	Drawing No
8Q	RH pump control circuit breaker	600.031
9Q	LH pump control switch	600.031
10Q	RH pump control switch	600.031
11Q	LH pump noise suppressor	600.031
12Q	RH pump noise suppressor	600.031
15Q	Main LP cock circuit breaker	600.031
16Q	Main LP cock control switch	600.031
17Q	Main LP cock	600.031
19Q	Bay tank transfer valve switch (locked in 2 positions)	600.032/3
20Q	Fuel tank crossfeed switch	600.032/3
21Q	Fuel tank crossfeed valve	600.032/3
22Q	Bay tank transfer valve	600.032/3
23Q	Two-way relay box	600.032/3
24Q	Fuel tank crossfeed system circuit breaker	600.032/3
26Q	RH fuel tank unit	600.032/3 — 600.034
27Q	LH fuel tank unit	600.032/3 — 600.034
28Q	Fuel low level warning light	600.032/3
29Q	Fuel gauge	600.034
31Q	LH pylon tank pressure switch	600.033
32Q	RH pylon tank pressure switch	600.033
33Q	Push-to-test button	600.034
34Q	Fuel gauge circuit breaker	600.034
35Q	LH wing and rear fuselage tank pressure switch	600.033
36Q	RH wing and rear fuselage tank pressure switch	600.033
37Q	66 UK gal (300 l) tank pressure switch	600.033
38Q	RH amplifier	600.034
39Q	LH amplifier	600.034
40Q	Fuel transfer indicator	600.033
41Q	Fuel transfer indicator circuit breaker	600.033
43Q	Fuel tank jettison system circuit breaker	600.035/3
44Q	Fuel tank jettison button	600.035/3 — 660.005/1
45Q	Fuel dump valve fuse breaker	600.036
46Q	Fuel dump valve relay	600.036
47Q	Fuel dump valve control button	600.036
48Q	III D Only. Repeating amplifier RH	600.034
49Q	III D Only. Repeating amplifier LH	600.034
50Q	Fuel remaining indicator circuit breaker	600.037
51Q	Fuel remaining indicator	600.037
52Q	Transmitter	600.037
53Q	Fuel remaining indicator amplifier	600.037 — 990.001
216Q	III D Only. LP cock control switch	600.031
228Q	III D Only. Fuel low level warning light	600.032
229Q	III D Only. Fuel gauge indicator	600.034

Ident	Description	Drawing No
240Q	III D Only. Fuel transfer indicator	600.033
244Q	III D Only. Fuel tank jettison button	600.035
251Q	III D Only. Fuel remaining indicator	600.037
1R	III O Only. Standby connector	650.005
2R	III O Only. Matching box	650.005
3R	III O Only. Radio selector box	650.002/1 — 650.005
4R	III O Only. Normal UHF antenna	650.001
5R	Microphone press-to-talk button (stick)	650.002/1 — 650.005
6R	Radio normal switch circuit breaker	600.011 — 650.005
7R	Radio preheating switch circuit breaker	600.011 — 650.005
8R	III D Only. Filter	650.005
8R	III O Only. Amplifier	650.002/1
9R	III O Only. Mixing amplifier filter	650.002/1
9R	III D Only. Junction box	650.005
10R	Microphone press-to-talk button (throttle)	650.002/1 — 650.005
11R	III O Only. Connector	600.069
11R1	III D Only. Connector	600.069
11R2	III D Only. Connector	600.069
12R	III O Only. Microphone	600.069
12R1	III D Only. Earphones	600.069
12R2	III D Only. Earphones	600.069
13R	III O Only. Throat microphone	600.069
13R1	III D Only. Microphone	600.069
13R2	III D Only. Microphone	600.069
14R	Coaxial connector	650.001
16R	Standby UHF circuit breaker	650.004
17R	UHF antenna	650.004
18R	UHF antenna	650.004
19R	Coaxial tee-connector	650.004
20R	Fin connector	650.004
21R	Power supply unit	650.004
22R	UHF transmitter receiver	650.004
23R	UHF control panel	650.004
25R	UHF circuit breaker	650.001
26R	UHF transmitter receiver	650.001
27R	Matching transformer	650.001
28R	UHF control panel	650.001
29R	III D Only. Power supply unit	650.001
30R	III D Only. Ground telephone door microswitch	650.001
31R	III D Only. Ground telephone connection	650.001
33R	III D Only. Coaxial connector	650.001
34R	Coaxial connector	650.004
35R	TACAN d.c. power circuit breaker	650.006/1/3

Ident	Description	Drawing No
36R	TACAN a.c. power circuit breaker	650.006/1/3
37R	TACAN transmitter-receiver	650.006/1/3
38R	Control panel	650.006/1/3
39R	Bottom antenna	650.006/1/3
40R	Top antenna	650.006/1/3
41R	Antenna relay switch	650.006/1/3
42R	Antenna selector switch	650.006/1/3
43R	III O Only. AC circuit breaker	650.007/3
43R	III D Only. Selector switch	650.005
44R	III O Only. AC circuit breaker	650.007/3
44R	III D Only. Rectifier	650.005
45R	III O Only. Connection box	650.007/3
46R	III O Only. Radio-altimeter indicator	650.007/3
47R	III O Only. Receiving antenna	650.007/3
48R	III O Only. Transmitting antenna	650.007/3
49R	III O Only. Transmitter-receiver	650.007/3
50R	III O Only. Antenna switch	650.007/3
52R	III O Only. Connection box	650.007/3
70R	Coaxial connector	650.006
71R	Coaxial connector	650.006
203R	III D Only. Selector box	650.004
205R	III D Only. Microphone press-to-talk button	650.005
223R	III D Only. UHF 2 control box	650.004
243R	III D Only. Radio selector box	650.005
1S	III O Only. Radar fuse breaker	600.011
2S	III O Only. Radar normal a.c. circuit breaker	600.012
3S	III O Only. Radar preheating a.c. circuit breaker	600.012
4S	III O Only. Radar normal preheating relay	600.012
5S	III O Only. DC supply fuse breaker	650.003/1
6S	AC supply fuse breaker	650.003/1
7S	Transponder	650.003/1
8S	IFF control box	650.003/1
9S	Transmitter transponder coaxial cable	650.003/1
10S	Antenna coaxial cable	650.003/1
11S	IFF antenna	650.003/1
12S	Distress switch	650.003/1
13S	III O Only. Programming box H	660.017 — 660.018
14S	III O Only. Programming box G	660.017 — 660.018
15S	III O Only. Radar circuit breaker	660.017
16S	III O Only. DC power relay	660.017
17S	III O Only. AC power relay	660.018
19S	III O Only. Echo unlock relay	660.006
20S	III O Only. Radar nose cone	660.017 — 660.018

Ident	Description	Drawing No
20S	III O Only. Special nose cone	660.024/3 — 660.033/3 — 894.002 — 894.003 (Photo Nose Cone A)
21S	III O Only. Antenna (not installed)	660.035
22S	III O Only. Pre-amplifier (not installed)	660.035
23S	III O Only. Coaxial connector	660.035
24S	III O Only. Coaxial cable (not installed)	660.035
25S	III O Only. Maintenance connector	660.017 — 660.018
26S	III O Only. Accelerometer	660.026 — 660.012
28S	III O Only. Radar scope	660.017 — 660.018
28S	III O Only. Camera control mount	660.033/3 (Photo NoseCone A)
29S	III O Only. Radar control stick	660.019
30S	III O Only. Failure relay	650.003/1
31S	III O Only. Control relay	650.003/1
32S	III O Only. Transponder test set	650.003/1
35S	Tail warning radar control circuit breaker	660.035
37S	Tail warning radar control	660.035
38S	Tail warning radar relay	660.035
39S	Tail warning radar indicator	660.035
40S	Receiver	660.035
42S	III O Only. Radar filter	660.017 — 660.018
44S	III O Only. Radar d.c. circuit breaker	660.017
45S	III O Only. Echo unlock line fuse	660.006 — 660.017
1V	Vario-alternator	600.006/1
2V	Energization box	600.006/1
3V	Detector box	600.006/1
4V	Protection panel	600.006/1
5V	Starting system circuit breaker	600.006/1
6V	Line contactor	600.006/1 — 600.012
7V	Resetting system circuit breaker	600.006/1
11V	External power receptacle	600.006/1 — 600.012
12V	External power receptacle contactor	600.006/1 — 600.012
13V	Overheat detector	600.006/1
14V	Inverter circuit breaker	600.008
15V	Inverter	600.008
16V	Inverter switch	600.008
17V	Noise suppressor	600.006/1
18V1	Test connection	600.006/1
18V2	Test connection	600.006/1
19V	Servo-control relay	600.008
20V	Servo-control detector	600.008
25V	III O Only. Voltage computer a.c. power circuit breaker	680.002/3
26V	III O Only. Voltage computer power supply unit	680.002/3 — 680.014/3
30V	Safety relay	660.006/1 — 600.012

Ident	Description	Drawing No
1Z	Failure warning panel	600.064/1/3 — 600.065 — 600.002 — 600.066 — 600.062 — 600.067 — 600.070
2Z	Indicator light circuit breaker	660.064/1/3 — 600.063 — 600.065
3Z	Failure warning light	600.064/1/3 — 600.065
5Z	Warning horn circuit breaker	600.065
6Z	Warning horn switch	600.065
7Z	Warning horn	600.065
8Z	Matching transformer	600.065
9Z	Transfer switch	600.002 — 600.006/1
11Z	Transformer rectifier	600.002
201Z	III D Only. Failure warning panel (rear cockpit)	600.070 — 600.002 — 600.064 — 600.065 — 600.066 — 600.067 — 600.068
203Z	III D Only. Warning horn cancel button (rear cockpit)	600.065

CHAPTER 12

FLIGHT INSTRUMENTS

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CHAPTER 12

FLIGHT INSTRUMENTS

Table 12-1 Function Of Components

Index No	Description	Characteristics and Functions
AERODYNAMICS PARAMETERS		
1C	Air data computer	The air data computer receives : a. Pitot pressure (T1) and static pressure (S2) from the nose pitot-static tube (7). b. Total temperature (Tt) from the total temperature probe (61C). From these parameters, it computes and transmits data to the ADC output multiplier (2C), or to the PHI computer (37F), or to aircraft systems (engine overspeed system, Mach warning light illuminating system).
2C	Output multiplier	The ADC output multiplier repeats and distributes the data from the air data computer (1C) and the incidence probe (12F). It computes and distributes the true angle of attack from the indicated angle of attack provided by the incidence probe.
61C	Temperature probe	This probe measures total temperature (Tt) data which it transmits to the air data computer (1C) in the form of electrical signals.
1(front cockpit) 4(rear cockpit)	ASI/Machmeter	Covers pressure-altitude range of 1000 to 80000 ft; it features : a. A pointer indicating IAS values from 80 to 850 knots. b. A dial showing Mach numbers from 0.7 to 2.5 M. c. A knob for adjustment of the index.
2(front cockpit) 5(rear cockpit)	Altimeter	Indicates pressures-altitudes from 1000 to 80000 ft; it features : a. A medium pointer graduation .... 100 ft b. A short pointer graduation .... 1000 ft c. A triangular pointer graduation 10000 ft. The triangular pointer displays a hatched area at altitudes below 16000 ft : d. A drum indicating the pressure setting between 905 and 1045 mb (26.72 and 30.86 in of mercury). e. A knob for pressure setting adjustments.
3(front cockpit) 6(rear cockpit)	Vertical speed indicator	Indicates vertical speeds from 0 to 2500 ft/min.
7	Nose pitot-static tube	Collects pitot and static pressures. a. Pitot (total) pressure : T1 to air data computer (1C), dash-pot IAS capsule and ASI/machmeter (4) of rear cockpit. b. Static pressure : (1) S1 to ASI/machmeter (1), altimeter (2) and vertical speed indicator (3) of front cockpit. (2) S2 to ASI/machmeter (4), altimeter (5), air data computer (1C) and auto-command amplifier box (120C) and vertical speed indicator for rear cockpit, Post-Mod 835. (3) S3 to auto-command amplifier box (120C). The nose pitot-static tube is heated by a heating resistor (14D).
6(III O) 8(III D)	Curved pitot tube	Collects pitot pressure T2 for the front cockpit ASI/machmeter (1). Heated by a heating resistor (12D).
5(III O) 9(III D)	Curved pitot tube	Collects pitot pressure T3 for the undercarriage system speed sensitive capsule (52G), the auto-command amplifier box (120C) and the engine approach speed control system. Heated by a heating resistor (13D).
ATTITUDE AND HEADING REFERENCES		
8F	Flux valve	Detects the horizontal component of the earth's magnetic field which it transmits to the Bezu output multiplier box (27F).
24F	Gyro centre control panel (also referred to as heading selector)	Includes : a. A fast erect pushbutton. b. A five-position heading selector.

Index No	Description	Characteristics and Functions
		c. A switch with an associated indicator light for heading transfer and fast gyromagnetic heading resetting.
25F	Electronic Unit	Performs two functions as follows : a. Supplies the gyro centre (26F). b. Amplifies the servo signals of the vertical and directional gyros of the gyro centre (26F).
26F	Gyro reference unit	This assembly consists of two gyros : a. A vertical gyro providing the roll and pitch references. b. A directional gyro providing the gyroscopic heading references.
27F	Bezu output multiplier box	Repeats and distributes roll, pitch and heading information to various aircraft installations and to the navigation and weapon systems. Computes gyromagnetic heading and transmits standby magnetic heading.
28F (front cockpit) 228F (rear cockpit)	Spherical indicator	Displays roll, pitch and heading information by means of a sphere which has three axis freedom and moves behind a miniature aircraft.Also provides slip indication.
37F	PHI computer	Performs the computations and switching necessary for navigation purposes.
38F (front cockpit) 238F (rear cockpit)	PHI indicator	Displays aircraft heading, as well as heading to and range from a selected station. Includes a switch for selection of the operating mode. Incorporates control for manual correction of displayed heading and range.
EMERGENCY ATTITUDE AND HEADING REFERENCES		
16F	Standby gyro horizon inverter	This inverter is supplied by the main d.c. bus; it provides 115 V, 400 Hz three-phase a.c. power for operation of the standby gyro horizons (17F-217F). Power output : 10 VA.
17F (front cockpit) 217F (rear cockpit)	Standby gyro horizon	Provides roll and pitch information; powered by inverter (16F).
30	Standby magnetic compass	Indicates magnetic heading.
31(front cockpit) 231(rear cockpit)	Accelerometer	Indicates aircraft acceleration from 10 g to -5 g; it features an indicating pointer and two maximum-minimum pointers. Resetting to zero is accomplished by pressing a button.
LIMITATION INDICATING SYSTEMS		
11E(front cockpit) 211E(rear cockpit)	MACH warning light	This light is controlled by the air data computer (1C); it illuminates when the total temperature sensed by the probe (61C) reaches 125°C.
11F	Incidence probe amplifier	Converts the electrical data received from the incidence probe (12F) into signals controlling the illuminating sequence of the indicator lights on the incidence indicators (13F-213F).
12F	Incidence probe	Senses aircraft incidence angle and converts this angle into an electrical control signal which it transmits to the incidence probe amplifier (11F) and the air data computer output multiplier (2C).
13F(front cockpit) 213F(rear cockpit)	Incidence indicator	Consists of three indicator lights (green-amber-red) with dimming provisions. The combined illumination of the lights provides a simple means of instant incidence angle readout.

GENERAL

Description

1201. The pupose of the flight instruments is to supply the pilot with all the data required to control the flight of the aircraft.

1202. The following systems are provided :

a. The aerodynamic parameters (total and static pressures and impact temperature) sensing system.

b. The roll, pitch and heading references determined by gyroscopic and magnetic instruments.

c. The emergency attitude and heading references determined by gyroscopic and magnetic instruments.

d. The limitation devices whereby warning is given to the pilot in the event of the aircraft being engaged in hazardous flying conditions.

PITOT-STATIC SYSTEM — III O ONLY

Description (refer to Figs 12-1 to 12-5)

1203. The pitot-static system supplies the following instruments :

a. ASI-machmeter (1).

b. Altimeter (2).

c. Vertical speed indicator (3).

d. Air data computer (1C).

e. SFENA auto-command box (120C).

f. Dash-pot jack IAS capsule (57).

g. Engine approach control unit.

h. Undercarriage warning system airspeed capsule (52G).

Pressure Inlets (refer to Fig 12-3)

1204. The following pressure inlets are provided :

a. Nose pitot-static tube (4) including the pitot inlet T1 and the static inlets S1, S2 and S3.

b. Curved pitot tube (5) inlet T3 — RH side, between frames 1 and 2.

c. Curved pitot tube (6) inlet T2 — LH side, between frames 1 and 2.

d. Static port (8) inlet S4 — LH side frame 37.

Installation Of The Nose Pitot-Static Tube (refer to Figs 12-4 and 12-5)

1205. The pitot-static tube can be mounted either :

a. On the radar nose cone, the pitot-static tube is directly mounted on the cylindrical end-fitting of the radome.

b. On the ballasted nose cone, the pitot-static tube is mounted at the end of a tubular boom.

1206. **Attachment to Radar Nose Cone (refer to Fig 12-4).** The pitot-static tube (4) for photo rec. attaches to the cylindrical end-fitting (22) of the radome through a sliding sleeve (21). The pitot-static system connections are made by conical couplings (24) and the electrical connections by two pins.

1207. The assembly is sealed by means of two rings (25), one installed on the end-fitting, the other on the pitot-static tube. Attachment of the end-fitting to the sleeve and clamping of the pitot-static tube to the end-fitting is by six countersunk-head screws (20 and 23) distributed every 120° in two parallel planes.

1208. **Attachment to Ballasted Nose Cone (refer to Fig 12-5).** The pitot-static tube is directly mounted on a boom extending forward of the nose cone; the method of attachment and the connections are the same as in the case of the radar nose cone except for four additional couplings provided at the rear of the boom to facilitate installation.

1209. **Attachment of Boom to Ballasted Nose Cone (refer to Fig 12-5).** The boom is attached to the ballasted nose cone as follows :

a. *Front Attachment.* A threaded sleeve (10) integral with the structure takes a screwed bushing (11) which clamps the boom through a split tapered bush (12). The bushing (11) is secured by a stop screw (13) retained by a ring (14). The assembly is sealed by a seal (15) accommodated in a circular groove.

b. *Centre Attachment.* A threaded sleeve (16) integral with nose cone frame A takes a screwed bushing (17) which clamps the boom through a split tapered bush.

c. *Rear Attachment.* A plain bushing (19), attached to the nose cone at frame C, acts as a guide and stop for the boom and determines its end position at the rear. A bonding braid is attached to the boom by means of a clip (18).

Installation Of Curved Pitot Tubes (refer to Fig 12-3)

1210. The curved pitot tubes are mounted at frame 1 with shims fitted between the tube flanges and the fuselage skin. Four bolts are used for attachment.

Connections (refer to Fig 12-4)

1211. At the nose cone-to-fuselage junction, the total and static lines are connected by means of quick-release couplings each consisting of :

a. A body (27) integral with the frame.

b. A sliding assembly (28) with seals (29).

c. A knurled screw (30) with a spring (31).

1212. When the nose cone is attached to the fuselage, the assembly mates with the nose cone system adapters (32) as a result of the pressure applied by the spring (31) through tightening of the knurled screw (30). Sealing is ensured by compression of seal (29) and the knurled screw is prevented from rotating by a spring-loaded lock (33). A shim riveted to the access door prevents the lock releasing.

1213. To provide the necessary flexibility, the fuselage system is connected to the nose cone system by hoses (34).

Table 12-2 Pitot-Static Connections to Equipments - III O Only

Equipment	Pressure Pitot	Pressure Static
ASI-machmeter (1)	T2	S1
Altimeter (2)		S1
Vertical speed indicator (3)		S1
Air data computer (1C)	T1	S2
Auto-command (120C)		S2
Altitude lock (in 120C)	T3	S3
Engine approach speed control unit	T3	S4
Undercarriage warning system air-speed capsule (52G)	T3	Ambient
Dash-pot jack IAS capsule	T1	Ambient

Pitot Heating (refer to Fig 12-3).

1214. **Purpose.** A built-in electrical heating resistor is provided to prevent pitot inlet icing. The system includes :

a. Nose pitot-static tube (4) ..Resistor 14D

b. Curved pitot tube (5) ....Resistor 13D

c. Curved pitot tube (6) ....Resistor 12D

1215. **Operation.** The PROBE HEAT switch (11D) controls pitot tube heating. Closing this switch causes :

a. Relay (8D) to open and extinguish the PITOT light on the failure warning panel (1Z).

b. Supply of current to heating resistor (12D) of the curved pitot tube (6).

c. Relay (9D) to close and supply, simultaneously, resistor (14D) of the nose pitot-static tube, through the solenoid of relay (10D) and resistor (13D) of the curved pitot tube (5).

d. Supply of current to the incidence indicator amplifier (11F).

1216. In case of failure of the nose pitot-static tube heating system, the relay (10D) solenoid, which is connected in series in the resistor (14D), is no longer supplied and causes the PITOT light to illuminate on the failure warning panel (1Z). The same warning is provided in the event of a failure of the curved pitot tube (6) heating system through the action of relay (8D). No warning is provided in case of failure of the curved pitot tube (5) heating system.

1217. The electrical heating circuit of the nose pitot-static tube (4) is connected by means of a connector (20S) mounted on a fitting screwed at the bottom of frames F and G.

Pitot-Static System Bleeds (refer to Figs 12-2 and 12-3)

1218. Bleeds are provided at the bottom points of the various total and static pressure lines as follows :

a. T1, T3, S2 and S3 on the RH side of the nose wheel well.

b. T2 between frames 1 and 2, on the LH side.

c. T3 and S4 at frame 25 in the port main wheel well.

Temperature Probe (refer to Fig 12-8)

1219. The probe temperature (61C) is located between frames 15 and 16 on the underside of the fuselage on the RH side. It is mounted on two shims (1) and (2) that conform to the fuselage contour. The flange of the probe and the shims are secured to the skin by four screws. The probe is provided with a positioning stud (4).

PITOT-STATIC SYSTEM — III D ONLY

Description (refer to Figs 12-6 and 12-7)

1220. The pitot-static system operates the following equipment :

a. ASI/Machmeter (1 — front cockpit 4 — rear cockpit).

b. Altimeter (2 — front cockpit 5 — rear cockpit).

c. Vertical speed indicator (5 — front cockpit 6 — rear cockpit).

d. Air data computer (1C).

e. Auto-command amplifier box (120C).

f. IAS capsule of the dash-pot (57).

g. Undercarriage warning system speed sensitive capsule (52G).

h. Engine approach speed control system.

Pressure Inlets (refer to Fig 12-7)

1221. The following pressure inlets are provided :

a. Nose pitot-static tube (7) including the pitot inlet T1 and static inlets S1, S2, and S3.

b. Curved pitot tube (8), inlet T2 — LH side frame 2.

c. Curved pitot tube (9), inlet T3 — RH side frame 2.

d. Static port (10), inlet S5 — RH side frame 2.

e. Static port (11), inlet S6 — LH side frame 37.

Installation Of Pitot-Static Tube On Nose Cone (refer to Fig 12-6)

1222. The pitot-static tube is attached to the front of the nose cone at frame 1 and to the rear of the nose cone at frame 2.

1223. **Rear Attachment (A).** On the rear face of the nose cone, frame 2, is a mounting flange (21) provided with a locking screw (22). The tubular section (29) of the pitot-static tube is centred by means of a tapered bushing (23) with a clamping nut (24). A screw (22) permits the pitot-static tube to be correctly set relative to the aircraft symmetry plane.

1224. **Front Attachment (B).** A mounting flange (25) is provided at the front of the nose cone fairing. The tubular section (29) of the pitot-static tube is centred by a tapered bushing (28) and tightened by means of a sleeve (26) fitted with a seal (27). The assembly is locked by tightening a pin (30).

1225. **Points C and D.** The tubular section (29) containing the pressure lines is extended by a spacer (37) retained by screws (31). The two pitot and the two static line connections are located in the centre of the spacer. The probe (32) is attached to the spacer by screws (33). The four lines are extended by flexible sleeves (34). In the rear part of the tubular section (29) the lines are clamped by two shim washers (35). The pitot-static tube is grounded by means of a strip (36) secured to frame 2.

Installation Of Curved Pitot Tubes

1226. The curved pitot tubes are installed at frame 1 by means of four bolts. Formed shims are fitted between the tube mounting flange and the aircraft skin.

Connections

1227. The pitot and static lines are interconnected by means of ARSAERO couplings and coupling blocks. Connections for the installation of the nose cone to the fuselage are described in Chapter 4.

Table 12-3 Pitot-static Connections to Equipments — III D Only.

Equipment	Pressure Pitot	Pressure Static
ASI/Machmeter (1)	T2	S1
ASI/Machmeter (4)	T1	S2
Altimeter (2)		S1
Altimeter (5)		S2
Vertical speed indicator (3)		S1
Vertical speed indicator (6)		S2
Air data computer (1C)	T1	S2
Auto-command amplifier box (120C)	T3	S2-S3
Engine approach speed control system	T3	S6
Undercarriage warning system speed sensitive capsule (52G)	T3	
IAS capsule of dash-pot (57)	T1	

Pitot Heating (refer to Fig 12-7)

1228. **Purpose.** A built-in electrical heating resistance is provided to prevent pitot inlet icing. The system includes :

a. Nose pitot static tube (7) ..Resistor 14D

b. Curved pitot tube (8) ....Resistor 12D

c. Curved pitot tube (9) ....Resistor 13D

1229. **Operation.** The pitot tube heating is controlled by the PROBE HEAT switch (11D). The closing of this circuit results in :

a. Opening the relay (8D) which cuts-off the power supply to the PITOT warning lights located on the failure warning panels (1Z front cockpit — 201Z rear cockpit).

b. Supplying the heating resistors (12D and 13D) located in the curved pitot tubes (8 and 9).

c. Closing relay (9D) which simultaneously supplies the resistor (14D) of the nose pitot-static tube (7) through the solenoid of the relay (10D) and the resistor (13D) of the curved pitot tube (9).

d. Supplying the incidence probe amplifier (11F) with d.c. power.

1230. In the event of failure of the heating system for the nose pitot-static tube (7) (resistor 14D), the solenoid of the relay (10D), connected in series with the resistor (14D), is no longer energized and causes the PITOT warning lights to illuminate on the failure warning panels (1Z front cockpit — 201Z rear cockpit). The same warning system is operated in case of failure of the heating system of the curved pitot tube (8); the relay (8D) in this case is operated. No warning is provided in the event of non-heating of the curved pitot tube (9).

Pitot-Static System Bleeds (refer to Figs 12-6 and 12-7)

1231. The following bleeds are provided at the bottom of the pitot and static pressure lines :

a. S1 and T2 on the front face of frame 2, lower section.

b. S2, S3, S5, T1 and T3 on the rear face of frame 10, lower section.

c. T3 and S6 between frames 23 and 24, lower section.

Temperature Probe (refer to Fig 12-8)

1232. The temperature probe (61C) is located between frames 15 and 16 on the underside of the fuselage, to the right. It is carried on two adapters (1) and (2) conforming to the fuselage contour. The flange of the probe and the adapters are secured to the skin by means of screws (3). A pin is provided for correct positioning of the probe.

AIR DATA COMPUTER

Description (refer to Fig 12-9)

1233. The air data computer (1C) determines the parameters listed below from the pitot pressure T1, the static pressure S2 (provided by the nose pitot-static tube) and from temperature information supplied by the total temperature probe :

a. Static pressure Ps.

b. Impact pressure qc.

c. Mach number M.

d. True airspeed Vt.

e. Pressure altitude H.

f. Total temperature Tt.

1234. Some of these parameters are directly transmitted by the air data computer, the others are transmitted to the output multiplier (2C) which distributes them in the form of electrical signals.

1235. The air data computer consists of the following units :

a. A static pressure detector.

b. An impact pressure detector.

c. A computation unit.

d. A power supply unit.

1236. Power supply for the air data computer is :

a. D.C. — 27 V from the manual load-shedding busbar via circuit breaker (62C).

b. A.C. — 115 V, 400 Hz single phase (phase B) via circuit breaker (63C).

1237. A full description of the air data computer system is contained in AAP7213.001-2-4, Section 2.9.

Operation (refer to Fig 12-9)

1238. The pitot and static pressures are directed to two groups of capsules. Each group forms a balance which is self-balanced by the variations in the ratio of the two balance beam lever arms. The following two forces are exerted at the ends of the balance beam :

a. The force due to the action of the pressure (variable force).

b. The force due to the action of a spring (constant force).

1239. To maintain a self-balanced position, the balance beam fulcrum is moved along the beam axis by a servo system which places the fulcrum in the position where the two forces are balanced. The measurements obtained are transmitted to the output multiplier and distributed to the various using systems through potentiometers.

Installation — III O Only (refer to Fig 12-10)

1240. The air data computer is mounted to a shock proof support (5), attached to the structure by four shock-mounts, in the equipment bay between frames 13 and 14. The air data computer is positioned on the support by two locating pins (7) and held in position by two screw-locks (6). Earthing of the unit is through the mounting system.

Installation — III D Only

1241. The air data computer is mounted in a housing provided in the centre of the rear cockpit slanting frame. The air data computer is centred on two studs located at the bottom of the mount and secured by means of screwed plates (refer to Chap 4 para 4201. The rear cockpit ejection seat must be removed to gain access to the air data computer.

OUTPUT MULTIPLIER

Description (refer to Fig 12-9)

1242. The output multiplier (2C) repeats and distributes the following information from the air data computer :

a. True airspeed Vt.

b. Mach number M.

c. Pressure altitude H.

d. Static pressure Ps.

e. Impact pressure qc.

1243. The output multiplier also computes the true angle of attack ($\propto$t) from indicated angle of attack ($\propto$i) sensed by the incidence probe and mach number (M). These references are transmitted to the following using circuits in the form of electrical signals :

a. Variable voltage signals by potentiometers.

b. Cut-out signals by contacts.

Installation — III O Only (refer to Fig 12-10)

1244. The output multiplier (2C) is mounted to the composite rack (1) located in the equipment bay between frames 14 and 15. It is guided into position by two rails (3) on the rack structure and held by two snap-fasteners (4) integral with the rack. Closing the snap-fasteners causes the electrical connections on the output multiplier to engage with the plugs at the bottom of the rack. The rack plugs are floating and are aligned with the output multiplier plugs by locating pins (2).

Installation — III D Only

1245. The output multiplier (2C) is installed in the upper part of the fuselage, to the rear of the slanting frame, between frames 15 and 17. It is attached to a platform and engaged on two longitudinal slide tracks and is held in position by two pins fitted through two mounting plates (refer to Chap 4, para 4204).

ATTITUDE AND HEADING REFERENCES SYSTEM

Description (refer to Fig 12-11)

1246. The gyroscopic reference system consists of the following units :

a. Gyro reference unit (26F).

b. Electronic unit (25F).

c. Bezu output multiplier box (27F).

d. Heading selector (gyro centre control panel) (24F).

e. Flux valve (8F).

f. PHI computer (37F).

g. PHI indicator — front cockpit (38F) — Rear cockpit (238F).

h. Station storage unit (39F).

i. Spherical indicator — front cockpit (28F) — rear cockpit (228F).

Gyro Reference Unit

1247. The gyro reference unit (26F) detects any deviation in pitch, roll and yaw. The corresponding information is transmitted in the form of electrical signals to the various systems, either directly through the electronic unit or through the bezu output multiplier box.

1248. The gyro reference unit basically consists of a directional gyro providing the gyroscopic heading references and a vertical gyro providing the roll and pitch references.

1249. These gyros are slaved by position pick-offs. Any position deviation is corrected by motors which move the gimbals according to the sensed position deviation. The gyros are also slaved to the vertical by mercury levelling switches.

1250. **Installation — III O Only (refer to Fig 12-12).** The gyro reference unit (26F) is mounted to a platform in the equipment bay between frames 12 and 13. The platform is shock mounted on a base which is attached to the structure at three points :

a. The forward point is a ball joint (2), integral with the gyro centre base, housed in a socket (3) attached to the bay floor. The ball joint is guided to the housing by a top hat section rail (1) riveted to the floor and forms the stationary support point of the gyro centre.

b. The rear points (4) and (5) consist of threaded pillars mounted on the floor; pillar height adjustment provides for initial settings as follows :

(1) Pitch .... 3° 28′ ± 5′ nose down.

(2) Roll .... ± 5′.

NOTE

Yaw setting is pre-set at ±10′.

1251. **Installation — III D Only.** The gyro reference unit is installed longitudinally between frames 23 and 24 at the lower part of the fuselage. It is attached at four points; two rear points provided with locking pins and two front points including special attaching bolts.

1252. The gyro reference unit, which is supplied with two mounting cradles, is correctly set when placed on the supports provided on the aircraft structure.

Electronic unit

1253. The electronic unit (25F) includes the gyro centre roll and pitch servo-amplifiers and the 400 Hz power supply required for its operation. The 400 Hz supply is also provided for the ball attitude indicators and the bezu output multiplier box.

1254. **Installation — III O Only (refer to Fig 12-12).** The electronic unit is accommodated in the equipment bay in front of frame 13 and is supported by a rack (6) which is attached to the structure by four shock mounts, two on the floor (7) and two on the bay RH side (8). The unit is held in position by two swivel locking levers (10) provided with hooks (11) integral with the unit. Closing the locking levers causes the electrical connectors on the gyro centre amplifier to engage with the plugs located on the rack. The rack plugs are floating and are aligned with the unit plugs by locating pins. Ventilation is provided by an air duct connected to the rear of the rack.

1255. **Installation — III D Only.** The electronic unit is located in the rear cockpit RH console between frames 12 b and 13. It is attached in a vertical position by four special safety locked nuts and provided with a bonding strip at its lower part. Access to the electronic unit is gained by removing the corresponding door (refer to Chap 3, door 533). Three electrical connectors are located at the rear of the electronic unit.

Bezu Output Multiplier

1256. The bezu output multiplier box (27F) performs the following three functions :

a. It computes the gyromagnetic heading.

b. It repeats and distributes roll, pitch and heading information to the required aircraft systems.

c. In the case of an alternator failure, it provides the power supply required to display the magnetic heading information, from the flux valve, by the moving compass card of the PHI indicators.

1257. The output multiplier distributes the following information :

a. Pitch and roll to the spherical indicators.

b. Gyromagnetic (or pure magnetic) heading to the gunsight, TACAN, PHI indicators and control unit and the spherical indicators.

1258. **Installation — III O Only.** The bezu output multiplier is mounted to the multiple rack (1) located in the equipment bay between frames 14 and 15. It is guided into position by two rails on the rack structure and held by two snap-fasteners (3) integral with the rack. Closing the snap-fasteners causes the electrical connectors on the output multiplier to engage with the plugs at the bottom of the rack. The rack plugs are floating and are aligned with the output multiplier plugs by locating pins (4). The rack is ventilated by opening (5).

1259. **Installation — III D Only.** The bezu output multiplier is installed in the front cockpit and constitutes the upper part of the control pedestal in front of the control stick. The output multiplier is centred on two studs in a support and held in position by a bolt which locks a folded plate fitting into a recess in the output multiplier. Electrical connections are made at the end of the box insertion motion (refer to Chap 4, para 4189).

Flux Valve

1260. The flux valve (8F) detects the direction of the earth's magnetic field by pendulously mounted internal coils. The magnetic heading is accurate only if the aircraft attitude remains near the horizontal position and the airspeed remains constant (no acceleration or deceleration).

1261. **Installation.** The flux valve is installed in the fin structure box between ribs 10 and 11 and is mounted on an adapter plate screwed to two supports secured to the aircraft structure. Access to the flux valve is gained by removing door 352, refer to Chap 3.

Control Panel (Heading Selector) (refer to Fig 12-11).

1262. The control panel (24F) is used to select various functions of the gyroscopic reference system and includes a five position heading selector having the following positions :

a. *NORMAL — NO TRA — DG :* These three positions have the same function. The heading fed by the output multiplier box to the various equipment is the gyromagnetic heading. The PHI computer receives the gyro heading added with the grivation set by the pilot on the PHI control unit.

b. *GYRO (FAIL) :* The magnetic heading is no longer monitored and is fed, in lieu of the gyromagnetic heading, to the same equipment as for the NORMAL position. This position is used to cut-off the gyro centre power supply without cutting-off the computing voltage.

c. *EMG AC Failure :* The output multiplier emergency a.c. supply is switched on and allows the pure magnetic heading to be transmitted to the PHI indicator compass cards. No other items of equipment are supplied.

1263. The control panel also has a TRANSFER switch, a fast erect pushbutton and an indicator light. The TRANSFER pushbutton type switch allows quick resetting of the ball attitude indicators on the gyromagnetic heading when the heading selector is in the NO TR, NORMAL or DG position.

1264. **Installation.** The gyro centre control panel (24F) is installed on the RH console of the front cockpit by means of screws.

Spherical Indicators (refer to Fig 12-15)

1265. The spherical indicators (28F front cockpit — 228F rear cockpit) repeat the information from the gyroscopic reference system to provide the pilots with a permanent and accurate display of the aircraft position in pitch, roll and heading through the complete freedom of the sphere. The spherical indicators include :

a. An index (5) at the bottom, which indicates bank angles.

b. A ball (6) which indicates the direction of the aircraft's apparent weight.

c. A knob (7) aligns the ball heading with the PHI in DG mode.

1266. **Installation.** The front cockpit indicator is mounted on the LH side of the instrument panel by means of screws and the rear cockpit indicator is mounted in the symmetry plane of the instrument panel by means of screws.

PHI Indicators

1267. The PHI indicators (38F front cockpit — 238F rear cockpit) permit dead reckoning navigation by displaying the same heading as the ball attitude indicators (28F front cockpit — 228F rear cockpit) by means of a compass card which moves opposite a fixed lubber line. The PHI indicators are mounted on the RH side of the instrument panels by means of screws. A description of the PHI navigation is given in Chap 14, para 1447.

EMERGENCY ATTITUDE AND HEADING REFERENCES

General (refer to Fig 12-16)

1268. These references are provided by the following instruments :

a. The gyro horizons and their inverter.

b. The standby magnetic channel.

c. The standby magnetic compass.

Gyro Horizons (refer to Fig 12-16)

1269. The purpose of the gyro horizons (17F front cockpit — 217F rear cockpit) is to provide roll and pitch references in case of failure of the gyro centre.

1270. Operation is based on the rigidity in space of a gyro slaved to the vertical by a ball erection system. The gyro horizon basically consists of a graduated drum (2) rotating about a horizontal axis. The drum is mechanically linked to the cage of the gyroscope for display of the pitch and roll positions. The pitch limitation of the instrument is 84°. There is no roll limitation.

1271. The horizon is represented by a line which separates the drum into two coloured zones, a light grey zone marked + for the sky and a black zone marked for the ground.

1272. Both of these zones are graduated every 10 degrees in pitch. The aircraft is represented by a miniature aircraft (3) integral with the instrument case. A control knob (4) is provided for vertical adjustment of the miniature aircraft. Aircraft bank angles are indicated by a roll index moving in front of fixed graduations.

1273. **Operation.** The gyroscope driving motor is supplied with 115V, 400 Hz three-phase power from the inverter (16F). Caging and resetting of the gyro are effected by pulling the control knob (4).

1274. Failure warning is provided by a flag (+) which appears on the dial in case of electrical supply failure. This warning device uses an electric motor (1) powered by the same source as the gyroscope driving motor.

Standby Magnetic Compass (refer to Fig 12-16)

1275. The standby magnetic compass is mounted at the top of the front cockpit windshield arch on the RH side to an adjustable support. The standby magnetic compass is of the moving-coil type and is filled with damping fluid. The brightness of the integral lighting system can be adjusted by a potentiometer located under the aircraft clock.

1276. **Compensation Adjustment.** To compensate for :

a. *A Error.* Rotate the bowl (8) by reference to the scale (9).

b. *B and C Errors.* Adjust the two compensating magnets by means of the corresponding screws (10) which are reached by rotating the cover (11).

Ball Mask On Spherical Indicator

1277. The ball mask is a turn time computer provided for evaluation of the time required to make a given heading change. The mask consists of two concentric discs; one disc is graduated in minutes (time) and the other disc is graduated in degrees over 360°. The computer is based on a speed of 300 knots, an altitude from 1500 to 25000 feet and a bank angle of 30°.

1278. The ball mask is used by aligning the existing heading with the time zero graduation and reading the time required to make the turn opposite the new desired heading; this flight parameter must then be adhered to when making the turn. To preclude the possibility of relying on erroneous indications, the mask is used to cover the ball in case of incorrect operation of the latter.

LIMITATION DEVICES

Incidence Indicating System (refer to Fig 12-17)

1279. This system indicates local angle of attack values to the pilots by causing indicator lights to illuminate in the incidence indicators (13F front cockpit — 213F rear cockpit) through the incidence probe amplifier (11F). Data readout is in the form of green, amber and red lighting combinations on the indicators located on the instrument panel.

1280. The incidence indicating system consists of the following units :

a. A PROBE HEAT switch (11D) located on the RH panel in the front cockpit.

b. An incidence probe (12F) mounted on the LH side between frames 1 and 2.

c. An incidence probe amplifier (11F) located on the rear top face of frame 2 in the aircraft symmetry plane; this amplifier is attached to the aircraft structure through a mounting plate provided with shock absorbers.

d. Two incidence indicators :

(1) The front cockpit indicator (13F) is located on the LH upper part of the instrument panel on the glare shield and is mounted in the vertical position (with the red light at the top and the green light at the bottom).

(2) The rear cockpit indicator (213F) is located on the upper part of the instrument panel in the aircraft symmetry plane and is mounted in the horizontal position (with the green light on the LH side and the red light on the RH side).

e. Two incidence test pushbuttons located on the LH side of the instrument panels, (14F) in the front cockpit and (214F) in the rear cockpit.

1281. **Operation.** The incidence data sensed by the probe (12F) is transmitted to the amplifier (11F) which combines the illumination of the lights on the indicators (13F and 213F).

1282. **Incidence Probe.** The incidence probe (12F) consists of a cylinder (1) with two rows of slots located along two different generating lines. The cylinder is connected to a cylindrical housing (2) divided into two chambers by a vane (3), each chamber being connected to one row of slots through a duct.

1283. The incidence is sensed by the probe as follows : when P1 differs from P2, the vane (3) rotates the cylinder until P1 and P2 equalize; the slots are then symmetrical relative to the direction of the airstream. In rotating, the cylinder (1) actuates two potentiometers (4) which feed electrical signals to the incidence probe amplifier (11F).

1284. **Incidence Probe Amplifier.** The incidence probe amplifier (11F) is an electronic selector which determines the operating cycles of the lights of the incidence indicators from the data supplied by the incidence probe (12F).

1285. **Incidence Indicators.** Each incidence indicator incorporates three lights : GREEN, AMBER and RED. These lights can be dimmed by means of a flap. If the airspeed drops below 250 kt, the lights come on in the following sequence :

a. GREEN.

b. GREEN + AMBER.

c. AMBER.

d. AMBER + RED.

e. RED.

CAUTION

If only the red light is on, the angle of attack limit is reached and must not be exceeded.

1286. To preclude icing, the incidence probe is electrically heated by a resistor (5) controlled by a built-in rheostat. pushbuttons (14F front cockpit-214F rear cockpit) permit testing of the incidence indicator lights for correct operation.

Aircraft Limit Indicating System (refer to Fig 12-8)

1287. The purpose of the aircraft limit indicating system is to avoid the detrimental effects of excessive total temperatures (Tt) due to excessive airspeeds. The system consists of a temperature probe (61C) located between frames 16 and 16 a on the bottom RH side of the fuselage; the probe is attached by means of screws and anchor nuts.

1288. **Operation.** The probe (61C) senses the total temperature (Tt) which it transmits to the air computer in the form of electrical signals which are used by the air data computer for computing the true airspeed for the navigation computer and the radar and controlling (through contacts) :

a. Engine overspeed system cut-in when Tt equals or exceeds 27°C.

b. Aircraft limit (MACH) warning light illumination when Tt equals or exceeds 125°C. Supply to the MACH warning lights is via the failure warning panels (1Z front cockpit — 201Z rear cockpit).

c. III O Only. Rocket motor cut-out when Tt equals or exceeds 144°C.

Accelerometers (refer to Fig 12-6)

1289. The purpose of the accelerometers (31 front cockpit — 231 rear cockpit) is to prevent safety limits from being exceeded by providing a permanent display of the positive and negative vertical acceleration values. These accelerometers indicate aircraft acceleration values from −5 g to +10 g and include an instantaneous acceleration pointer and two maximum-minimum pointers. Zero resetting of the maximum-minimum pointers is accomplished by means of a pushbutton.

CHAPTER 13

FURNISHING AND AIR CONDITIONING

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FURNISHING AND AIR CONDITIONING

Table 13-1 Function Of Components

Index	Description	Characteristics and Functions
		CABIN TEMPERATURE CONTROL SYSTEM
2H	Control panel	The control panel carries : a. Master ON-OFF switch. b. AUTO-MANUAL dual-range rotary selector. c. EMERGENCY-COLD switch directly controls closing of the temperature control valve.
4H	Master valve	Electrically-operated valve slaved to position of ON-OFF switch on control panel. Opens or closes the conditioning circuit.
5H	Temperature amplifier	Compares the signals received from the control panel with data furnished by the cabin and duct temperature sensors. Transmits the resultant signals to the temperature control valve (8H) and the warning circuit.
6H	Duct temperature sensor	Includes a thermistor calibrated at 3°C (low temperature limit slaving) and a thermostat calibrated at 60°C (high temperature warning).
7H	Cabin temperature sensor	Includes a thermistor (cabin temperature slaving) and a cabin thermostat calibrated at 32°C (warning system).
8H	Temperature control valve	Distributes the engine bleed air between the hot air duct and the cooling loop. The degree of opening of this valve is controlled by the amplifier (5H).
9H	Giffard valve	Electrically-operated valve; closed when the main undercarriage is retracted.
17H	Limitation amplifier (incorporated in 5H for III D).	Associated with the amplifier (5H), this maintains the air temperature slightly over 0°C.
3 (III O) 26 (III D)	Water separator	This is crossed by the conditioning air; its purpose is to cause precipitation of the water in suspension to improve demisting.
22	Venturi	Limits the air flow in high-temperature areas to 45 lb/min (20.5 kg/min) in full hot position.
23	Dual heat-exchanger	Consists of two exchangers series connected in the ram air circuit, a main exchanger at the front and a primary exchanger at the rear. It lowers cabin air temperature through thermal exchange with ram air bled from the LH boundary layer bleed.
24	Turbo-compressor	Consists of two units as follows : a. A compressor which pressurizes the loop crossing the exchanger and the evaporator thereby increasing the cooling capacity. b. A turbine which is supplied with loop outlet air and drives the compressor. The energy derived from the air flow also provides additional cooling through expansion.
25	Water evaporator	Consists of a tank with an immersed tubular core assembly through which the conditioning air is circulated.Capacity : 1.1 U.K. gal (5 l) — Usable capacity : 1.0 U.K. gal (4.6 l).Lowers the air temperature to a value close to boiling point (which varies with altitude).
29	Air inlet valve	Manually operated valve through which the conditioning air is admitted directly to the cabin. The closed position provides maximum demisting of windshield and side window panels.
		CABIN PRESSURE CONTROL SYSTEM
23H	Cabin pressure contactor	Controls cabin pressure drop; causes illumination of CAB P light on the warning panel when cabin altitude exceeds 30000 ± 500 ft (9000 ± 200 m).
24H	Altitude switch	Holds the overpressure/underpressure valve open at aircraft altitudes below 6500 ft (2000 ± 150 m).

Index No	Description	Characteristics and Functions
25H	Overpressure/underpressure valve	Opens if cabin pressure differential exceeds a predetermined value following failure of the pressure regulator. Opens if cabin pressure differential drops below a pre-determined value. Remains open at aircraft altitudes below 6500 ft (2000 m).
2	Ram air valve	Manually operated valve interlinked with the cockpit seal valve. When in the open position, it allows ram air into the cockpit.
9	Pressure reducing valve	Lowers the pressure of the air bled from the engine compressor (P2). Calibration pressure : 17.4 lb/in² (1.2 bar).
10	Canopy seal valve	Manually operated 3-way valve allowing canopy seal inflation and deflation; it is combined with the butterfly valve (2).
11 and 12	Safety valve	Calibration pressure :18.8 lb/in² (1.3 bar).Regulates the canopy seal pressure according to altitude changes and protects the seal against bursting in case of deterioration of the pressure reducing valve.
44	Pressure regulator	Adjusts the cabin pressure according to a given schedule.
	EQUIPMENT TEMPERATURE CONTROL SYSTEM — III O ONLY	
13H	Master valve	Electrically-operated, slaved to position of master switch (11H); opens or closes the temperature control circuit.
14H	Amplifier	Compares the signals received from the reference resistor (incorporated in the amplifier) with the data furnished by the duct temperature sensor (15H) and transmits the resultant signals to the temperature control valve (16H). Provides for temperature control on two different temperatures (switch 32H) according to whether the radar is installed or not.
15H	Duct temperature sensor	Consists of a thermistor which measures the temperature of the air flow and a thermostat (warning circuit).
16H	Control valve	Electrically operated valve which distributes the engine bleed air between the hot-air duct and the cooling loop. The degree of opening of this valve is controlled by the amplifier (14H).
38	Venturi	Limits the air flow in high-temperature areas to 45 lb/min (20.5 kg/min) in the full hot position.
43	Air-to-liquid heat exchanger	Cools radar FHS conditioning fluid through thermal exchange with equipment conditioning air.
39	Air-to-air heat exchanger	Lowers the conditioning air temperature through thermal exchange with ram air from the RH boundary layer bleed.
40	Turbo-compressor	Consists of two units as follows : a. A turbine which is supplied with loop outlet air and drives the compressor. The energy derived from the air flow also provides additional cooling through the expansion process. b. A compressor which pumps cooling air and discharges it into the Giffard valve of the heat-exchanger to increase the flow.
50 and 51	Wire gauzes	Incorporated upon installation of the couplings, these protect the turbo-compressor inlets (24 and 26).
	ANTI-G SYSTEM	
20	Anti-g valve	Ensures pilot's flying suit inflation in positive-g flight.
	OXYGEN SUPPLY SYSTEM	
19H (III O) 219H (III D)	Control panel	The control panel carries : a. Two oxygen supply selectors : (1) N-100% selector : N = Automatic dilution 100% = Pure oxygen (2) N-EMG selector : N = Main regulator EMG = Emergency regulator (5 mb pressure differential). b. Aircraft oxygen bottle pressure gauge with 0-1/4-1/2-3/4 graduations. c. Blinker indicating main regulator operation.
20H (III O) 202H (III D)	Pressure switch	Connected into the emergency regulator servo system. Cancels main regulator failure detection and blinker operation if control panel is in EMG position (vented to atmosphere).

Index No	Description	Characteristics and Functions
21H (III O) 212H (III D)	Regulating system including :	
22H (III O) 222H (III D)	Electronic failure detector	Connected to blinker circuit. Detects main regulator failure by switching off the blinker switch, and causes operation of OXY REG light and warning horn through pressure switch (20H).
13 and 14	Pressure reducing valve	Reduces aircraft oxygen system pressure from 2175 to 72 lb/in^2 (150 to 5 bars).
34	Emergency oxygen bottle	Capacity :0.4 litre compressed to 2465 lb/in^2 (170 bars). Ensures oxygen supply in the event of zero flow from aircraft circuit or after ejection.

GENERAL

1301. The missions and the performance characteristics of the MIRAGE call for internal air conditioning provisions so as to permit a number of vital functions to be fulfilled eg Pilot's safety and operation of certain equipment items.

1302. The MIRAGE is designed for operation within an altitude range between 0 and 59000 feet and a speed range between 0 and Mach 2.

1303. The temperatures measured at zero speed for standard atmosphere conditions are as follows :

H = 0 T = +15°C
H = 36 000 ft T = −56.5°C
H = 59 000 ft T = −56.5°C

As the kinetic temperature varies as the square of the aircraft speed, the temperature is then equal to :

+23°C at H = 36 000 ft and M = 0.9
+117°C at H = 36 000 ft and M = 2

It is therefore obvious that the inside of the cabin and a number of equipment items need to be temperature controlled.

1304. Another consequence of high airspeeds is the acceleration that may be reached during steep manoeuvres. To prevent the pilot from being seriously affected by accelerations, flying suit bladders need be inflated to a pressure varying as a function of the acceleration, to prevent a restriction of blood circulation in the pilot's lower limbs.

1305. The pressures measured for standard atmospheric conditions are as follows :

H = 0 P = 1013 mb
H = 36 000 ft P = 225 mb
H = 59 000 ft P = 75 mb

Knowing that the human body requires a minimum pressure of 196 mb, a cabin pressurization system, operated in connection with a temperature control system, is therefore necessary to maintain a constant pressure differential relative to the outside pressure. This is achieved by means of an inflatable seal between the cabin and the canopy.

1306. A breathing oxygen system is required to maintain a correct oxygen supply to the pilots. This oxygen system is also provided in case of cabin decompression or ejection and operates as an emergency pressurization system by establishing a positive pressure in the oxygen mask (low altitude flights) or in the pressure suit (high altitude flights).

1307. The relative humidity of the air requires the windshield and the pilots' helmet visors to be demisted. A forced ventilation system using ram air demists the cabin walls and absorbs calories from the heat exchangers and cools certain equipment items and compartments exposed to engine heating. For III O, a ground conditioning system is fitted to maintain the cockpit and certain items of equipment under given temperature conditions during standby for scramble take-off.

1308. The aircraft is fitted with the following :

a. Cabin temperature control system.
b. Cabin pressure control system.
c. Anti-g system.
d. Oxygen system.
e. Demisting and ventilating system.
f. Equipment temperature control system.
g. Ground conditioning system.

EJECTION SEAT

Description (refer to Figs 13-1 and 13-2)

1309. The aircraft is equipped with a Martin Baker OM6 ejection seat in both front and rear cockpits. At the start of a seat ejection sequence, a time delay unit ensures a one second lag between canopy jettisoning and seat ejection. If the canopy fails to eject, the top of the seat crashes through the plexiglass canopy. Front seat ejection automatically triggers IFF distress signal transmission. A full description of the ejection seat is contained in DI(AF) AAP 7291.021-2M.

Installation (refer to Fig 13-3)

CAUTION

Do not swivel the seat forward on the lower attachment point as damage to the rocket pack may occur.

1310. The front and rear cockpit ejection seats are installed in the same manner. The fixed portion of the seat (1), consisting of the slide rails and the ejection gun, carries two drilled bosses (2 and 3) which mate with two fittings (4 and 5) bolted to the cockpit structure. The attaching parts consist of bolts (6), recessed washers and nuts (7).

1311. Two fittings (8) on the cockpit floor accommodate the leg strap links (9) which are secured by pins (10) held in position by a ball type locking system.

1312. A cable (11) connects the seat face blind to the canopy jettison mechanism through an adjustable fork fitting (12) and a locked screw-type end-fitting (13) at the seat end and through a ring (14) at the canopy mechanism end.

1313. When simultaneous canopy and seat ejection takes place, the bell-crank (15) travels upwards to release the ring (14). The high altitude equipment attached to the seat is connected to the aircraft oxygen system through a quick-release coupling (F). Electrical and radio connections are made through a quick-release coupling (C). The seat connector is fitted with a clamp and a cord to break the connection at the time of ejection.

AIR CONDITIONING

General (refer to Figs 13-4, 13-8, 13-9, 13-11 and 13-12)

1314. An automatic air conditioning system maintains the cabin temperature at the temperature selected on the front cockpit control for most operational conditions. If the kinetic heating exceeds the capacity of the cooling system, the cabin temperature is stabilized between 30 and 32°C.

1315. A manual temperature control in both cockpits and an emergency control in the front cockpit are provided in case of failure of the automatic temperature control system.

1316. Because of the differences in the air conditioning installation between III 0 and III D, each installation has been covered separately as follows :

a. *For III O :*

(1) Cockpit temperature control — paras 1317 to 1332.

(2) Equipment temperature control — paras 1333 to 1348.

(3) On ground air conditioning — paras 1349 to 1356.

b. *For III D :*

(1) Cockpit temperature control — paras 1357 to 1367.

(2) Nose cone and TACAN air conditioning — para 1368

(3) On ground air conditioning — para 1369.

Cockpit Temperature Control — III O Only (refer to Figs 13-5 and 13-6)

1317. The hot air under pressure, which is bled from the engine (pick-off N, circuit F), is first cooled in the RH pre-cooler. It then flows through the basic conditioning system consisting of :

a. A hotair line (F1) incorporating the temperature control valve (8H).

b. A cooling loop (F2) parallel connected on either side of the temperature control valve.

1318. The function of the temperature control valve is to control the proportion of hot air to cooled air delivered into the system. The position of the control valve depends on control signals from the pilot, temperature and temperature derivative data furnished by two sensors.

1319. The installation is monitored by means of a failure warning system. In case of failure of the automatic system, the temperature can be controlled by the manual control system.

1320. **Hot Air Line (F1).** The hotair line consists of :

a. A venturi (22) whose function is to prevent excessive air flows from causing too high a pressure in the cabin and also overspeeding of the turbine unit under certain flight conditions.

b. A master valve (4H) controlled by the ON-OFF master switch (34) on control panel (2H). This valve opens or closes the conditioning system and enables the system to be cut-out.

c. A temperature control valve (8H) slaved to amplifiers (5H) and (17H).

d. A water separator (3) which removes the moisture in suspension in the conditioning air.

e. A non-return valve (28) which prevents sudden cabin decompression following failure or leakage in the conditioning system, or in the event of an engine flame-out.

f. A duct sensing unit (6H) which measures the blown air temperature.

g. A first air diffuser fitted with a valve (29) manually controlled through a slide rod (30); when this valve is closed, all the air is directed onto the front and side windshield panels (demisting).

h. A line (31) through which air is blown over the pilot's feet.

i. A distribution gallery (32).

j. Upsteam of the venturi (22), a tapping for pressurization of the radar nose cone (circuit C), and the ground conditioning Giffard valve (circuit B).

k. Upstream of the master valve (4H), a tapping for supplying compressed air to the canopy seal (circuit D), and the anti-g valve (circuit E).

l. At the bottom of the cockpit, a connection for the ground condition line.

1321. **Cooling Loop (F2).** The air taken upstream of the temperature control valve (8H) flows through the following units :

a. The rear chamber of a dual air-to-air exchanger (23) which lowers the temperature of the conditioning air by thermal exchange with ram air from the outside. This ram air is bled from the central duct of the LH boundary layer bleed and is discharged overboard via a Giffard valve (33) and through a scoop located under the LH air intake duct at frame 14.

b. The compressor stage of a turbo-compressor (24) (through filter (50)) which increases the air pressure in the system to produce a higher expansion ratio in the turbine stage.

c. The front chamber of the heat exchanger (23).

d. A water evaporator (25) which lowers the temperature of the conditioning air by thermal exchange with the water it contains. When the air temperature exceeds the water boiling point (which varies with altitude) the water evaporates and is discharged overboard through a duct run to the bottom of the fuselage at the front of the undercarriage well.

e. The turbine stage (26) of the turbo-compressor (through filter (51)) where the air is cooled by expansion. Air is supplied to this turbine from the compressor located upstream. The cooled air flows to the hot air line through the water separator (3).

1322. **Electrical Temperature Control Circuit.** The electrical temperature control circuit includes :

a. A control panel (2H) which carries :

(1) An ON-OFF master switch (34) controlling the master valve (4H).

(2) A dual-range rotary selector (35). In the AUTO range, the selector controls a comfort temperature adjusting potentiometer. In the MANUAL range, it cuts out the automatic control circuit. The MANUAL position is obtained by turning the adjusting knob 180 degrees. Automatic operation is the normal mode, manual operation being resorted to only in the event of failure of the automatic control system. The MANUAL range has three positions; a stable mid-position and two side position. The position to the right is unstable and is used for sending hot control signals, the position to the left is stable and is used for sending cold control signals.

(3) EMERGENCY COLD switch (36) which, when operated, closes the control valve (full cold) irrespective of the position of the setting knob. This switch is normally locked in the upward position.

(4) Control panel lighting lamp (37).

b. Two temperature sensors as follows :

(1) The duct temperature sensor (6H) which consists of a group of sensitive resistors which measure the temperature derivative, a thermistor which measures the temperature for the limitation circuit and the duct thermostat of the warning circuit.

(2) The cabin temperature sensor (7H) consists of the cabin temperature measuring thermistor and the cabin thermostat of the warning circuit.

c. A temperature control amplifier (5H) which receives temperature information from the sensors (6H) and (7H), compares it with the temperature the pilot has selected and allowing for errors, sends resultant control signals to the temperature control valve (8H) and the warning system.

d. Limiting amplifier (17H) prevents icing of the water separator by limiting the blown air temperature to a value slightly above 0°C.

e. Warning system which warns the pilot of excessive duct or cabin temperatures by a light (CAB T) located on the failure warning panel (1Z). This light operates in conjunction with the warning horn (7Z) and the FAIL light (3Z).

1323. **Automatic Temperature Control.** With the ON-OFF switch (34) on control panel (2H) at ON, the master valve (4H) is open and the conditioning air is allowed into the system. The rotary selector (35), in the AUTO range is set at the desired comfort temperature. Since this selector is linked to the potentiometer, it adjusts the value of the reference resistor. The degree of opening of the temperature control valve (8H) then varies according to kinetic heating.

1324. When temperature due to kinetic heating is lower than the desired temperature, the conditioning air flows through the following units :

a. The OPEN temperature control valve (8H). When this valve is open, the cooling loop is no longer supplied due to the high pressure loss; all the air is delivered into the hot air line and hence the blown air is at maximum temperature.

b. The water separator (3).

c. The non-return valve (28).

1325. The air then flows over the duct temperature sensor (6H) and is distributed through :

a. The air diffuser through the valve (29).

b. The pilot's feet warming lines (31).

c. The distribution gallery (32).

1326. When the temperature due to kinetic heating is greater than the desired temperature, the cabin temperature sensor (7H) and duct temperature sensor (6H) cause the actuator of the temperature control valve (8H) to operate through the slaving amplifier (5H). As the valve closes, there is an increasing pressure loss in the hot air line and more air is passed through the cooling loop. The conditioning air flows through the following units :

a. The rear chamber of dual heat exchanger (23).

b. The compressor stage of the turbo-compressor (24).

c. The front chamber of the dual heat exchanger (23).

d. The evaporator (25).

e. The turbine stage of the turbo-compressor (26).

1327. The air is then discharged into the hot air line through the water separator (3). Maximum cooling is provided when the temperature control valve (8H) is fully closed.

1328. On the ground, the dual heat exchanger (23) is not sufficiently supplied with cooling air to provide satisfactory thermal exchange. The Giffard outlet (33) at the rear of the exchanger creates a depression which accelerates the cooling air flow through the exchanger. This system is controlled by the Giffard valve (9H) which is slaved to the undercarriage (undercarriage down : valve open).

1329. **Manual Temperature Control (refer to Fig 13-8).** If the cabin temperature drops below the desired temperature :

a. Turn the rotary selector (35) to MANUAL on the control panel (2H) (for complete isolation of the automatic system).

b. Hold the rotary switch in the HOT position until a correct temperature is obtained. The temperature control valve (8H) is energized for the time the switch is held in the HOT position.

1330. If the cabin temperature rises abnormally, turn the rotary selector (35) to MANUAL and set the selector to the COLD position (to the left) to close the temperature control valve (8H). When the control valve reaches its fully closed position, all the air flows through the cooling loop, but if the duct temperature falls below +3°C, the duct sensing unit will control the position of the valve through the limiting amplifier (17H) regardless of the position of the selector.

1331. **Emergency Cold Warning System (refer to Fig 13-9).** If the temperature continues to rise despite the operation of the manual control, the warning system is operated when the thermostat of the duct sensor (6H) detects a temperature above 60°C or the thermostat of the cockpit sensor (7H) detects a temperature above 32°C.

1332. When the warning system operates, setting switch (36) to the EMERGENCY COLD position (direction of arrow) causes the actuator of the temperature control valve (8H) to be directly energized, the valve remains in the closed position and all the air passes through the cooling loop and provides maximum cooling.

Equipment Temperature Control — III O Only (refer to Figs 13-11 and 13-12)

1333. The air from tapping (P) on the engine is cooled through the rear chamber of a dual pre-cooler unit (circuit G), the front chamber of this unit is used for fuel tank (M) and hydraulic reservoir (L) pressurization.

1334. The system consists of two circuits in parallel, as in the case of the cockpit conditioning system, ie a direct hot air duct and a cooling loop, but it has no evaporator. There is also a warning and temperature control circuit.

1335. **Hot Air Duct (G1).** The direct circuit consists of :

a. A venturi (38) acting as a flow limiter.

b. A master valve (13H) controlled by a switch (11H).

c. A control valve (16H) which distributes the total airflow between the hot air duct and the cooling loop under the control of the automatic control circuit. The control valve is a slow-acting valve (about 30 seconds required to travel from full hot to full cold) that may also be closed manually (emergency control).

d. A duct temperature sensor (15H) which measures the blown-air temperature and provides for temperature control on two different temperatures according to whether the radar is installed or not (switch 32H).

e. An air-to-liquid heat exchanger (43) for cooling the radar conditioning fluid.

f. Ducts which convey the air to :

(1) Equipment bay (conditioning of gyro amplifier, TACAN and RPH output multiplier.

(2) MATRA R530 missile (through a self-sealing valve).

(3) Tail warning radar bay.

(4) Stand-by UHF bay.

1336. **Cooling Loop (G2).** The air taken upstream of the control valve passes through the following cooling elements :

a. An air-to-air heat exchanger (39) which is supplied with ram air from the RH boundary layer bleed. This air is returned to atmosphere through a scoop located below the fuselage on the RH side. This exchanger differs from the cabin system exchanger in that it has only one compartment.

b. A turbo-compressor. The turbine (42) derives its energy from the conditioning air fed by the compressor (40) which draws air from the secondary air outlet duct of the air-to-air heat exchanger; the air is compressed and then re-injected into the same duct (41), thereby accelerating (by Giffard effect) the air flow through the exchanger. This arrangement permits cooling during ground running and taxiing. From the turbine outlet, the air is delivered into the direct circuit, downstream of the control valve (and upstream of the temperature sensor).

1337. **Electrical Temperature Control Circuit.** This circuit includes :

a. A switch (11H) (on the RH side of the cockpit). When this switch is set to ON, the master valve (13H) opens and the system starts functioning.

b. A switch (12H) which acts on the control valve (16H) through an amplifier (14H). The valve closes when the switch is set to COLD.

c. A temperature sensor (15H) in the radar exchanger inlet duct. This sensor includes the warning circuit thermostat and the blown-air temperature measuring thermistor.

d. A temperature control amplifier (14H).

1338. **Warning System.** Should the blown-air temperature become excessive (50 ± 3°C), an EQ.T red light will illuminate on the warning panel (1Z) together with the FAIL light (3Z). This system is not connected to the warning horn.

1339. **System Control (refer to Figs 13-12 and 13-13).** The control principle differs from that employed in the cabin conditioning system. In this case, the items of equipment involved are capable of withstanding large temperature variations for short periods of time. There is no need, therefore, for a control system as accurate and responsive as for the cabin. In this instance, it is the blown-air temperature which is controlled and not the ambient temperature prevailing in the equipment compartments.

1340. The blown-air temperature varies the resistance of the duct sensor thermistor (15H). This resistance is compared with a pre-adjusted resistance which determines the desired value. The error signal is transmitted to the amplifier (14H) which dispatches a control current to the motor of the control valve (16H).

1341. The blown-air temperature must be held between + 5°C and + 15°C. In the event of failure of the automatic control system, the control valve is closed by moving the EMG COLD switch in the direction of the arrow (downwards). All the air then flows through the cooling circuit.

1342. **Equipment Bay.** The equipment bay is ventilated by the conditioning air evacuated from the cabin by the pressure regulator and the relief valve. This bay is further ventilated by the air which escapes from the equipment conditioning system after flowing through the heat exchanger (43), the gyro amplifier, the TACAN and the gyro centre output multiplier.

1343. **Rear Lateral Bays.** The rear lateral bays (tail warning radar bay and stand-by VHF bay) are ventilated directly by the equipment conditioning system. The air is taken upstream of the radar heat exchanger by a pipe which divides into two branches, between frames 15 and 16. These two branches are routed through the RH and LH wing-to-fuselage fillets and then directly into the rear bays. Ventilation air is fed to the MATRA R530 missile from a tapping provided with a self-sealing coupling (112) on the RH system (forward of main wheel well).

1344. **Radar Cooling (refer to Fig 13-14).** The radar is conditioned by circulating FHS fluid which is cooled through thermal exchange in the heat exchanger (43) mounted at the rear of frame 10, in the nosewheel well.

1345. The connections between the radar nose cone and the fuselage are made by flexible pipes and self-sealing couplings (84). Sealed connections (83) are used for routing the pipes through frames 2 and 10. The fluid return pipe crosses the cockpit between frames 5 and 6; at this location, the pipe is fitted with a non-return valve (97) mounted between two tappings to which the ground conditioning lines connect.

1346. The FHS fluid is circulated by a pump which is part of the radar nose cone unit. Mounted on the heat exchanger outlet duct is a bleed pipe fitted with a valve enabling a ground mechanic to detect the presence of water in the conditioning system.

1347. **Radar Pressurization.** This is ensured by a regulator supplied from pressure tapping (N) on the engine (circuit C) through a filter. The function of this regulator is to hold the pressurization pressure at 2.2 lb/in^2 abs (980 mb abs).

1348. The filler and the regulator attach to the RH side of the nose undercarriage well, between frames 14 and 15. Between the regulator and frame 2 the pressure is ducted by a pipe with couplings at frames 2 and 10. The radar to aircraft connection is made by a flexible pipe and a self-sealing coupling.

On Ground Air Conditioning — III O Only (refer to Figs 13-14 and 13-15)

1349. On the ground, while the aircraft is on stand-by with the engine shut down, a special air conditioning unit can be connected to the aircraft to condition the cabin, certain items of equipment and the radar. The unit also ventilates the pilot's flying suit.

1350. **Description.** A ground conditioning connector connects to the aircraft through a sealed box (78), at the bottom of the cockpit (between frames 7 and 10); this box carries four self-sealing valves as follows :

a. Two (right and left, (89 and 90)) for radar FHS fluid.

b. One (rear, (88)) for pilot's suit ventilation.

c. One (front, (87)) for cabin conditioning.

1351. The connector (85) is held in place by two roller type jaws (91) clamped by a spring (94) and located by three pegs (92). A door (93) enables the opening in the fuselage underside to be closed when the connector breaks away. The closing of this door is combined with the motion of the two jaws (91) which, in their initial movement, unlock and release the connector.

1352. The complete system is controlled by an electric motor (14M) provided with limit switches. The ground conditioning connector is automatically ejected when the pilot presses the starter button (7K). An electrical control (13M), in the nosewheel well on the RH web between frames 10 and 11, enables the mechanic to open the door and lock the connector; this control also permits the connector to be ejected and the door to close. A red light (18M) marked GRD CONN (on the RH panel) extinguishes when the connector is ejected and the door closed.

1353. A second connector connects to the aircraft (nosewheel well, frame 13) through a quick-disconnect and is used for conditioning the equipment bay.

1354. **Operation.** When the connector is connected to the ground conditioning unit, the self-sealing valves are open and the following systems are supplied :

a. *Pilot's System.* The conditioning air is admitted directly into the cockpit through the three distribution galleries :

(1) On to the front and side windshield panels.

(2) Over the pilot's feet.

(3) Into the direct diffuser.

The cabin system non-return valve (28) (frame 10) prevents the air flowing into the normal aircraft system.

b. *Pilot's Flying Suit Ventilation.* The air is circulated into the pilot's suit and evacuated through a number of orifices provided at different points on the suit.

c. *Radar System.* The FHS cooling fluid, admitted through the RH valve, flows through the radar heat exchanger to the radar and returns to the ground unit through the LH valve (direct return to the ground unit is prevented by non-return valve (97)). The fluid flows through the non-return valve when the connector is ejected. A gauge (96), located between frames 1 and 2 on the RH side, is connected into the radar inlet line and permits checking of the cooling fluid pressure.

1355. **Ejection Of Connections (refer to Fig 13-15).** Opening action is initiated by pressing button OP on the mechanic's box (13M). Ejection of the connection and closing of the door can be achieved by pressing either button CL on the mechanic's box or the pilot's starter button (7K). Electric motor (14M) is energized by a relay (16M) (in the relay box) and operates until the limit switches open. These limit switches also control the GRD CONN warning light (18M) which is a press-to-test type indicator. Relay (16M) is self-holding when the closing is initiated through button (7K).

Cockpit Temperature Control — III D Only (refer to Figs 13-16 to 13-21)

1356. The hot air under pressure bled from the engine (pick-off N — system C) is first cooled in the RH pre-cooler. The system consists of the following units :

a. A shut-off valve (4H).

b. A venturi (22) which limits air flow rates.

c. A conditioning assembly entirely contained in the aircraft nose cone that consists of a hot air duct (C1) incorporating a temperature control valve (8H) and a cooling loop (C2) connected in parallel.

1357. The temperature control valve (8H) controls the hot air/cooled air mixing ratio. Its position is determined by control signals from the pilots and temperature information furnished by the duct and cabin temperature sensing units (6H and 7H).

1358. **Cooling Loop (refer to Figs 13-16 and 13-17).** The hot air taken upstream of the temperature control valve (8H) flows through the following elements :

a. The rear section of a dual air-to-air heat exchanger (23). The function of this exchanger is to reduce the conditioning air temperature through thermal exchange with ram air taken from outside the aircraft.

b. The compressor stage of a turbo-compressor (24) which increases the air pressure in the system to produce a higher expansion ratio in the turbine further downstream.

c. The front section of the above mentioned heat exchanger (23).

d. A water evaporator (25) designed to lower the conditioning air temperature through thermal exchange with the water it contains.

e. The turbine stage of the turbo-compressor (24) in which the air is cooled by expansion. Air is fed to this turbine from the compressor situated upstream.

1359. At the outlet of the cooling system, the conditioned air flows through a water separator (26); it is then conveyed to the cabin through a non-return valve (27). It is first distributed to the windshield (front and side panels) by a three-piece duct (28) and then to the rear of the canopy by two lateral ducts (35). An additional outlet (30), fitted with a mechanically operated valve (29), enables the windshield air flow to be controlled to provide for maximum de-icing whenever required (this valve can be controlled from either cockpit).

1360. **Electrical Temperature Control Circuit (refer to Figs 13-18 to 13-20).** The electrical temperature control circuit includes :

a. A front cockpit control panel (2H) including :

(1) An ON-OFF master switch (35) for control of the system shut-off valve.

(2) A dual-range rotary selector (36). In the AUTO range, the selector controls an ambient temperature adjusting potentiometer. In the MANUAL range, it cuts-out the automatic control circuit. The MANUAL position is obtained by turning the selector 180 degrees (This position is used only in the event of failure of the automatic system). The MANUAL range has a central position, which is stable, and two lateral positions which are unstable. The RH position for hot control signals and the LH position for cold control signals.

(3) An EMERGENCY COLD switch (38) which, when operated, closes the temperature control valve (full cold) irrespective of the position of the temperature selector. This switch is normally safety-locked in the up position.

(4) A control panel lighting lamp (37).

b. A rear cockpit control panel (202H) including :

(1) A selector switch (39) allowing manual control from the rear cockpit and cutting-out the automatic control circuit.

(2) A three-position switch (40); this switch has two unstable positions which permit hot control signals (RH position) and cold control signals (LH position) to be transmitted. Manual control from the rear

cockpit is possible when the switch (39) is in the MANUAL position. The front cockpit control panel has no effect in this case.

c. Two temperature sensing units :

(1) A duct temperature sensing unit (6H) consisting of a sensitive resistor group which measures the temperature derivative, a thermistor which measures the temperature for the limiting circuit and a duct thermostat for operation of the warning system.

(2) A cabin temperature sensing unit (7H) consisting of a thermistor which measures the cabin temperature and a cabin thermostat for operation of the warning system.

d. A temperature amplifier (5H). This amplifier is fed with information from the temperature sensing units (6H and 7H); it compares this information with the pilots' control signals and, taking the errors into account, transmits control signals to the temperature control valve and the warning system. The temperature of the blown air is kept to a value slightly above 0°C to preclude water separator icing.

1361. **Warning System.** This system is provided to warn the pilots of excessive duct or cabin temperatures and consists of :

a. In the front cockpit :

(1) A CAB T warning light on the failure warning panel (1Z).

(2) The FAIL warning light (3Z).

b. In the rear cockpit : A CAB T warning light on the failure warning panel (201Z).

c. A warning horn (7Z) that operates if the warning lights illuminate.

1362. **Automatic Temperature Control (refer to Fig 13-18).** The air conditioning system is set into operation by placing the front cockpit control panel (2H) ON-OFF switch to the ON position to open the shut-off valve (4H). The desired ambient temperature is then selected by means of the rotary selector in the AUTO range.

1363. When the temperature due to kinetic heating is lower than the desired temperature, the open temperature control valve (8H) allows most of the conditioning air to flow to the hot air duct; the cooling loop is not supplied due to the high pressure loss and maximum heating is provided.

1364. When the temperature due to kinetic heating is higher than the desired temperature, the cabin temperature sensing unit (7H) and the duct temperature sensing unit (6H) feed information to the amplifier (5H). This amplifier electrically controls the actuator of the temperature control valve (8H) which opens partially. As the pressure loss increases in the hot air duct, a larger air flow is allowed through the cooling loop. Maximum cooling is provided when the valve (8H) is completely closed.

1365. **Manual Temperature Control (refer to Fig 13-19).** The manual temperature control system is designed for use in case of failure of the automatic control system and consists of :

a. *In the Front Cockpit.* A rotary selector which is placed on MANUAL to isolate the automatic control system and the related servo system. The actuator of the temperature control valve (8H) is then controlled by momentarily switching the rotary selector until the desired temperature is obtained; to the right to increase heating and the left to increase cooling.

b. *In the Rear Cockpit.* A control panel selector switch which is placed in the MANUAL position and the temperature adjusted by the three-position switch; to the right for increased heating and to the left for increased cooling.

1366. **Emergency Cold Warning System — (refer to Fig 13-20).** The warning circuit is energized when the duct sensing unit (6H) senses a temperature above 60°C or the cabin sensing unit (7H) senses a temperature above 32°C.

1367. When the warning system operates, setting switch (36) to the EMERGENCY COLD position (direction of arrow) causes the actuator of the temperature control valve (8H) to be directly energized, the valve remains in the closed position and all the air passes through the cooling loop and provides maximum cooling.

1368. **Nose Cone And TACAN Air Conditioning — III D Only (refer to Fig 13-16).** At the outlet of the water separator is a tapping which supplies, through a non-return valve (32), the TACAN and the elements located at the rear of the nose cone through a nozzle (33). At the end of this system is a quick-release coupling (34), which is accessible after opening the nose cone, that connects to the ground conditioning unit. The purpose of the non-return valve (32) is to prevent ground conditioning air from flowing to the cabin.

On Ground Air Conditioning — III D Only

1369. On the ground, the dual heat exchanger is not supplied in order to ensure a sufficient thermal exchange. The Giffard outlet (36) located on the rear section of the heat exchanger is intended to accelerate the cooling air flow through the exchanger by creating a suction effect. The system is controlled by a Giffard valve (9H), a relay (46G) and the position of the undercarriage. With the undercarriage extended, the Giffard valve is in the open position.

CABIN PRESSURIZATION SYSTEM

Description (refer to Fig 13-21)

1370. When the pressurization system is in operation, the cabin altitude never exceeds 30000 feet, even with the aircraft flying at its ceiling height. The cabin pressurization system includes :

a. A pressure regulator (44) which is adjusted to maintain a pressure differential (relative to outside pressure) conforming to a given schedule.

b. An overpressure/underpressure valve (25H) which opens to compensate for excessive positive or negative pressures.

This valve incorporates a device to allow cabin ventilation at low altitude.

1371. The overpressure/underpressure valve is slaved to an altitude switch (24H) which causes the valve to open at an altitude below 6500 ft and a cabin pressure switch (23H) connected to the electrical warning circuit.

Operation

1372. The cabin pressurization system operates as follows :

a. At aircraft altitudes up to 6500 ft, the overpressure/underpressure valve (25H), controlled by the altitude switch (24H), remains open.

b. At aircraft altitudes between 6500 and 18000 ft, the overpressure/underpressure valve closes. The pressure regulator (44) maintains a constant cabin pressure corresponding to an altitude of 6500 ft.

c. At aircraft altitudes above 18000 ft, the pressure differential (P = cabin pressure − outside pressure) remains constant and equal to :

(1) For III O : 4.611 ± 0.14 lb/in^2 (318 ± 10 mb).

(2) For III D : 3.9 ± 0.15 lb/in^2 (269 ± 10 mb).

Table 13-2 Cockpit Structure Sealing - III O

Location	Index	Description	Sealing Method				Remarks
			Sealing Compound	Seal	Packing Gland	Sliding Assembly	
Frame 2	1	Shield door		X			
	2	Radar conditioning	X				
	3	Radar coaxial	X				
	4	Radar wiring	X				
	5	Pitot-static connections	X				
	6	Fresh-air supply	X				
	7	Radar pressurization	X				
LH side	8	Brake chute control	X				
		Canopy control bell-crank	X				
RH side	9	Canopy control lever		X			
Cockpit top	10	Windshield panels	X				
	11	Canopy		X			
Floor	12	Ground conditioning unit	X				
	13	LH sealed box roll control		X			Seal on bearing
		throttle control			X		
	14	Emergency undercarriage control	X				
	15	Cabin air conditioning pipe	X				
	16	Anti-g valve (pipe)	X				
	17	Parachute control			X		
	18	Oxygen pipe	X				
	19	Rudder controls				X	
	20	Electric connectors and wires	X				
	21	Radar coaxial	X				
	22	Coupling block (radar conditioning, braking, pressurization)	X				
	23	Parking brake	X				
	24	Wave guide	X				
	25	Coupling block (ASI and canopy inflation)	X				
	26	Pitch control (in RH box)		X			Seal on bearing
	27	Cabin pressurization connection	X				
	28	Cabin pressure test connection	X				

Table 13-3 Cockpit Structure Sealing - III D

Location	Index No	Description	Sealing Method				Remarks
			Sealing Compound	Seal	Packing Gland	Sliding Assembly	
Frame 2	1	Shield door		X			
	2	Pitot-static line couplings	X				
	3	Ram air inlet	X				
	4	Conditioning air	X				
	5	Sealed electrical connectors	X				
	6	Feeders	X				
LH side	7	Brake chute control axle-pin	X				
		Canopy control bell-crank	X				
RH side	8	Canopy control lever		X			
Upper part	9	Windshield panel	X				
	10	Canopy		X			
Floor	11	Coupling block (brake lines)	X				
	12	Electrical connectors and feeders	X				
	13	Pitot-static system drains	X				
	14	Canopy seal inflation system coupling	X				
	15	Electrical bulkhead connectors	X				
Frame 10	16	Flight control pressure sealed unit	X				
	17	Throttle control			X		
	23	Inflation connection	X				
	24	Pressure test connection	X				
Slanting frame	18	Emergency undercarriage control		X			
	19	Emergency brake control		X			
	20	Brake chute control		X			
	21	Oxygen coupling	X				
	22	Rudder control				X	
	28	Regulator pressure connection	X				
Pocket-recess	25	Sealed electrical connectors	X				
in-slanting	26	Static line couplings	X				
frame	27	Anti-g system coupling	X				

Failures

1373. If the cabin altitude exceeds 30000 ft, the cabin pressure switch (23H) closes the warning circuit which causes the CAB P warning light (on 1Z and for III D 201Z) and the FAIL light (3Z) to illuminate and the warning horn (7Z) to sound.

CABIN SEALING

Cockpit Structure (refer to Figs 13-22 to 13-24)

1374. The cockpit structure riveting is sealed by a special sealing compound. Sealing methods for the passages of the various controls, pipes and wires through the cockpit walls are listed in Table 13-2 for III O and Table 13-3 for III D.

Canopy (refer to Fig 13-25)

1375. The canopy seal is inflated by air tapped (circuit D for III O and system F for III D) from the cabin air conditioning system, up-stream of the master valve (4H). The canopy seal inflation system includes the following units in succession :

a. A filter.

b. A non-return valve (8) which isolates the upstream system during pressure testing on the ground.

c. A pressure reducing valve (9) calibrated at 17.4 lb/in^2 (1200 mb).

d. A control lever unit (5) with three marked catches to hold the lever (6) in the desired position. The lever (6) is also used for operating the ram air valve (2).

e. A three-way canopy seal valve (10) controlled by the lever (6); it incorporates a non-return valve (11) which closes in the event of supply failure.

f. A safety valve (12) calibrated at 18.8 lb/in^2 (1300 mb).

g. The inflatable rubber seal (13).

1376. **Canopy Sealing Lever Operation.** The canopy sealing lever operates as follows :

a. With the control lever on position 1; the position of the canopy seal valve (10) allows the canopy seal to vent to atmoshphere and the canopy seal inflation system inlet is closed.

b. With the control lever on position 2 or 3; the canopy seal vent is closed. Pressure is applied to the seal through the non-return valve (11). The safety valve (12) protects the seal in case of failure of the pressure reducing valve or under certain engine operating conditions at altitude.

Sealing Tests

1377. **Cabin.** Two connections are located on the rear face of the slanting frame to provide for sealing tests. Access to the inflation and pressure test connections is gained through the nose wheel well.

1378. **Canopy Seal.** An inflation connection (14) and a pressure test connection (15) are provided for sealing tests of the canopy seal on the ground.

ANTI-G SYSTEM

Description (refer to Fig 13-25)

1379. A branch line from the canopy seal inflation system conveys compressed air to an anti-g valve (20) in each cockpit. This valve is connected to the pilot's suit through a hose fitted with an automatic release coupling (in case of ejection). It is located on the LH side of the cockpit and is accessible to provide for testing (on the ground or in flight).

Operation

1380. The anti-g valve (20) is designed to supply and regulate the anti-g suit inflation pressure according to positive vertical accelerations of the aircraft. In the absence of acceleration, the pressure in the suit is equal to the cabin pressure. The valve has two different pressure stages as follows :

a. *First Stage.* The purpose of the first stage is to lower the supply pressure to a relative pressure of 21.8 lb/in^2 (1.5 bar).

b. *Second Stage.* Supplies the pilot's suit and permits venting of the suit.

1381. Each pressure stage is provided with a relief valve as a protection against excessive pressures. A test button is located at the top of the anti-g valve to enable the pilot to check his suit at any time for correct connection and inflation.

OXYGEN SYSTEM

General (refer to Figs 13-26 to 13-30)

1382. The oxygen system is designed to supply oxygen under maximum safety conditions in both low altitude and high altitude missions. The system consists of :

a. A normal high pressure supply in the form of two gaseous oxygen bottles located between frames 15 and 16.

b. For each pilot :

(1) A regulator assembly attached to the seat.

(2) A control panel.

(3) A pilot's low altitude or high altitude personal equipment.

(4) A warning circuit.

Description (refer to Figs 13-26 to 13-30)

1383. The oxygen system assembly consists of the following units :

a. A bottle charging connection (1) with a filter (2). This connection is located at the lower part of the RH air intake duct between frames 13 and 14.

b. Two bottles (4), housed in a cabinet attached to the rear face of frame 15, with couplings (3). These couplings are fitted with :

(1) Flow limiters (7) which close the bottle when the pressure drops below 43.5 lb/in^2 (3 bars).

(2) Filters (5 and 9).

(3) Non-return valves (6 and 8).

c. A bulkhead connector (10).

d. Forward of the bulkhead, two similar independent installations.

1384. Each installation consists of :

a. A valve (11 and 12) and a pressure reducing valve (13 and 14) located on the LH side of the cockpit.

b. An oxygen regulator assembly (21H and for III D 221H) mounted on the seat.

c. A connector (11R for III O and for III D 11R1 and 11R2) on the seat.

d. A control panel (19H and for III D 219H) incorporating a blinker, situated on the RH console.

e. A warning circuit.

1385. Each oxygen regulator assembly, slaved to the control panel and to cabin altitude, is grouped into a single compact unit attached to the LH side of the seat.

1386. **Oxygen Regulator Assembly (refer to Fig 13-29).** The oxygen regulator assembly consists of :

a. A normal regulator (41) with automatic dilution system.

b. An emergency regulator (42).

c. An emergency supply consisting of an oxygen bottle (34) of 0.4 litre capacity, charged to a pressure of 2466 lb/in^2 (170 bars).

d. A pressure reducing valve (35), which lowers the emergency oxygen bottle pressure to 47.9 lb/in^2 (3.3 bars), and a pressure gauge (38). The 8 min OXY lights on the failure warning panels are operated by either the valve or the gauge.

e. An altitude selector (36) with two positions, HA — LA (high altitude — low altitude). The selector is set, by the pilot on entering the cockpit, in accordance with the type of mission involved.

f. A helmet visor seal inflation valve (39).

g. A HA test button (37).

h. An emergency bottle charging connection (40).

1387. **Control Panel (refer to Fig 13-27).** The control panel (19H and for III D 219H) carries :

a. A N-EMG (normal-emergency) selector which controls the following functions :

(1) In position N, the normal regulator is in operation.

(2) In position EMG, the emergency regulator is cut-in.

b. A N-100% (normal-100%) dilution control selector which operates as follows :

(1) In the N position, automatic dilution variation with altitude.

(2) In the 100% position, pure oxygen supply.

c. A high pressure gauge which indicates the quantity of oxygen remaining in the aircraft bottles.

d. A blinker type indicator which is controlled by a switch incorporated in the regulator assembly and indicates operation of the normal regulator only.

1388. **Pilots' Oxygen Supply System (refer to Figs 13-26, 13-27, 13-29 and 13-30).** The pilots' oxygen supply system comprises :

a. The pilot-to-seat connector (11R). The cover, which is connected to the pilot, separates automatically during the seat/pilot separation phase in the event of ejection. Manual separation is achieved by using the handle provided on the cover.

b. The chest connector which carries a valve for normal connection with the mask (LA) or the helmet and flying suit (HA). In normal operation, the connector cover remains attached to the seat; the pilot connects the line or lines at the chest connector on strapping in.

1389. **Warning Circuit (refer to Fig 13-26).** The warning circuit consists of :

a. An electronic failure detector (22H and for III D 222H).

b. A pressure switch (20H and for III D 220H) which cancels failure detection in the EMG position.

c. Two lights on the failure warning panel (1Z and for III D 201Z) as follows :

(1) An OXY REG light which indicates failure of the normal regulator; this light also indicates that the helmet visor seal valve is closed.

(2) An 8 min OXY light which indicates that oxygen is being supplied from the emergency bottle and that only 8 minutes oxygen is available; this light also indicates that the emergency bottle is closed.

1390. These lights are operated in conjunction with the front cockpit FAIL warning light (3Z) and with the audio warning horn (7Z).

Operation (refer to Figs 13-29 and 13-30)

1391. The description of the oxygen system covers the supply for either pilot. The oxygen system is supplied from two pressure sources in parallel :

a. The normal source (aircraft bottles).

b. The emergency source (emergency bottle).

CAUTION

Do not open the valve (35) of the emergency bottle (34) before opening the valve (11) controlling the aircraft bottles; otherwise, oxygen from the emergency bottle will pass into the aircraft system. The reverse is to apply when the system is switched to OFF — ie, valve (35) is to be closed before closing valve (11).

1392. In normal operation, oxygen is delivered to the system from the aircraft bottles only. The pressure from these bottles, as delivered by the pressure reducing valve, 73 lb/in^2 (5 bars) is higher than that supplied by the emergency bottle which is 47.9 lb/in^2 (3.3 bars). The emergency reducing valve (35) is unable to deliver pressure and acts as a pneumatic valve on the emergency bottle (34).

1393. When the normal system pressure drops down to 47.9 lb/in^2 (3.3 bars), the emergency reducing valve starts to deliver pressure thereby causing the non-return valve (51) to close. This process occurs in the event of ejection or of depletion of the aircraft oxygen bottles.

Low Altitude Operation (refer to Fig 13-29)

1394. The pilot must wear a low altitude flying suit. Open the aircraft system valves (11 and 12) and then the emergency system valve (35). Set the altitude selector (36) to LA (low altitude) and the helmet visor seal valve (39) to CLOSED.

1395. **Normal Operation.** For normal operation, set the NORMAL — EMERGENCY selector to N and set the NORMAL — 100% selector to N.

1396. The normal regulator (41) will then supply oxygen at a pressure of 73 lb/in^2 (5 bars). This pressure is applied through the NORMAL — EMERGENCY selector to pneumatic valve (43) of the emergency regulator whose supply line is closed off as a result. It is also applied to the pneumatic actuator (44) of the dilution system. Normal regulator operation actuates the air pressure switch controlling the blinker.

1397. The pilot breathes an air and oxygen mixture, the oxygen percentage of which is regulated by an altitude capsule (45); this percentage increases with altitude but never reaches 100% (for a cabin altitude of 30000 ft).

1398. **100% — Emergency Operation.** For 100% — emergency operation set the NORMAL — EMERGENCY selector to EMG and the NORMAL — 100% selector to 100%.

1399. The normal regulator and the emergency regulator are in operation, the dilution system cuts out and the pilot is supplied with pure oxygen.

13100. **Switch Over To Emergency Bottle.** The pilot switches over to the emergency bottle in the following circumstances:

a. Depletion of aircraft bottles When the normal system reduced pressure drops to 47.9 lb/in^2 (3.3 bars), oxygen flows from the emergency pressure reducing valve and the non-return valve closes to isolate the NORMAL and EMERGENCY systems. The residual pressure remaining in the main system is applied to the pneumatic valve (43) of the emergency regulator and to the pneumatic actuator (44) of the dilution system, with the result that :

    (1) The emergency regulator is inoperative.

    (2) The dilution system continues to operate.

    (3) The blinker continues to operate.

b. Failure of a normal system high pressure line or ejection. There is no more residual pressure and the dilution system and emergency regulator actuators are open to atmosphere, with the result that :

    (1) The dilution system cuts out.

    (2) The emergency regulator operates in parallel with the normal regulator.

    (3) The blinker ceases to operate (air pressure switch open to atmosphere).

NOTE

When the emergency bottle pressure is down to 1958 lb/in^2 (135 bars), the 8 min OXY light illuminates and the warning horn sounds.

13101. **Failures.** The pilot is warned by the OXY REG warning lights on the failure warning panels (1Z and for III D 201Z). In the event of failure of the normal regulator, the blinker contactor becomes inoperative and causes :

a. If the blinker is visible; the normal regulator, which is open, delivers oxygen continuously and the pressure differential is limited to permissible values by the safety valve (46). No action required by the pilot.

b. If the blinker has moved out of sight; the normal regulator is closed and oxygen delivery ceases. Switch over to the emergency regulator is required. This is accomplished by setting the selector on the control panel to EMG, whereby the emergency regulator supply valve actuator (43) is opened to atmosphere and the dilution system cuts-out.

13102. **Cabin Decompression.** In the event of cabin decompression at altitude, the normal regulator operates at a pressure differential which varies with altitude from a height of 38000 ft. At about 50000 ft, this pressure differential may attain 0.58 lb/in^2 (40 mb) and is bearable for only a very short time.

13103. The pressure differential is dependent on the closing of valves controlled by an altitude capsule and is limited by safety valve (46). Altitude capsule (45) cancels dilution from 31000 ft.

13104. **Ejection.** The emergency bottle (34) takes over from the aircraft bottles. There is no self-sealing device on the ejection connections and, therefore, the pneumatic actuators controlling dilution and the emergency regulator valve are open to atmosphere. The regulators deliver pure oxygen at a pressure differential varying between 0.05 and 0.58 lb/in^2 (3.5 and 40 mb), depending on altitude.

High Altitude Operation (refer to Fig 13-30)

13105. The pilot must wear a high altitude flying suit. Set the altitude selector to HA (high altitude). Set the control panel selectors to EMG and 100% and the helmet visor seal valve (39) to OPEN.

13106. **Normal Operation.** The dilution and emergency regulator supply valve actuators are open to atmosphere (selectors at positions EMG and 100%). The emergency regulator operates in parallel with the normal regulator and supplies oxygen at a pressure differential of 0.05 to 0.08 lb/in^2 (3.5 to 5.5 mb). The blinker is inoperative as its air pressure contactor is open to atmosphere.

13107. **Switch-Over To Emergency Bottle.** With the selectors set at EMG and 100% at the beginning of the flight; the dilution system is cut-out, the blinker is inoperative and the emergency regulator operates in parallel with the normal regulator.

13108. When the emergency oxygen bottle pressure is down to 1958 lb/in^2 (135 bars), the 8 min OXY light illuminates and the warning horn sounds.

13109. **Failures.** In the event of failure of either regulator, while both are in service, the delivery from the remaining regulator is sufficient to cater for normal oxygen supply to the pilot and no warning of a failure is given.

13110. **Cabin Decompression.** In the event of cabin decompression, the altitude valves maintain a pressure differential related to altitude. With the altitude selector at position HA, the emergency regulator delivers an overpressure which blocks the normal regulator to prevent it from delivering above 38000 ft.

13111. A constant absolute pressure, equal to that obtained at 38000 ft, namely 2.9 lb/in^2 (200 mb), is maintained in the pilot's flying suit by the automatic overpressure device in the emergency regulator.

13112. **Ejection.** During HA ejection, oxygen is supplied to the pilot from the emergency supply as described for cabin decompression, para 13110 and LA ejection, para 13104.

13113. **Function Test.** With the pilot wearing his stratospheric suit, and with the selectors at EMG and 100%, pressing the emergency regulator test button should cause the regulator to deliver an overpressure and inflate the HA pressure suit. This shows that the HA suit is properly connected and that the emergency regulator is operative.

Installation (refer to Fig 13-28)

13114. All the components of the installation are connected by copper tubing with olives (13 and 14) and captive nuts (15).

13115. The bottles (4) are placed in their cabinet after removing the door (16), which is accessible through the nose wheel well. They fit into cut-outs lined with a rubber seal (17) at the top and are held in position by two bolted brackets (18) with rubber pads at the bottom. The charging connection (1), attached to the structure, is blanked off by a sealed plug (21).

13116. Each regulator assembly (21H and for III D 221H) has quick-disconnect features, as follows :

a. A tapered lower end (22) that enters a gusset (23) attached to the seat.

b. A top end (24) that is recessed to accommodate a swivel bolt (25) and attached to the seat.

c. Clamping is accomplished by means of a washer and a castellated nut (26) locked by a safety pin (27).

13117. The regulator assembly is connected to :

a. The supply system by a connection (28) which separates when ejection takes place. This connection carries :

(1) An extractor cable (29) attached to the structure.

(2) An electric cable (30) connected to the failure detection circuit.

(3) An oxygen supply hose (31).

(4) A hose (33) that matches the system to the type of supply.

b. The oxygen mask by hoses through the connector (11R for III O and for III D 11R1 and 11R2).

c. The quick-release connector (radio and helmet visor demisting system connections) by an extractor cable.

DEMISTING

Windshield Demisting (refer to Fig 13-25)

13118. A scoop at the bottom of the fuselage between frames 1 and 2 collects ram air and conveys it into the cabin. The system comprises :

a. A sealed connection at frame 2 (RH side).

b. A butterfly valve (2) with a built-in non-return valve; this valve is connected to the control lever quadrant (5) through a Teleflex control.

c. A demisting duct through which the air is blown on to the front and LH windshield panels. Demisting is provided when the control lever (6) is in position 1 or position 2.

13119. **Normal Demisting (refer to Figs 13-6 and 13-8).** Normal demisting is ensured by the cabin conditioning air. Maximum demisting is provided by placing the control lever of the air inlet valve (29) in the open position. In this position, all the conditioning air enters the cabin through the front and rear distribution ducts.

13120. **Emergency Demisting (refer to Figs 13-5 and 13-25).** The emergency demisting system is controlled by a single lever (6) which controls both the ram air supply and the inflation of the canopy pressure seal.

Pilots' Helmet Visor Demisting (refer to Figs 13-30 and 13-31)

13121. The installations provided for the IIIO and IIID pilots are similar. Each installation is fully electrically controlled by means of a three-position switch (2D and for III D 202D) marked EMG-NORM-OFF. This switch is located on the RH side of the cockpit. When necessary, the pilot's helmet visor may be demisted by setting the switch to NORM. Heating is accomplished by means of a visor resistor (5D for III O and 5D1 and 5D2 for III D) through a fixed resistor (3D and for III D 203D).

13122. In the event of failure of the normal circuit, the control switch is set to EMG which :

a. Isolates the circuit of the fixed resistor which is energized through circuit breaker (1D).

b. Closes the emergency circuit which is directly energized through fuse breaker (4D).

AIR-TO-AIR EXCHANGER — III O ONLY

Description (refer to Fig 13-32)

13123. The air-to-air exchanger is designed to lower the conditioning air temperature through thermal exchange with ram air bled from the air intake duct.

13124. The exchanger consists of a cylindrical body (1) of welded construction with stiffening baffles and conditioning air inlet and outlet flanges (2) at the bottom. The two main baffles (3) at the front and rear carry rims (4) to accommodate the Giffard duct (5) and the cooling air inlet duct (6). The three intermediate baffles (7), separated from one another by welded plates (8), are drilled with holes to permit conditioning air circulation. The cooling air passes through tubes (9) which are crimped to the main baffles.

Operation

13125. The cooling air is circulated through the exchanger tubes (9). The conditioning air passes around the exchanger tubes and is deflected from one compartment to the next by baffles (7).

Installation (refer to Fig 13-32)

13126. The exchanger is attached to the aircraft through two gussets (10) by means of flexible clamps (11) tightened by screws (12).

13127. The ram air inlet duct (6) and the Giffard duct (5) are attached to the exchanger by rigid clamps (13) secured by a screw and nut (14).

13128. The conditioning air inlet and outlet are connected to the aircraft system by a semi-flexible SEMCA coupling which is attached to flange (2) by two half-rings (19) and a flexible clamp (20). Sealing is ensured by an 0 ring (21). The coupling is made up of:

a. A nozzle (15).

b. An 0-ring seal (16).

c. A washer (17) held by a lock ring (18) with three pegs.

EVAPORATOR

Description (refer to Fig 13-33)

13129. The evaporator is an air-to-water exchanger designed to lower conditioning air temperature through thermal exchange with the water it contains. Its water capacity is 1.1 U.K. gal (5 litre).

13130. The evaporator consists of a tank (1) of welded sheet metal construction which carries :

a. Externally :

(1) An air tube (2) screwed in position with an end-fitting to take a SEMCA coupling.

(2) A steam outlet orifice (3).

(3) A fixing rod (4) with two swivelling end-fittings (5), which is welded to a reinforcement plate (6).

(4) A filler port with a plug (7).

(5) An air outlet tube (8) with an end-fitting to take a SEMCA coupling.

(6) A drain plug (10).

b. Internally :

(1) A group of tubes (11) crimped into two base plates (12) and welded in position at either end of the evaporator.

(2) A steam outlet tube (13) with a welded deflector (32).

(3) Two formed sheet metal stiffeners (33) welded to the evaporator.

Operation

13131. The conditioning air flows through the evaporator tubes (11) which are immersed in water. When the conditioning air temperature exceeds water boiling point (which varies with altitude) the water evaporates, thereby absorbing calories and thus restoring the air temperature to a value close to that of the water boiling point.

Installation (refer to Fig 13-33)

13132. The conditioning air inlet and outlet tubes (2 and 8) are connected to the aircraft system through a semi-flexible SEMCA coupling (14) consisting of a nozzle (15) with seal (16), a washer (17) and a lock ring (18). The connection is sealed off by an 0-ring seal (19), half-rings (20) and a flexible screw-clamp (21).

13133. The steam outlet orifice (3) is connected to the aircraft system by a rigid pipe (9) attached to the structure and sealed by an O-ring seal (23).

13134. The evaporator is attached to the aircraft structure by :

a. At the front : by connecting steam outlet (3) to rigid pipe (9).

b. At the rear : by assembling the swivelling end-fittings (5) to gussets (24) on the structure. The necessary rigidity is provided by the screws and nuts (25).

DUAL HEAT EXCHANGER

Description (refer to Fig 13-34)

13135. This air-to-air exchanger is designed to lower the conditioning air temperature through thermal exchange with ram air from outside.

13136. The unit consists of two exchangers connected in series into the ram air circuit, namely a primary exchanger (1) and a main exchanger (2). These exchangers are in the form of welded sheet metal cylinders attached to each other by screws (3). There are two main baffles (4), one at the front and one at the rear. The air to be cooled is ducted through the unit by internal baffles (5) separated by welded plates (6) (one baffle in the primary exchanger and three in the main exchanger).

13137. Running across the unit are tubes (7), crimped to the main baffles, through which the cooling air passes. Conditioning air inlet and outlet flanges (8) are provided on one side of each exchanger. At the front of the main exchanger and at the rear of the primary exchanger, there are rims which accommodate the cooling air inlet and outlet ducts.

Operation

13138. The conditioning air is ducted by the baffles around the tubes through which the cooling air passes. The same cooling air first passes through the primary exchanger and then through the main exchanger.

Installation

13139. The dual heat exchanger is installed similarily to the air-to-air exchanger, refer to paragraph 13126.

13140. The primary exchanger (1) and main exchanger (2) which are integral with each other, are attached to gussets by means of flexible clamps. The ram air inlet and the Giffard duct are secured to the dual exchanger by rigid clamps. The conditioning air inlets and outlets (8) are connected to the aircraft system by semi-flexible SEMCA couplings.

AIR-TO-FLUID HEAT EXCHANGER — III O ONLY

Description (refer to Fig 13-35)

13141. The air-to-liquid (FHS) exchanger is designed to cool the radar FHS conditioning fluid through thermal exchange with the equipment conditioning air whose temperature at the exchanger inlet should be approximately 10°C.

13142. The exchanger is a welded light alloy sheet metal cylinder (1) with stiffening baffles. The two main baffles (2), at the front and rear, carry a rim (3) to accommodate the cooling air inlet and outlet ducts. The inner baffles (4) are staggered.

13143. Running across the unit are tubes (5) which admit the cooling air; these tubes are crimped to the main baffles. The cylindrical body also carries two conditioning fluid inlet and outlet flanges (6).

Operation

13144. The cooling air flows through the tubes (5) around which circulates the conditioning fluid. The fluid is deflected from one compartment into the next by the action of the baffles.

Installation

13145. Installation is similar to that for the air-to-air exchanger (refer to para 13126 except that the conditioning fluid inlet and outlet connections (6) carry Quinson type swivel couplings (7) for connecting to the lines (10) of the aircraft system. Sealing is ensured by ARSAERO crimped rings (8) and nuts (9).

WATER SEPARATOR — III O ONLY

Description (refer to Fig 13-36)

13146. The water separator is designed to lower the percentage of free water contained in the conditioning air.

13147. The water separator consists of a cylinder (1) which is internally lined with foam rubber (2) and houses a special felt tube (3). The air is admitted at one end of the tube; the other end of the same tube is closed by a valve (4) which opens in case of freezing of the liner.

13148. In normal operation, the moisture contained in the air forms small droplets which collect together and enlarge, first when crossing the felt liner and then when striking the inner wall of the separator. The water is discharged through a drain (5), located at the bottom of the evaporator, via a vent in the fuselage LH side. The separator is externally heat insulated with polyurethane.

Installation

13149. The separator is attached under the roof of the nose wheel well LH side, behind frame 10. It is held on a sheet metal support (7) by a strap (6) tightened by screws. The front coupling is fitted into a rim (8) at frame 10, with seals (9). The connections to the system are made by semi-flexible SEMCA couplings (10) with seals.

WATER SEPARATOR — III D ONLY

Description (refer to Fig 13-37)

13150. The purpose of the water separator, which is connected at the outlet of the conditioning assembly, is to reduce the water content in the air distributed to the cabin and the nose cone.

13151. The water separator basically consists of a body in two parts (1 and 2) each carrying a nozzle for connection to the system. The two parts are connected by screws and flanges on a cut-out partition (3). In the centre of the partition is a spring-loaded valve (4) which opens in the air flow direction. Attached to the front face of the partition is a cylindrical perforated plate support (5) the opening of which is aligned with the cut-outs in the partition. The attaching screws of this support are also used for attachment of a deflector (6) to the rear face of the partition. The support is provided with a Terylene liner (7) and a nylon envelope (8) held in position by a retaining ring, a binding and a large diameter spring.

13152. Both parts (1 and 2) of the body includes a connecting plug (9). Part (2) also incorporates a water discharge coupling (10) at its lower part. The water separator assembly is externally provided with heat insulating material (11) retained by lacing and binding cord.

Operation

13153. The droplets in suspension in the conditioning air first enlarge when passing through the liner (7 and 8) and then strike the rear wall of part (2) of the body. The released water is discharged to the outside of the aircraft through a line connected to the discharge coupling (10).

13154. In case of icing in the separator, excessive pressure in part (1) of the body causes the valve to open, which allows the conditioning air to flow to part (2) of the body.

Installation

13155. The water separator is attached to a sheet metal support (13) at the rear of the nose cone by means of two straps (12) with screws. Connection to the air conditioning system is made with semi-flexible SEMCA couplings and seals. A banjo union (14) is used for connection of the water discharge line.

CHAPTER 14

RADIO SYSTEMS

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CHAPTER 14

RADIO SYSTEMS

Table 14-1 Function of Components

Index No	Description	Characteristics and Functions
37F	PHI computer	Consists of electronic and mechanical sub-assemblies (which perform the computation and switching necessary for navigation purposes) and memory units (used for navigation resetting).
38F front cockpit 238F rear cockpit	PHI indicators	Include : a. A moving compass card, a pointer and a distance counter showing the bearing, distance and heading information furnished by the computer (37F). b. A mode selector, the following three positions of which are used : (1) SET .... Resetting. (2) DR .... Dead reckoning navigation. (3) TCN .... TACAN. c. A BRG knob for bearing resetting. d. A DIST knob for distance resetting. Provided with edge lighting adjustable through the PHI rheostat.
39F	PHI control unit	Includes : a. A housing for the station storage unit. b. A green press-to-test sequence light. c. A WIND SPEED button for manual setting of wind speed. d. A WIND DIRECTION button for manual setting of wind direction. e. A GRIVATION setting button. f. A 12-position station selector. Provided with edge lighting adjustable through the PHI rheostat located on LH console.
1R(III O)	Ground telephone connection	With the switch on the radio selector box (3R) in the TB position; permits communication between pilot and ground crew
2R	Ground interphone matching box	Includes a three-position selector switch : A (British), F (French), US (American) to align the characteristics of the headset used with those of the amplifier.
3R(III O) 43R (front cockpit III D) 243R (rear cockpit III D)	Radio selector box	Contains : a. A dual-range selector with : (1) Range N (normal) : 4 positions. (2) Range EMG (emergency) : 3 positions. b. Ground interphone circuit indicator light. c. Two illumination lights for the controls. d. Three AF volume adjustment potentiometers :ICS — UHF 1 (green) — UHF 2 (red).
4R	UHF 1 (green) antenna	Serves for both transmission and reception on unit (26R).
5R, 10R, and 205R	Microphone press-to-talk buttons	Permit transmission through the unit selected at the radio selector boxes (43R and 243R) and transmission for position identification through the IFF (1P).
6R	TACAN-SW switch	Permits reception of TACAN missiles (MATRA R550 Post-Mods 1112 and 1113).
8R (III D)	Noise suppressor	Suppresses any interference in the electrical power supply to the flight crew interphone (9R, 43R and 243R).
8R (III O)	Mixing amplifier	Amplifies all audio channels except the warning horn and tail warning radar circuits.
9R (III D)	Junction box	Interconnects the radio selector boxes (43R and 243R). Receives channels :UHF 1, UHF 2, audio warning horn, TACAN (MATRA R550 Post-Mods 1112 and 1113) Includes : a. One preamplifier and one amplifier for the flight crew interphone. b. Two separate amplifiers for reception of the audio warning horn.
17R 18R	UHF 2 (red) antennas	Interconnected by a coaxial tee-connector (19R); coupled in parallel; serve for both transmission and reception on unit (22R).
21R	Power supply unit	Supplies HT and LT currents to unit (22R).

Index No	Description	Characteristics and Functions
22R	UHF 2 (red) transmitter-receiver	Operates on 20 preset frequencies between 225 and 399.95 MHz.
23R Front cockpit 223R rear cockpit	UHF 2 (red) control boxes	Selected at the selector box (203R) (III D Only) permits, either from the front cockpit or the rear cockpit, switching on, frequency selection and volume adjustment of unit (22R).
26R	UHF 1 (green) transmitter-receiver	Communications transmitter-receiver operating within frequency range from 225 to 399.95 MHz.
27R	Isolating transformer	Eliminates circulating currents between input transformers of units (26R) and (22R) during simultaneous transmission.
28R	UHF 1 (green) control box	Controlled from the front cockpit only; permits switching on, frequency selection and volume adjustment of unit (26R).
29R (III D)	Power supply unit	Supplies HT and LT currents to unit (26R).
30R 31R (III D)	Ground telephone door micro-switch Ground telephone connection	With the ground telephone door open and the connection established, permits : a. Telephone communications between the two pilots and the ground crew (provided the UHF 1 (green) set is in operation) without depressing the microphone press-to-talk button). b. Ground testing MATRA R550 (Post-Mods 1112 and 1113).
37R	Transmitter-receiver	Transmitter : operates within a frequency range from 1025 to 1150 MHz with 126 transmission channels. Receiver : operates within a frequency range from 962 to 1024 MHz and from 1151 to 1213 MHz with 126 reception channels.
38R	TACAN control unit	Includes : a. Two coaxial knobs for : (1) Switching on and operating mode selection. (2) Adjustment of identification signal volume. b. Two coaxial rotary selector switches for channel selection (selected channel appears in two windows). c. Two edge illumination lights.
39R 40R	Antenna	Similar antennae operating separately according to the position of the antenna selector switch (41R).
41R	Antenna selector switch	Controlled by the antenna selector (42R), connects one of the two antennae to unit (37R).
42R	Antenna selector	Controls the antenna selector switch (41R) as a function of rèceived signal strength.
203R (III D)	Selector box	Controlled from the rear cockpit only; permits selection of the front cockpit UHF 2 control box (23R) or rear cockpit UHF 2 control box (223R).
7S	IFF coder transmitter-receiver	The transmitter enables pulse trains to be transmitted for identification purposes. The receiver receives interrogation pulses. The transmitter operates in the 1080 to 1100 MHz range and the receiver in the 1018 to 1042 MHz range.
8S	IFF control box	Includes : a. On the control side : (1) A five-position selector : OFF — STDBY — LOW — NORM — EMERGENCY. (2) Two switches (modes 2 and 3). (3) A three-position selector (transmission mode selection). (4) Two lighting bulbs. b. On the coding side : (1) Two switches (modes 1 with 32 combinations and 3 with 64 combinations). (2) A lighting bulb.
11S	IFF antenna	Serves for both transmission and reception.
12S	Distress switch	Provides for automatic transmission of distress signals in case of ejection of the seat.

DESCRIPTION (refer to Figs 14-1 to 14-3)

1401. The radio system consists of the following major components :

a. Radio communications :

(1) Main UHF.

(2) Standby UHF.

(3) Ground interphone.

b. Radio navigation :

(1) Tacan.

(2) Navigation computer.

c. Operational radio equipment :

(1) Tail warning radar (optional fit).

(2) IFF.

1402. These individually controlled installations are linked to a common circuit terminated by the earphones and microphone assembly of each pilot. A ground telephone connection (31R) provides for telephone communications between the pilots and the ground crew.

COMMON CIRCUIT.

Description (refer to Figs 14-4 to 14-6)

1403. The common circuit :

a. Provides the link between :

(1) The above mentioned circuits.

(2) The missile audio circuit.

(3) The TACAN circuit.

(4) For III D, the two pilots for telephone communication.

b. Is permanently connected to :

(1) The audio warning horn.

(2) The tail warning radar audio circuit (if fitted).

1404. The common circuit consists of the following units :

a. For III D Only, two radio selector boxes (43R front cockpit — 243R rear cockpit) and for III O, a radio selector box (3R).

b. For III D Only, a junction box (9R).

c. For III D Only, a noise suppressor (8R).

d. TACAN — SW switch (6R).

e. A microphone press-to-talk button (5R and for III D 205R) on control stick hand grip in both cockpits.

f. One front cockpit and one rear cockpit connector (47Y and for III D 47Y1 and 47Y2) for aircraft-to-seat connection.

g. One front cockpit and one rear cockpit connector (11R and for III D 11R1 and 11R2) on the ejection seat for seat-to-pilot connection.

h. One front cockpit and one rear cockpit connector for connector-to-headset connection.

Radio Selector Boxes — III D Only (refer to Fig 14-6)

1405. The radio selector boxes (43R front cockpit — 243R rear cockpit) permit separate or simultaneous operation of all radio communications systems by each pilot. They include :

a. A two-sector selector switch having three normal and four emergency positions.

b. Three potentiometers, which only operate in the N sector, enable volume adjustment of :

(1) UHF 1 (green dot).

(2) UHF 2 (red dot).

(3) Flight crew interphone (ICS).

c. A built-in reception mixing amplifier bypassed when on EMG sector.

d. A built-in relay controlled by the microphone buttons to switch over from interphone or reception to transmission on UHF 1 and UHF 2 sets.

1406. The radio selector boxes are provided with edge lighting the intensity of which can be adjusted by the Internal Red rheostat located on the LH consoles.

1407. With the selector switch of the radio selector boxes in the N (normal) position (positions 1-2-3), amplification is provided by the built-in reception mixing amplifier for the following channels :

a. Flight crew interphone.

b. UHF 1 (green).

c. UHF 2 (red).

d. TACAN (or MATRA R550, Post-Mods 1112 and 1113).

1408. The TACAN-SW switch (6R) located on the LH console permits TACAN (or MATRA R550, Post-Mods 1112 and 1113) reception through the flight crew interphone.

1409. **Selector Switch — Normal Operation.** The functions of the selector switch normal positions are :

a. Red dot : UHF 2 (red) set transmission plus all receptions in parallel.

b. Green dot : UHF 1 (green) set transmission plus all receptions in parallel.

c. Green and red dots : Simultaneous transmission on UHF 1 and 2 plus all receptions in parallel.

1410. **Selector Switch — Emergency Operation.** The functions of the selector switch emergency positions are :

a. ICS : operation as flight crew interphone.

b. Green dot : Communications on UHF 1 (green) set only.

c. Red dot : Communications on UHF 2 (red) set only.

d. TAC : Operation TACAN (or MATRA R550 Post-Mods 1112 and 1113) mode only (TACAN-SW switch (6R)).

Radio Selector Box — III O Only (refer to Fig 14-7)

1411. The radio selector box (3R) permits separate or simultaneous operation of the various units listed above. It includes :

a. A seven-position selector having four normal positions (N) and three emergency positions (EMG) The N and EMG sectors are separated by neutral positions.

b. A light indicating that the ground interphone is energized.

c. Two bulbs for edge lighting of the controls. Their brilliance is adjusted by means of the Internal Red lighting rheostat on the port console.

1412. **Selector Switch — Normal Operation.** The functions of the selector switch normal positions are :

a. Green dot : Main UHF transmission (green set) + all audio circuits in parallel.

b. Red dot : Standby UHF transmission (red set) + all audio circuits in parallel.

c. Red and green dots : Simultaneous transmission from main and standby UHF + all audio circuits in parallel.

d. TB : Ground interphone + missile audio circuits.

1413. **Selector Switch — Emergency Operation.** The functions of the selector switc.. emergency positions are :

a. Green dot : Separate operation of the main UHF.

b. Red dot : Separate operation of the standby UHF.

c. TAC : Separate operation of the TACAN.

1414. **Audio Channels.** On the normal (N) positions of the selector switch on the radio selector box, the standby UHF, main UHF, TACAN and missile audio channels are amplified by amplifier (8R). On the normal position TB, the missile heating signal can also be received.

1415. Emergency (EMG) positions are not routed through amplifier (8R). The warning horn and tail warning radar audio channels are not amplified and can be heard on all the selector positions including all neutral positions.

Junction Box — III D Only

1416. The junction box (9R) provides interconnection between the radio selector boxes (43R and 243R). Its input channels are : UHF 1, UHF 2, audio warning horn, TACAN or (or MATRA R550, Post-Mods 1112 and 1113) as selected by the TACAN-SW switch (6R). The junction box includes :

a. One preamplifier and one amplifier forming the flight crew interphone amplifying network.

b. Two amplifiers used as separator amplifiers for audio warning horn reception.

1417. The audio warning horn is audible on all positions of the selector switch on each radio selector box, including the neutral positions.

Flight Crew Interphone — III D Only (refer to Figs 14-5 and 14-6)

1418. The flight crew interphone is operational as soon as the main d.c. bus is energized by the battery, generator or ground power unit. It is not necessary to press a microphone button to operate the flight crew interphone in any position within the N (normal) sector and in the ICS position within the N sector and in the ICS position within the EMG (emergency) sector on the radio selector boxes. The volume can be adjusted only in the positions within the N sector by means of the ICS potentiometer on each radio selector box.

Ground Telephone — III D Only (refer to Fig 14-8)

1419. The ground telephone provides communication between both pilots and the ground crew. Communication requires :

a. Operation of the UHF 1 (green) set and of the flight crew interphone.

b. Connection of the ground telephone connection (31R) located beneath the fuselage (frame 25 — LH side). In the open position, the ground telephone connector door actuates a micro-switch (30R) which permits :

(1) Communications between the two pilots and the ground crew.

(2) Ground testing of MATRA R550 (Post-Mods 1112 and 1113) missiles.

Ground Telephone — III O Only (refer to Fig 14-9)

1420. The aircraft circuit is used to connect the common circuit to the ground connection; the amplifier is part of the ground circuit. The installation consists of :

a. A matching box (2R).

b. A ground connection (1R).

c. The common circuit.

1421. The matching box (2R) matches the characteristics of three different headsets to that of the amplifier by a three-position switch (1). The various positions are as follows :

a. A — British microphone and earphone circuits directly connected.

b. F — French microphone signal amplified (by the pre-amplifier) and ear-phone circuit taken through the auto-transformer.

c. US — USA carbon type microphone excited through EXCITATION resistance capacitor circuit. Earphone circuit taken through auto-transformer.

1422. The ground connection (1R) is attached to a spring-loaded door. When the aircraft leaves the parking area, the connection disconnects and the door closes under the action of the spring.

1423. **Operation.** With the switch on matching box (2R) at the position corresponding to the type of headset employed, and with the ground connection plugged in, set the switch on the radio selector box (3R) to TB this causes the red light to illuminate and the system is operational. There is no need to press the microphone button for speech transmission and UHF signals are no longer received. With the switch in any position other than TB, there is no reception through the ground interphone and the red light extinguishes (even with the ground amplifier connected).

1424. Current is supplied to the ground interphone circuit through a pre-heating relay (39P) that is activated when switch (40P) is set to PRE-HEAT. This permits the ground telephone circuit to be supplied from the pre-heating bus. Relay (39P) ceases to be energized when switch (40P) is set to Normal and supply to the ground interphone circuit is supplied automatically by the main bus.

RADIO SYSTEMS EQUIPMENT LOCATION

1425. The location of the major items of equipment for the radio and navigation systems is given in Table 14-2.

Table 14-2 Location Of Radio And Navigation Components

Component	Location — III D	Location — III O
COMMON CIRCUIT		
Matching box (2R)		Port wing root lower section between frames 26 and 27.
Radio selector box (3R)		Port console
Microphone press-to-talk buttons (5R and 205R)	Front and rear cockpit control stick hand grips. Control stick hand grip	
Microphone press-to-talk button (10R)	Front cockpit throttle control lever Throttle control lever	
Amplifier (8R)		In the port console
Noise suppressor (8R)	Centre section, frame 10	
Junction box (9R)	Centre section, frame 10	
Filter (9R)		Port console, frame 4
Radio selector box (43R)	Front cockpit instrument panel	
Radio selector box (243R)	Rear cockpit control pedestal	
Headset connector	On seat personnel equipment connector	On seat personnel equipment connector
Ground Telephone Connection (1R)		On a spring door at frame 35 on the LH side.
Ground Telephone Connection (31R)	Below the fuselage at frame 25 on the LH side.	
MAIN UHF (GREEN) INSTALLATION		
Transmitter-receiver (26R)	LH side bay, frames 33 to 35	Equipment bay, tilting rack
Control box (28R)	Front cockpit LH console	LH console
Power supply unit (29R)	LH lower section, frames 16 and 17	

Component	Location — III D	Location — III O
Matching transformer (27R)	LH side bay, frame 35	Equipment bay, tilting rack
Antenna (4R)	Fin tip	Fin tip
STANDBY UHF (RED) INSTALLATION		
Transmitter-receiver (22R)	RH side bay, frames 33 to 35	LH rear lateral bay tilting rack
Control boxes (23R and for III D, 223R)	Front and rear cockpit LH console	(23R) LH console
Selector box (203R)	Rear cockpit LH console	
Power supply unit (21R)	LH side, frame 17	LH side, frame 16
Antenna (17R)	Fin leading edge	Fin leading edge
Antenna (18R)	Fin trailing edge	Fin trailing edge
TACAN SYSTEM		
TACAN — SW switch (6R)	Front cockpit LH console	
Transmitter-receiver (37R)	Nose cone	Equipment bay, multiple rack
Antenna selector switch (41R)	Lower bay, frame 21	Lower bay, frame 21
Antenna selector (42R)	Nose cone	Equipment bay, multiple rack
Antenna (39R)	RH lower surface wing fillet, frame 29	RH lower surface wing fillet, frame 29
Antenna (40R)	Fuselage upper surface	Fuselage upper surface
Control unit (38R)	Front cockpit RH console	Cockpit RH console
PHI NAVIGATION SYSTEM		
Rheostats (41L and 241L)	LH console	centre console
Computer (37F)	Frames 3 to 5, Lower section	Radio bay
Control unit (39F)	Front cockpit RH console	RH console
Indicator (38F)	Instrument panel RH side	Instrument panel RH side
Indicator (238F)	Instrument Panel RH Side	
IFF		
Transponder (7S)	Nose cone	LH side of equipment bay, between frames 12 and 13.
Control panel (8S)	Front cockpit RH console	RH console

Component	Location — III D	Location — III O
Distress switch (12S)	Frame 10 front face, Upper section	Frame 10 front face
Antenna (11S)	Nose undercarriage rear door	Nose undercarriage rear door
Transponder test set (32S)		Radio equipment bay

RADIO COMMUNICATIONS INSTALLATION

Description

1426. The UHF sets permit communications in the 225 to 399.95 MHz band. There are 3500 available frequencies with a 50 kHz inter-channelspacing. There are two separate circuits, main UHF (green) and standby UHF(red).Each of these circuits allows transmission and reception from a control box and is connected to the common circuit.

1427. An additional receiver, preset to a fixed frequency selected between 238 and 248 MHz, permits permanent reception on this fixed frequency which is known as the guard frequency. This receiver is incorporated in unit (26R).

1428. For III D, energizing of the UHF 1 (green) set and selection of its frequencies can only be performed from the front cockpit. Consequently, the rear pilot can use the UHF 1 (green) set only on the frequency selected by the front pilot.

1429. For III D, energizing of the UHF 2 (red) set and selection of its frequencies can be performed from both cockpits by means of a selector box (203R) in the rear cockpit which selects either the front cockpit control box (23R) or the rear cockpit control box (223R).

Main UHF Circuit (Green) (refer to Figs 14-8 and 14-10)

1430. The main UHF circuit includes

a. The main UHF (green) transmitter-receiver (26R) incorporating a transmitter and two receivers.

b. Control box (28R).

c. Antenna (4R).

d. Isolating transformer (27R).

e. For III D Only, a power supply unit (29R).

1431. The main UHF (green) transmitter-receiver (26R) consists of :

a. Communications transmitter-receiver.

b. Guard receiver preset to a single frequency between 238 and 248 MHz capable of permanent reception.

c. AF amplifier used by both receivers (guard and communications) to amplify the output signal.

NOTE

The AF amplifier of the receiver is also used in connection with the ground telephone when the ground telephone connection access door is open (microswitch 30R) and connection is made to the ground telephone connection.

d. Power supply unit which supplies HT and LT current to the above three items (unit (29R) for III D).

1432. The control box (28R), front cockpit only, is used to switch on the equipment, set the frequency and adjust the volume.

1433. **Main UHF — Switching On (refer to Figs 14-8 and 14-10).** The equipment is switched on by the four-position switch (8). The action of each switch position is :

a. OFF — All equipment OFF.

b. MAIN — Transmitter and communications receiver ON.

c. BOTH — Transmitter, communications receiver and guard receiver ON.

d. ADF — Not used.

1434. **Main UHF — Frequency Setting.** Frequency setting is by three pushbuttons marked M, P and G. M is for manual setting, P is for automatic setting of a preset frequency and G is for transmission and reception on the guard frequency on the communications transmitter-receiver.

NOTE

The guard receiver is operative when the selector switch is set to BOTH, even if the pushbuttons (M, P and G) are not depressed.

1435. Manual setting is achieved by depressing button M and using the five rotary switches to select any of the 3500 available frequencies :

a. Switch 1 — one graduation = 100 MHz.

b. Switch 2 — one graduation = 10 MHz.

c. Switch 3 — one graduation = 1 MHz.

d. Switch 4 — one graduation = 0.1 MHz.

e. Switch 5 — one graduation = 0.05 MHz.

1436. Automatic setting is selected by depressing button P. A drum (6) with 20 indexed positions permits automatic frequency selection by rotation opposite a fixed index. The 20 frequencies corresponding to the 20 positions are preset and indexed from 1 to 20. For frequency seeking, the controls act on the communications transmitter-receiver only and not on the guard frequency.

1437. **Volume Adjustment.** Volume adjustment is provided by potentiometers (7) on the radio selector boxes.

1438. **Control Box Lighting.** Edge lighting of the controls is ensured by four lamps (9) energized through the Internal Red rheostat on the port console. Lighting intensity is adjusted by two rheostats located in the box; one rheostat controls the two lower lamps and the other the two upper lamps.

1439. **A2 Modulation.** In the A2 mode, 1000 Hz is obtained by pressing the TONE button on control box (28R) (used for position finding).

1440. **Antenna.** The antenna (4R) is located at the top of the fin, under a fibreglass fairing. The unit is independent of the aircraft structure. Its SWR is less than two in the range involved.

1441. **Isolating Transformer (27R).** This transformer isolates the mid-point connections of the modulation transformers of both sets in simultaneous transmission conditions.

Standby UHF Circuit (Red) (refer to Figs 14-11 and 14-12)

1442. The standby UHF circuit includes :

a. A power supply (21R).

b. A transmitter — receiver (22R).

c. Two control boxes (23R front cockpit — 223R rear cockpit).

d. Two antennae (17R and 18R).

e. For III D Only, a selector box (203R).

1443. The communications transmitter-receiver (22R) covers the same frequency band as the main UHF but at lower power. It is connected to an AF amplifier which amplifies the receiver output signals.

1444. The control boxes incorporate the function switch and the automatic frequency setting and volume adjustment controls.

1445. **Standby UHF — Switching On.** The equipment is switched on by the three-position switch (14). The action of each switch position is :

a. OFF — All equipment OFF.

b. NORMAL — Transmitter-receiver ON.

c. ADF — Not used.

1446. **Frequency Setting.** Automatic frequency setting is accomplished by a 20-position drum (15). Each position corresponds to one of 20 frequencies, indexed from 1 to 20, which are preset prior to the mission.

1447. **Volume Adjustment.** Volume adjustment is accomplished by potentiometer (16).

1448. **Control Box Lighting.** The controls are edge-lit by two lamps (17 and 18) energized through the Internal Red rheostat on the port console. Lighting intensity is adjusted by means of a rheostat located in the box.

1449. **A2 Modulation.** A2 modulation (1000 Hz) is obtained by pressing the TONE button (19) on either control box (used for position finding).

1450. **Antennae.** Two antennae are used with the standby UHF; A fin leading edge antenna (17R) and a fin trailing edge antenna (18R). The antennae are connected in parallel through a coaxial tee-connector (19R) located upstream of an impedance matching transformer. The SWR for these antennae is less than two.

1451. **Selector Box — III D Only.** The selector box (203R) is fitted with a two position selector. One position is identified FNT and the other by an arrow symbol. These positions determine the operation and information frequency setting of the standby UHF as follows :

a. FNT position — from the front cockpit control box (23R).

b. Arrow symbol position — from the rear cockpit control box (223R).

1452. The intensity of the control box edge lighting can be adjusted by the Internal Red rheostat.

RADIO NAVIGATION SYSTEM

TACAN (refer to Figs 14-13 and 14-14)

1453. The TACAN installation is a radio-navigation system through which the aircraft is supplied with magnetic bearings and the distance from a selected ground beacon. The indications are given up to 292 NM in direct line of sight. The ground beacon is identified by its call sign issued in Morse code every 37.5 seconds. These indications are read off the PHI indicators (38F front cockpit — 238F rear cockpit) after setting the selector on the PHI to TCN. A full description of the TACAN can be found in AAP 7213.001-2-4, Section 2.7.

1454. The TACAN installation consists of :

a. A transmitter-receiver (37R).

b. An antenna selector system (42R).

c. An antenna selector switch (41R).

d. A control box (38R).

e. An upper antenna (40R) and a lower antenna (39R).

f. A TACAN-SW selector switch (6R) on the front cockpit LH console, ref para 1408.

1455. **Transmitter-receiver.** The transmitter-receiver is equipped with 126 working channels having 1 MHz separation. Interrogation frequencies are in the 1025 to 1150 MHz band whilst receiver frequencies are in the 962 to 1024 MHz and 1151 to 1213 MHz bands.

1456. The transmitter-receiver is cooled through a duct tapping air from the cabin air conditioning system. On the ground with the engine stopped, the TACAN system must not be energized if a ground cooling unit is not available.

1457. **Control Box.** The control box (38R) includes :

a. Two coaxial rotary selector switches (1 and 2) for channel selection. Switch (1) sets the tens and switch (2) the units which appear in two windows.

b. A two coaxial knob :

(1) The lower knob (3) is used for operating mode selection; its posi-

tions are as follows :

(a) OFF — Off.

(b) ST-BY — Heating.

(c) REC — The receiver only is operating. The set receives bearing and identification signals from the beacon.

(d) NOR — Both the transmitter and the receiver are operating. Bearing and range signals are provided.

(e) T/R. Transmitter-receiver on.

(f) SHORT — Range finding time is reduced.

(2) The upper knob (4) controls the volume of the station identification audio signal when switch (6R) is in the TACAN position.

c. Two edge illumination lights (5) the intensity of which can be adjusted by the Internal Red rheostat on the LH console.

1458. **Antenna Selector.** To ensure the best possible transmission and reception is obtained, the antenna selector (42R) selects the antenna which is best located relative to the TACAN beacon. The selector is adjusted to operate at a minimum signal strength pick-up threshold.

1459. When the signal transmitted by the selected antenna becomes weaker than the threshold value, the selector causes switching over to the other antenna. If the signal transmitted by the second antenna is stronger than the threshold value, this antenna remains connected. If the signal is insufficient, switching continues until a strong enough signal appears at one of the two antennae.

1460. **Antenna Selector Switch.** The antenna selector switch (41R) is controlled by the antenna selector (42R) to connect one of the two antennae (39R or 40R) to the transmitter-receiver (37R).

1461. **Equipment Installation (refer to Figs 14-1 and 14-14).** The main components of the TACAN system are installed as follows :

a. *Transmitter-receiver (37R) :*

(1) For III O Only. The transmitter-receiver is installed on the RH side of the multiple rack (1) in the equipment bay between frames 14 and 15. The rear face slides in two guide rails (2) riveted to the multiple rack and the bottom face is located by two pins (3). Transmitter handling, installation and plugging-in is facilitated by a handle (4) which is fitted with a locking device and includes two rollers (5) which engage in two hooks (6) riveted to the rack. Connection to the electrical and conditioning systems is achieved through two connectors (7 and 8) on the bottom section of the rack.

(2) For III D Only. The transmitter-receiver is mounted in the nose cone as described in Chap 4, para 4184.

b. *Antenna Selector (42R) :*

(1) For III O Only. The antenna selector is secured on the rack LH side by angle section (9) at its lower part and two screws (10) at its upper part.

(2) For III D Only. The antenna selector is mounted to the equipment rack in the nose cone forward of the transmitter-receiver.

c. *Antenna Switch (41R).* The antenna switch is secured by two screws to the LH web of the accessory gear box compartment level with frame 21.

d. *Control Panel (38R).* The control panel is secured by four screws to the front part of the front cockpit RH console.

e. *Antennae.* Each antenna is secured by eight screws to the structure as follows :

(1) The upper antenna (40R) to the fuselage top part on the aircraft centre line between frames 23 and 24.

(2) The lower antenna (39R) beneath the starboard lower surface wing fillet between frames 28 and 29.

PHI Navigation System (refer to Figs 14-15 to 14-17)

1462. The PHI navigation system enables dead reckoning navigation by preparing and supplying the pilot with the range and bearing of a selected station using the following data :

a. Airspeed and heading supplied by the air data computer and gyro centre.

b. Preset co-ordinates of the departure base.

c. Wind speed and direction settings.

1463. A full description of the PHI navigation system is contained in AAP 7213.001-2-4, Section 2.6.

1464. The main components of the PHI navigation system and their functions are as follows :

a. Navigation computer (37F). The computer carries out the various navigation computation and switching and provides the necessary memory functions for navigation resetting.

b. Navigation indicator (38F front cockpit — 238F rear cockpit). The navigation indicator comprises the following units :

(1) A mobile compass card moving in front of a lubber line to indicate the aircraft heading according to the heading selector position (gyro unit control panel).

(2) A pointer moving in front of the mobile compass card indicating the bearing to the selected station or

TACAN beacon or the mean antenna azimuth.

(3) A steering index (bug) giving the same information as the pointer. When in the ANT-AZ position, of the function selector switch, it indicates the bearing of the selected station.

(4) Two reference marks located at 60° on either side of the lubber line that show the maximum scan angle of the radar antenna.

(5) A range meter with three scale drums that indicate the distance in NM to the selected station or TACAN beacon.

(6) A function selector switch marked : IN, SET, D/R, TCN and ANT-AZ. The following switch positions are used :

(a) SET (reset). On this position, the pilot can correct the distance and bearing data displayed on the indicators, thus allowing computer resetting.

(b) DR (dead reckoning). On this position, the data computed from the true airspeed, the aircraft grid heading, the wind setting and the selected station co-ordinates are displayed.

(c) TCN (TACAN). On this position, the indicator displays TACAN information, ie, distance and magnetic bearing of the selected TACAN station.

(7) Two setting knobs, DIST and BRG used to correct distance and bearing information when the function selector switch is in the SET position.

c. Control panel (39F) which contains :

(1) A station selector.

(2) A wind velocity manual setting knob.

(3) A wind direction manual setting knob.

(4) A grivation setting knob.

(5) A sequence light that illuminates when the computer is resetting. This is a press-to-test type indicator.

(6) A station storage unit.

d. Station Storage Unit. The station storage unit is a rectangular shaped box plugged in to the control panel in which the cartesian co-ordinates of the eleven selected stations are preset in relation to the origin of the PHI grid (generally the departure base). The 22 adjusting potentiometers are guarded. Additional stations can be obtained by changing the station storage unit in flight.

1465. **Equipment Installation (refer to Figs 14-2 and 14-15).** The main components of the PHI navigation system are installed as follows :

a. Navigation computer (37F).

(1). For IIIO Only. The navigation computer is located in the equipment bay between frames 14 and 15 on the same multiple rack as the multiplier units (2C) and (27F).Four attaching lugs (1) are secured by nuts and bolts (2) to fittings provided on the rack structure.

(2). For IIID Only.The navigation computer is located at the bottom of the fuselage between frames 2 and 5. Mounting details are given in Chap 4, para 4199.

b. Navigation indicator (38F front cockpit - 238F rear cockpit). The navigation indicator is located on the RH upper section instrument panel for both cockpits and is held in position by three screws.

c. Control panel (39F). The control panel is mounted on the front cockpit RH console and is attached by four quick release fasteners.

d. Station storage unit. THe station storage unit plugs into the front face of the control panel and is held in position by a fitting on the panel face.

OPERATIONAL RADIO INSTALLATIONS

Tail Warning Radar (Optional Fit)

1466. The tail warning radar is an optional fit equipment and is intended to activate a warning system which informs the pilot that the aircraft has been illuminated by radar at regular intervals for more than two seconds. This information is given in the form of an audio signal and a combination of lights. The UHF reception band is 8500 to 11000 Hz

1467. The equipments listed below are the main items required for tail warning radar :

a. An antenna pack (21S) at the fin trailing edge, consisting of an antenna, a motor and an alternator.

b. A receiver unit (40S) in the lower compartment on the RH side of frame 33.

c. A warning panel (39S) on the RH side of the instrument panel.

d. A video pre-amplifier (22S) in the fin.

e. A radar detector relay (38S) in the central box located in the equipment bay (III O Only).

IFF Installation (refer to Figs 14-18 and 14-19)

1468. The IFF installation provides automatic identification of the aircraft in response to interrogation from a ground radar station; the information being transmitted in coded form. In addition to its identification function, this equipment is also used for air traffic control in peacetime.

NOTE

The circuits and controls related to mode C are not described.

1469. The IFF set receives a group of coded spacing RF pulses and sends a reply selected from among the various combinations corresponding to the interrogation mode. The replies consist of 1090 MHz RF pulses; the interrogation is made at a frequency of 1030 MHz.

1470. The system consists of the following :

a. A transmitter-receiver (7S).

b. A control unit (8S).

c. Two suppressor coaxial cables (9S and 10S).

d. An antenna (11S).

e. A distress switch (12S).

f. A failure relay (30S) (caution light).

g. A control relay (31S) (I/P mode).

h. A transponder test set (32S).

1471. **Control Unit (refer to Fig 14-19).** The control unit (8S) includes the following controls and indicators :

a. A master switch (1) with five positions :

(1) OFF : — System off.

(2) STBY : — Standby.

(3) LOW : — Reduced sensitivity.

(4) NORM : — Normal sensitivity.

(5) EMER : — Distress. To select the EMER position, the master switch (1) must be pulled to override the stop.

b. Mode selector switches (2) (modes 1, 2, 3 and C) these switches are used for :

(1) Selection of the corresponding modes (ON position).

(2) Checking the operation on the corresponding modes in the absence of interrogation (TEST position).

(3) Switching off the corresponding modes (OUT position).

c. Mode 1 code selector switches (6).

d. Mode 3 code selector switches (3).

e. Test light.

f. A MONITOR control switch (4) with three positions; RAD TEST, MONITOR and OUT. This switch provides :

(1) Visual operating check of the various modes when the mode switches are in the TEST position.

(2) On the ground, in the RAD TEST position, operating check of the IFF in reply to a given ground interrogation ($T = 6.5\ \mu sec$).

(3) Visual monitoring of the IFF reply to a ground interrogation during transmission-reception in the MONITOR position.

g. Three edge illuminating lamps (5).

1472. **IFF Antenna.** The cylindrical (dish type) antenna (11S) is mounted on the nose undercarriage door by means of twelve screws.

1473. **Distress Switch.** The distress switch (12S) is attached to frame 10 through a bracket secured to the aircraft structure by screws. The switch is connected to the bracket by a cord.

1474. The distress switch is operated in the event of operation of the ejection seat. It causes transmission of distress replies in modes 1, 2 and 3/A and the IFF receiver operates in the normal sensitivity mode.

CHAPTER 15

SAFETY INSTALLATIONS

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CHAPTER 15

SAFETY INSTALLATIONS

Table 15-1 Function of Components

Index No	Description	Characteristics and Functions
12B Front cockpit 212B Rear cockpit	Wing store jettison buttons	Depressing one of these buttons closes the electrical circuit and causes : a. Jettisoning of the wing pylon tanks by firing of the corresponding initiators. b. Firing (inert) of wing mounted missiles.
49F-III O Only	Gyro centre failure warning light	Illuminates if the gyro centre power supply fails
2M Front cockpit 202M Rear cockpit	Fuselage store jettison buttons	Pressing one of these buttons closes the electrical circuit and causes : a. Jettisoning of the fuselage pylon tank. b. Jettisoning of the bombs (inert) through firing of the initiators in the racks. c. III O Only : Jettisoning of the rocket motor pack. d. Jettisoning of the missile and its pylon.
5M-III O Only	Retraction squib	When energized by pressing button (2M), this squib fires an explosive actuator, causing the rocket motor drive shaft to be retracted.
7M-III O Only	Shaft retracted microswitch	Energized simultaneously with (5M), this microswitch permits the supply of current to the ejector squibs (8M and 9M) after drive shaft retraction.
8M and 9M III O Only	Ejector squibs	Energized by (7M); these squibs fire the jettisoning initiators; the combustion gases are then ducted to the ejectors.
44Q Front cockpit 244Q Rear cockpit	Fuel tank jettison buttons	Pressing one of these buttons closes the electrical circuit and causes : a. Jettisoning of the wing pylon tanks. b. Jettisoning of the fuselage pylon tank.
1Z Front cockpit	Failure warning panel	This panel includes 18 warning lights (two lights are not used). The RH and centre row are connected to an audio warning system. In case of failure of one of the monitored circuits, the corresponding warning light illuminates. The panel also incorporates : a. Three resetting pushbuttons ...... Battery, Generator, Alternator. b. Three switches .... Battery, Generator, Alternator. c. One DAY-NIGHT selector switch. d. One test button.
201Z Rear cockpit	Failure warning panel	Identical to 1Z but does not include the battery, generator, alternator switches and resetting pushbuttons.
3Z Front cockpit	FAIL warning light	This light is illuminated whenever a failure is indicated on the warning panels (1Z and 201Z). Depressing the FAIL warning light causes it to extinguish and silences the warning horn. It is then reset to indicate any other possible failure.
7Z	Audio warning horn	This horn is simultaneously operated with the RH and centre light rows of the failure warning panels (1Z and 201Z). It is connected to the radio junction box (9R) and transmits an intermittent signal which is heard by both pilots through their earphones.
203Z Rear cockpit	Cancel horn button.	Depressing this button cancels the audio warning horn and resets it to indicate any other possible failure.

GENERAL

In Flight Safety Devices

1501. The following safety devices are provided :

a. Failure warning systems (red lights, horn).

b. Emergency controls (switches, handles).

c. Automatic circuits (electrical, hydraulic).

d. Guards and lockwires safeguarding switches against inadvertent operation.

e. Dinghy puncturing device. For III D, a dingy puncturing device is fitted in each cockpit.

On Ground Safety Devices

1502. The following safety devices are provided :

a. In the cockpit :

(1) Safety pins and plain pins.

(2) An escape knife for use in case of an accident on the ground making it impossible to open or eject the canopy. For III D, an escape knife is fitted in each cockpit.

b. On the outside of the aircraft :

(1) Protective devices and servicing equipment fitted with red streamers.

(2) Various warning signs on the aircraft external skin.

Windshield De-Icing and De-Misting

1503. The conditioning air is blown over the front and side windshield panels by diffusers. A description of windshield de-icing and de-misting is contained in Chap 13.

Table 15-2 Safety Features and Related Equipment

Related Equipment	Failure	Warning Device	FAIL light	Horn 7Z	Emergency or Safety Device
Flying aids	Autocommand failure	AC light on the failure warning panel (1Z), (and for III D 201Z)	+	+	Disengagement by AC pushbutton on panel (43C) (front cockpit) or emergency disengagement by 134C front cockpit and 334C rear cockpit.
	Altitude monitor failure				Autocommand disengagement or altitude monitor disengagement by Alt pushbutton on panel (43C) (front cockpit).
	Pitch damper failure Yaw damper failure	DAMP light on (1Z) (and for III D 201Z)	+	+	Disengagement by pushbutton P on 43C. Disengagement by pushbutton Y on 43C.
	Roll stabilizer	DAMP light on (1Z) (and for III D 201Z)	+	+	Disengagement by pushbutton R on 43C, trigger 86C (and for III D 286C) on stick hand grip or for III O pushbutton 115C.
	Stick jammed in pitch				Emergency disengagement by switch 134C (and for III D 334C) (electro-valve 135C).
Pitch and roll servos	No 1 hydraulic system	HYD 1 light on 1Z (and for III D 201Z)	+		Emergency hydraulic pump (automatic).
	No 1 hydraulic system + emergency system	HYD. 1 light +EMG. HYD. light on 1Z (and for III D 201Z)	+	+	Dual supply valve.
	No 2 hydraulic system	HYD 2 light on 1Z (and for III D 201Z)	+	+	Nil.
Airbrakes	Airbrakes not locked	Light 37C (and for III D 237C)	+		
Undercarriage (main wheels)	Overheating	Thermocolor paint			Fuse preventing tyre bursting.
Undercarriage system	Undercarriage not locked down	Red light 29G (and for III D 229G) Undercarriage down lock audio checking tone	+		Undercarriage emergency extension handle.
	Undercarriage not down (Throttle retarded and IAS<Kt)	Flashing light (red) 27G (and for III D 227G)			
	Locked down, not moved aft	Flashing light on 29G (green) (and for III D 229G)			
Brake system	Hydraulic system failure	HYD. 1 light on 1Z (and for III D 201Z)	+		Emergency brake handle.
Dinghy	Inadvertent inflation of dinghy in flight				Dinghy puncturer (Front and rear cockpits).
Canopy	Canopy not locked	CAB. P light on 1Z (and for III D 201Z)	+	+	
Emergency canopy jettisoning system	Seat curtain fails to jettison canopy (in flight)				Canopy jettison levers (front and rear cockpits).

Related Equipment	Failure	Warning Device	FAIL light	Horn 7Z	Emergency or Safety Device
Emergency canopy jettisoning system	On the ground				LH and RH handles under red plexiglass covers
Canopy locking	Canopy not locked	CAB. P light on warning panel 1Z (and for III D 201Z)	+	+	Locking lever.
Canopy actuating system	Canopy jamming or unlocking system failure				Escape knife (front and rear cockpits).
Fuel supply system	130 gal (600 l) fuel remaining	Red light 28Q (and for III D 228Q), on	+	+	
Rocket motor — III O Only	Fire	Red light 39K on			Rocket motor safety switch 29K.
Engine (dry)	Fire	Red light 4J (and for III D 204J) on			Main LP cock switch 16Q (and for III D 216Q).
Engine (afterburner)	Fire	Red light 7J (and for III D 207J) on			Afterburner cock switch 13K (and for III D 213K).
Engine afterburner operation	Afterburner ignition failure	Amber light 15K (and for III D 215K) does not illuminate			A/B emergency button 17K.
Engine operation	Engine flameout				In-flight relight switch 8K.
Fuel supply system	No fuel transfer from external tanks	Fuel transfer indicator 40Q (and for III D 240Q)			Fuel dump pushbutton 47Q.
Engine fuel control	Abnormal engine speed	RPM indicator 1E (and for III D 201E)			Emergency fuel control switch 27K.
Engine operation	Oil pressure drop	OIL light on warning panel 1Z (and for III D 201Z)	+	+	
Fuel supply system	Fuel pressure drop	FUEL light on warning panel 1Z (and for III D 201Z)	+		
Rocket motor — III O Only	Malfunctioning on the ground				Ground safety through connector 33K.
No 1 hydraulic system	Pressure drop	HYD. 1 light on warning panel 1Z (and for III D 201Z)	+		Emergency hydraulic system (automatic).
No 2 hydraulic system	Pressure drop	HYD. 2 light on warning panel 1Z (and for III D 201Z)	+	+	
Emergency hydraulic system	Pressure drop	EMG. HYD light on warning panel 1Z (and for III D 201Z)	+	+	
Automatic loadshedding (a.c. and d.c.)	Generator failure	GEN light on warning panel 1Z, (and for III D 201Z)	+	+	Generator reset button on warning panel 1Z.
Alternator busbar	Alternator failure	ALT light on warning panel 1Z (and for III D 201Z)	+	+	Alternator reset button on warning panel 1Z.
Battery busbar	Battery failure	BATT light on warning panel 1Z, (and for III D 201Z)	+		Battery reset button on warning panel 1Z.
Nose pitot-static tube or curved tube	Pitot heating failure	PITOT light on warning panel 1Z (and for III D 201Z)	+		
Aircraft skin	Excessive kinetic heating	Red MACH light 11E (and for III D 211E)			
Cockpit air conditioning system	Excessive temperature	CAB. T light on warning panel 1Z (and for III D 201Z)	+	+	Cockpit temperature control 2H.

Related Equipment	Failure	Warning Device	FAIL light	Horn 7Z	Emergency or Safety Device
Equipment air conditioning system	Excessive temperature	EQ. T light on warning panel 1Z (and for III D 201Z)	+		Switch 12H.
Cockpit pressurization system	Cockpit pressure drop	CAB. P light on warning panel 1Z (and for III D 201Z)	+	+	
Oxygen supply system	Oxygen pressure drop	8 min. OXY light on warning panel 1Z (and for III D 201Z)	+	+	
Oxygen supply system	Failure of normal regulator	OXY REG light on warning panel 1Z (and for III D 201Z)	+	+	Selector on control panel 19H (blinker masked).
Radio installation	Aircraft in distress				Simultaneous transmission and reception on both sets (by selection at radio mode selector box (3R) for III O (43R front cockpit and 243R rear cockpit for III D).
IFF installation	Ejection caution light	IFF light on warning panel 1Z (and for III D 201Z)	+	+	Automatic switch over to distress transmission through switch 12S.
Gyro centre — III O Only	Power supply failure	Light 49F			
Fire control circuits					Ground firing prevention through contactor 45G on undercarriage leg and relay 46G — Ground safety connector 90Y.
Gun circuits					Isolation of gun fire control circuit through GUNS safety switch on panel 2A.
	Jamming				Pyrotechnical recocking : a. Automatic b. Through connector 5A (LH gun) or 6A (RH gun).
Rocket circuits					Isolation of rocket fire circuit through ROCKETS safety switch on panel 2A.
Bombing circuits					Isolation of bomb release circuit through BOMBS safety switch on panel 2A.
Missile launch circuits					Isolation of missile launch circuit through MISSILE safety switch on panel 2A.
Underwing stores	Distress				Safe jettisoning of stores by depressing jettison button 12B (and for III D 212B).
Underfuselage stores	Distress				Jettisoning of stores by depressing jettison button 2M (and for III D 202M).

FAILURE WARNING SYSTEMS

Failure Panel Warning Lights (refer to Fig 15-1)

1504. Most warning lights are grouped on a panel (1Z) (and for III D 201Z) located on the right of the instrument panel in both cockpits. The front cockpit panel also carries the control and resetting pushbuttons for the electric generating units.

1505. Incorporated in the failure warning panels are a dimming system and a push-to-test button. The panels are supplied with power from the main busbar and include the lights shown in Table 15-3.

Table 15-3 Failure Panel Warning Lights

System Monitored (Light Identification)	Detection Unit	Index
BATT	Make and break switch	12P
HYD. 1	Pressure switch	27C
PITOT	Warning relay	8D and 10D
FUEL	Pressure switch	2E
AUTO. TRIM	Relay in autocommand amplifier	120C
EQ.T.	Sensor	15H
*GEN	Make and break switch	2P
*HYD. 2	Pressure switch	28C
*8 min. OXY	Pressure gauge	21H front cockpit 22H rear cockpit
*OIL	Pressure switch	on engine
*AC	AC warning relay	132C
*CAB.T	Duct and cabin sensors	6H and 7H
*ALT	Line contactor	6V
*EMG.HYD.	Pressure switch	29C
*OXY.REG	Pressure switch and electronic slaving	20H, 22H front cockpit 222H rear cockpit
*IFF	Relay	30S
*DAMP	Pitch and yaw warning relay	80C and 81C
*CAB.P	Cabin pressure capsule	23H and 40H

*These lights are connected to the warning horn (7Z)

1506. Each warning light illuminates when the associated detecting unit is connected to ground. In addition, the following warning lights are carried on the instrument panel :

a. A FAIL light (3Z) which illuminates when one of the warning lights on the failure panel illuminates.

b. A U/C light (27G) and for III D (227G) associated with the undercarriage.

c. A MACH light (11E) and for III D (211E) warning the pilot that there is a serious danger of exceeding the aircraft limitations.

d. For III O Only, a GYRO light (49F) which comes on in the event of gyro centre power supply failure.

1507. A DAY-NIGHT switch provides two different intensities for the warning lights on 1Z and 201Z and warning lights FAIL, U/C, MACH and GYRO. In the NIGHT position, this switch connects the various warning lights in series with dimming resistors.

Warning Horn (refer to Fig 15-1)

1508. The warning horn consists of a vibrator system including a relay and cell assembly which operates in conjunction with the centre and RH row of lights carried on the warning panels. The installation includes a warning box (7Z) on the front face of frame 10 and a switch (6Z) on the RH panel.

1509. The warning horn causes an interrupted tone to be heard in the earphones. This tone cannot be confused with any radio signal as it is much louder than any reception or noise. It is audible whatever the position of the radio mixer box selector switch. The pilot may silence the horn by pressing the FAIL light (3Z), in the front cockpit, which extinguishes. The warning horn can be silenced by pressing pushbutton 203Z in the rear cockpit. The horn and warning light are then ready to give further warning in the event of other failures. The warning horn can be tested by pressing the button provided on the warning panels.

1510. For III D, the warning horn operates if the canopy is not down and correctly locked.

EXTERNAL STORES EMERGENCY JETTISONING SYSTEM (refer to Figs 15-2 to 15-4)

General

1511. There are three jettisoning systems, all activated by an electrical circuit controlled by a pushbutton in each cockpit. The pushbuttons are fitted with a guard cover through which a safety pin, on the safety pin cord, is fitted when the aircraft is on the ground.

1512. A jettisoning pushbutton panel is fitted to the LH side of the instrument panel in both cockpits. The jettisoning circuits comprise :

a. A jettison control panel fitted with three pushbuttons controlling the circuits.

b. The various electrical circuits.

c. The firing systems.

NOTE

The bomb racks under the fuselage and the tank pylons under the wings cannot be jettisoned.

Operation

1513. **Wing Stores Jettisoning (refer to Figs 15-2 and 15-4).** The electrical circuit is closed by pressing the wing pushbutton (12B) front cockpit or (212B) rear cockpit. This causes the following simultaneous actions :

a. The external tanks are jettisoned by firing of the corresponding squibs in the pylons.

b. The wing mounted missiles are fired in a safe condition.

1514. **External Stores Jettisoning (refer to Figs 15-2 to 15-4).** The electrical circuit is closed by pressing the TANK pushbutton (44Q) front cockpit or (244Q) rear cockpit. This causes the following simultaneous actions :

a. The external wing tanks are jettisoned.

b. The external fuselage tank is jettisoned by firing the corresponding squibs.

1515. **Fuselage Stores Jettisoning (refer to Fig 15-4).** The electrical circuit is closed by pressing the FUS pushbutton (2M) front cockpit or (202M) rear cockpit. The effect is as follows according to the stores carried :

a. Bomb release (inert) by firing of the squibs in the corresponding pylons.

b. III O Only, rocket motor pack jettisoning through firing of squibs (8M) and (9M).

c. Jettisoning of the missile and its pylon by firing of the corresponding squib.

d. Jettisoning of the underfuselage external tank by firing of the corresponding squib in the CRP pylon.

Rocket Motor Pack Jettisoning — III O Only (refer to Figs 15-3 and 15-5)

1516. **Description.** The jettisonable rocket motor pack is anchored at eight points in the box structure at the bottom of the fuselage. The jettison system is of the electro-mechanically controlled pyrotechnical type and consists of :

a. A rocket motor drive shaft retraction mechanism (1) comprising :

(1) Explosive actuator (2).

(2) Two bellcranks (3 and 4).

(3) Link rod (5).

(4) Bearing housing (6).

(5) Microswitch (7M).

b. A rocket motor equipment pack comprising :

(1) Two squibs (8M and 9M).

(2) Eight ejectors (7) fitted with an expendable piece (8) and a piston (9) with a ball type locking system (10).

(3) A line connecting the squibs to the ejectors.

1517. **Operation.** Jettisoning is initiated by pressing button (2M) front cockpit or (202M) rear cockpit marked FUS; the circuit closes and current is suppled to :

a. The shaft retracted microswitch (7M) whose contacts are thus armed.

b. The squib (5M) of explosive actuator (2).

1518. When the squib is fired, actuator (2) causes rotation of the bell-crank (3) which frees link rod (5). The action of springs (11) causes bell-crank (4) to exert a pull on the bearing housing (6) of shaft (1) which, as a result, disengages from the rocket motor. The retracted shaft microswitch (7M) closes at the end of travel point and fires the squibs (8M) and (9M).

1519. When the squibs are fired, the combustion gases are delivered to the ejectors (7). The gas pressure forces piston (9) downwards to release the locking balls (10) and the expendable piece (8) is ejected with the rocket motor pack.

CANOPY JETTISONING — III O ONLY

Description (refer to Figs 15-6 to 15-8)

1520. The ejection system is essentially a pyrotechnic device controlled as follows :

a. Normally, by pulling the face blind (1) provided on the seat or the alternate firing handle.

b. In an emergency, through lever (2) on the LH side of the cockpit or, on the ground, through handles (3) in the compartment located aft of frame 10 on both sides.

1521. The installation consists of :

a. A pyrotechnical system comprising :

(1) A canopy jettison breech (4) fitted with a firing cam (5), on the forward face of frame 10.

(2) A double barrel actuator (6) in the upper box structure of frame 10.

(3) Two double barrel actuators (7) connected by a bell-crank (8) and a link rod (9) to the canopy locking hooks (10).

(4) Actuator (32) filled with nitrogen. The pressure is maintained in one of the chambers of the actuator by a capsule (33).

(5) Rigid pipes (11) interconnecting the four actuators.

b. A control system comprising :

(1) A bell-crank and link rod assembly (12) on the forward face of frame 10, which takes cable (13) connected to the seat ejection control, cable (14) connected to lever (2) and cable (15) attached to cam (5)

of the explosive actuator (4).

(2) A bell-crank and torque shaft assembly (16) on the rear face of frame 10, which takes cables (17) connected to the RH and LH handles (3) (external controls).

(3) A connecting rod (18) featuring a removable head, which is locked by a shouldered pin (19) and a series of balls (20).

(4) An unlocking sector (21) with a spring (22).

(5) A bell-crank (23).

Operation (refer to Figs 15-7 and 15-8)

1522. **Canopy Down and Locked.** The canopy is locked by the hooks (10). Jettisoning is initiated by pulling the seat face blind (1), one of the handles (3) or by pushing on lever (2).

1523. **Ejection.** Cam (5) disengages, thereby releasing piston (24) which, under the action of spring (25), detonates cartridge (26). The gases, due to the explosion, escape through lines (1) leading to the actuators (6,7 and 32). Piston (27) of actuator (7) operates bell-crank (8) which, through link rod (9), actuates hook (10). In the second part of its stroke, piston (27) uncovers the supply port to piston (28) which then forces the canopy away from the fuselage.

1524. The canopy tilts under the action of actuator (32). Actuator piston (34) is filled with nitrogen and the pressure is maintained by a capsule (33). Gas pressure causes needle (35) to pierce capsule (33) and the pressure thus released actuates piston (34). Piston (29) of actuator (6) rotates the bell-crank (23) to release sector (21) which is then forced against its stop by the action of spring (22).

1525. As rod (18) moves upward, the sector (21) acts on the shoulder pin (19) to cause the withdrawal of balls (20) and, as a result, unlocking of the removable head of rod (18). The canopy tilting motion rotates bell-crank (3). When this bell-crank reaches its end of travel point the canopy disengages from its rollers (31), pivots on the end of bell-crank (30) and ejects from the fuselage. The handles (3) (LH and RH) are recessed into the fuselage and concealed by red plexiglass covers which must be smashed in to gain access to the handles.

Canopy Jettison Breech Unit (refer to Fig 15-13)

1526. The canopy jettison breech unit is a pyrotechnic device with a mechanical firing system that is attached to the front face of frame 10 by four screws. The breech unit is connected to various actuators by pipes and consists of :

a. An elbow connection (1) with a lock-nut (2).

b. A cylindrical body (3) with integral mounting lugs.

c. A firing system comprising :

(1) A breech (4).

(2) A firing pin (5) fitted with a roller (6).

(3) A spring (7).

(4) A threaded spring retainer (8).

(5) A firing cam (9).

(6) A cartridge (10).

1527. **Operation.** When cam (9) is pulled downward by the control cable, it forces the firing pin (5) to the rear against the pressure of spring (7). When the cam releases, the firing pin (5) is freed and strikes the detonator of cartridge (10) which then explodes. The gases produced are ducted forward through the elbow connection (1).

Canopy Jettison Pneumatic Jack (refer to Fig 15-9)

1528. The purpose of this jack, which is charged with nitrogen to a pressure of 2600 lb/in^2 (180 bars), is to open the canopy up to the point where it is disengaged from its articulation system and can be pulled away by the slipstream. Operation of the canopy jettison pneumatic jack is initiated by the pyrotechnic jettison system.

1529. The jack consists of the following components :

a. A hollow piston (1) which forms the nitrogen chamber and is provided with a screwed sealed base (2) at its lower part. This base incorporates a brazed brass disc (3).

b. A charging connection (4), a stop cock (5) and a ball (6) at the upper part of the assembly.

c. A cylinder (7) housing the piston (1); at the bottom of the cylinder is a firing piston (8) retained by a spring (9) and a coupling (10) for connection to the pyrotechnic system.

1530. **Operation.** The gases resulting from explosion of the cartridge are directed to the firing piston (8). The spring is compressed and the disc (3) is pierced. The released nitrogen fills the cylinder and causes extension of the jack. A vent orifice (15) in the cylinder wall provides for free extension of the jack.

1531. **Installation.** The assembly is attached by a nut (11) and a check nut (12) screwed on the threaded portion of the cylinder either side of a support fitting (13) integral with the aircraft structure and is centred by means of a guide washer (14).

CANOPY JETTISONING — III D ONLY

Description (refer to Figs 15-10 to 15-12)

1532. The ejection system is essentially a pyrotechnic device controlled as follows :

a. Normally, by pulling the face blind (1) of the front cockpit or rear cockpit ejection seat.

This control also causes ejection of the respective seat.

b. In an emergency the canopy can be jettisoned either by :

(1) By pulling one of the two internal jettisoning levers (2) on the LH side of each cockpit.

(2) By pulling one of the two external red handles (3) located in the pocket-recesses to the rear of the slanting frame.

1533. The installation consists of :

a. A pyrotechnic system comprising :

(1) A gun (4) fitted with a firing cam (5) in the front cockpit.

(2) A gun (6) fitted with a firing cam (5) in the rear cockpit.

(3) A valve box (8).

(4) A release microswitch box (51M).

(5) Two initiators (52M and 54M) on the LH side, and two initiators (53M and 55M) on the RH side, on the canopy hinge arm (10).

(6) A battery (50M) in the front cockpit.

(7) A resistor (57M).

(8) Two dual actuators (11) connected to the front hooks (12) by a bell-crank (13) and linkrod (14) assembly. This assembly is connected by a rod (15) to bell-crank (16) and linkrod (17) assembly which attaches to the rear hooks (18).

(9) Rigid pipes (19) interconnecting the various system components.

b. A control system comprising a bell-crank and linkrod assembly (20) (in the front cockpit) which receives :

(1) The cable (21) connected to the ejection seat (front cockpit).

(2) The fork (22) attached to the cam (5) of the gun (4) (front cockpit).

(3) The cable (23) connected to the lever (2) (front cockpit).

(4) The cable (24) connected to the lever (2) (rear cockpit).

(5) Two bell-cranks (25) which receive the cable (26) connected to the ejection seat curtain (rear cockpit), the fork (27) attached to the cam (5) of the gun (6) (rear cockpit) and cables (28) connected to the LH and RH handles (3) (emergency external control).

c. A Canopy Hinge Assembly Comprising :

(1) The canopy hinge arms (10) with a removable end-fitting (29) locked by a shouldered pin (37) and balls (38).

(2) Two compensating cylinders (32). The upper part of these cylinders is connected to the structure through clamps and elastic cords to prevent them from tilting toward the ejection seat in case of canopy jettisoning.

(3) Two adjustable stops (33).

(4) Two rollers (34) secured to the aircraft structure.

Operation — (refer to Figs 15-11 and 15-12)

1534. **Canopy Locked In The Closed Position.** Jettisoning is initiated by activating the ejection seat controls, or by pulling either one of the handles (3) or one of the levers (2).

1535. **Jettisoning By Using The Front Cockpit Face Blind Or One Of The Internal Levers.** Pulling the front cockpit face blind or one of the internal levers causes the front cockpit jettison breech to operate. The cam (5) disengages and releases the piston (35) which, under the action of its spring, detonates the cartridge (36). The gases caused by the explosion escape through the rigid pipes (19) and are applied to the dual actuators (11) and the release microswitch box (51M) through the valve box (8).

1536. The following actions occur simultaneously :

a. The initiators (52M and 53M) are fired by the cell (50M) circuit or the initiators (54M and 55M) are fired by the aircraft battery circuit through the resistor (57M). Under the pressure of the gases released by the initiators, the shouldered pin (37) moves back and releases the balls (38), thereby unlocking the removable end-fitting (29) of the hinge arms (10).

b. The piston (39) of the actuator (11) operates the bell-crank (13) which, through the connecting rod (15), unlocks the hooks (12 and 18). In the second part of its stroke, the piston (39) uncovers the supply orifice to the piston (40) which then forces the canopy away from the fuselage. Slipstream is finally used to take the canopy away from the aircraft.

1537. **Jettisoning By Using The Rear Cockpit Face Blind Or One Of The External Handles.** Pulling the rear cockpit face blind or one of the external handles causes the rear cockpit jettison breech to operate. The jettison breeches (4 and 6) operate in the same manner and perform the same functions as described in paras 1535 and 1536.

NOTE

The emergency external handles (3) only unlock the canopy; they do not eject it. After pulling one of these handles, the canopy is to be opened by hand. This action is made easier by the compensating cylinders (32).

1538. **Canopy Ejection.** At the end of its rotation motion, the canopy disengages from the rollers (34). The canopy is then supported by the adjustable stop (33) which pivots and lifts the canopy which separates from the aircraft

NOTE

The external handles (3), recessed into the fuselage, are concealed by red plexiglass covers which must be smashed to gain access to the handles.

Canopy Jettison Breech Unit (refer to Fig 15-13)

1539. For III D, two canopy jettison breech units are fitted; one to the front face of frame 10, and one to the front face of the sloping frame. The units function as described in paras 1527 and 1528.

GROUND SAFETY FEATURES

Description

1540. When the aircraft is on the ground, safety pins are fitted to certain equipment controls to prevent inadvertent operation. The safety pins are attached to nylon cords, two for III O and four for III D. When the pins are not in use, they are stored in the cockpit

Safety Pins In The Front Cockpit (refer to Figs 15-14 and 15-15)

1541. The following safety pins are fitted :

a. Airbrake control (1).

b. Parachute control (2).

c. Undercarriage control (3).

d. External stores jettison control (4).

e. Stand-by gyro erect knob (5).

f. Gyro centring switch (6) — III O Only.

g. Canopy locking control (7) — III O Only.

Safety Pins In The Rear Cockpit (refer to Fig 15-15)

1542. The following safety pins are fitted :

a. Airbrake control (8).

b. Undercarriage control (9).

c. Stand-by gyro erect knob (10).

d. External stores jettison control (11).

Ejection Seat Safety Pins (refer to Fig 15-15)

1543. The following safety pins are fitted to the front and rear seats :

a. Seat pan firing handle (12).

b. Manual override handle (13).

c. Drogue gun (14).

d. Remote rocket initiator (15).

e. Ejection gun time delayed firing mechanism (16).

f. Face screen firing handle (17).

g. Canopy gun (18).

Safety Pins — Stowage

1544. The seat pan handle, face screen handle and manual override handle safety pins are stowed in special pin brackets in the front and rear cockpits. The front cockpit bracket is located adjacent to the EMG COLD switch on the RH cockpit wall and the rear cockpit bracket is located below the emergency brake handle on the front of the RH console. The remaining seat pins are stowed in the pouch on the LH side of the ejection seat. The cockpit safety pins are stowed in the case provided for this purpose.

PYLON TANKS — FUEL DUMP VALVE

Description and Operation (refer to Fig 15-16)

1545. The fuel dump valve permits rapid dumping of the fuel contained in the pylon tanks. The installation comprises :

a. A circuit-breaker (45Q).

b. A relay (46Q).

c. A pushbutton (47Q) on the LH panel (front cockpit only).

d. A fuel dump system in the pylon tanks.

e. An electrical circuit.

1546. Pressing pushbutton (47Q) closes relay (46Q) which causes :

a. Energization of the circuit through the connectors (87Y and 88Y) of the wing pylon tanks and through the pylon of the fuselage pylon tank.

b. Instant opening of the fuel dump system.

CRP 18 PYLON EJECTOR

Description (refer to Fig 15-17)

1547. The CRP 18 pylon ejection is a pyrotechnically operated mechanical device that permits anchoring and ejection of the pylon tanks. The unit consists of :

a. A cylinder (1) machined with a port for insertion of the ALKAN tool (2); this port is normally closed by a plug.

b. Two squibs (3).

c. A nut for attachment to the pylon (4).

d. Four locking petals (5).

e. A shouldered taper plunger (6).

f. A spring (7) and a push-rod (8).

g. An ALKAN tulip (9) (on the tank).

h. A position feeler (10).

i. A stepped screw (11).

Operation

1548. **Anchoring.** Lower the plunger (6) using ALKAN tool (2) until the locking petals (5) retract. Fit the ALKAN tulip (9) in position. The plunger will then move up under the pressure applied through push-rod (8) and spring (7). The locking petals (5) are pushed outwards and engage the recess in tulip (9) which locks the system. Check the position of the plunger by means of feeler (10).

1549. **Ejection.** Firing of the squibs (3) causes :

a. Downward travel of plunger (6) due to the pressure of the gases.

b. Retraction of locking petals (5).

c. Ejection of the tank by the action of push-rod (8) and plunger (6) which is ejected with the store.

CHAPTER 16

OPERATIONAL EQUIPMENT

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OPERATIONAL EQUIPMENT

Table 16-1 Function of Components

Index	Description	Characteristics and Functions
	AIR DATA SYSTEM	
1C	Air data computer	Using total and static pressures, and impact temperature, this unit computes Ps, qc, P, Z, M and TAS which are fed to the aircraft systems.
2C	Air data computer output multiplier	Parameters as computed by 1C are supplied to all the equipment concerned. Purpose of the multiplier is to have these parameters produced in sufficient number, multiplied and distributed as pulsed inputs to the equipment.
	GYRO CENTRE — HEADING	
8F	Magnetic detector (flux valve)	Supplies magnetic heading to the output multiplier (27F)
24F	Control panel	It includes the following : a.A heading selector. b.A warning light and a switch used when a transfer gyro is available. c.A fast erection switch that provides fast slaving of the gyro reference unit (GRU) to local vertical.
25F	Electronic unit	Contains all the amplifying, supply, correction, control and monitoring systems of the GRU.
26F	Gyro reference unit (GRU)	Detects pitch, roll or yaw motions and transmits the resultant outputs via output multiplier (25F) to the various equipment concerned.
27F	Output multiplier	Feeds gyro-magnetic heading information Reproduces and feeds roll, pitch and heading information Provides power necessary for magnetic heading reproduction if the alternator fails (emergency power system).
48F	Gyro centre switch	Operates the gyro control loop.
	SIGHTING SYSTEM	
25A	Sight relay box	This box is electrically connected to the gyro centre, air data computer output multiplier and range setting unit to feed the sight head with control signals from these units.
26A	Voltage regulator	Supplied with 28 V d.c. power from the aircraft electrical system. This regulator supplies the different regulated voltages necessary for sighting system operation; its output voltage is adjustable from 21.5 to 22.5 V.
27A (Front cockpit only)	Gyro caging button	Provided that radar lock-on is not achieved, while operating in the air-to-air gun firing mode, it controls the setting of fixed corrections (target and gravity drop) in the sight for 820 ft (250 m) range. When radar lock-on is achieved and the range is less than 6560 ft (2 km) the sight sensitivity is reduced.
29A	Sight head	Used in conjunction with the radar, the air computer and the gyro centre. Operates as a flight indicator for navigation, as a command synthetizer for air-to-air missile tracking and firing, and also as a conventional gyro gun sight for gun firing.
4B	Armament selector switch	Initiates and controls the sighting system operation.
	Gunsight dimmer rheostat	Controls the brightness of reticules and coloured lights.
14B	Heading error switch	Cuts off heading error displayed on the sight at the discretion of the pilot.
	SIGHT CAMERA	
14A	Camera pushbutton	When the firing mode has been selected by the pilot on the armament control panel (4B), this pushbutton closes the power supply circuit of the camera (15A) when the firing trigger is depressed up to its first detent position (hard point).

Index	Description	Characteristics and Functions
15A	Camera	This is an electrically operated and automatically adjusted camera. Frame size : 16 mm — Fixed framing rate of 16 frames/second. Variable aperture setting controlled by a photo-cell. Variable exposure from 1/100 to 1/1200 second according to ambient luminosity. The camera is controlled from a panel (17A).
16A	Photo-cell	This cell is adjustable on the ground according to the film emulsion speed by means of a knob which is not accessible in flight. When the camera is not energized, the cell is protected by a hood.
17A	Control panel	The camera is energized through the control panel which includes : a. A camera On-Off power relay. b. Two exposure setting relays controlled by the photo-cell. c. A three-position delayed stop setting control knob : 0 — 2.5 — 5 seconds.
RADAR INSTALLATION — III O ONLY		
13S 14S	H. programming unit G. programming unit	Programs the different scanning modes and feeds them to the antenna servo-mechanisms.
20S	Radar nose cone	Pressurized and cooled compartment housing the complete radar assembly (antenna, transmitter-receiver ranging circuits, computers, servo mechanisms) and also accommodating the pitot head.
20S	Photo nose cone	Compartment designed to accommodate the photographic system equipment and panoramic camera, the air conditioning system, the electrical circuit and the pitot head.
26S	Accelerometer	Measures and indicates the load factor necessary for navigation command computation.
28S	Radar indicator	Converts and displays radar information mainly during the target search phase.
28S	Camera control mount	Receives the control panel (72M).
29S	Radar control stick	Contains all provisions for antenna scanning control.
TAIL WARNING RADAR (OPTIONAL FIT)		
21S	Antenna unit	Contains : a. A motor driving the antenna and coupled to an alternator. b. A 20 Hz rotating antenna that scans the radar zone of effectiveness. c. An alternator supplies 90° out-of-phase voltages to operate the receiver.
22S	Pre-amplifier	Amplifies UHF pulses detected by the rotating antenna and feeds them into the receiver video-amplifier.
35S	Relay circuit breaker	Energizes through switch (37S) the warning radar relay (38S).
36S	Motor circuit breakers	One on each phase of the inverter circuit, feeds the antenna motor through the warning radar relay (38S).
37S	Switch	Energizes the tail warning radar relay.
38S	Tail warning radar	Controlled by (37S), it applies the aircraft power voltage to receiver (40S) and energizes the antenna motor.
39S	Warning panel	Comprises three warning lights, each corresponding to a detection sector. They light to show the direction of the detected radar signal; a push-to-test button is provided to check proper operation.
40S	Receiver	Receives radar pulses, converts them into signals and directs them to the warning panel (39S) and the audio warning circuit (radar audio monitoring).
NAVIGATION SYSTEM		
37F	PHI computer	Consists of electronic and mechanical sub-assemblies which perform all computations and switchings necessary for navigation purposes and memory units used for navigation resetting.
38F (front cockpit) 238F (rear cockpit)	PHI indicator	Provides visual indication of aircraft heading and heading and distance to a selected station. A function selector switch is provided to select the mode of operation. It allows for manual corrections of the displayed heading and distance information.

Index	Description	Characteristics and Functions
39F	Control panel	Comprises the wind setting and station grivation controls together with the housing for the station storage unit.
	Station storage unit	Determines the Cartesian co-ordinates of stations to be used for navigation.
	TACAN SYSTEM	
37R	Transmitter-receiver	By means of interrogation and received pulses, it measures, computes and supplies to the pilot the magnetic bearing and distance information concerning a selected ground beacon.
38R	Control panel	Includes a five-position selector switch, a volume control knob, a channel setting system and two edge-lighting lamps.
39R 40R	Upper antenna Lower antenna	Identical antennae operating singly according to the position of switch (41R).
41R	Antenna switch	Switches in the antenna providing the best signal.
42R	Antenna selector	Monitors and selects the antenna providing the best signal and controls switch (41R).
	DOPPLER SYSTEM — III O ONLY	
53F	Transmitter-receiver	Supplies the pilot and various systems with drift and ground speed information.
55F	Control	Includes : a.A drift and ground speed indicator. b.A five position selector switch. c.A memory indicator. d.Two setting knobs.
	RADAR-ALTIMETER SYSTEM — III O ONLY	
49R	Transmitter-receiver	Transmits a frequency-modulated signal and measures the altitude with the reflected signal.
46R	Indicator	Supplies the pilot with accurate altitude information
	ARMAMENT/RADAR CONTROL PANEL	
4B	Armament selector	Comprises : a.A rotary switch which selects the weapon system circuits associated with the specific weapon firing modes. b.Controls for radar (III O Only) missiles and bombs.
72 M (Optional fit — III O Only)	Camera control panel	Comprises : a.Manual setting switches. b.A frame counter. c.A cloud coverage ratio chart. d.A power switch. e.A camera control switch.
75 M (Optional fit — III O Only)	Camera	Horizon-to-horizon (180°) panoramic camera. An independent optical system takes pictures of the counters displaying the aircraft co-ordinates and heading while the camera is filming. Camera speed : 1 to 6 frames per second. Shutter speed : 1/500 to 1/5000 second.

GENERAL

1601. The operational equipment is designed to facilitate operational missions of the following types :

a. Interception with or without rocket motor.

b. Air interdiction.

c. Ground attack and tactical support.

1602. The operational equipment includes :

a. The air data computer and gyro centre which are essential to aircraft control and have a primary role in the weapon system operation.

b. The sighting system which operates as a flight indicator for navigation and serves as a command synthetizer for air-to-air missile firing. The sighting system operates as a conventional gyro gun sight for air-to-air firing and provides a fixed sighting reference for air-to-ground gun firing, rocket firing and dive bombing.

c. The radar system which is fundamental for target acquisition, tracking and firing control.

d. The tail warning radar (optional fit) which is designed to warn the pilot that the aircraft has been continuously illuminated by radar for more than two seconds.

e. The navigation computer which provides the pilot with independent dead reckoning. It is also used as a TACAN indicator.

f. The armament/radar control panel which selects the weapon to be used and operates the complete weapon system depending on the selected weapon.

g. Doppler system — III O Only.

h. Radio-altimeter system — III O Only.

AIR DATA SYSTEM AND ATTITUDE AND HEADING SYSTEM

1603. The air data computer (1C), output multiplier (2C) and attitude and heading references system are described in Chap 12. A full coverage of the units is given in AAP7213.001-2-4 in the following sections.

a. Air data computer and output multiplier in section 2.9.

b. Gyro centre in section 2.8.

SIGHTING SYSTEM

Sight Head

1604. **Description (refer to Fig 16-1).** The sight head operates as :

a. A flight indicator for navigation; reproducing the attitude information in roll and pitch and the heading.

b. A radar command signal synthetizer for air-to-air missile tracking and firing.

c. A conventional gyro gun sight for air-to-air gun firing.

d. A fixed sight set to the weapon selected : air-to-ground gun firing, rocket firing and dive bombing.

1605. The sight head incorporates :

a. A semi-reflecting mirror.

b. Two sets of indicator lights; the RH indicators relay information from the radar.

c. A centre support to accommodate the sight recording camera (15A).

d. Controls for :

(1) Operating the heading/range scale shutter (4) at the RH side lower part.

(2) Selecting the RH or LH indicator lights (5) at the LH side lower part.

(3) Manual gravity drop control rheostat (6) at the LH side.

1606. **Installation.** The sight head (29A) is located at the upper part of the instrument panel and secured to the cockpit structure by means of a milled light alloy bracket (1). The bracket is attached to the structure by two screws (2).

1607. The sight head is attached to the support; its lower part by a flanged bolt (3) inserted in a threaded hole in the sight head. This system permits azimuth adjustment. The initial setting is 5° nose down. Two sockets (7) at rear of the sight provide for electrical connection.

1608. A full description of the optical sight is contained in AAP 7213.001-2-4, section 2.4.

Voltage Regulator

1609. The voltage regulator compensates for any change in the electrical system power voltage and maintains a constant output into the sighting system. The voltage regulator (26A) is located in the cockpit at the rear of the seat on LH side of frame 10 for III O and on the forward face RH side of the slanting frame for III D.

Junction Box

1610. The junction box (25A) receives commands and information for the sight and converts this data into currents capable of being accepted by the gyro elevation, gravity drop heading and sensitivity circuits. These signals are also fed to the course and heading distance servo-mechanisms. The junction box is secured to frame 10 by means of a bracket which is attached to the structure by four bolts. Attachment to the bracket is made by two captive nuts.

Miscellaneous Controls

1611. The following miscellaneous controls are provided :

a. The armament/radar control panel.

b. The throttle (at the fore and centre parts).

c. The instrument panel (HDG ERR switch).

Sight Camera

1612. **Description (refer to Fig 16-3)** In case of gun or missile firing, depressing the control stick trigger also operates the sight camera.

1613. The purpose of the sight camera is to film the sighting and firing run made by the pilot (front cockpit only). The sight camera installation consists of the following units :

a. A camera (15A) engaged on the sight head (29A).

b. A photo-cell (16A) mounted in front of the sight head.

c. A control panel (17A) bolted to the slanting frame.

d. A camera pushbutton (14A) located on the front cockpit control stick hand grip.

1614. **Operation.** The sight camera installation is energized as soon as the selector on the armament control panel (4B) is placed in any position other than OFF. The film drive motor is started as soon as the firing trigger is squeezed. The protective hood of the photo-cell (6A) retracts automatically.

1615. Exposure is automatically set; in case of failure of the automatic system, it can be manually adjusted by the pilot by means of a knob located on the camera; exposure can be adjusted from 1/100 to 1/1200 second.

1616. The photo-cell can be adjusted on the ground according to the emulsion speed of the film used. It adjusts the aperture of the camera shutter according to ambient luminosity through the control panel (17A).

NOTE

The photo-cell (16A) is energized through the camera (15A). Consequently, when the camera is not installed, the circuit is interrupted and the hood covers the photo-cell, which prevents deterioration of the latter as the result of continuous operation.

1617. **Marking.** A triangular index is marked on the film by the armament box relays (3B and 68A) when the firing trigger is depressed.

1618. **Overrun System.** Stopping of the camera can be delayed (recording of firing results) through selection of an overrun time (0 — 2.5 — 5 sec) by means of a knob located on the control panel (17A).

1619. **Installation.** The camera is locked on a quick-release mount on the rear face of the sight head (29A). Cameras are interchangeable. The photo-cell (16A) is held in position by two bolted slide tracks on the front face of the sight head (on the instrument panel glareshield).

RADAR SYSTEM — III O ONLY

Description (refer to Figs 16-4 to 16-6)

1620. The CYRANO II radar displays on the scope (28S) and on the sight head (29A), the information needed by the pilot to select different attack procedures :

a. Hostile search.

b. Tracking.

c. Entry into firing zone.

d. Break-away.

1621. The radar system receives information from the air data computer and the gyro centre (aircraft parameter). Electrical power is derived from the a.c. and d.c. aircraft power systems.

1622. The radar system consists of :

a. A nose cone (20S), which is pressurized and cooled, housing the following :

(1) Mobile antenna unit.

(2) Transmitter-receiver unit.

(3) Antenna servo-mechanisms.

(4) Range finding unit.

(5) Firing computer.

(6) Navigation computer.

(7) Pressurization and cooling system.

b. Radar indicator (28S) containing :

(1) A scope on which radar information and echoes are displayed. The scope brightness may be adjusted using two superimposed polaroid screens — one of which can be swivelled using an index. Opposite these polaroid screens, a removable vizor facilitates the use of the scope regardless of the external light brightness.

(2) Lighting and brightness controls.

(3) Indicator lights indicating the selected mode of operation of the radar.

(4) Warning lights indicating the condition of some radar equipment.

(5) The FREEZE and AUTO RESET buttons and the 15 NM toggle switch.

c. Radar control stick (29S) : The radar control stick contains all necessary provisions to control the antenna scanning pattern as follows :

(1) Antenna elevation control (23 and 24).

(2) Azimuth control.

(3) Range finding control.

(4) The antenna lock-on lever (25).

(5) The antenna unlocking lever (26).

(6) The gain control (27) and the maximum gain pushbutton (28).

(7) The antenna scanning selector (29).

(8) The operation mode selector switch air-to-ground (30).

d. Two programming units which are located in the cockpit.

e. An accelerometer which is located in the lower fuselage bay in front of frame 23.

f. Miscellaneous controls which are located on :

(1) Armament/radar control panel.

(2) Sight head.

(3) PHI indicator.

1623. A full description of the radar is contained in AAP7213.001-2-4, section 2.3.

Installation

1624. **Radar Nose Cone (refer to Fig 16-4).** The radar nose cone is interchangeable with a ballasted dummy nose cone. The forward section of the radar nose cone is made of laminated fibre glass and supports the pitot head. The radar nose cone is attached by :

a. Two locating pins (12) that facilitate positioning of the radar nose cone on the airframe structure.

b. Four captive axle-pins (3) securing the nose cone to fittings (4) integral with the aircraft structure. The fittings are bolted to webs between frames 1 and 2. Access to the axle-pins can be gained through access doors (5). The axle-pins are secured by angle plates (6) bolted to the structure.

1625. The radar nose contains :

a. At the lower section of the rear flange :

(1) Six electrical connectors (7) (including one test connector).

(2) One pressure control valve inlet (8).

(3) One self-sealing valve (9) (pressurization system).

(4) One bonding connection (10).

(5) One coaxial connector (11).

(6) One inflation valve (12) for the radar nose seal.

(7) Two self-sealing valves (13) (cooling system).

b. Four pitot/static inlets (14) at the upper part.

c. A pressure relief valve (15) at the LH side.

1626. The radar pressurization system which originates in the cockpit conditioning system (refer to Chap 13) incorporates a pressure regulator at the rear of frame 17 and a filter. The fluid is cooled through an air-liquid heat exchanger which uses the cooling air from the equipment conditioning system. Pressurization and cooling systems are connected to the radar nose cone by means of flexible hoses and self sealing valves. Electrical connection is provided by mobile connectors. Access can be gained through an access door situated at the lower part of the fuselage.

1627. **Radar Indicator (refer to Fig 16-5).** The radar indicator (28S) is housed in a metal unit (16) which is located along the cockpit centre line, between frames 2 and 5 and mounted at an angle in the centre console. The radar indicator is mounted immediately below the instrument panel.

1628. The unit (16) is attached to the instrument panels as follows :

a. By one of the screws (50) attaching the BEZU ball (28F) on the LH instrument panel.

b. By a special screw (51) on the RH instrument panel.

1629. Both screws are attached to two special supports (52 and 53) riveted on the unit. The complete assembly, unit and instrument panel are shockmounted as follows :

a. At the front, two shockmounts (18) connecting the unit to the rudder pedals support cross-bar.

b. At the rear, four shockmounts (19) connecting the instrument panel support beam to the cockpit consoles.

c. The radar indicator is held in the unit by two locating pins (20) at the bottom of the unit (these pins also position the electrical connector) and by two screws (21 and 22) at the front upper part of the radar indicator. One of these screws (21) is used as an extractor.

d. The electrical connection is provided by a plug mounted at the bottom of the units.

1630. **Radar Control Stick (refer to Fig 16-6).** The radar control stick is secured by four screws (31) to a support (32) inside the LH console, behind the throttle. A spring hook holds the stick at maximum left deflection when in the rest position. Electrical connection is made by a connector (33) at the lower part of the support.

1631. **Programme Units.** Located at rear of the cockpit LH and RH sides the programme units are bolted to frame 10 by four attachment lugs integral with the units. Electrical connection is provided by mobile connectors.

1632. **Accelerometer.** The accelerometer is installed vertically near the aircraft centre line by four screws to a support integral with web 22. Electrical connection is made by a mobile connector.

1633. **Scope Camera.** The scope camera is secured to the scope by the LH instrument panel mounting screw. Electrical connection is made by connector (47S).

TAIL WARNING RADAR (OPTIONAL FIT)

Description (refer to Fig 16-7)

1634. The MIRAGE III can be equipped with a tail warning radar. Space provision has been made in the airframe and aircraft wiring has been routed to allow for installation. The tail warning radar informs the pilot that the aircraft has been illuminated steadily by radar for more than two seconds. This information is displayed by warning lights and duplicated by an audio signal. The tail warning radar system includes the following equipment :

a. An antenna unit (21S) containing an antenna, a driving motor and an alternator.

b. A video preamplifier (22S) housed near the detecting unit.

c. A receiver set (40S) which converts detected pulses into signals to the warning panel.

d. A warning panel (39S) with three warning lights, each associated to a detected sector. A test button located at the centre of the light panel is provided to check proper operation of the lights.

e. Warning radar relay (38S) which is energized by the d.c. power main bus, it feeds :

(1) DC power to receiver (40S).

(2) Three-phase a.c. power to the antenna unit (21S) from the alternator bus.

f. A control switch.

Installation

1635. **Antenna Unit (refer to Fig 16-7).** The antenna unit is mounted on the upper end of the fin trailing edge below the navigation light and above the rudder. It is housed in a fairing which includes a protruding radome (this fairing also houses the aft navigation light). The unit is attached on the fin leading edge box assembly. Electrical power is supplied by a plug.

1636. **Preamplifier.** The preamplifier (22S) is screwed to a rigid support bracket which in turn is riveted between both skin structures of the fin box assembly between the upper stiffener and the main spar, above rib 19.

1637. **Receiver Unit.** The receiver unit (40S) is mounted on a rigid support (4) riveted to the access door of the aft starboard bay (5) aft of frame 33. The receiver is attached by quick release type fasteners which consist of two studs on the front and two screw clamping brackets on the rear. Electrical power is supplied by a plug.

1638. **Warning Panel.** The warning panel (39S) is installed on the RH side post of the windshield. An angle plate (9) attaches the support bracket to the warning panel.

1639. **Warning Radar Relay.** The warning radar relay (38S) is installed inside the armament control box.

1640. **Control Switch.** The control switch (37S) is mounted on the upper RH part of the RH side instrument panel.

PHI NAVIGATION UNIT

1641. For description, installation and operation of the PHI navigation unit refer to Chap 14.

TACAN SYSTEM

1642. For description, installation and operation of the TACAN system refer to Chap 14.

DOPPLER SYSTEM — III O ONLY

Description (refer to Fig 16-8)

1643. By using the Doppler effect, this system supplies the pilot and various aircraft units with drift and ground speed information. The Doppler system comprises :

a. A transmitter-receiver (53F).

b. A control panel (55F).

c. An antenna (54F) and its radome.

d. A wave-guide.

e. A supply transformer (52F).

Transmitter-Receiver

1644. The transmitter-receiver (53F) is installed on a support rack located between frames 14 and 15 at the rear part of the nose wheel well. The rack is located by two pegs (5) and secured by screws (12) to an upper plate (8). The system is provided with four shockmounts (10) installed on brackets (16) attached to the structure.

1645. The upper part includes two ventilation orifices (13), an access hole for the electrical plug board (7) and a wave-guide connection (9). The transmitter/receiver is centred by spring-loaded pegs (6) secured at the bottom of the container retained in its housing by knurled nuts (15) locked by leaf springs (14). Bonding is provided by a contact blade (17).

Control Panel

1646. The control panel (55F) is installed on the pedestal and secured by four screws. It comprises :

a. A drift and ground speed indicator.

b. A five position selector switch.

c. A MEMORY indicator.

d. Two setting knobs.

Antenna and Radome

1647. The antenna (54F) is installed on a support in the bay between frames 2 and 6 under the cockpit and secured by three screws (4). Four screws (3) and two pins (2) ensure attachment and location of the antenna.

1648. The laminated glass radome is screwed to the frame of the bay door with the radome protruding slightly from the fuselage profile. It forms a metal frame and fairing assembly. A hole to drain water condensation is drilled in the radome.

Wave Guide

1649. The wave-guide assembly connects the antenna to the transmitter-receiver and is attached by supports all along its route. The wave-guide has a sealed connector where it passes through the cockpit floor at frame 10. Connection at the transmitter-receiver end is achieved through a flexible quick-release coupling.

Supply Transformer

1650. The supply transformer (52F) is secured by screws inside the navigation rack between frames 14 and 15 at the front top LH side.

RADIO-ALTIMETER SYSTEM — III O ONLY

Description (refer to Fig 16-9)

1651. The radio altimeter system supplies accurate altitude information both on an indicator and through the illumination of coloured lights on the sight head. The system consists of :

a. Transmitter-receiver (49R).

b. Two antennae (47R and 48R).

c. An indicator (46R).

d. A switching box (50R).

e. Two connecting boxes (45R and 52R).

Installation

1652. **Transmitter-Receiver.** The transmitter-receiver is secured by four screws (1) under a cross member to angle brackets (3) integral with the structure.

1653. **Antennae.** The antennae (47R and 48R) are each secured by six screws (6) to a plate (7) under the fuselage between frames 14b and 14c on either side of the centre line.

1654. **Indicator and Boxes.** The indicator and connecting boxes are installed as follows :

a. The indicator (46R) is screwed to the RH instrument panel.

b. The connecting box (52R) is screwed to the radio equipment bay door.

c. The connecting box (45R) is attached at the front of the pedestal through an extrusion at the lower part and two screws at the upper part.

d. The switching box (50R) is attached by four screws (4) to cross member (5) which is integral with cross member (2) and secured at each end by angle brackets (3).

ARMAMENT/RADAR CONTROL PANEL — III O ONLY

Description

1655. The armament/radar control panel is located on the cockpit RH side console and secured to the console by four screws. It enables the pilot to select a weapon, thereby defining the weapon system mode of operation. The control panel has four sectors :

a. Main sector.

b. Missile sector

c. Bomb sector.

d. Radar sector.

1656. **Main Sector.** The main sector contains :

a. An eight position selector switch with the following positions:

(1) Three AIR-AIR positions : GUNS, S.W. and MISS.

(2) One G/S OFF position : no mode selected.

(3) Three AIR-GROUND positions : G.R.M. (guns rockets, missile) H.E. bombs and S.S which is not used.

(4) One AIR-GROUND navigation position used in straight and level flight. On this position the sight head conveys aircraft altitude and heading information and also heading error in flight towards a selected station.

b. One firing mode selector switch selecting either SALVO or SINGLE firing of missiles, rockets or bombs. On indroduction of Mods 1112 and 1113, the SINGLE position gives priority to the LH Matra R550 firing circuit and the SALVO position gives priority to the RH Matra R550 firing circuit.

c. One dimmer rheostat knob located in the centre of the eight position selector switch which adjusts the brightness of the sight head reticules and indicator lights.

1657. **Missile Sector.** The missile sector includes two switches. STND-BY is used for missile preparation and preheating and MISS-AUTO FIRE is used for firing of the MATRA R530 missile.

1658. **Bomb Sector.** The bomb sector includes two switches and a selector as follows :

a. NOSE switch having two positions :

(1) NOSE : activation of nose fuse arming wire circuits (live release, instantaneous explosion).

(2) SAFE : non-activation of the arming wire circuits (safe release).

b. TAIL switch having two positions :

(1) TAIL : activation of the tail fuse arming wire circuits (live release, delayed explosion).

(2) SAFE : non-activation of the arming wire circuits (safe release).

c. A selector with three positions :

(1) FUS : selection of fuselage circuit.

(2) F + W : selection of fuselage + wing circuit.

(3) WINGS : selection of wing circuit.

1659. **Radar Sector.** The radar sector includes :

a. One three-position switch operating the radar : OFF, STND-BY and TRX.

b. One EMG TRX pushbutton to reset radar transmission.

c. Three two-way switches :

(1) NORM-JAM enables the radar and missile homing device to lock onto jamming from a target.

(2) NORM-RANGE. The radar indicates along the aircraft centre line the distance to ground in AIR-GROUND mode or distance to locked-on target in AIR-AIR mode.

(3) GUARD PLANE switch which has two position; SINGLE for single plane contour mapping and DUAL for dual plane contour mapping.

ARMAMENT/RADAR CONTROL PANEL — III D ONLY

Description

1660. The armament/radar control panel fitted to III D is less complex than that fitted to III O. The panel has three sectors :

a. Main sector. The main sector is similar to that described for III O, refer to para 1655, except that the MISS position for AIR-AIR is not fitted.

b. Missile sector. The missile sector has a single switch that is used for missile preparation and preheating.

c. Bombs sector. The bombs sector has a switch with three positions :

(1) SAFE : non-activation of the arming wire circuits (safe release).

(2) DELAY : activation of the tail fuse arming wire circuits (live release, delayed explosion).

(3) INST : activation of the nose and tail fuse arming wire circuits (live release, instantaneous explosion).

PHOTOGRAPHIC SYSTEM — III O ONLY (OPTIONAL FIT)

Description (refer to Fig 16-10)

1661. The photo nose cone can be installed in place of the radar nose cone on the MIRAGE III O aircraft. The nose cone is fitted with a panoramic camera permitting vertical shooting with a 180° field of view (complete horizon). A pitot-static pressure head extends the cone. The junction, between frame 9 of the cone and frame 1 of the fuselage, is sealed by a rubber extrusion integral with aircraft frame 1. The photo nose cone uses the same attachment points as the radar nose cone.

1662. Lead ballast plates are used to compensate for the difference in weight between the photo and radar nose cones. A wide opening provided at the lower section of the cone accommodates a door (19) for access to the camera. The external surface of the cone is painted matt black. The nose cone includes the following equipment :

a. A camera system.

b. An air conditioning system.

c. An electrical system.

1663. Leading particulars;

(1) Overall length including pitot probe : 3.34 m.

(2) Overall height : 0.765 m.

(3) Overall width : 0.748 m.

(4) Weight : 230 kg (507 lb) with cg location 670 mm in front of the rear of frame 9.

(5) Weight without camera and mount: 191 kg (421 lb) with cg location 615 mm in front of the rear of frame 7.

Nose Cone Structure (refer to Figs 16-10 and 16-11)

1664. The structure is of the stressed skin type and includes two sections :

a. Body (1), from nose cone frame 1 to nose cone frame 9.

b. Nose cone (2), from pitot-static pressure head to nose cone frame 1.

1665. **Body.** The body consists of stringers, webs and light alloy skin panels. The wide opening of the door has required reinforcement of the structure for stress distribution in this area. Forward, a steel plate is mounted on the rear surface of frame 5 and covers its lower quarter section. Aft, a half frame 7 located 20 mm in front of frame 8 reinforces the box-type structure formed by frames 8 and 9. Stringers (51), (54) and (57) are reinforced to form a box-structure strengthening the door frame. The door frame has a doubler from frames 4 to 9, on the outside, and from frames 5 to 9 on the inside. The lower box-structure between RH and LH stringers (54) and between frames 7 and 9 is reinforced by an internal steel sheet. The skin is made of two half shells riveted to light alloy frames and screwed to steel frames. Two hoisting fittings (3) are provided at the upper section, between stringer (51) and stringer (52a), forward of frame 8.

1666. **Nose Cone.** The nose cone (2) is a sheet metal element stiffened at its tapered section by two frames. The cone is attached by screws on its periphery to frame 1 of the body. It is terminated by a tapered section (5) accommodating the pitot-static pressure head (6).

1667. **Installation of Pitot-Static Pressure Head.** The pitot-static pressure head (6) crosses a threaded bush (7) mounted from the inside, a tightening cone (8) mounted from the outside and a nut (9) to which is installed a check nut (10) fitted with a seal (11). The shouldered rear section of the head is provided with a castellated nut (12) fitted with a tightening cone (8) screwed in a flange (13) in the centre of frame 2.

1668. **Installation of Pitot-Static Pressure Probe.** The probe is mounted from the outside in the end section of the head and is attached by six screws to the head tube. Sealing is ensured by an O-ring housed in a circular groove in the probe.

1669. **Door Structure.** The door (19) incorporates a centre well for access to the camera (75M). The lower section of the well is closed by a V-shaped window (20).

1670. The door is a box structure made of frames and internal and external skin panels. The front box includes three frames provided with flanged holes that form a duct for ventilating the window. Frame 1 is attached to two door goose-neck hinge fittings (21) which are attached by pip-pin fasteners to clevises (22) secured to frame 5. The centre section of the frame accommodates a screwed door which, when removed, gives access to the attachment points of four thermostats. A flanged hole is provided for passage of the ventilating air. The rear surface of frame 2 is fitted with four thermostat supports (23) retained by screws and captive nuts; the rear surface of frame 3 carries a window cooling baffle (24). Each side box assembly contains a door locking system. Door riveting is sealed. The external skin is provided with four attachment points (25) for the window protective cover.

1671. The door is sealed by a rubber seal (26) integral with the structure of the door frame.

1672. **Door Window.** The external skin of the door accommodates a V-shaped window (20) composed of two window panes arranged 122° from each other. Forward and aft, the window is fitted in a metal frame made of angles (27) integral with the structure and steel extrusions (28) attached by screws and either anchor or removable nuts. The required space between these parts is maintained by a light alloy shim (29) at the rear. The seal (30) is made of tecsil material. The window panes are kept in contact with each other by two side seals (31) embedded in steel extrusions (32) screwed to the structure.

1673. **Heat Insulation.** The heat-insulating material (14) covers the internal surfaces of the cone from frame 2 to frame 9. These surfaces are insulated with 5, 10 and 15 mm thick insulating material. The ballast weights (15) and (16) as well as beams (17) and (18) are also heat-insulated. The inside of the cone is sprayed matt black.

1674. **Bonding (refer to Fig 16-12).** Bonding leads are provided at four places :

a. Fuselage-to-nose cone by a nut mounted on a threaded peg (48) integral with the box structure between frames 8 and 9.

b. Door-to-nose cone by screws between the LH goose-neck fitting and the door structure.

c. Pitot-static pressure head-to-nose cone frame 2 by screws on the frame front surface and a clamp on the pressure head.

d. Camera mount-to-nose cone, between the LH web (65) of the camera mount fitting and the door structural frame.

Equipment (refer to Figs 16-12 and 16-13)

1675. **Nose Cone Structual Fittings and Features.** The nose cone is provided to accommodate the equipment of the various systems. The cone is fitted with supports and two beams to which the equipment is attached. To avoid external heat radiation, the equipment components are insulated from the cone internal skin by cork wedges. Access to the equipment is gained through the door and, for some elements, after removal of the camera complete with its mount.

1676. Beam (17) is attached aft of frame 5 and beam (18) is attached in the boxes of frames 8 and 9. Frame 9 carries the following :

a. Nose cone-to-fuselage attaching and centring fittings (33).

b. Pitot-static pressure pipe unions which are from top to bottom :

(1) LH side : S1 and S3.

(2) RH side : T1 and S2.

c. A connector support (34) which also carries the two self sealing unions (35) of the hydraulic system, the stowage support for the self sealing union (94) of the aircraft radar pneumatic system and three stopper stowage clips. The clips are distributed as follows :

(1) On the external surface, clip (36), mounted on the RH side, supporting the stopper of the self sealing union (35) and clip (37), mounted on the LH side, supporting the stopper of the self sealing union (35).

(2) At the upper section of the support, clip (38), mounted on the LH side, supporting the stopper of the self sealing union (94) of the aircraft radar pneumatic system.

1677. The connector support (34) upper section, accommodates the accumulator support consisting of folded sheet metal to which a cradle is attached. The accumulator support is attached by screws to the face and flange of the connector support. This support is provided with two side stiffeners.

1678. Frame 5 carries the clevises (22) that attach the door goose-neck fittings. The camera mount hinge fittings (39) are attached to the lower section of frames 7 and 8. A bungee cord attaching fitting (40), secured to the ballast weight mounting plate, is installed at the upper section forward of frame 8. The 49 kg (108.03 lb) front ballast weight (15) is located between frames 5 and 6. The 21 kg (46.30 lb) rear ballast weight (16) is installed between frames 7 and 8. These weights are attached by screws through mounting plates integral with angles attached to stringers and frames to form a box structure.

1679. **Fitted Equipments.** The equipment includes the following :

a. A fan (59M) installed on two supports composed of two vertical sheets forming a bracket between frames 3 and 4.

b. A heater (58M) attached by clamps and a support to the internal skin (lower section) along the beam centre line. On installation, a shim is inserted between the support and the structure to adjust the alignment with the ventilating tube.

c. A nozzle (41) attached by a clamp (42) to the heater outlet.

d. An electric pump (65M) attached to the rear face of beam (18) at frame 9.

e. A by-pass valve (64M) screwed to a support at the upper section of beam (18).

f. A cooler (42) attached by a clamp to a support under beam (18).

g. A pressure relief valve (43) attached by screws to a support, between frames 7 and 8, at the LH upper section. This valve is connected to the hydraulic fluid drain tube (44) by a nylon hose (45).

h. A relay box (46) attached by screws to a mounting plate integral with frames 6 and 7.

i. A temperature probe (66M) mounted in the door front box, aft of the thermostats.

j. A DCU8 unit (78M). Before installation in the nose cone, the unit is fitted with an angle bracket to its upper face. The unit is located between frames 4 and 5, on the LH side and attached as follows :

(1) Front section : by locating pins plugged in an angle on frame 4.

(2) Rear section : by two anchor nuts between the angle bracket and the beam front face.

k. A junction box (77M) located between frames 4 and 6, on the RH side. The box is attached as follows :

(1) Front section : by two locating pins plugged in an angle on frame 4.

(2) Rear section : by one locating pin. The attachment is completed by two locks mounted on stirrups on the beam rear face.

l. A DAR-1 unit (79M) located on the camera rear face to which it is retained by two quick-release fasteners.

m. An accumulator (120) attached by two clamps to its cradle located at the upper section of the connector support.

1680. **Electrical Connectors.** The electrical connectors include the following :

a. Coupling connectors connecting the nose cone to the aircraft which are grouped on the connector support (34). The item numbers of these connectors are as follows (from left to right) :

(1) At the upper section :
20SE, 20SC, 20SB and 20SA.

(2) At the lower section :
20S and 20SD.

(3) Connector 128Y on the RH side is not mounted to the connector support.

b. The connectors that unplug for removal of certain equipment are arranged as follows :

(1) On a support (47) riveted to frame 4 at the RH lower section of the nose cone. The item numbers of these connectors are from bottom to top :

(a) 142Y for the pitot-static system (resistor 14D).

(b) 59M for the fan.

(c) 58M for the heater.

(2) At the bottom of frame 5, on the heater support RH side : connector 140Y for disconnection of the thermostats and temperature probe (66M) to allow for removal of the door.

1681. **Camera Mount (refer to Figs 16-12 and 16-13).** The camera is retained on its mount by four screws (49) self-locked in the camera body. The mount is attached to a supporting fitting hinged on the structure. The mount is tilted in the lower position to install the camera cartridge (76M). A bungee cord (50) supports the camera and mount to facilitate removal or installation of the cartridge.

1682. The camera mount includes the following :

a. A front box (51) provided with a handle (52) incorporating two holes (53) for attachment of the camera. On the inside, this box is fitted with a system locking the mount to the nose cone structure side beams.

b. Two side arms (55) closing the box ends; each arm is provided with a hole (56) for passage of the locking bolt (57). A stop screw (58) attached at the lower section is used to limit the mount travel in the up position and facilitate camera mount locking to the structure. Two rear intermediate fittings (59) are attached to the support fitting by four pip-pins (60). This fitting incorporates the rear attachment holes (61) for the camera.

1683. The support fitting is formed by a steel box structure (62) comprising two steel end plates (63). The upper section mounts in the clevises of the intermediate fittings (59) and the lower section is attached through ball joints (64) on fittings (39).

1684. The RH plate (63) carries a ball joint (66) accommodating a link-rod (72) to maintain the mount in the down position. Two webs (65) welded to the front face of the box are used to attach the compensating bungee cord (50) by a quick-release fastener.

1685. The bungee cord (50) is made of circular rubber strands with a loop at each end through which are fitted quick-release fasteners. A handle is formed below the lower loop to permit the cord to be expanded for removing the camera mount.

Access Door and Camera Mount Operation (refer to Figs 16-12 and 16-13)

1686. The access door and camera mount operate as follows :

a. *Camera Mount in the Up Position, Door Closed.* In up position, the camera mount is locked sidewise by the bolts (57) engaging the fittings (95) on the door frame. When the door is closed, the door box frame 1 presses against the seal (68) of an adjustable tube (69) mounted on a flanged nozzle (41) providing air flow between the heater and the door box. The tube (69) is retained by three adjusting plates (70), arranged 120° apart, attached by screws and nuts to nozzle (41). A seal (71), is mounted between the nozzle and the tube to seal the joint.

b. *Door Open, Camera Mount in the Down Position.* The door is unlocked by actuating the two rear plungers of the side locks. The door opens forwards and the goose-neck fittings (21) on the door come into contact with the rubber stops located inside the clevises (22). The door is balanced by its weight. The camera mount is unlocked by actuating the plunger on the lock located at the lower section of the front box (51). The handle (52) mounted on the front face facilitates handling of the mount and camera, the weight of which is compensated for by bungee cord (50). A link-rod (72), stowed in a clip (73) on the nose cone RH side, holds the mount in the down position. This rod is provided with two hinge ends (74); one is attached to a ball joint integral with the structure, the other is attached to the ball joint mounted on the RH side of the support fitting.

AIR CONDITIONING

1687. **Description (refer to Figs 16-14 to 16-16).** The camera requires a blown air temperature of between 25°C and 55°C. The air conditioning system is installed in the nose cone and is self-contained. The air conditioning system is switched on by setting the radar selector (4B) to STND-BY or TRX. The air in the nose cone, blown by a fan (59M), flows through a cooler (42) and a heater (58M). Both units are controlled by thermostats that adjust the temperature.

1688. The air conditioning system includes the following :

a. An air-FHS heat exchanger (42).

b. A ventilation duct (80).

c. A centrifugal fan (59M).

d. An electrical heater (58M).

e. A warning thermostat (81).

f. Four thermostats controlling the air temperature (60M, 61M, 62M, and 63M).

1689. The air is blown on either side of the window through the structure of the nose cone access door. The air is circulated around the camera to maintain it within the desired temperature range. A by-pass, tapped between the fan (59M) and the heater (58M), is used to blow air on the fan motor through a ventilating pipe (82).

1690. The heat exchanger (42) uses the hydraulic fluid of the radar exchanger (83) as a coolant; the fluid is circulated by a pump (65M). A bypass valve (64M), controlled by the temperature regulating system, permits the heat exchanger (42) to be by-passed when the air blown in the nose reaches a given temperature. A pressure relief valve (84), calibrated at 130.5 lb/in² (9 bars) protects the system; however, the valve starts bleeding off the pressure as soon as the pressure attains 116 lb/in² (8 bars). An accumulator, by-passing the main system above the pump, provides for correct pressurization of the hydraulic system and prevents cold fluid being supplied to the pump. The air chamber is charged to 23.2 lb/in² (1.6 bar) through an inlet (123). One end of the accumulator is fitted with an inlet (121) connected to the system and an outlet (122) which is blanked by a cap.

1691. A general description of the accumulator is given in publication AAP 7844.001-3 — Radar accumulator.

1692. **Heat Exchanger.** The heat exchanger (42) consists of a cylindrical body crossed by tubes crimped at the ends to two walls. Inside are arranged baffle plates. The body incorporates the coolant inlet and outlet. The air crosses the exchanger in the tubes surrounded by the fluid flowing through the internal baffles.

1693. **Ventilation Duct.** The ventilation duct (80) is composed of three tapered elements, from the heat exchanger to frame 7, from frame 7 to frame 5 and from frame 5 to the fan. Each element consists of two half shells. The installation is to be performed in the direction of flow; the first is fitted on the exchanger where it is retained by four screws, the second element at frames 7 and 5 where it is maintained by flanges and the third element on the fan to which it is attached by a clamp.

1694. **Centrifugal Fan.** The centrifugal fan (59M) provides air flow in the nose cone and has the following characteristics :

a. Power supply : 200 V 400 Hz three-phase a.c.

b. Operating speed : 11300 rpm.

c. Normal usable power : 1000 W on ground.

1695. **Electrical Heater.** The electrical heater (58M) consists of a circular copper body in which is brazed a bundle of small copper tubes of hexagonal cross section. The body is surrounded by electrical resistors. A warning thermostat (81) is located near the heater body. The assembly is covered with a 15 mm thick fibreglass coat. The air flows through the tubes where it is heated by convection.

1696. **Warning Thermostat.** The warning thermostat (81) is located near the heater body and is adjusted to close when the temperature exceeds 120°C.

1697. **Control Thermostats.** The control thermostats (60M, 61M, 62M and 63M) are located at the heater outlet, these thermostats are adjusted as follows :

a. 60M : Opens when the temperature exceeds 30°C.

b. 61M : Opens when the temperature exceeds 35°C.

c. 62M : Closes when the temperature exceeds 50°C.

d. 63M : Closes when the temperature exceeds 45°C.

1698. **Electric Pump.** The electric pump circulates fluid between the exchangers (42) and (83) and has the following characteristics :

a. Power supply : 27 V d.c.

b. Operating speed : 12000 rpm.

c. Power : 200 W.

d. Rate of flow : 2.20 U.K. gal/min (10 litre/min.).

e. Pressure : 65 lb/in² (4.5 bars).

1699. **By-pass Valve.** The by-pass value (64M) is an electrically operated plug valve having the following characteristics :

a. Maximum operating pressure : 130.5 lb/in² (9 bars).

b. Fluid used : H-515 (MIL-H-5605E).

c. Operating voltage : 27 V d.c.

d. Rotation : 90°.

e. Operating time : 2 seconds.

16100. **Regulation (refer to Fig 16-15).** The fan (59M) and the pump (65M) are controlled from the radar control unit. The regulating system is operated by four thermostats; 60M and 61M for heating control and 62M and 63M for by-pass valve (64M) control. An indicator (74M), indicates the temperature of the air blown around the camera. The indicator is connected to the temperature probe (66M) located in the door box structure. A red warning light (73M), operated by the thermostat (81), indicates to the pilot that the temperature in the heater (58M) exceeds 120°C.

16101. **Relay Box.** The relay box (46) contains the following electrical components :

a. Power supply fuses (50M, 51M, 52M, 53M, 54M and 55M).

b. Heating relay (56M).

c. Valve control relay (57M).

d. Capacitors (67M, 68M, 69M, 70M and 71M).

16102. **Relay Box Operation (refer to Figs 15-13 and 16-16).** The current supplying fuse (55M) is fed to the terminals of :

a. Thermostats (60M) controlling the following by acting in response to temperature changes :

(1) Heating relay (56M) energizing the resistors of heater (58M).

(2) Power supply of thermostat (61M) holding relay (56M) energized.

b. Thermostat (62M) controlling the following by acting in response to temperature changes :

(1) The relay controlling valve (57M) providing for operation of by-pass valve (64M).

(2) Power supply of thermostat (63M) holding relay (57M) energized.

16103. The fuse (55M) supplying the thermostats and relays is protected by five capacitors (67M, 68M, 69M, 70M and 71M) to prevent the thermostat from arcing.

16104. Fig 16-16 shows the electrical sequences controlled by the thermostats when the temperature increases or decreases.

16105. **Instrument Panel and Control Pedestal Installation (refer to Fig 16-17).** The following equipment is installed in the cockpit to control and operate the camera :

a. A ballasted support located in the centre of the instrument panel in place of the radar scope, the ballasted support accommodates the control unit (72M) on its front face. The support is made of riveted steel sheets with lightening holes in the upper and lower surfaces. The rear face carries connector (28S) and four locating pins (75) plugged into guides on the structure. Each side carries a ballast plate (76) to compensate for the difference in weight from the radar scope. The electrical wiring (77) is retained by fasteners (78) screwed at the upper section of the support and connects the nose cone overheat warning light (73M) to connector (28S). The unit is attached to the instrument panel by two captive screws (79) on its front face. The ballasted support carries the following :

    (1) Control unit (72M) : The control unit is provided with a quick-release connector (141Y) and is mounted to the quick-release connector support on the instrument panel LH side.

    (2) The nose cone overheat warning light (73M) mounted on the control unit (72M).

b. A temperature indicator (74M) located on the control pedestal on the right of the PHI lighting rheostat.

c. Electrical wiring which connects :

    (1) Connector (141Y) and the indicator connector to sealed connector (1Y) at frame 2, on the LH side.

    (2) Sealed connector (1Y) to connector (128Y) at frame 1 and to connector (20SC) on support (34) in the nose cone.

    (3) Connector (28S) to sealed connector (1Y) and to connector (20SA) on support (34) in the nose cone.

System Operation

16106. The aircraft is provided with a FAIRCHILD panoramic camera for recording navigation data. The camera accurately locates the frames in relation to the aircraft position and heading. For detailed operation of the camera, refer to AAP 7581.002-3M.

16107. The panoramic camera uses a 5 in film and incorporates a device that automatically controls exposure and provides image motion compensation. The camera produces 180° photographs by rotating a double prism in front of the lens. When shooting, an independent optical system records the counters showing the aircraft coordinates and heading.

16108. The camera operates automatically with a speed varying from 1 to 6 frames per second. The shutter speed is from 1/500 to 1/5000 second. For a maximum cycle of 6 frames/ second, the overlap is 56% at an altitude of 250 ft and a speed of 600 kts.

16109. **Operation and Use of FAIRCHILD KA 56 Camera.** The operation and use of the camera (75M) requires two distinct systems; one for operating the camera and one for recording the data.

16110. **Control Unit Operation.** The control unit (72M) carries the following control and indicating components :

a. On the left, two MANUAL INPUT selectors (85) and (86) for manual setting :

    (1) Selector (85), altitude above the ground : FEET × 1000.

    (2) Selector (86), ground speed : KNOTS × 100.

b. On the right, from top to bottom :

    (1) A FRAMES EXPOSED counter (87) and a FAIL warning light (88).

    (2) A CLOUD BELOW chart (89) giving the cloud coverage ratio, graduated 4/8 — zero — 2/8.

    (3) A POWER — OFF switch (90) with a signal light (91).

    (4) An OPERATE — OFF switch (92), controlling camera operation, with a signal light (93).

16111. **PHI-to-Camera Connection.** The PHI-to-camera connections include the following :

a. A DCU8 matching unit (78M).

b. A DAR-1 recorder (79M).

16112. Interconnection between the PHI computer (37F) and the output multiplier (27F) and the matching unit (78M) and the recorder (79M) is made through the following connectors :

a. 20SD connected to connector 2Y on frame 2 RH side.

b. 20SE connected to connector 1Y on frame 2 LH side.

16113. **Data Recording.** The recorded data are as follows :

a. The N/S and E/W grid co-ordinates of the aircraft position whatever the operating mode of the Doppler radar (the co-ordinates are computed from the ground speed when the Doppler radar operates normally and from true air speed and the wind set when the Doppler radar is in memory).

b. Aircraft Grid Heading

16114. The data transmitted by the PHI computer and the gyro centre output multiplier are presented in the form of numerals in windows and are recorded by the camera. This display is controlled by a matching unit (78M) which converts the data into signals for the recorder. A Parameter recorder (79M) which presents the information in the form of numerals.

16115. **Transformation of Grid Co-ordinates.** The N/S and E/W grid co-ordinates of the actual aircraft position relative to the actual computed postion after selection of the departure or intermediate base (including the wind components when the aircraft position is determined without the Doppler radar), are transmitted by the PHI computer. These data are amplified to supply motors (96), which are driven to the postion determined by the signal transmitted from the PHI computer. A loop feed back provides for stability. The motors mechanically drive synchro-transmitters (97), which transmit the co-ordinates to two step-by-step motors (98) located in the recorder (79M). The step-by-step motors drive the counters through reduction gears.

16116. **Transmission of Aircraft Grid Heading.** The aircraft grid heading, in the form of a synchro signal from R.P.H. output multiplier, is supplied to a synchro-receiver in the matching unit. The position variation signal is amplified and applied to a motor (99) which aligns the synchro-receiver with the position of the gyro centre synchro-transmitter. The signal is delivered through the R.P.H. output multiplier. The motor mechanically drives a synchro-transmitter (100) providing a signal to a step-by-step motor (101) located in the recorder (79M). The motor drives the heading counter through a reduction gear.

16117. **Data Display.** The data are presented in the form of numerals by three counters which can be reset manually. The counter display ranges are 0 to 999 NM for the co-ordinates and 0 to 360° for the heading. Bulbs (102) are supplied from the 115/5 V step-down transformer for counter lighting.

Nose Cone Air Conditioning

16118. Before flight, in hot weather (static temperature above 25°C), do not leave the photo nose cone exposed to the sun radiation which could result in starting a mission with too high an ambient temperature in the nose cone (specially when the aircraft is to be flown at high speed and low altitude). When the aircraft is to be parked out of a hangar, a reflecting sunshade must be placed above the cone.

16119. During flight, two indicating elements are available to the pilot to monitor the operation of the air conditiong system as follows :

a. A red warning light illuminates when the temperature in the body of the electrical heater increases abnormally indicating a failure of the fan. This causes the heater, which is no longer ventilated, to rapidly overheat. If this occurs the pilot must immediately cut-off the electrical power supply to the nose cone.

b. An indicator graduated from +10° to +70°C permits the temperature of the air blown on the window and the camera to be monitored.

KALORI DATA COLLECTING SYSTEM

Description

16120. KALORI is the name used for a system that collects aircraft operational data for assessing operational effectiveness and to assist in developing improved procedures and techniques. KALORI consists of a wide angle camera, a precision clock and a radar transponder by which the aircraft position, height and attitude with respect to time can be recorded.

16121. KALORI is carried in a dummy sidewinder missile mounted to the wing outboard pylon. Loading procedures are detailed in AAP 7213.00-33, Weapons Loading Manual.

CHAPTER 17

ARMAMENT

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CHAPTER 17

ARMAMENT

Table 17-1 Function of Components

Index	Description	Characteristics and Functions
2A	Armament safety panel	Includes guns, rockets, bombs and missiles safety switches.
3A	LH gun firing relay	Energized through closing of 2A, this relay transmits the electrical power supply to the LH gun (9A) through the junction box (7A).
4A	RH gun firing relay	Energized through closing of 2A, this relay transmits the electrical power supply to the RH gun (10A) through the junction box (8A). It also cancels the anti-stop system in the sight during the firing period.
5A	LH gun arming connector	This connector permits the LH gun (9A) to be pyrotechnically recocked on the ground through a shunt (2A must be closed).
6A	RH gun arming connector	Same function as 5A but for the RH gun.
7A	LH gun junction box	This box houses the pyrotechnical recocking time delay relay as well as the various electric connections between the gun and the control circuit.
8A	RH gun junction box	Same as 7A.
11A	Gun/Missile firing button	When pressed, this button initiates firing and causes a triangular index to appear on the sight camera film.
013A	Post-Mod 926 Pre-Mods 1112 and 1113. Sight override button located on the throttle control lever.	This pushbutton enables rapid change from missiles to guns.
013A	Post-Mods 1112 and 1113. Rapid gun/MAGIC missile (RG/M) button.	In addition to the Mod 926 functions, initiates fast switching of the MATRA R550 preparation circuits.
25A	Sight relay box	This box is electrically connected to the gyro centre, air data computer output multiplier and range setting unit to feed the sight head with control signals from these units.
26A	Voltage regulator	Supplied with 28V d.c. power from the aircraft electrical system. This regulator supplies the different regulated voltages necessary for sighting system operation; its output voltage is adjustable from 21.5 to 22.5V.
27A	Gyro caging button (front cockpit only — on throttle control lever)	In the air-to-air gun or rocket firing mode, this button permits the sensitivity current to be changed. It performs the following functions when depressed : a. Stiffening of the gyroscope (to increase sensitivity current). b. Suppression of incidence correction. c. Post-Mods 1112 and 1113, with SW selected on the armament control panel or pushbutton (013A) pressed, reduces the MATRA R550 homing head to narrow scan.
29A	Sight head	Located in front of the pilot's eyes, the sight head permits sighting through a semi-reflecting glass for air-to-air gun tracking and firing as well as for air-to-ground gun, rocket and conventional bomb firing.
43A	Missile selector switch	When set to the MATRA position, permits MATRA pre-firing preparation, missile monitoring, firing and jettisoning.
45A	A.C. Isolation relay (Post-Mod 919)	Prevents overheating of the MATRA missile computer power transformer and associated components including the harmonization unit.
49A	Armament safety relay	When in the de-energized position, this relay ensures closing of the missile audio circuit.
55A	Harmonization box	In conjunction with the radar, ensures the missile homing device operation.
56A	Computer	Computes proportional navigation co-efficient and fuse ignition delay data for transmission to the missile.

Index No	Description	Characteristics and Functions
65A	LH missile heater relay	In the preparation position, this relay permits energization of the LH heater circuit. Supplies missile heating for the MATRA R550 (Post-Mods 1112 and 1113).
66A	RH missile heater relay	Same as LH heater relay.
67A	Post-Mods 1112 and 1113 Preparation relay	Energized from the ST-BY ON switch on the armament control panel, this relay ensures missile preparation through heater relays (65A) and (66A).
68A	Post-Mods 1112 and 1113 R550 missile safety relay	Energized from the armament control panel mode selector in the S.W. AIR-TO-AIR position, this relay ensures closing of the R550 missile safety circuit.
69A	LH missile firing relay	Energized by the firing circuit, this relay closes the LH missile ignition circuit.
70A	RH missile firing relay	Energized by the firing circuit, this relay closes the RH missile ignition circuit.
72A	Post-Mods 1112 and 1113 Missile prelaunching relay	Energized from the armament control panel mode selector in the S.W. AIR-TO-AIR position, this relay ensures starting of the prelaunching sequence of both missiles.
74A	Missile volume potentiometer	This potentiometer is used to adjust the volume of the locked on missile audio signal.
75A	Post-Mods 1112 and 1113 Aircraft connector/LH missile launcher connection	This connection is provided by the loading devices : CES 3 and ADP 4.
76A	Post-Mods 1112 and 1113 Aircraft connector/RH missile launcher connection	This connection is provided by the loading devices : CES 3 and ADP 4.
93A	Post-Mods 1112 and 1113 Firing timer relay	Energized through depression of the button (2B) this relay holds the firing relay energized for 3 seconds.
113A	Post-Mods 1112 and 1113 LH missile locked on light	Located on the LH glareshield, this light indicates the following : a. When flashing : LH missile homer locked on. b. When continuously on : LH missile gyro-aligned function achieved.
114A	Post-Mods 1112 and 1113 RH missile locked on light	Located on the LH glareshield, this light indicates the following : a. When flashing : RH missile homer locked on. b. When continously on : RH missile gyro-aligned function achieved.
115A	Post-Mods 1112 and 1113 Rapid gun/MAGIC missile (R G/M) light	Located on the LH glareshield, this light indicates selection of the R G/M function (lighted) and enables cancellation of the R G/M function.
123A	Post-Mods 1112 and 1113 Transfer box	Located in the radio compartment, this box enables firing of the RH or LH missile if only one missile has its gyro aligned upon firing, and in case of misfiring of the first missile.
02B	Post-Mods 1112 and 1113 Armament master switch	Located on the LH Instrument panel, this switch ensures the continuity of the R550 missile safety circuit.
03B	Post-Mods 1112 and 1113 Armament master light	This light indicates closing of the R550 missile safety circuit.
2B	BRM firing button	This button causes missile firing provided all safety systems are overriden, inclusive of the undercarriage down firing prevention system.
3B	Rocket/Bomb firing relay	Energized by actuation of the button (2B), this relay controls the power supply to the rocket and bomb firing circuits.
4B	Armament/Radar control panel	Includes weapon selectors, firing mode selectors, as well as missile preparation, auto fire, conventional bomb release, sight and radar.
6B	Bomb emergency relay	Energized by the release on SALVO (armament control panel 4B) through the master switch (2A), this relay permits salvo release of the front and rear bombs (front bomb released 0.3 second after the rear bomb).
	Pylon	This pylon, which cannot be jettisoned, is used for connection between the aircraft system and the bombs. It contains the normal release relay, the emergency salvo release relay, the under fuselage store jettison relay and the arming solenoids.

Index No	Description	Characteristics and Functions
12B/212B	Front/Rear cockpit — Wing store jettison buttons	These buttons permit emergency jettisoning of under-wing stores. The missiles are jettisoned safe (without the pylons).
14B	Heading error switch	This switch is used to feed to the sight, through a potentiometer, the measurement of the error between the true heading and the heading read on the instruments.
57F	Range setting unit	This unit enables the approximate aircraft-to-target range to be set by the pilot; it directly feeds the sight with gravity drop correction information necessary for accurate firing.
44G	Ground firing connector	This connector permits the energization circuit of 46G to be shunted for performance of a number of tests on the ground.
45G	Undercarriage slaving microswitch	Actuated by the LH undercarriage leg, this microswitch allows energization of relay (46G) and button (2B) when the undercarriage is retracted.
46G	Ground operation prevention relay	When de-energized, this relay prevents continuity of a number of electrical circuits when the undercarriage is extended. Energized when the undercarriage is retracted, or when the ground firing shunt is in position, it allows continuity of these circuits.
39P	Preheating relay	When de-energized, this relay ensures continuity of the R550 and Sidewinder missile system preparation and safety circuits.
3R	Radio selector box	This box ensures continuity of the missile audio circuit.
30R	Ground telephone door micro-switch	This micro-switch is closed when the ground connector door is open to supply the circuit of 31R.
1R-III 0 31R-III D	Ground telephone connection	This connection enables the Sidewinder missile audio circuit to be checked on the ground through a headset.
4S	Radar preheating relay	When de-energized, this relay ensures a.c. power supply to the R550 missile system.

GENERAL (refer to Figs 17-1 and 17-2)

1701. This chapter gives a brief description of the armament systems that can be carried by the MIRAGE. All systems described are optional fits and are selected to suit the planned mission. It is not possible to simultaneously fit all of the systems described in this chapter. The systems are :

a. Two 30 mm guns.

b. Two combined rocket launchers/fuel tanks capable of carrying 18 rockets each.

c. Two MATRA R550 missiles (Post-Mods 1112 and 1113).

d. Two 400 kg STRIM bombs or Two 500 lb bombs (U.S. or U.K.) or One 1000 lb bomb (U.S. or U.K.).

e. III O Only, one MATRA R530 missile.

f. III O Only, two bomb carrier tanks (Post-Mod 406).

1702. For III D, only the front cockpit control stick hand grip is provided with firing controls.

INSTALLATION OF STORES

Gun Bay (refer to Fig 17-3)

1703. The gun bay is located at the bottom of the fuselage between frames 17 and 20 and enclosed laterally by two web assemblies. The gun bay incorporates the following units :

a. Gun pack attachment fittings (2).

b. Gun pack guide fittings (3) on frames 17 and 20.

c. Three hoisting cable attachment fittings (4) at the top of the bay.

d. An electrical connector (5) on the RH web.

Under Fuselage Stores (refer to Figs 17-3 and 17-4)

1704. There are two under fuselage attachment points (A and B) located on the aircraft centre line, point A rear of frame 20 and point B at frame 26. The attachment points consist of :

a. A threaded housing (16) for the attachment device tulip-fitting.

b. Two bearing faces (7) for locating the pylon transversely.

c. A ball-joint housing (8) for locating the pylon laterally.

1705. When not in use, the threaded housings are blanked off by a screwed plug at point A and a protective fairing secured by three screws (12) at point B.

1706. The following items are also provided under the fuselage :

a. An electrical connector (122Y) forward of frame 23 for the tandem bomb pylon connection.

b. For III O Only, self-sealing coupling (112) aft of frame 24 for connecting the MATRA R530 missile pylon.

c. A panel (13) incorporating :

(1) For III O Only, two electrical connectors (91Y and 92Y) and three coaxial connectors for the MATRA R530.

(2) For III D Only, an electrical connector (92Y).

d. A ball-joint housing (11) aft of panel (13) for locating the missile pylon laterally.

Under-wing Stores (refer to Fig 17-5)

1707. The inboard attachment points at rib 4 are, from front to rear :

a. A ball-joint housing (20).

b. A self-sealing pressurization coupling (21).

c. A fuel coupling (22).

d. A ball-joint housing (23).

e. An electrical connector (88Y RH wing, 87Y LH wing).

f. A threaded housing (24) taking the tulip-fitting of the attachment device. When not in use, the housing can be blanked off by a plug (28).

g. A rear centring ball-joint housing (25).

1708. The outboard attachment points at rib 6 are from front to rear :

a. A threaded housing (26) (pylon front attachment) consisting of a fitting integral with the front spar.

b. An electrical connector (85Y).

c. A threaded housing (27) (pylon rear attachment). This housing is machined in the outboard elevon jack attachment fitting.

GUNNERY SYSTEM

General (refer to Figs 17-6 and 17-7)

1709. The aircraft is fitted with two automatic 30 mm guns installed in a gun pack. The 30 mm gun is an automatic, revolving drum, electrically-fired, gas-operated gun using ammunition belts with detachable links. The position of the moving parts is controlled by a chain of contacts.

1710. The gun pack also houses the ammunition feed and link recovery boxes, the hoisting device, a cooling system and the electric circuit.

Electrical Description (refer to Fig 17-6)

1711. The electrical circuit includes :

a. A ground firing prevention circuit common to all firing controls and operated by microswitch (45G) and relay (46G). This circuit can be closed by plugging a ground firing shunt into connector (44G) (port undercarriage well) to enable ground tests to be performed.

b. A firing control, pushbutton 11A on the control column hand grip which is used for both gun and missile firing.

c. Relays (3A) and (4A) in the armament box.

d. Two LH (7A) and RH (8A) junction boxes, in the gun pack.

e. Two gun recocking connectors 5A LH and 6A RH on the gun pack.

f. Gun pack safety pack (6A).

g. Post-Mod 926, a pushbutton (013A) on the throttle handle.

Operation (refer to Figs 17-6 and 17-17 to 17-19)

1712. When safety switch (2A) is closed (and safety connector (44G) is shunted for ground tests) relay (46G) allows energizing of firing button (11A). Depressing the firing button activates firing relays (4A and 3A) thus energizing the time delay relay. When the tube is positively locked and the slide in contact, the relay sets the firing pin in contact and firing starts.

1713. Relay (3A) is provided with an output to the engine fuel dipper unit which prevents surging or flame-out under certain firing conditions. Relay (4A) is provided with an output to the sight relay box which by-passes the sight gyro anti-toppling device during firing.

1714. Post-Mod 926, pushbutton (013A) on the front of the throttle controls a relay (RL1) in the armament control panel (4B) to permit rapid change over from missiles to guns in close combat conditions. Depressing pushbutton (013A) results in :

a. The gunsight mode to change to GUNS irrespective of the selection on the armament control panel, excepting G/S OFF.

b. For III O only :

(1) The radar switching to the AIR-AIR GUNS mode.

(2) If 013A is held depressed, the radar switching to AIR-AIR ranging.

1715. The action of pushbutton (013A) is cancelled by moving the selector on the armament control panel to another sector.

1716. The circuit diagram for relay (RL1) in the armament control panel is given in Fig 17-19 Post-Mod 929 and Figs 17-17 and 17-18 for Post-Mods 1112 and 1113.

1717. Post-Mods 1112 and 1113 pushbutton (013A) initiates additional MATRA R550 functions as described in Para 1761.

Automatic Recocking

1718. If firing is interrupted, with the gun parts forward and a shell in place (defective cap), recocking is automatically performed after 0.3 second by a time delay relay located in the gun junction box.

Pyrotechnical Recocking On The Ground

1719. Pyrotechnical recocking on the ground is performed by closing the master switch (2A) and shorting one of the connectors (5A) or (6A) to operate the gun through a complete cycle by igniting a special cartridge.

Installation of Guns (refer to Fig 17-7)

1720. For attachment of the gun pack to the aircraft, refer to Chap 4, para 4153.

1721. Three attachment points are provided to attach each gun to the gun pack. The attachment points are :

a. *Front Mount (30).* The gun barrel is fitted through an adjustable ring which is bolted to frame 15 and supports the gun blast tube.

b. *Centre Mount (31).* The centre mount consists of a ball and socket assembly. A ball member (33) secured to the gun barrel is engaged in a socket (34) integral with a bracket (35) attached to the gun pack. A handle (36) fitted with a safety catch locks the assembly in position and provides for quick installation and removal.

c. *Rear Mount (32).* The gun pack is fitted with two adjustable attachment fittings (37); elevation adjustment is by means of screws (38) and azimuth adjustment by adding shims (39). Two plain pins (40 and 41) orientated towards the rear are secured to the above fittings. Each gun is provided with two bored swivel bosses (42) engaging on the plain pins (40 and 41).

1722. **Case Ejection Chute.** A tube (43) connects the breech to the external ejection point in the aircraft skin and guides the empty cases to outside the aircraft.

1723. **Gas Deflectors.** The gas deflectors are located at the gun blast tube outlets between frames 13 and 14. The gas deflectors are intended to avoid the risk of engine surging or flame-out when firing the guns at high altitude and low IAS and under certain load factors. The gas deflectors consist of a roll-formed sheet element (100) onto which are welded four fins (101). The assembly is attached to the structure by means of six screws (102) with shims (103).

BOMBING SYSTEM

Description (Refer to Figs 17-8 to 17-11)

1724. The ALKAN bomb rack enables two 880 lb STRIM bombs or two 500 or one 1000 lb U.S. or U.K. bombs to be carried under the fuselage. Provision is made for either salvo or single bomb release. For single release, the rear bomb is dropped first. Each bomb is normally fitted with a nose fuse and a tail fuse.

1725. Two release modes are provided :

a. Instantaneous (both fuses armed).

b. Delayed (tail fuse only armed).

1726. A jettisoning system enables both bombs to be dropped in an unarmed condition.

Electrical Installation (refer to Fig 17-8)

1727. The electrical installation consists of :

a. A relay control circuit (common to the rocket circuit).

b. A ground firing prevention system common to all firing systems (microswitch 45G and relay 46G) which can be shunted through connector (44G) for ground tests.

c. A firing button (2B) (bombs — rockets).

d. The armament safety switch panel (2A).

e. The armament control unit (4B).

f. A multiple relay (3B) in the armament box.

g. The bomb emergency relay (6B).

h. A safety circuit including a switch on the safety panel (2A).

i. A priming and release circuit which activates the bomb release initiators either through relay (3B) or through relay (6B) depending on the position of the SINGLE-SALVO selector.

j. An emergency circuit including :

(1) A pushbutton (2M).

(2) A relay (A) enabling bombs to be dropped in an unarmed condition.

Operation

1728. Bombing selection is controlled by the armament control unit (4B) as follows :

a. Selector switch in the FUS position; fuselage mounted bombs selected.

b. For III O Only. Selector switch in the WINGS position; wing mounted bombs selected.

c. For III O Only. Selector switch in the F + W position; fuselage and wings bombs selected.

d. SINGLE-SALVO selector set as required.

1729. **Single Bombing.** Depressing the firing button (2B) energizes the firing relay (3B) which closes. This applies 27V power to the release initiator of the rear bomb through the delayed action reversing switch (57). The reversing switch remains in this position as long as the firing button is kept depressed.

1730. As soon as the firing button is released, the reversing switch solenoid is no longer energized and the switch contacts reverse. When the button is depressed again, the 27V power is applied to the release initiator of the front bomb through a 0.3 second time delay relay located in the beam.

1731. **Salvo Bombing.** Depressing the firing button (2B) energizes the relay (6B) which closes. The position of the relay (6B) is determined by the mode selected on the release mode selector on the armament control panel (4B). The 27V power is first applied to the release initiator of the rear bomb. Release of the rear bomb reverses the contacts of the instant action reversing switch (59) located under the fuselage. The front bomb release initiator is then energized provided the firing button is kept depressed. As the initiator is energized through a time delay relay, the front bomb is released 0.3 second after the rear bomb.

1732. **Bomb Fuse Arming.** Three positions can be selected on the armament control panel (4B) :

a. *Safe.* Fuses unarmed.

b. *Delay.* Tail fuse armed.

c. *Instantaneous.* Nose and tail fuses armed.

1733. **Emergency Jettisoning.** Depressing the jettison button (2M) results in salvo jettisoning of the bombs and opening of the two fuse arming circuits (through relay A). The bombs are dropped safe, the front bomb being released 0.3 second after the rear bomb through the action of the time delay relay.

Installation (refer to Fig 17-11)

1734. The rack is attached to the fuselage underside by the following :

a. Two roller devices hooking and locking the rack at points A and B.

b. Two centring ball-joints; one at frame 20 (50) and one at frame 26 (51) which fit into two sockets (18 and 22) integral with the aircraft close to the roller locking devices. The ball-joint at frame 20 allows longitudinal play.

c. Transversal centring is provided by four plates (two at frame 20 (20) and two at frame 26) coming in contact with the four corresponding plates under the fuselage.

1735. Electrical connection is provided by connector 122Y at frame 23. The rack aft end includes a fairing to streamline it with the rear bay tank.

Underwing Bombing System — III O Only

1736. Mirage III O aircraft with Mod 406 incorporated are capable of carrying two RPK 10-0 bomb carrier tanks attached to the inboard under-wing stations. For information on the RPK 10-0 tank, refer to AAP 7271.299-3 and AAP 7271.299-2M.

1737. Bomb release and fusing is controlled by the armament control unit as described in para 1724. Electrical supply to the bomb carriers is provided by 87Y (LH) and 88Y (RH).

1738. When the under-wing bomb carrier tanks and the twin-bomb under fuselage pylon are fitted, for a SALVO release, a time delay relay in the armament control unit (4B) introduces a delay between release of the first fuselage bomb and release of the first wing bomb. The rear fuselage bomb is released singly then 0.3 second later the first wing bomb and the front fuselage bomb are released at the same time.

NOTE

Mod 752 must be incorporated before the RPK bomb carrier tanks and a twin-bomb pylon can be used at the same time. RPK bomb carrier tanks may be used without a twin-bomb pylon on Pre-Mod 752 aircraft.

MATRA R550 MISSILE SYSTEM

Description (refer to Figs 17-14 to 17-16, 17-18 and 17-19)

1739. The MATRA R550 Air-to-Air missile system is introduced by Mods 1112 and 1113 and consists of two missiles mounted one to each outboard wing pylon by means of adapters and missile launchers. The missiles are fitted with an infra-red homer the sensing cell of which is cooled by the expansion of nitrogen which is stored under pressure (maximum value of 325 bars at 20°C) in a cylinder in the missile launcher. Provision is made for launching either the LH or RH missile, as required. There is also provision made for safe jettisoning of the missiles in an emergency. A full description of the MATRA R550 Missile system is given in AAP 7352.035-2-2 ANCP.

Installation

1740. The missiles are mounted at the outboard wing station by means of :

a. CES 3 wing pylon.

b. ADP 4 adapters.

c. Type 40 missile launchers.

1741. In addition to carrying the missiles the mounting system performs the following functions :

a. Connection of the various circuits to the missiles for heating, preparation, pre-launch and presence.

b. Launching pulse.

c. Missile jettison pulse in case of emergency.

d. Cooling of the infra-red (IR) cell.

e. Target search control.

f. Control of the safety devices against inadvertent ignition on the ground and in flight.

1742. **CES 3 Wing Pylon.** The wing pylon is attached to the wing by two threaded bolts (40 and 42). The front bolt (40) can be aligned through spherical rings which float freely between two washers. Electrical connection is provided by two connectors, an upper connector (41) for connection to the aircraft connector and a lower connector (44) for connection to the ADP 4 adapter connector. The CES 3 wing pylon cannot be jettisoned.

1743. **ADP 4 Adapter.** The ADP 4 adapter is attached under the CES 3 wing pylon by two bolts (46 and 48) access to which is gained through two doors attached by screws. Electrical connection is provided by three connectors, the upper one (47) attached to a support for the CES 3/ADP 4 connection and the two lower connectors (50) for the ADP 4/LM40 connection. The adapter cannot be jettisoned.

1744. **Type 40 Missile Launcher.** The missile launcher is attached under the ADP 4 adapter by two floating bolts (7 and 14) having a 30 inch (762 mm) centre-to-centre distance. Electrical connection is provided by three connectors, two upper connectors for attachment to the corresponding adapter connectors and the third connector at the front of the cord allowing connection to the missile. The missile launcher cannot be jettisoned.

1745. **MATRA R550 Missile.** The missile is attached to the missile launcher by means of three fittings, two inner fittings (7 and 10) and one outer fitting (12) on the rearmost section. Interconnection with the missile launcher is provided by a cord (23) with an electrical plug (24) for connecting to the corresponding missile launcher receptacle. The cord also provides continuity of the nitrogen system. At the other end the cord is fitted to a receptacle (21) secured to the missile by four shear screws. A stud (22) locks the receptacle to the missile launcher pull-out lever. A safety pip-pin (12) fitted with a streamer (13) locks the front step (32) in the down position to prevent the missile being launched in the case of accidental ignition on the ground.

Operation

1746. The missile heating circuit is automatically energized on switch-on of the aircraft electrical power supply system, provided the NORMAL/PREHEATING selector switch is at NORMAL. The relays (65A) and (66A) are energized and the heating circuit is closed. Relays (65A and 66A) can be energized only if missiles are carried; the missiles being used for the earth return for the relays.

1747. The preparation function of both missiles is performed as soon as the armament control panel STND-BY switch is turned ON through the relay (67A) which is energized and relays (65A) and (66A). The pre-launching sequence of both missiles is simultaneously performed as soon as the mode selector is set to the AIR/AIR : SW position through the relay (72A) which is energized. The SINGLE-SALVO release mode selector, on the armament control panel (4B) is used, through the relay (1) or (8) of the transfer box (123A), to give priority to either missile, with automatic transfer if the missile which has priority has not locked on. In the SINGLE position priority, for the LH and in the SALVO position, priority for the RH missile.

1748. **Post-Mods 1112 and 1113 III D Only (refer to Fig 17-20).** A transfer box (1) is fitted to the upper ventilation duct (2) in the pressurization bay and is connected to the armament box by electrical connector and loom 123A.

1749. **Post-Mods 1112 and 1113 III O Only (refer to Fig 17-21).** A transfer box (1) is fitted at the LH front of the radio equipment bay and is connected to the armament box by electrical connector and loom 123A.

1750. **Single Firing (refer to Figs 17-2, 17-14 and 17-15).** For single firing the following actions are required :

a. On the armament control panel (4B) set :

    (1) The mode selector A/A S.W.

    (2) The release mode selector SINGLE.

    (3) The STND-BY ready switch ON.

b. On the armament master switch panel (2A) set the MISSILES switch to ON.

c. On the LH instrument panel set the armament master switch (02B) to ON and check that the armament master light (03B) illuminates.

1751. The missile ready information is displayed on the sight to indicate that the missile circuit is ready for firing. If both missiles are locked on, priority is given to the LH missile (SINGLE position). LH missile lock-on is indicated to the pilot by the LH 550 light (113A) either flashing at 4 Hz (homer locked on) or continuously illuminated (homer locked on and gyro aligned).

1752. In the transfer box (123A) the relay (4) is energized, which closes the audio circuit through relay (65A) and applies power to the R550 missile audio module. LH missile lock-on is indicated by an audio signal in the audio circuit of the pilot's headset, which is either a 800 Hz audio signal, 4 Hz chopped (homer locked on), or 800 Hz audio signal continuous (homer locked on and gyro aligned). The missile volume potentiometer (74A), on the LH console, is used to adjust the volume of the audio tone which is supplied through the common circuit. The relay (2) is energized, which causes relay (1) to be energized and the LH missile firing circuit closed in the transfer box.

1753. When the pilot presses the firing button (2B), the firing timer (3S) relay (93A) is energized and causes the firing relay (69A) to be energized and apply 28V power to the LH missile firing circuit. Firing of the LH missile automatically causes transfer to the RH missile circuit.

NOTE

During firing, continuously pressing button (2B) ensures self-holding of the relays enabling closing of the LH missile firing circuit; transfer to the RH missile circuit can be performed only after having released the button and the 3 second time delay of relay (93A) has elapsed. This self-holding system therefore does not allow salvo firing.

1754. RH missile lock-on to the target is indicated by the illumination of the RH 550 light (114A) in the same way as described for the LH missile. In the transfer box (123A) relay (5) is energized, which causes the audio circuit to be closed through relay (66A) and power is applied to the R550 missile audio module. RH missile lock-on is indicated by an audio signal in the same way as for the LH missile. Relay (7) is energized, which causes relay (8) to be energized and the RH missile firing circuit closed in the transfer box.

1755. Pressing button (2B) again, 3 seconds later due to the time delay of the relay (93A), causes relay (70A) to be energized through the RH missile firing circuit of the transfer box and apply 28V power to the RH missile firing circuit.

NOTE

If automatic transfer does not occur after LH missile firing, the release mode selector must be placed in the SALVO position to ensure RH missile firing.

1756. **Switchover to RAPID AIR-TO-AIR MAGIC Missile (refer to Figs 17-17 to 17-19).** Regardless of the position of the mode selector, except G/S OFF and of the STND-BY ready switch. Pressing the rapid gun/MAGIC missile button (013A), on the throttle, energizes relay (RL1) of the armament control panel (4B). Relay (RL1) is self-energized through a normally closed contact of the R G/M annunciator (115A).

1757. When relay (RL1) is energised the following actions occur :

a. Annunciator (115A) on the LH glareshield illuminates to show that the system is in operation.

b. The gunsight changes over to GUNS and, for III O only, the radar changes to AIR-AIR GUNS.

c. All MISS STBY functions for MATRA R550 are initiated, provided that the MISSILES master switch is set to ON.

d. The missiles commence a wide scan and will lock-on to any IR target within their field of view.

e. For III O only, if pushbutton (013A) is held in, the radar scans in AIR-AIR ranging. When 013A is released, the radar either remains locked on or reverts to AIR-AIR sweep if a lock-on is not taken.

1758. To cancel the RAPID AIR-TO-AIR function, it is necessary either to depress the annunciator button (115A) or to move the armament control panel selector to another sector to de-energize relay (RL1) and thus revert to the function selected at the armament control panel.

1759. **Missile Jettisoning.** Safe missile firing is performed by depressing the WING emergency jettison button (12B) which also jettisons the wing stores.

MATRA R530 MISSILE SYSTEM — III O ONLY

Description (refer to Figs 17-12 and 17-13)

1760. The aircraft is capable of carrying one Air-to-Air MATRA R530 missile. There are two different versions which differ in the principle of operation of the homing device. Missile firing is controlled either manually by the pilot or automatically by the radar and computer.

1761. Firing of the missile is possible only after preparation, warm-up, harmonization and locking on to the target (indicated by the audio circuit). A circuit is provided for jettisoning of the missile and launching pylon. The installation consists of :

a. A missile launching pylon.

b. An electrical preparation and control circuit.

c. The harmonization and computation box.

d. The MATRA computer.

Installation

1762. The missile launcher is mounted in the aircraft symmetry plane under the fuselage at point B. It consists of a streamlined body incorporating devices for attachment to the aircraft and suspension of the missile as well as aircraft-to-missile interconnecting circuits. The launcher is fitted with an ALKAN ejector which provides both the attachment points and a means of jettisoning the missile. Attachment at point B is similar to that of the twin-bomb rack. Centring at point (11) is provided by a ball (78) fitting in a mating socket (25).

1763. At the rear of frame 24, the pylon is connected to the aircraft equipment conditioning system through a self-sealing air valve (112) used for cooling.

1764. The pylon is connected to the aircraft radar and electrical circuits by two multi-pin and coaxial quick-release connectors (91Y and 92Y) attached to door (13). The electrical preparation and control circuit consists of :

a. A warm-up circuit.

b. A missile fitted circuit.

c. A firing circuit.

d. A distress jettisoning circuit.

1765. The harmonization and computation circuits comprise :

a. The MATRA harmonization box (55A).

b. MATRA computer (56A).

c. The circuits for interconnection with the a.c. power supply circuits, preparation and control circuit, radar nose cone and missile launcher.

Operation

1766. The missile selector switch (43A) on the armament junction box is set to the MATRA position which permits operation of the following systems :

a. Pre-heating under ground alert conditions.

b. Missile preparation.

c. Listening-in.

d. Harmonization and firing computation. Post-Mod 919, the a.c. power supply to computer and harmonization unit is controlled by relay (45A).

e. Manual firing.

f. Automatic firing.

g. Distress jettisoning.

1767. **Preheating Under Ground Alert Conditions.** With the missile STND-BY switch in the OFF position on the armament control panel (4B) and an external power unit connected to the aircraft power receptacle (17P) when switch (40P) is placed in the PREHEAT position, relay (39P) isolates the aircraft circuit. This allows the preheating circuit to be supplied from the external power receptacle (17P) through the missile selector switch (43A) and relay (46A).

1768. **Missile Preparation.** After removing the external power supply and switch (40P) is set to OFF, the weapon selector is placed at AIR-TO-AIR and the missile STND-BY switch to ON. This cancels all ground firing prevention devices. Pre-mod 919, the harmonization box (55A) and the computer (56A) are energized as soon as the a.c. power system is under load; Post-Mod 919, the a.c. supply is via relay (45A) which is controlled by the missile selector switch (43A). Relay (39P) reverses and the control circuit is connected to the aircraft electrical circuit. Relays (42A) and (46A) are energized and the warm-up and operating circuits are supplied via fuse (20A) which also supplies the sight junction box and causes its missile indicator lights to illuminate.

1769. **Listening-In.** With the safety switch (2A) set to ON, relay (49A) is energized, the audio circuit is closed and connects the harmonization box (55A) to the radio switching circuit. The audio level can be adjusted by means of the volume potentiometer (74A).

1770. **Harmonization and Firing Computation.** Harmonization and firing computations are achieved by the harmonization box (55A) and the computer (56A) and supplied to the missile using the following parameters prepared by the radar :

a. Aircraft/target relative position and velocity.

b. Altitude.

1771. Locking of the missile on to the target is indicated by a steady tone in the earphones.

1772. **Manual Firing.** With the AUTO FIRE switch in the OFF position the missile is fired by the pilot pressing the firing trigger to actuate button (11A). Relay (50A) is energized for three seconds by relay (93A). The ignition circuit supplied from (20A) is closed and has a branch circuit leading to the harmonization box.

1773. **Automatic Firing.** With the AUTO FIRE switch set to ON, the missile is fired by a signal furnished by the radar (in connection with the computer) which energizes relay (48A). The energizing circuit of relay (50A) is closed. The firing sequence is then similar to that for manual firing.

1774. **Distress Jettisoning.** The jettisoning circuit is directly supplied from the battery when the jettisoning button (2M) is closed. The missile is jettisoned safe together with the launching pylon; there is no self-destruction device.

MATRA R530 LIVE MISSILE FIRING RECORDING EQUIPMENT — III O ONLY

Description

1775. Post-Mod 958, a SFIM paper trace recorder is fitted to enable accurate assessment of the performance of the missile detonation delay unit on MATRA R530 missiles. The recorder is fitted before missile loading is carried out for a live firing sortie.

1776. The recorder is mounted to a base plate and a support plate located forward on the equipment bay floor LH side. The recorder is removed on completion of the sortie.

1777. For mounting and wiring diagram details refer to AAP 7213.003-100, Mirage Modification No 958.