PART 1

DESCRIPTION AND MANAGEMENT OF SYSTEMS

List of Chapters

PART 1

Chapter 1—ELECTRICAL SYSTEM

Contents

Description

1 General

(*a*) Power for the electrical services is normally obtained from an alternator and a generator, mounted on an air turbine motor which is driven by air tapped from the engine compressors. The generator supplies DC at 28 volts to the DC busbar and the alternator supplies 3 phase AC at 200 volts, 400 Hz to the AC busbar.

(*b*) A standby generator is also provided. This generator is driven by an air turbine using air tapped from either or both engine compressors, and supplies DC at 28 volts to the DC busbar when the generator is switched into service.

AL 13

(*c*) A 24 volt ~~25~~ 18 amp hour battery is connected to the DC busbar by an isolating switch and is charged by whichever generator is running. It provides standby power for all DC services and, via a type 100A inverter, standby AC for certain of the AC-operated instruments and services.

(*d*) A 24 volt 0·4 amp hour emergency battery supplies the emergency lighting for the cockpit, the E2B compass and the standby direction indicator. The battery is not connected to the DC busbar and is not charged from the aircraft system.

(*e*) Switching for both AC and DC operated services is 28 volt DC controlled.

2 Air turbine motor

The motor is supplied with air from a duct fed by both engine compressors. When the air supply is sufficient, motor speed is controlled by an integral governor so that alternator output is maintained at a constant frequency. The unit incorporates three switches:

Overspeed switch: if turbine speed becomes excessive, the switch closes to close the main turbine shut-off cock to cut-off main generator and alternator output. The shut-off cock can only be reset on the ground.

Nozzle stall switch: if air pressure falls below the minimum value to maintain turbine speed, the switch operates to bring on the TURB warning and open the alternator contactor (AC warning on). The switch resets when air pressure increases.

Underspeed switch: duplicates nozzle stall switch operation as motor speed reduces, if nozzle stall switch fails to operate.

3 Main generator supply

(*a*)

(i) Power for the DC operated equipment is supplied by a brushless, 28 volt, 200 amp generator. Its output is automatically controlled by a control and protection unit. A fuse protects the busbar from generator faults.

(ii) The generator comes on line automatically when an engine is accelerated to approximately 40% RPM and achieves full output at about 50% RPM. When the RPM of both engines falls below 35% approximately, or if the generator voltage falls appreciably for any other reason, the differential relay operates to take the generator off-line.

◀(ii) Before an engine is started, the STANDBY GEN switch must be at NORMAL. When starting from the battery, the generator warning on the SWP goes out when the output of the generator is between 15 and 25 amps, ▶ followed by the generator warning on the AWP when voltage increases to 25·5 volts. Removal of the generator warning on the AWP indicates that the generator has taken over the full load on the busbar. With a ground DC supply connected, the generator warning on the AWP goes out when the main contactor closes as the generator runs up. The generator warning on the SWP may go out or it may remain on, depending on the relative outputs of aircraft and ground supply; if the warning remains on, it should go out when the external supply is removed.

(iii) The DC equipment is protected by an overvoltage relay and an undervoltage relay. If the overvoltage relay operates or if the undervoltage relay operates when the turbine is running at normal speed, the generator is shut down. If undervoltage is caused by the turbine underspeeding, the undervoltage relay operates to light the generator warning on the AWP but does not shut down the generator.

(iv) A facility is embodied for resetting the generator in the event of transient failure occurring.

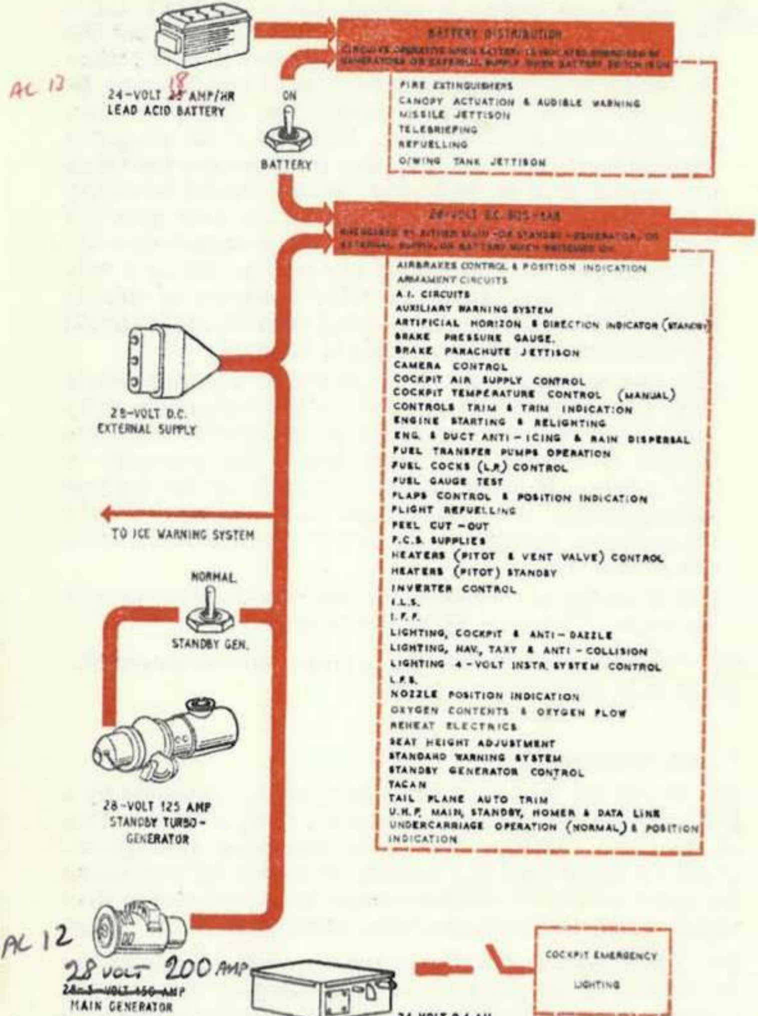
AL9 (b) ~~(c)~~ Operation of both the inertia crash switches automatically isolates the generator field.

4 Alternator supply

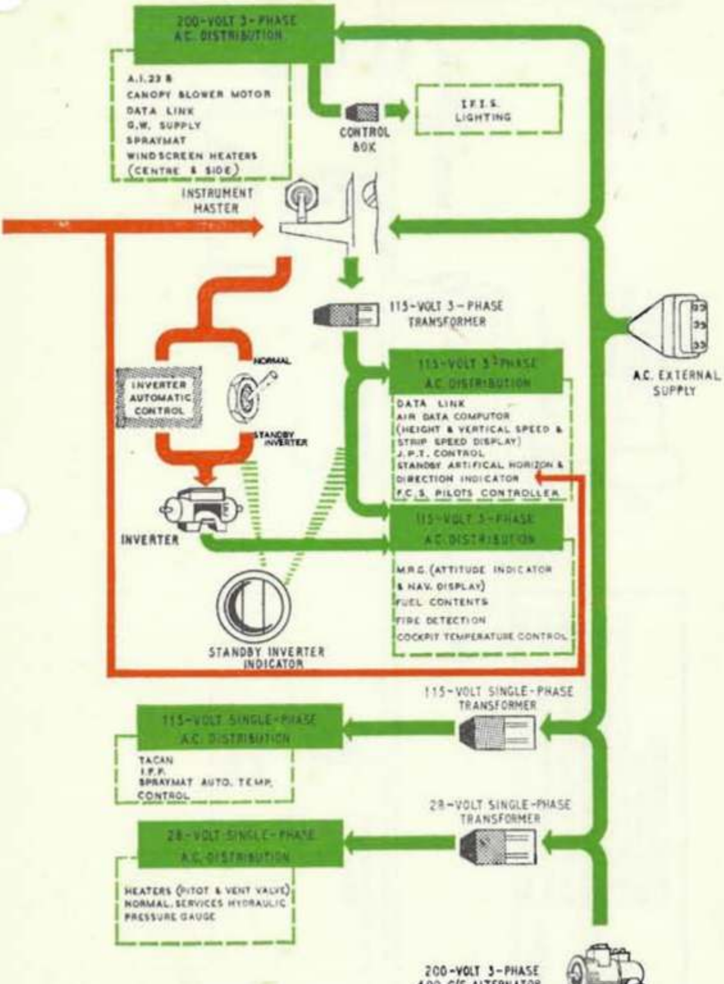
(*a*) Power for the AC operated equipment is provided by a 200 volt, 400 CPS, 3 phase alternator having a rated output of 20 KVA. Voltage control is automatic throughout. Output is maintained at a constant frequency by controlling the speed of the air turbine motor by a governor system which adjusts the position of the nozzle vanes of the motor.

(*b*) The supply is distributed either at 200 volts from the AC busbar, or modified via voltage changing transformers to 115 volts single and 3 phase and 28 volt single phase to operate the various groups of instruments and equipment.

(*c*) (i) An external AC supply may be connected to provide AC prior to the alternator coming on line after engine start.



ELECTRICAL POWER DISTRIBUTION



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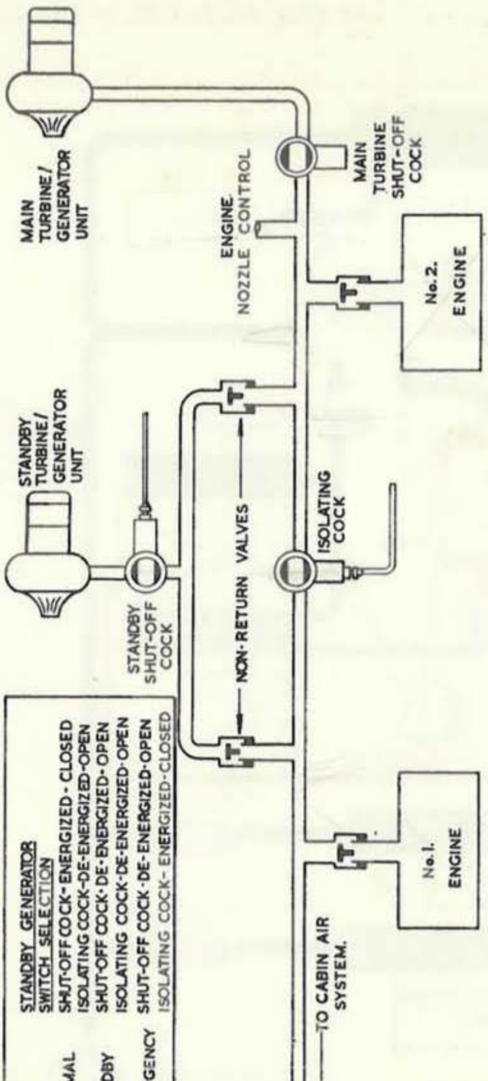


STANDBY GENERATOR SYSTEM

(ii) When an engine is started and RPM increased to air turbine control speed, the alternator main contactor closes to connect alternator output to the AC busbar provided phase sequencing is correct. At the same time, the external AC supply, if used, is automatically isolated. The alternator should remain on line provided at least one engine is maintained at or above 58% RPM.

(*d*) The AC equipment is protected against overvoltage, undervoltage, earth faults and incorrect phase sequencing by:

(i) An overvoltage relay which de-energises the alternator field. This relay can only be reset by shutting down the air turbine.

(ii) An undervoltage phase sequence unit which breaks the main contactor operating circuit. In this case, AC is restored if conditions return to normal.

(*e*) A facility is provided for resetting the alternator in the event of a transient failure occurring.

(*f*) Operation of both inertia crash switches automatically isolates the alternator field.

5 Standby Generator Supply

(*a*) Switching the standby generator on de-energises the main generator contactor and disconnects the main generator supply from the busbar. Standby supply comes on line when standby generator voltage is slightly in excess of busbar voltage.

(*b*) Maximum output of the 28 volt, 3·5 kW generator is limited to 2·8 kW (100 ampere) by the restricted size of the turbine wheel.

6 Instrument Supply

(*a*) For supply purposes, the aircraft instruments are divided into two groups: one group has an alternative supply from a Type 100A inverter, the other has no alternative AC supply (see para 6 (*f*)).

(*b*) A 115 volt, 3-phase transformer, fed from the aircraft AC supply normally supplies both groups of instruments and its output is monitored by an undervoltage/phase sequence unit.

(*c*) An INSTRUMENT MASTER switch controls switching of the instruments.

(*d*) If a transformer output fault is sensed by the undervoltage/phase sequence unit, the services listed in para 6 (*f*) (ii) are supplied by the Type 100A inverter which is started automatically. ▶◀ The services listed in para 6 (*f*) (i) are isolated from the faulty supply. ▶◀

(*e*) If required, the Type 100A inverter may be switched on manually. With STANDBY INVERTER selected, those instruments which have a standby AC supply (para ◀6 (*f*) (ii)), are fed by the inverter. The remaining services are isolated. ▶

(*f*) (i) The following AC operated instruments and services, under the control of the INSTRUMENT MASTER switch, are supplied from normal sources and have no standby supply:

Automatic control system

Air data computer

JPT control

Standby artificial horizon and direction indicator (automatic changeover to DC if AC supply fails).

(ii) The following AC instruments and services, under the control of the INSTRUMENT MASTER switch, are supplied from normal sources and are also supplied from the Type 100A inverter in standby conditions:

MRG and attitude indicator

Navigation display

Fire detection system

Cockpit temperature control (auto)

Fuel contents gauges

CONTROLS AND INDICATORS

7 Alternator Controls

A resetting facility is provided in the alternator circuit to enable the alternator to be brought back on line following a transient failure. Control is by an AC reset button adjacent to the voltmeter.

8 Generator Controls

(*a*) A DC reset button is positioned adjacent to the voltmeter for use in resetting the generator following a transient failure.

(*b*) A 3-position, NORMAL/STBY/LIFT FOR EMERG (guarded), GENERATOR switch is on the right console. The switch controls the solenoids of the shut-off cock and isolating cock in the bleed air system as follows (refer to Standby Generator System diagram):

NORMAL	—	Shut-off cock closed — Isolating cock open
STBY	—	Shut-off cock open — Isolating cock open
LIFT FOR EMERG	—	Shut-off cock open — Isolating cock closed —

(*c*) When NORMAL is selected, 28 volt DC is provided by the main generator. At STBY or LIFT FOR EMERG, 28 volt DC is provided by the standby generator and the main generator is automatically isolated.

Note: The LIFT FOR EMERG position should not be selected if the No 2 engine is not running, particularly at low altitude. Selection would result in loss of air supply to the engine nozzle control. The nozzle of the No 1 engine would open to the maximum position and its thrust would be reduced by 40% approximately at full throttle.

9 Battery Controls

(*a*) The battery is connected to the DC busbar when the BATTERY switch on the starter panel is set to on (up).

(*b*) With the BATTERY switch off, the battery is isolated from the DC busbar but the following services, permanently connected to the battery, remain operative:

Overwing fuel tank jettison (inoperative)
Fire extinguisher system
Canopy operation and audible warning
Missile jettisoning
Telebriefing
Refuelling (but see Part 1, Chapter 2, para 9 (*c*))
Inertia crash switches

10 Instrument Supply Controls

(*a*) The instruments are controlled by an INST MASTER switch on the starter panel.

(*b*) The 100A inverter, for use in the event of failure of the normal AC supplies, can be switched on manually by a NORMAL/STANDBY INVERTER switch on the left coaming panel. With the switch at NORMAL, the AC operated instruments are supplied from the AC busbar. When the switch is set to STANDBY INVERTER, the services listed at para 6 (*f*) (ii) are operated by a supply from the 100A inverter. The services listed at para 6 (*f*) (i) are isolated. The switch should only be used for testing or if the instruments are not operating correctly on normal supply.

(*c*) A magnetic indicator on the left coaming panel, shows black when the AC instruments are being supplied normally and white/ON when the changeover relay has operated to start up the 100A inverter.

11 Voltmeter

A voltmeter, fitted on the cockpit right shroud, continually shows the DC busbar voltage. The indicator is divided into a red sector for voltages of 15 to 22 volts, an amber sector for voltages between 22 and 25, a white sector for voltages between 25 and 29 and a further red sector for voltages above 30 volts.

12 Supply Failure Warnings

Warnings on the SWP and AWP appertaining to electrical failures are:

SWP	*AWP*	*Effect of Indication*
GEN	GEN	Both generators off-line
—	GEN	Main generator off-line Standby generator on-line
—	AC	Alternator off-line
—	AC and TURB	Alternator main contactor tripped due to under-speeding of the turbine or operation of nozzle stall switch
—	AC, TURB and GEN	Alternator main contactor tripped due to under-speeding of turbine or operation of nozzle stall switch. Generator output reduced below 25·5 volts
GEN	GEN, AC, TURB	Failure of main turbine (Standby generator also off-line or not selected)

Management of the System

13 Pre-flight and starting

(*a*) Normally, both external AC and DC supplies should be connected. In the operational role an AC supply is used to provide a 5 minute warm-up period for the AI equipment before take-off. Up to four minutes of this period can be supplied by the external AC, the remainder by the alternator after starting. If the alternator comes off-line after starting, however, the whole of the 5 minute period must be recommenced before the equipment becomes operational. Therefore, do not reduce both engine RPM simultaneously to below 58% (65% if duct lip anti-icing is in use).

(*b*) With no external AC supply connected, the 100A inverter will start up as soon as the DC busbar is energised if the INST MASTER switch is ON (but see Part 1 Chap 8 para 2(*b*)(ii)).

(*c*) If an external DC supply is not connected, the battery must support all the aircraft loads including engine starting and the 100A inverter load until ▶◀ the GEN warning on the AWP is extinguished, when the main generator takes over the full load.

(*d*) Up to 70% RPM may be required to extinguish the TURB and AC warnings. Thereafter maintain 58% RPM or above.

(*e*) When the engines have been started, check the functioning of the standby generator system as follows:—

(i) Run the No 1 engine at 65% RPM with No 2 engine at 50% RPM. Check that both GEN warnings are out.

(ii) Select STBY on the GENERATOR switch and check that the GEN warning appears on the AWP, indicating that the main generator has gone off-line; momentarily, the GEN warning should appear on the SWP, the voltmeter should register a slight fall and the attention light and audio warning should come on prior to the standby generator coming on-line. If the GEN warning does not appear momentarily on the SWP, it signifies that the standby generator has been running prior to selection. When the GEN warning on the SWP is out, check that the voltmeter reads 28 volts.

(iii) Reselect the GENERATOR switch to NORMAL and check that the GEN warnings are out. Check that the voltmeter reads 28 volts.

(iv) Increase No 2 engine RPM to 58% and then return No 1 throttle to idle.

(*f*) Maintain No 2 engine at 58% RPM or above to maintain AC supplies.

14 In flight

(*a*) (i) The air turbine maintains its governed speed at 58% RPM or above. This RPM corresponds to idling RPM at approximately 15 000 ft/250 kts. At idling RPM below this altitude the air turbine speed reduces, the alternator comes off-line and, the GEN warning appears on the AWP.

(ii) To prevent oil starvation of the air turbine gear box, negative G flight must not be sustained for longer than 15 seconds.

(iii) Check the voltmeter periodically; voltage should be indicated in the white sector.

(*b*) Due to limited cooling, the standby generator is to be used only if the main generator fails.

15 After landing and shut-down

(*a*) Maintain the No 2 engine at the fast idling position when taxying to keep AC on line.

(*b*) When shutting down switch off all electrical services and then put the BATTERY switch to off.

Malfunctioning

16 AC supply failure

(*a*) AC supply failure is indicated by the AC warning on the AWP. If this warning appears alone, ▶◀ an attempt to reset the system should be made by pressing the AC reset button; more than one attempt to reset may be made. If power cannot be restored, a return to base should be made with the loss of AC-operated services except those provided for by the standby inverter.

(*b*) If intermittent AC supply failure occurs or the AC reset button is pressed to restore AC power, an output surge from the JPT controllers may occur at JPT in excess of approximately 550°C. This could result in a reduction of fuel flow to the engines with the consequent transient loss of thrust. Therefore, if AC supply becomes intermittent or if repeated operation of the AC reset button is necessary, it is recommended that the JPT control switches are set to OFF.

17 Generator Failure

(*a*) Failure of the main generator is indicated by the GEN warning on the SWP and the GEN warning on the AWP: Attempt to reset the generator by pressing the DC reset button. If power cannot be restored, switch the standby generator into service. When the standby generator comes on-line, the GEN warning on the SWP goes out but the GEN warning on the AWP remains. DC services will be unaffected provided the standby generator is functioning correctly.

(*b*) (i) If neither generator is on-line, both GEN warnings will remain. In this event all DC loads will be borne by the main battery. The life of the battery depends on its capacity, charge and rate of discharge. Complete failure of all electrically-operated services, except those supplied by the emergency battery, will occur once the battery has discharged. It should be noted that AC services, which are controlled by DC relays, will also fail when the battery has discharged.

(ii) Reliance may be placed on the battery to supply DC power to all relevant services provided the reading on the voltmeter is above 22 volts, although some services will be affected, eg slow trimmer response, when voltage comes below 23 volts. The battery will rapidly discharge when voltage falls below 22 volts and action should therefore be taken to place the aircraft in a position for landing before this occurs.

(iii) Emergency drills in the event of double generator failure are given in the Flight Reference Cards. It should be noted that the heaviest electrical load is the DC fuel pumps and these should be switched off immediately. They should not be switched on again since the starting load is likely to discharge the battery rapidly.

(*c*) With the DC pumps switched off, the fuel collector boxes depend solely on gravity feed and consequently there is a risk of fuel starvation in flight conditions which prevent gravity flow. Generally, maintenance of a nose-up attitude ensures gravity flow and where possible, this practice should be adopted. This may, however, run counter to the need to descend quickly due to weather conditions and battery state and where this is the overriding factor, the risk of fuel starvation during a rapid descent may have to be taken. If possible, the rapid descent should be interrupted by adopting a nose-up attitude for at least 15 seconds to refill the collector boxes. With the DC pumps off, about 100 lb/side gauged fuel is unusable.

18 Generator Overvolting

(*a*) If a voltmeter reading of 30 volts or more is sustained, select STBY on the standby generator switch; the GEN light appears on the AWP.

(*b*) (i) If, after taking the above action, the standby generator also overvolts at a later stage in the flight, switch the standby generator switch back to NORMAL and select the BATTERY switch off. Subsequently, should the main generator overvolt and come off line, the shut off cock will automatically open and allow the standby generator to run up. As the voltage from the standby generator reaches 28 volts, the shut off cock will electrically energise shut. This will close down the standby generator again, causing all electrics to come off line and allowing the sequence to start again. This cycle will repeat itself every 2 seconds giving the symptoms of electrical oscillations or power surges and failures. To recover DC control in this situation, *select* the standby generator. Put the BATTERY switch on again when in the circuit.

(ii) Electrical services will not be affected immediately. The actions will prevent overcharging of the battery which may otherwise be damaged by overheating.

Note: With brushless type generators fitted, any overvoltage above 30 volts approximately should cause the generator to come off-line. However, if the overvoltage protection system fails, the indications and the actions to be taken are as given above.

19 Main Air Turbine Malfunction

(*a*) *General*

Failure of the main air turbine is indicated by the GEN warning appearing on the SWP and the GEN, AC and TURB warnings on the AWP. Selecting STBY on the standby generator switch restore DC but supplies to the AC services are not available except for those provided via the standby inverter.

(*b*) *Underspeeding*

(i) If underspeeding occurs, the AC and TURB warnings appear on the AWP. Additionally, if turbine speed falls sufficiently, the GEN warning also appears on the AWP, brought on by the generator undervoltage relay; if this condition is prolonged the battery may discharge unless the standby generator is brought into use.

(ii) A probable cause of underspeeding is engines idling at low altitude. Increasing engine speed should remove the warnings and restore conditions to normal.

(*c*) *Overspeeding*

If the fault is due to the turbine wheel speed increasing above a datum RPM, an overspeed switch operates to close the main turbine shut-off cock in the air duct to the turbine, thus stopping the turbine. As the cock can only be reset on the ground, the turbine is inoperative for the remainder of the flight. Indications of this condition and the action to be taken are similar to those of para 19 (*a*).

(*d*) *Burst Air Duct*

(i) A severe burst in the air duct between No 1 engine and the main air turbine is indicated by the GEN warning on the SWP, the GEN, TURB and AC warnings on the AWP and additionally, by the CPR warning on the SWP and/or loss of jet pipe nozzle control. Furthermore, hot air leaking from the burst air duct may trigger the fire warning system. Nevertheless, the failure may also manifest itself simply as a loss of thrust without the attendant warnings.

(ii) If the jet pipe nozzles are at the fully open position and loss of thrust has occurred, ensure that both engines are running and then select EMERG on the GENERATOR switch. The isolating cock in the air duct between No 1 and No 2 engines closes and the standby generator comes on line to restore DC electrical supplies.

(iii) The position of the burst in the air duct can now be identified. If the burst has occurred on the No 1 engine side of the isolating cock, the CPR warning and the GEN warning on the AWP remain and action must be taken for cockpit pressurisation failure. Nozzle control will be restored and, if approximately 65% RPM or above is maintained on No 2 engine, AC and DC power is available; subsequently, No 2 engine should not be shutdown otherwise nozzle control will be lost on No 1 engine. If the burst is on the No 2 engine side of the isolating cock, however, loss of nozzle control, with the possible serious reduction in thrust, remains and all AC services will be lost except for those provided by the standby inverter. Experience suggests that the nozzles can be expected to remain in the position they were at the time of failure of the bleed air supply. However there is a remote possibility that they may blow open under exhaust gas pressure if control is not restored.

(iv) Emergency action to be taken in the event of air turbine malfunction is given in the Flight Reference Cards.

20 Instrument Supply Changeover in Flight

(*a*) If intermittent instrument supply changeover occurs, indicated by fluctuations of the STANDBY INVERTER magnetic indicator, set the NORMAL/STANDBY INVERTER switch to STANDBY.

(*b*) Whenever the STANDBY INVERTER indicator changes to white/ON whilst autostabilisation or any mode of the flight control system is engaged, the flight control system should be switched off in the order, MASTER switch, STAB switch and the stick switch.

PART I

Chapter 2—FUEL SYSTEM

Contents

Description

1 Fuel tanks

(a) Internal fuel tanks

Internal fuel is carried in integral tanks in the wings and flaps. Each wing tank system comprises a leading edge tank and a two-compartment main tank. The outboard compartment of each main tank serves as the collector box from which during normal operation, all LP fuel is delivered to the engine HP pumps and reheat pumps. The flap tank fuel is transferred automatically by air pressure into the wing tank system. All tanks are vented outward and inward, but outward venting is prevented during negative G flight.

(*b*) *Ventral tank*

Extra fuel may be carried in a jettisonable ventral tank. Fuel from this tank is transferred automatically by air pressure into the wing tank system. Outward and inward venting is provided but outward venting is prevented during negative G flight.

◀2 Fuel feed

(*a*) *General*

Fuel is initially taken from the wing tank system. After 120–160 lb. of fuel has been delivered from the wing tanks, float switches and refuel/transfer valves control the sequence of fuel feed so that fuel is normally used in the order, ventral tank, flap tanks and wing tanks.

(*b*) *Wing tanks feed*

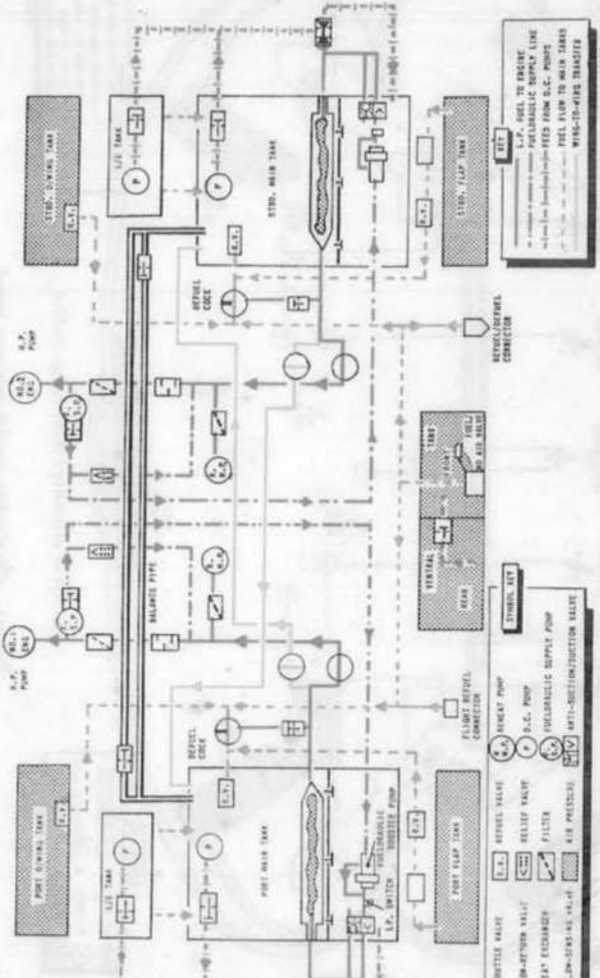
Fuel from the main compartment of the wing tanks is normally fed by gravity into the collector box (outboard compartment). Additionally, DC transfer pumps feed fuel from the leading-edge tank and main compartment of the wing tank to maintain the collector box full. The collector box fuel is delivered to the engine HP pump and reheat pump by a fueldraulic booster pump, via a recuperator and an LP cock.

(*c*) *Ventral tank feed*

The ventral tank feeds automatically into the wing tank system by air pressure when the wing tank float switches operate to open a Mk. 44 refuel valve in the wing tanks, i.e. when 120–160 lb. of wing tank fuel has been used. When the ventral tank empties, a Mk. 7 valve in the tank closes, which operates electrically a Mk. 27 valve to permit transfer of fuel to commence from the flap tanks.

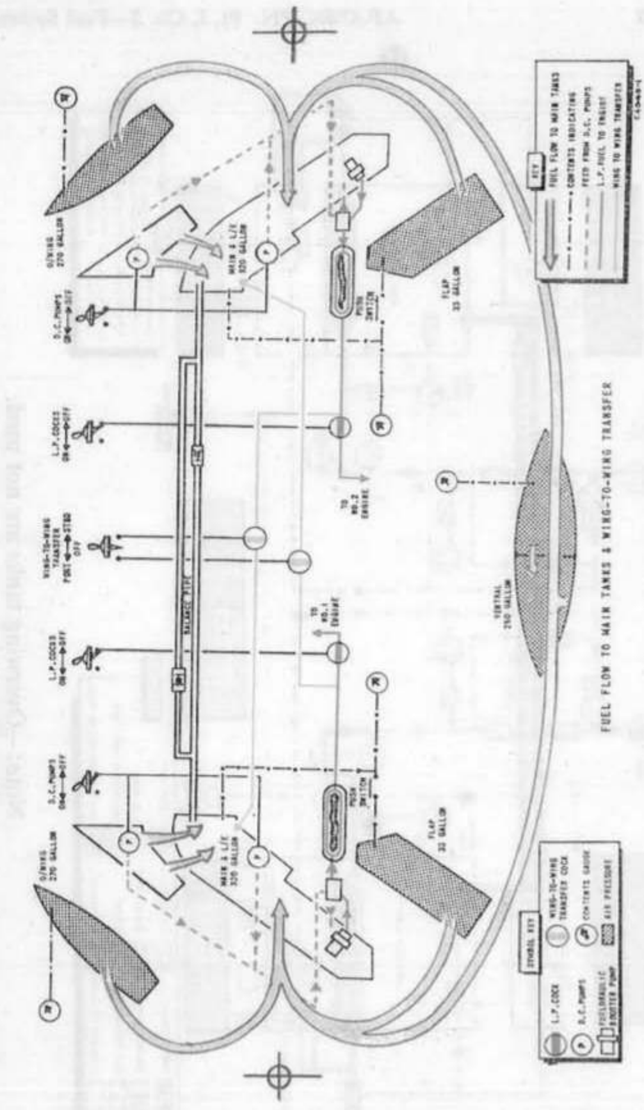
(*d*) *Flap tanks feed*

The flap tanks automatically feed into the wing tank system by air pressure when the Mk. 7 valve in the ventral tank closes, i.e., ventral tank empty, provided that the Mk. 44 refuel valve is open. Flap transfer rate is slow and▶



Note:—Overwing tanks are not used.

FUEL SYSTEM



Note:—Overwing tanks are not used

SIMPLIFIED FUEL SYSTEM

it should be noted that in the flaps down configuration, any fuel remaining in the flaps, up to 200 lb per tank, may be trapped and thus be unusable.

3 Fueldraulic System

(*a*) A fueldraulic supply pump (FSP), mounted on the engine wheelcase, pumps fuel under pressure to drive a fueldraulic booster pump in the collector box. The FSP is primed by fuel tapped from the LP line to the engine, which is supplied initially either by the DC pumps (when the engine is at rest) or if the DC pumps are inoperative, by LP fuel drawn under gravity from the collector box by the HP fuel pump. In the latter case it is necessary for the engine to be rotating and for the shuttle valve to be in the position which opens the collector tank line to the HP pump. Exhaust fuel from the booster pump motor is directed into the output line from the booster pump.

(*b*) The fueldraulic pumps are double-ended and are fitted in such a manner that one end is always immersed. In negative g conditions, fuel pressure will not fall below a safe value. However, for air turbine lubrication reasons, negative g flight must not exceed 15 seconds.

(*c*) A pressure switch, downstream of the booster pump, operates if the difference between tank pressure and pump delivery pressure falls to $5{\cdot}75 \pm 0{\cdot}75$ PSI. Operation of the switch lights the fuel warning on the AWP and trips reheat, if in use.

4 DC Transfer Pumps

Two DC transfer pumps are fitted in each wing, one in the leading-edge tank and one in the main wing tank to transfer wing fuel into the collector box. If a booster pump fails or is not running, a pressure sensitive shuttle valve diverts the output of the DC transfer pumps into the engine feed line.

5 ▶◀ *Not used.*

6 Wing-to-Wing Transfer System

(*a*) Provision is made for cross-transferring fuel from one wing to the other. Each wing tank is connected to the

LP line in the opposite wing and an isolation cock is in each cross-transfer line. When the cock is selected open, the fuel pressure from the LP line on one side causes fuel to transfer to the wing tank on the other side. The operation is completely independent of the LP cocks in the engine feed lines.

(*b*) A balance pipe between the wing tanks prevents overfilling and venting during cross-transfer. Until mod 2383 is embodied, however, care should be taken not to overfill, otherwise fuel venting may occur.

7 Fuel Tank Capacities

The usable fuel capacities are:

Tank	*Capacity Gallons*	*Capacity — Pounds*	
		AVTUR 8·0 *lb/gall*	*AVTAG* 7·7 *lb/gall*
Main and leading edge	2 × 320	2 × 2560	2 × 2464
Flap	2 × 33	2 × 264	2 × 254
▶◀ Ventral	1 × 250	1 × 2000	1 × 1925
Total fuel	966*	7728	7439

* ▶◀ Approximately 7 gallons/side extra gauged fuel available when Mod 2459 is embodied.

8 Air-to-Air Refuelling System

The aircraft may be fitted with a detachable air-to-air refuelling probe positioned beneath the port mainplane. A refuelling pipe from the probe leads to the fuel system via a non-return valve. Refuelling is controlled by a switch in the cockpit which, when set to FL REFUEL, opens a valve to release air pressure from the ventral and flap tanks and all refuelling valves are opened to refuelling flow until their respective tanks are full. Each refuelling valve closes when its 'tank full' float switch operates and an appropriate light on a panel in the cockpit goes out.

Note: Account should be taken of the possible loss of use of the ventral and flap tank fuel on completion of air-to-air refuelling, due to the possible malfunction of the air pressure release valve.

9 Ground Refuelling System

(*a*) A pressure refuelling point, under a panel on the port side of the fuselage, permits simultaneous refuelling of all tanks including the ventral tank. The act of opening the panel operates a micro-switch which changes the fuel electrical circuit from its flight condition to the ground refuelling condition. The micro-switch, together with fuel level switches in the tanks, controls the operation of the refuelling valves and the tank indicator lights.

(*b*) Above the refuelling panel are seven tank-full indicator lights which illuminate when the panel is removed and go out individually when the associated tank-full switch operates.

(*c*) All tanks should be depressurised prior to refuelling. This is done by setting the battery switch on and selecting FL REFUEL on the air-to-air refuelling switch for a period of at least 30 seconds. This switch must be returned to off when depressurisation is complete.

Controls and Indicators

10 LP cocks

Two switches, FUEL COCK No 1 (port) and FUEL COCK No 2 (starboard), on the starter panel control the position of the LP cocks. With the switch up the cock is open. The switches may be set to on by a ganging lever or individually operated as desired. When Mod FC089 is embodied the switches lock in the on position to prevent inadvertent selection to 'off'; the switch toggles must be pulled to clear the lock before 'off' selection can be made.

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11 DC pumps

Two switches, DC PUMPS PORT and DC PUMPS STBD, are on the starter panel; each controls the two DC transfer pumps in the associated wing tank system. With the switches at up the pumps are running provided a DC electrical supply is available. The switches may be set to on by the ganging lever or individually operated as desired.

12 Wing-to-wing transfer

(*a*) A three-position P/off/S — TRANSFER switch and associated indicator light on the starboard instrument panel controls the transfer of fuel from one wing tank to the other. When P (port) is selected, the starboard isolation cock is opened and, provided pressurised fuel is available in the starboard LP line, fuel will transfer to the port wing. A selection to S (starboard) will similarly transfer fuel to the starboard wing. The indicator light only shows that transfer has been selected. Monitoring of the fuel gauges is the only way of ascertaining that a transfer is taking place.

(*b*) When the P or S selection is made, the adjacent indicator light illuminates. The rate of fuel transfer is about 210 lb/minute at 100% RPM, 160 lb/minute at 85% RPM, if the fueldraulic pump is supplying the transfer pressure. Transfer rate by the DC pumps alone with the engine shut down is about 75 lb/minute. Because the transfer rate can be low, cross-transferring should be started as soon as possible after engine failure. When the switch is returned to the off position, the light goes out momentarily, illuminates again for 2 or 3 seconds, then goes out.

(*c*) If the DC pumps *only* are transferring fuel, with the aircraft in straight and level flight, there may be up to 400 lb of fuel which cannot be transferred.

13 Fuel contents gauges

(*a*) Three capacitor contents gauges are on the starboard instrument panel. The two MAIN tank gauges indicate the fuel contents of the respective wing and leading-edge tanks; indication of flap tank contents is given on the same gauges as an increase in the contents reading but only during transfer of the flap tank or when a FUEL CONTS T/E button on the starboard console is pressed. A VENTRAL gauge registers ventral tank contents.

(*b*) A FULL/EMPTY — GAUGE TEST switch, spring-loaded to centre-off, on the starboard console may be used to check the serviceability of the contents gauges. When FULL or EMPTY is selected, all gauges should register full or empty respectively.

(*c*) The gauges require 115 volt AC, controlled by the INST MASTER switch, for their operation. If the normal AC supply fails the gauges will continue to operate from the standby inverter supply.

14 Fuel pressure warnings

(*a*) FUEL 1 and FUEL 2 warnings are on the AWP. The FUEL 1 warning comes on when the output of the fueldraulic booster pump in the port main tank falls to $5\frac{3}{4} \pm \frac{3}{4}$ PSI above tank pressure, due to pump failure or lack of fuel in the collector tank. The FUEL 2 warning is similarly energised if the starboard fueldraulic pump output falls. A FUEL 1 or FUEL 2 warning automatically cancels the corresponding reheat.

(*b*) PUMPS P and PUMPS S warnings are on the AWP. Each warning is energised via an associated pressure switch, positioned in the line between the DC transfer pumps and the shuttle valve in each wing. Thus, a PUMPS P or PUMPS S warning can occur when:

(i) Both DC pumps fail or are uncovered.

(ii) One pump becomes uncovered when the other has failed.

into malfunctions

(*c*) Since the collector tank is at the rear of the wing, the failure of a front DC pump is of greater significance than the failure of a rear pump. In flight conditions where the front pump is uncovered and the rear pump immersed but inoperative (ie nose-up or accelerating in level flight) the associated PUMPS warning light will appear but the fuel will have flowed into the collector tank of its own accord. However when the opposite conditions prevail (usually in the descent) fuel collects in the front of the wing; should the front DC pump be inoperative the fuel cannot be transferred either to the collector box or to the engine. The significance of a PUMPS warning therefore, has to be interpreted in relation to aircraft attitude and the prevailing flight conditions.

15 Vent valve heating

All vent valves are heated by 28 volt AC supplies; control is by a PITOT HEATER — NORMAL/OFF/STANDBY switch on the starter panel. Vent valve heating is not available at the OFF or STANDBY positions.

16 Ventral tank jettison

A black and yellow striped handle at the bottom of the port instrument panel, when pulled, jettisons the ventral tank. The handle also incorporates a guarded pushbutton for jettisoning the guided missiles. Both facilities may be operated independently or together.

NOTE: When the ventral tank is jettisoned, the tank contents gauge 'freezes' at the value indicated at the time of jettison.

17 Air-to-air refuelling controls

(*a*) A FL REFUEL/NORMAL switch below the port shroud, when selected to FL REFUEL, controls the air-to-air refuelling operation as described in para 8. Note that the ventral, and flap tank fuel will not feed if the switch is left at FL REFUEL during normal flying.

(*b*) An air-to-air refuelling panel is positioned to the left of the light fighter sight. On the panel are seven windows which light green when FL REFUEL is selected and go out when the associated fuel tank is full or if NORMAL is selected at the air-to-air refuelling switch. The panel incorporates a light dimmer control. The windows are labelled, left to right, WG PRT, FLP PRT, OW PRT, VEN, OW STB, FLP STB and WG STB; the OW PRT and OW STB windows are to be disregarded.

(*c*) A PROBE LIGHT — ON/off switch on the port coaming panel controls the probe lighting. Note that there is no supply to the probe lighting unless the NAV LTS switch is set to ALL ON or TIPS.

18 Overwing tank jettison and ventral emergency transfer switches

An OVERWING TANK — JETN/OFF switch and a VENTRAL TANK EMERGENCY TRANSFER switch are on the port console. These switches are associated with overwing tank operation and are therefore inoperative.

Normal use of the System

19 Starting the engines

(*a*) Check the fuel contents gauges by selecting EMPTY at the GAUGE TEST switch and noting that all contents gauges register zero.

(*b*) Start the engines with both FUEL COCK switches and both DC PUMPS switches on (up). Check that the FUEL warnings go out by 38% RPM.

20 In flight

(*a*) Out-of-balance of main tank contents can be corrected by selective switching of the wing-to-wing TRANSFER switch.

(*b*) (i) As soon as an engine is shut down, fuel should be transferred to the side feeding the live engine in order to keep the fuel distribution as required. Note that the FUEL PUMPS switch of the shut-down engine must be left ON to facilitate fuel transfer.

(ii) With the aircraft in straight and level flight both DC pumps are uncovered when the fuel level in the tanks associated with the stopped engine have fallen to approximately 400 lb gauged fuel. This fuel can be transferred only if the LP cock is open and the booster pump is being driven by the windmilling engine.

(*c*) No attempt should be made to transfer fuel to or from a side which is showing either a PUMPS or a FUEL warning, until diagnosis of the problem is complete.

21 Flight with low fuel contents

When landing with indicated fuel below 400 lb per side, care should be taken to avoid side-slip as this can result in an increase in unusable fuel.

22 After flight

If the wing-to-wing transfer system has not been used in flight, it should be checked as follows:

(*a*) After stopping No 1 engine, note the fuel contents and select the transfer switch to P.

(*b*) When the fuel gauges confirm that fuel has transferred to the port tanks, select the switch off for 5 seconds and then select S.

(*c*) After stopping No 2 engine, check that fuel has transferred to the starboard tanks and select the transfer switch off.

Malfunctioning of the System

23 Fuel pressure warning

(*a*) *FUEL* 1 *or FUEL* 2 *warning*

(i) The warning can result from failure of the fueldraulic booster pump or lack of fuel in the collector box. In the case of booster pump failure, the DC transfer pumps are capable of supporting engine operation up to 40000 ft, 0·9M, and 85% RPM. If it is necessary to transfer fuel from the side having only DC pumps, the corresponding engine should be set to idling before transfer begins.

(ii) The collector box remains topped-up under gravity as long as the aircraft retains a nose-up attitude. In nose-down attitudes, the collector box is emptied by the fueldraulic booster pump at a rate corresponding to engine demand, unless output from the DC pumps (particularly, in this attitude, the front pump) is sufficient to replenish it.

See 14(b)

◀(*b*) *PUMPS P or PUMPS S warning*

A steady PUMPS P or PUMPS S warning signifies a pressure drop in the combined DC pumps output line. While this could indicate the simultaneous failure of both pumps, it is much more likely to indicate the uncovering of one pump due to fuel movement in the tanks, following failure of the other pump. This necessitates further investigation:

Flight condition	*Probable situation if light on*
1 Level acceleration High nose attitude without acceleration	Front pump uncovered, rear pump failed
2 Nose-down attitude (no acceleration) Rapid deceleration	Front pump failed, rear pump uncovered

(i) Condition 1 requires no special action; re-adopting a nose-down attitude re-immerses the serviceable pump to provide a continuous, if reduced, supply of fuel to the collector box.

(ii) Condition 2, if maintained, leads to the emptying of the collector box; the immediate action is to throttle the affected engine to idle and adopt a 7° nose-up attitude for at least 15 seconds to allow the collector box to refill under gravity. This action should extinguish all warnings but must be repeated at intervals during recovery if the risk of flame extinction on that engine is to be avoided. If the warning persists, carry out the subsequent actions detailed in the Flight Reference Cards.

(*c*) The output of a single DC pump is limited and is incapable of sustaining simultaneous collector box replenishment and fuel transfer. Therefore should a warning appear while fuel transfer is in progress, transfer should be stopped (fuel transferred into a malfunctioning side may not be recoverable).

(*d*) An engine which is being sustained by a single DC pump following a FUEL warning should be run at idling speed only for the services it provides.

(*e*) It is possible for a single DC pump to provide enough pressure to hold the shuttle valve over, thus preventing the associated HP pump from drawing fuel from the collector ▶

tank under gravity. In this event, if the collector box has previously emptied and the fueldraulic system has not been re-primed by the DC pump output, the engine will flame out if the supply to the DC pump is interrupted (eg by a change of aircraft attitude or velocity). This situation can be resolved by deliberately shutting down and then relighting the engine.

24 Wing-to-Wing Transfer Cock Failure

(*a*) If a cock fails in the open position, constant cross-transfer will occur. The asymmetry can be stopped by selecting cross-transfer in the opposite direction, and the fuel balanced by differential throttle.

(*b*) If a cock fails in the closed position, cross-transfer from that side cannot be achieved. Fuel asymmetry may be corrected by differential throttle.

(*c*) During any malfunction of the wing-to-wing transfer system, the transfer indicator light should be ignored. Fuel gauge readings are the only reliable method of ascertaining whether or not transfer is taking place.

25 Fuel Contents Gauge Failure

If the electrical power supply to a gauge fails the pointer freezes at the reading indicated at the moment of failure. Confirmation of power supply failure can be obtained by selecting the spring-loaded GAUGE TEST switch on the starboard console to FULL or EMPTY. The pointers of serviceable gauges move in sympathy with the selection: the pointer of an unserviceable gauge will not move. Other types of failure, such as an internal short circuit in the tank sensor unit, may result in the gauge reading incorrectly or falling to zero or below, even though it operates correctly on test.

26 Ventral and Flap Tanks Flow Failure

Failure of the ventral and flap tank fuel to transfer will result in a progressive aft movement of the CG as fuel is used from the wing tank systems. As soon as the failure is recognised, handling should be restricted to gentle manoeuvres. If possible, the aircraft should be landed before the CG moves beyond the aft limit.

27 Vent Valve Failure

An interconnecting pipe between the left and right wing tanks ensures that the air pressure in both tanks remains equal even if a vent valve fails in one of the wing tanks. Thus, fuel asymmetry should not occur in the event of a vent valve failure.

PART 1

CHAPTER 3 — ENGINE CONTROLS AND INDICATORS

Contents

DESCRIPTION

1 Avon ▶◀ Mk 302 Engines

(*a*) Each engine is a 16-stage axial-flow gas turbine. The ◀Avon Mk 302 develops approximately 12,600 lb static thrust at sea level without reheat, and approximately 16,300 lb with full reheat. The engines are mounted in▶ the fuselage, the lower engine being designated No 1 and the upper engine No 2.

(*b*) The main engine systems include:

A high pressure fuel system

A reheat system

A liquid fuel starting system

Relighting facilities

A self-contained oil system

JPT control

Engine anti-icing system.

▸◂

(*c*) Each reheat/engine installation comprises the engine itself, an intermediate jet pipe and a reheat jet pipe assembly.

2 Airflow Control System

The airflow through the compressor is controlled by varying the engine speed: the compressor stages are matched to give the highest efficiency in the normal operating range. To maintain stable airflow, hydraulically controlled variable-angle intake guide vanes and an air bleed valve are fitted; the system is fully automatic.

3 Engine Fuel System

(*a*) *Fuel Feed System*

Fuel from the main fuel system is delivered to the HP fuel pump through a filter and the pump output is fed to the burners through a fuel-cooled oil cooler and a proportional flow control unit; the latter unit incorporates the combined throttle valve and HP cock which is controlled by the pilot's throttle lever.

(*b*) *HP Fuel Pump*

The HP fuel pump consists of two variable stroke pumps contained within a common housing. Either pump of the dual unit is capable of delivering sufficient fuel for approximately 100% RPM to be obtained. The output of the pump is controlled by a servo system in response to signals from the engine speed governor and the flow control unit.

(*c*) *Fuel Control System*

The fuel control system contains the following controls:

(i) *Maximum RPM Governor*

The maximum RPM of the engine is governed automatically. The governed speed, however, varies with altitude and ambient temperature and, under extreme conditions, manual control may be necessary to prevent the operating limitations being exceeded.

(ii) *Altitude Sensing Unit (ASU)*

The ASU regulates the fuel pump delivery flow in accordance with the intake air pressure to satisfy the engine fuel requirements at varying aircraft speeds and altitudes.

(iii) *Altitude Idling Valve*

This valve provides a minimum limit to the fuel flow when the throttle valve is closed to the idling position thus preventing excessively low idling speed at altitude.

(iv) *Acceleration Control*

This control automatically limits the rate of increase of the fuel flow to the burners during rapid throttle movements under all conditions.

(v) *JPT Controller*

This control automatically prevents the JPT rising above the maximum limitation. Once the maximum JPT is attained and the temperature controller is in operation, the control, by trimming the fuel flow, prevents JPT rising above the maximum permissible value. Maximum RPM, therefore, varies according to ambient air temperature, altitude and speed. Under extreme conditions manual control may be necessary to prevent the operating limitations being exceeded.

(vi) *Fuel Flow Limiter*

The maximum fuel flow is limited by means of an acceleration control unit (ACU) back stop to prevent overstressing due to high intake pressure; this reduces maximum RPM particularly at low altitude and high speed.

4 Jet Pipe Nozzle

(*a*) The aft end of the reheat jet pipe assembly incorporates a fully variable orifice nozzle, consisting of sixteen interlocking flaps. Gas stream pressure provides the nozzle opening force and the flaps are closed or held fixed by eight screw jacks, driven and synchronised by an annular gear. An air motor, powered by engine compressor air, drives the annular gear in response to mechanical inputs from an air motor control unit (AMCU). One part of the AMCU controls nozzle position in the cold power range: the remainder is associated with reheat operation.

(*b*) In cold power, the nozzle can adopt either of two positions, controlled by engine RPM. The nozzle is partly opened (cruise position) from idling up to 97% RPM ▶◀ at which setting it is closed automatically and remains closed up to maximum cold power. On decelerating, the nozzle remains closed until RPM reaches 89%. ▶◀ These are sea level figures and tend to increase with altitude.

(*c*) With reheat selected, nozzle area varies with throttle lever movement, from part open at minimum reheat to fully open at maximum reheat.

(*d*) Two nozzle position indicators are on the right instrument panel.

Reword the following:

5 Reheat System During cert stages of flight it, may be preferable to leave throttle at max R/H since nozzle cancelling may fail.

(*a*) *General*

(i) Reheat is used to augment engine thrust by injecting fuel into the jet pipe and burning the fuel in the unconsumed portion of the air passed through the engine. Combustion of the injected fuel raises the temperature and efflux velocity of the exhaust gas and increases the thrust.

(ii) Fuel supply for reheat is taken from the aircraft LP system and fed to the reheat burners by an air turbine pump, utilising air tapped from the associated engine compressor. The reheat burners form three concentric rings, the two inner rings are fed from one pipe and supply the fuel requirements at low reheat selections. For supply at higher selections, fuel is supplied to all three rings of the burner.

(iii) Reheat is ignited by a 'hot streak' ignition method in which a metered quantity of fuel is injected into one of the engine flame tubes, where it ignites. The resultant streak of flame passes through the turbine and ignites a further quantity of fuel injected into the engine exhaust unit. This streak of burning fuel passes down the intermediate jet pipe and ignites the fuel spray from the reheat burners.

(iv) Metering of reheat fuel is automatic. Movement of the pilot's throttle lever changes the nozzle area and thus the pressure ratio across the turbine. The change of pressure ratio is sensed by a reheat fuel control unit which positions a throttle valve to increase or decrease reheat fuel to restore the pressure ratio.

(v) A reheat trip system provides automatic cancellation of reheat in the event of certain malfunctions of the system. Indication of reheat trip is by TTC 1 and TTC 2 warnings on the AWP which operate if reheat fails to light or if, following satisfactory lighting, a malfunction results in automatic cancellation. The TTC warnings also operate during normal reheat lighting to indicate the progress of light-up.

◀ (vi) Reheat lighting above 35,000 feet is improved by delaying full opening of the nozzle until reheat has lit. ▶

(vii) Fuel consumption with reheat in operation, especially at medium and low altitude, is extremely high; it is more than double the consumption at maximum cold power.

(*b*) *Operation*

(i) Reheat is selected by rocking the throttles outboard at the maximum cold power position and then moving them smoothly and quickly forward to the maximum reheat position. During reheat light-up, or cancellation, there may be a transient RPM overswing (5 seconds limit) to 106% ▶◀.

(ii) ▶◀ On selection of reheat below 35,000 feet, the nozzles open fully with a momentary pause at an intermediate position. On selection above 35,000 feet, the nozzles open to the intermediate position for 15 seconds while light-up occurs, and then open to maximum area. When selection is made, the TTC warnings come on to indicate initiation and when the 'hot streak' commences, the TTC warnings go out. If reheat fails to light, the warnings re-appear.

(iii) After light-up, reheat may be reduced to the value required. Reheat is fully variable between maximum and the minimum available (approximately 18% of maximum ▶◀) and is proportional to throttle movement.

(iv) Apart from the initial selection, movement of the throttles over the reheat range should be made slowly and smoothly.

(v) Minimum reheat can be retained at engine speeds down to 96% RPM ▶◀ by holding the throttle levers outboard when throttling back: a mechanical stop limits rearward travel of the throttle levers with reheat selected. Reheat can be cancelled by rocking the throttles inboard at any position between maximum cold power and this stop.

(vi) When using maximum reheat with the correct nozzle position indication, the JPT should be stable at or near the operating limit, the RPM possibly being reduced due to the action of the top temperature control.

(*c*) *Reheat Failure*

(i) Reheat failure is indicated by the associated TTC warning failing to go out or going out and coming on again during light-up, or by TTC operation when reheat is in operation. A TTC warning can only be cancelled by moving the throttle into the cold power range; reheat cannot be relit whilst a warning remains. Monitor JPT during reheat selection, since a low JPT is the most reliable indication of reheat failure.

Note: If a TTC warning appears when reheat is not selected, a DC power supply failure to the reheat circuit is indicated. The nozzle moves automatically to the cruise setting and reheat cannot be selected.

(ii) Automatic reheat trip operation cancels reheat and closes the nozzle after any one of four conditions occurs: no single reheat fault can cancel reheat on both engines. The fault conditions are:

Failure to light

Reheat extinction

Maximum JPT exceeded by 60°C when JPT control is in operation

Fuel system failure resulting in FUEL 1 or FUEL 2 warning.

If automatic cancellation does not occur, select the throttle of the affected engine to full cold power to close the nozzle and to regain full cold thrust.

(iii) On selection of reheat, a time switch is started so that, if reheat fails to light, a time delay of between 3·5 and 7 seconds occurs (depending on the cause of failure) before automatic cancellation. If, on selection, reheat fails to light in 5 seconds, cancel the selection, wait 2 seconds and reselect. If reheat again fails to light within the recommended lighting range, further attempts are unlikely to be successful and the system should be considered unserviceable.

(iv) In fault conditions other than failure to light, the TTC warning appears 2 seconds before the reheat trip occurs. This is due to a delay device in the trip circuit, introduced to prevent reheat extinction when firing missiles. If a TTC warning comes on but normal conditions are restored within 2 seconds, reheat remains lit and the TTC warning goes out. If automatic cancellation occurs, reheat may be reselected; if automatic trip again occurs, no further reheat selections should be made.

(v) If, after cancellation of reheat, the nozzles remain fully open, a substantial loss of thrust in cold power occurs.

(*d*) *Throttle Seizure in Reheat*
If a throttle seizes in the reheat range, force should be applied to the throttle lever. This force stretches a telescopic rod in the reheat control run allowing the gate switch to be reached; reheat cancels and the nozzle closes. The lever then operates normally in the cold range but it is recommended that reheat should not be used again during that flight.

6 Engine Starting System

(*a*) Each engine is started by an iso-propyl-nitrate (Avpin) liquid fuel starter, which uses the gases from the decomposition of the fuel to drive the starter turbine which is connected to the engine by a reduction gear box. The starters use a common 3-gallon fuel tank installed in the fuselage spine. Normally a total of six engine starts may be obtained.

(*b*) Pressing the starter button initiates a timed sequence of operations as follows:

(i) The starter motor combustion chamber is scavenged by compressed air which is supplied by a combined fuel/air pump.

(ii) A fuel charge is pumped into the combustion chamber.

(iii) The mixture of fuel and air is ignited by two high-frequency igniter plugs and combustion is sustained by decomposition of the injected fuel.

(iv) The starter turbine turns the engine and at the same time the engine HE ignition plugs are energised to light up the engine.

(v) When the engine reaches self-sustaining speed, a switch operates to shut down the starter system.

(*c*) If an engine fails to start, limitations on subsequent attempts to start that engine are imposed, depending on the type of failure. The limitations are as follows:

(i) If, after pressing the starter button, starter combustion is not sustained (A or B type failures), or even if there is no apparent effect, a waiting period of at least 1 minute should be allowed before making a further attempt to start. This is to ensure that Avpin fuel has drained from the starter combustion chamber. A maximum of three attempts only may be made.

(ii) If, after a normal starter combustion cycle, the main engine fails to reach self-sustaining speed, it is necessary to wait until the engine has stopped turning before making a further attempt. Additionally in this type of failure, heat soakage of the starter may prejudice the next start. Therefore the interval between attempts should be as close as possible to 1 minute. A maximum of two such attempts to start may be made. A cooling period of 60 minutes must then be allowed before investigation or a further starting attempt is made. The cooling requirement is considered satisfied if the final permitted starting cycle results in an engine start provided that the engine is allowed to run for a period of 15 minutes.

(iii) If, after a normal starter combustion cycle, the main engine fails to rotate, a further attempt to start must not be made. Examination or investigation must not be made in the vicinity of the engine starter until a cooling period of 60 minutes has elapsed. The starter must then be removed for investigation.

WARNING: On no account are the engine starting buttons to be pressed (either in the air or on the ground) when the engines are rotating.

7 Relighting

With the engine master switch on, pressing the relighting button bypasses the normal starting sequence and energises the ignition unit which in turn operates the high-energy igniter plugs to ignite the fuel spray. The system incorporates a time switch which holds ignition on for a period of approximately 30 seconds. Whenever the armament firing trigger is pressed and provided the undercarriage is up, the relighting circuits are automatically energised.

8 Oil System

Each engine has its own independent integral oil system of 12·5 pints capacity. One pressure and five scavenge pumps maintain a continuous circulation through a fuel-cooled oil cooler and filter to the engine bearings and gears.

9 Ice Warning

The ice warning system measures the temperature and moisture content of the air flow through the engine air intake duct. Should icing conditions occur, a warning appears on the AWP. The system is supplied from the generator busbar. Therefore, if neither generator is running, the system will be inoperative even though the 28 volt DC busbar is energised. The system is not at present cleared for use and is inoperative.

10 Anti-icing

(*a*) *General*

(i) Anti-icing is provided by hot air and electrical heating systems. The hot air heating system gives protection to both engines and the engine duct lip, and the electrical heating system protects exposed surfaces in the engine duct. ▶◀

(ii) The individual systems are initially controlled by a single switch in the cockpit. Operation of the switch energises a relay to connect DC power to initiate operation of each engine system, the duct lip system and the Spraymat system.

(*b*) *Engine Anti-icing*

On each engine, hot air tapped from the compressor is directed by an electrically-operated gate valve to flow into the front bearing support struts, the intake guide vanes, the starter fairing and the starter exhaust pipe. The air then passes into the engine.

(*c*) *Duct Lip Anti-icing*

Compressor bleed air, controlled by an electrically-operated butterfly valve, is ducted to a manifold from where it is divided to flow through sections of the duct lip and escape through small vents into the duct.

(*d*) *Spraymat Anti-icing*

(i) The leading edges of the upper and lower radome bullet struts and the areas about the engine duct bifurcation are protected by the Spraymat system. Power for heating is provided by the 200 volt, 3-phase, 400 Hz, AC supply. AC power may be switched off automatically either by operation of a temperature controller within the system or the tripping of the relay described in para 10 (*a*) (ii).

(ii) The temperature controller in the system is ineffective when the engines are not running. To prevent damage to the Spraymat elements, an AC supply must not be connected to the system with DE-ICING ON selected whilst the engines are at rest.

(*e*) *Cockpit Indication*

A switch in the duct lip butterfly valve is in circuit with a magnetic indicator in the cockpit. The switch operates in the last few degrees of movement at either end of the range of the valve on opening or closing.

(*f*) *Temperature and Pressure Switches*

Downstream of the duct lip butterfly valve are two switches designed to protect the system against excessive temperatures and pressures. Operation of either switch trips the relay (para 10 (*a*) (ii)) which switches off DC power to all the systems. In the engine and duct lip systems the hot air supply is shut off by closing of the valves and in the Spraymat system a contactor is tripped to disconnect the AC electrical supply. The temperature sensing switch is self-resetting, therefore the systems will resume operation after a trip if the temperature drops. The pressure-sensing switch however, is not self-resetting and, if the switch operates, anti-icing will not be operative again until the switch has been reset on the ground.

(*g*) *Malfunctioning*

(i) With electrical supplies available, if the duct lip butterfly valve fails closed when DE-ICING ON is selected, duct lip anti-icing is inoperative but engine anti-icing and Spraymat function. Since the anti-icing indicator is controlled by the duct lip butterfly valve, it shows black (off) in this case. The automatic protection switches are inoperative, therefore it is essential to remember to switch off anti-icing when clear of icing conditions.

(ii) If the duct lip butterfly valve fails open when anti-icing is switched OFF, duct lip anti-icing remains operative and the indicator remains at I; engine anti-icing and Spraymat, however, are switched off. Flight conditions producing high ram temperatures may lead to deterioration of the hose couplings and consequently should be avoided.

CONTROLS AND INDICATORS

11 Throttles/HP Cocks

Mk5 different

(*a*) The throttle valve and HP cock of each engine are combined in one unit. The outboard lever controls No 1 engine and the inboard No 2. The No 2 throttle lever incorporates an airbrake control switch and an RT press-to-transmit switch.

(*b*) With the throttle fully back the HP cock is closed and when set to the first OPEN position, the HP cock is opened. Further forward movement to the IDLING position sets self-sustained engine RPM and at the second OPEN position, maximum cold running is selected.

(*c*) To select reheat, the throttle is rocked outboard out of the cold running range and fully opened. Reheat is cancelled by rocking the throttle inboard at any position between the maximum cold power position and a preset mechanical stop (96% approximately).

(*d*) On No 2 throttle only is a fast idling stop (58% RPM approximately). This stop can be cleared by operating a PRESS, FAST IDLING STOP RELEASE lever on the No 2 throttle.

(*e*) A stop at the IDLING position prevents further rearward movement of the throttle levers. To select HP COCKS OFF from IDLING, the SHUT-DOWN lever at the rear of the throttle box must first be pressed forward.

(*f*) When flying with the No 2 engine shut-down, there is a possibility that forward movement of No 2 throttle lever in the HP cock range could cause the idling stop to be depressed. In these circumstances, there would be nothing to prevent the No 1 throttle being moved back into the HP cock closed position when throttling back. ▶◀ To prevent this, an additional stop ensures that the No 2 throttle is positively held at the HP cock closed position until cleared by operating the PRESS FAST IDLING STOP RELEASE.

(*g*) A THROTTLE SERVO—ENGAGE/DISENGAGE lever on the side of the left console engages auto throttle control. (See Part 1, Chap 14).

12 Engine Starting and Relighting Controls

(*a*) The main starting controls consist of:

(i) An ENG MASTER switch on the starter panel.

(ii) An ENGINE STARTER ISOLATION switch and No 1 and No 2 ENGINE START buttons, on the right console. Engine starter operation is only possible when the ENG MASTER switch is in the on position (up) and the isolation switch is forward. After starting, the isolation switch must be set to the aft position to prevent inadvertent operation of a starter.

(*b*) Two buttons, No 1 and No 2 RELIGHT are on the left console: the ENG MASTER switch must be on to relight an engine in flight.

13 Jet Pipe Temperature Control

Jet pipe temperature control is brought into operation by two switches, labelled JPT CONTROL No 1 and No 2 outboard of the throttles. Each switch has an AUTO and OFF position. With a switch at AUTO, maximum JPT control is in operation and automatic reheat cancellation occurs if the maximum JPT is exceeded by 60°C. With a switch at OFF, there is no JPT control and reheat is not cancelled automatically if excess JPT is experienced. The JPT control switches are normally wire-locked at AUTO.

14 Engine Instruments

(*a*) Each engine has a percentage RPM gauge, a JPT gauge and a nozzle position indicator on the right instrument panel.

(*b*) *Jet Pipe Nozzle Indication*

▶◀

Each indicator features two arcs, the lower for the non-reheat range and the upper for the reheat range. A specific mark at the end of each arc denotes the maximum dry power and maximum reheat nozzle positions, ie nozzles closed and fully open respectively.

(*c*) *Oil Pressure Warning*

A pressure switch is fitted in the oil system of each engine. If the pressure falls below 20 PSI the pressure switch closes and an OIL 1 or OIL 2 warning, as appropriate, appears on the AWP. The warning may remain after engine start until 45% RPM has been obtained.

15 Anti-icing Controls and Indicators

(*a*) A 3-position DE-ICING ON/OFF (guarded)/RAIN DISPL switch is on the left console. When DE-ICING ON is selected, engine anti-icing for both engines, Spraymat and ▶◀ duct lip anti-icing are operative. Selection to RAIN DISPL brings into operation the rain dispersal system, (see Part 1, Chapter 11).

(*b*) A magnetic indicator adjacent to the switch indicates white/I with anti-icing in operation, white/R with rain dispersal in operation and black with both systems switched off.

(*c*) An ICE warning on the AWP comes on whenever icing conditions are detected, provided a generator is on line. The system is not cleared and is inoperative at present.

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PART 1

CHAPTER 4 — ENGINE FIRE PROTECTION SYSTEM

Contents

1 Fire Zones

Each engine bay is divided into two fire zones by a firewall between the engine compressor section (zone 1) and the turbine and exhaust section (zone 2). The space surrounding the jet pipes of both engines is known as zone 3 and it is separated from the other zones by vertical and horizontal firewalls. Each fire zone has a separate ventilating system.

2 Fire Extinguishers

(*a*) Two dual-headed methyl bromide fire extinguishers are installed in the fuselage. The port extinguisher serves each zone 2 and the starboard extinguisher each zone 1. There is no extinguisher for zone 3. Extinguisher operation is effected by depressing either of two indicator switch units on the SWP. When either switch unit is depressed, both extinguishers are completely discharged into zones 1 and 2 of the appropriate engine bay. Telltale indicators, one on each side of the ventral tank refuelling light on the port side of the fuselage, show a reddish-brown colour if the extinguishers have been discharged electrically.

(*b*) Each extinguisher is fitted with an over-temperature safety device which causes the contents to discharge overboard should the temperature in the vicinity of the bottle exceed 175°C approximately. Two indicators, one port and one starboard, located just aft of the No 1 engine longeron show green when no discharge has occurred. If the contents of a bottle have been released overboard, the green indicator cover will have blown off and a bright red bowl under the cover will be disclosed.

3 Fire Detection

(*a*) A FFFD fire detection system is fitted in all fire zones. There is a separate Firewire circuit for each engine bay and two circuits, one to each jet-pipe, in zone 3. The elements are connected to the input channels of a relay unit which controls the relevant warnings on the SWP. The INST MASTER switch must be on for the detection circuits to be operative.

(*b*) When mod 2511 is embodied, fire detection is provided by a Triple-FD system. When tested, unserviceability of this system is indicated either by the warnings not illuminating or failing to extinguish when the test button of the SWP is released. Testing of the system should not be done in the air as there is a possibility that a fire warning, triggered by the test facility, might remain on when the test button is released, even though the system is serviceable. Testing should be done before engine start and immediately after landing when optimum conditions for moisture contamination of the firewire have been experienced.

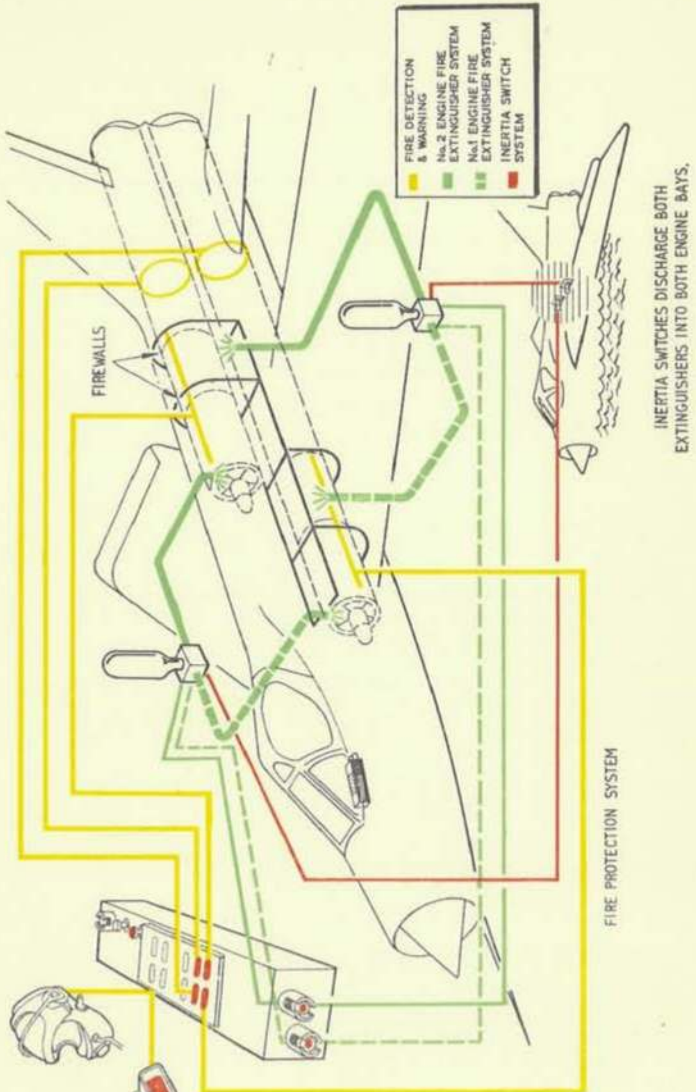
(*c*) The fire detection system is supplied with AC from the alternator output and has a standby supply from the type 100A inverter. Thus, a warning initiated on one AC source is continued if change-over to the other source occurs, with a momentary interruption at the instant of changeover. Because of this interruption, cancelled attention-getters will be re-activated if their originating warning is still present at the time of change-over.

4 Fire Warnings

Warnings of a fire in zone 1 or 2 of the No 1 engine is given by a FIRE 1 warning on the SWP and the illumination of the F1 light in the fire extinguisher switch unit; warning of a fire in zone 1 or 2 of the No 2 engine is by a FIRE 2 warning on the SWP and the illumination of the F2 light in the fire extinguisher switch unit. Warning of excessive temperature or fire in zone 3 is given by a RHT 1 and/or RHT 2 warning on the SWP.

5 Inertia Crash Switches

Two inertia crash switches are in circuit with the fire extinguisher system. If both crash switches operate, both extinguishers are discharged into zones 1 and 2 of each engine. The crash switches also operate to isolate certain electrical supplies (see Part 1 Chap 1).



FIRE PROTECTION SYSTEM

PART I

Chapter 5—HYDRAULIC SYSTEM

Contents

Description

1 General

(*a*) The hydraulic installation comprises three separate systems:—

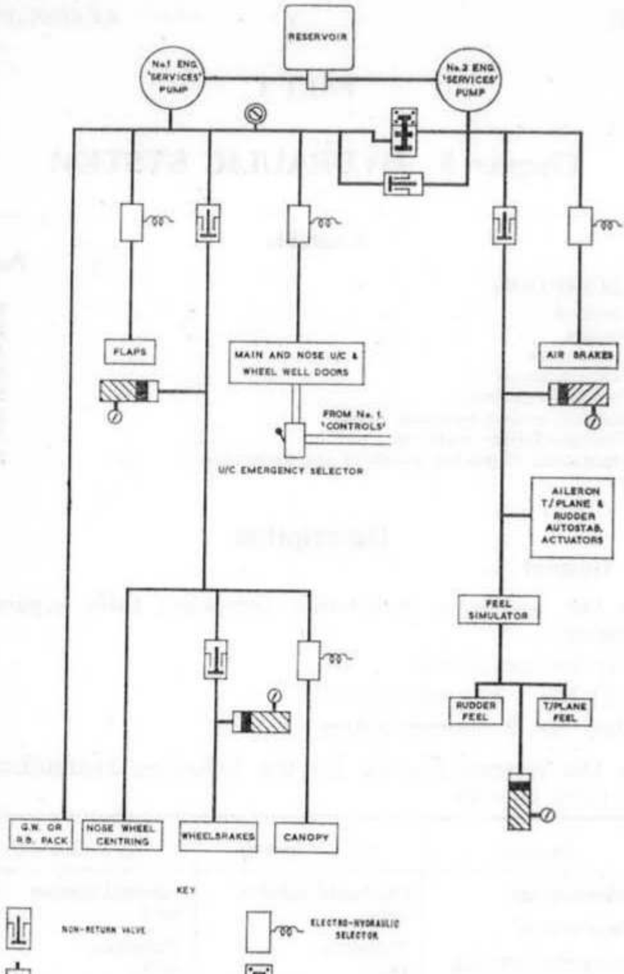
(i) Services system

(ii) No. 1 controls system

(iii) No. 2 controls system

(*b*) The systems provide for the following hydraulically operated services.

Services	*No. 1 Controls*	*No. 2 Controls*
Undercarriage	Outboard aileron PFCU	Inboard aileron PFCU
Wheel brakes	Tailplane PFCU (starboard motor)	Tailplane PFCU (port motor)
Nose-wheel centring	Rudder PFCU	Rudder PFCU
Air brakes	Brake parachute compartment doors	
Flaps	UC emergency lowering	
Feel System		
Autostabiliser actuators		
Canopy		
GW or rocket pack		



SERVICES HYDRAULIC SUPPLY

2 Pumps

(*a*) Power is provided by four engine-driven pumps, two mounted on each engine wheelcase. The forward pumps on each engine jointly power the services system; the aft pump on No 1 engine powers the No 1 controls system and the aft pump on No 2 engine the No 2 controls system.

(*b*) A hand-pump for ground operation only of the services system is behind an access panel on the port side of the fuselage. A stowage for the hand-pump handle is in the port wheel well.

3 Reservoirs

The pumps draw hydraulic fluid from three reservoirs, one to supply the services system requirements and one each for the No 1 and No 2 controls system. The reservoirs are pressurised with air tapped from the engine compressors. An auxiliary reservoir for the No 1 controls system is fitted to provide an additional reserve for emergency lowering of the undercarriage. The auxiliary reservoir operates in parallel with the main No 1 controls reservoir. Mod 4749 introduces reservoirs fitted with an anti-G valve to prevent accumulated air from entering the hydraulic pump from the reservoirs when the aircraft is in a negative-G condition.

4 Accumulators

(*a*) *Services accumulators*

Four accumulators in the services system store pressure for operation of the wheel brakes, canopy raising or lowering, nosewheel centring, autostabiliser actuators, feel simulator and feel units.

(*b*) *Controls system accumulators*

Two accumulators in each controls system store pressure for all services in the controls system. These provide for high rates of normal operation of the control surfaces and for limited emergency operation.

5 Pressure regulator

It is necessary to maintain a minimum pressure of 2700 PSI when the guided missile pack alternator motor is running. If the airbrakes are selected the pressure may reduce to below 2700 PSI. A pressure regulator and non-return valve are therefore interposed between the delivery lines of the No 1 and No 2 services system pumps. If the No 2 pump delivery pressure falls through selection of airbrakes the pressure regulator operates and the non-return valve closes, thus ensuring that the No. 1 pump output pressure is not reduced. The pressure at the missile pack alternator motor, which is supplied from the No. 1 pump, is therefore protected.

6 Services system pressure gauge

A HYD pressure gauge on the port cockpit shroud indicates services system pressure. When the pumps are running the ◀pressure reading should be 3,000 ± 250 PSI. When▶ a service is selected which has a high fluid demand, the reading will fall rapidly and then gradually rise to 3,000 PSI approximately.

7 Pressure failure warning

(*a*) A pressure switch is fitted in the pressure line of both No. 1 and No. 2 controls system. If line pressure falls to 1,750 PSI the switch closes and a HYD 1 or HYD 2 warning as appropriate, appears on the AWP. If both pressure switches close an additional HYD warning appears on the SWP.

(*b*) Failure of the services system is indicated when the ◀services pressure gauge reading falls to and remains at▶ zero (red sector of gauge). The gauge requires a 28-volt single phase AC supply and if this fails the needle on the gauge will fall below zero into the white sector on the gauge.

8 Operation of guided weapons and airbrakes

One engine must be maintained at or above fast idling speed for the satisfactory operation of the guided missiles; this must be No. 1 engine if the airbrakes are to be used with guided weapons switched to ARM. Loss of electrical supplies to the guided weapons will result if, when switched to ARM, flaps or undercarriage are operated, or if the airbrakes are operated when flying on No. 2 engine only.

PART I

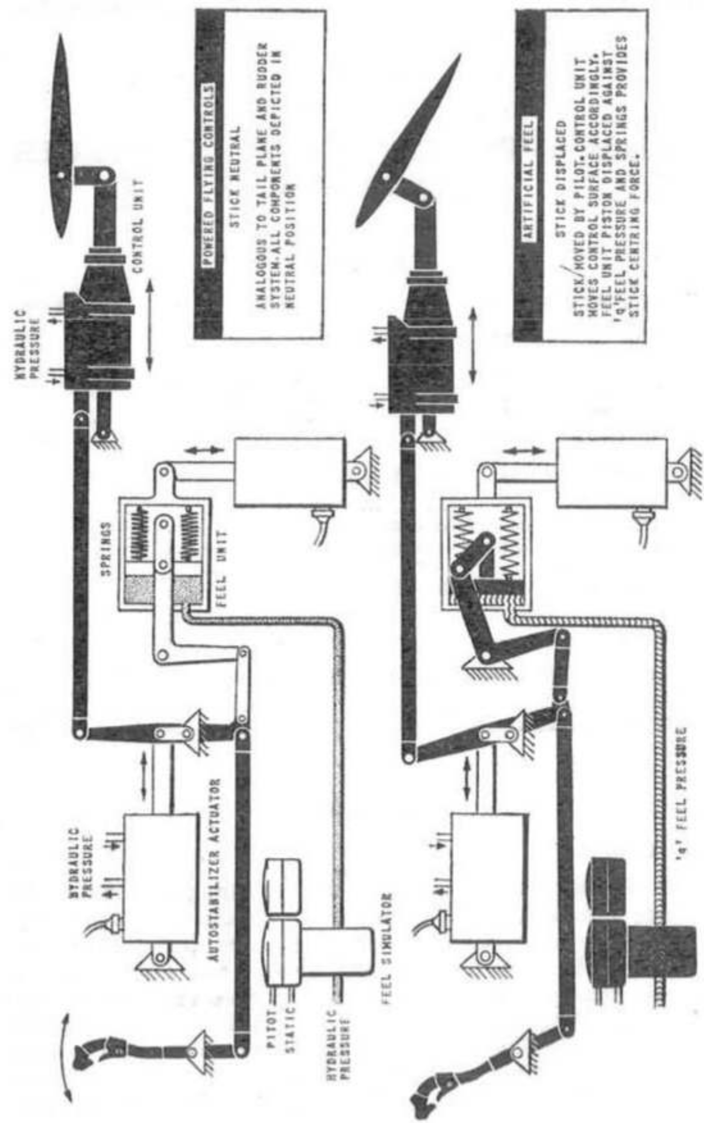
Chapter 6—POWER FLYING CONTROLS AND TRIMMERS

Contents

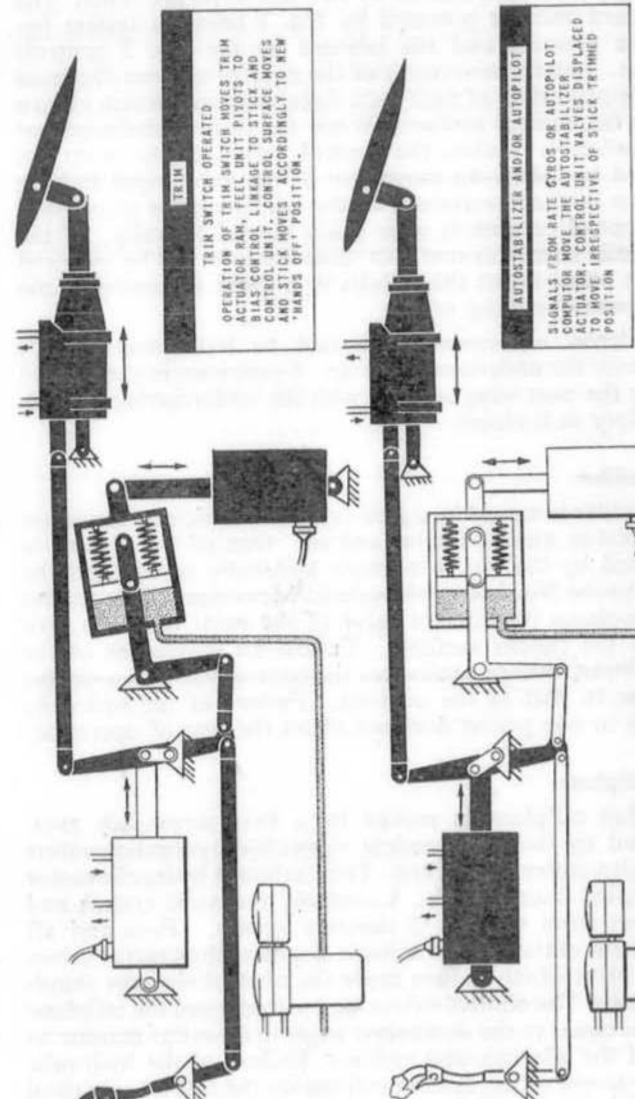
Description

1 General

The ailerons, tailplane and rudder are fully power operated flying controls. The ailerons and rudder are moved by jack-type powered flying control units (PFCU) and the tailplane by a twin screw-jack PFCU. The PFC units are mechanically actuated from the control column and rudder bar. Hydraulic power for the operation of the control surfaces is duplicated so that failure of one controls system does not result in loss of aircraft control. As the controls are irreversible, artificial feel is provided in each system. Trimming is effected by electrically operated actuators, which, by moving the control runs, displace the PFCU and adjust the cockpit controls to the required position.



TYPICAL POWER CONTROL SYSTEM



TYPICAL POWER CONTROL SYSTEM

2 Ailerons

(*a*) Each aileron is moved by two jack-type PFC units. The outboard PFCU is powered by No. 1 controls system hydraulic pressure and the inboard by the No. 2 controls system. Lateral movement of the control column displaces the control valves of each PFCU simultaneously which in turn move the control surface. When the desired deflection of the surface is reached the control valves of the PFCU are centred by follow-up movement of the valve input linkage and no further movement of the surface takes place until the control column is once again moved laterally. If the hydraulic supply to one PFCU fails the unit will be motored by the other PFCU; this results in a slight reduction in the maximum operating rate.

(*b*) Aileron movement is limited to half-travel ($\pm$ 8°) whenever the undercarriage is up. A restrictor in the control run in the port wing engages with the undercarriage wheel-well door as it closes.

3 Rudder

The rudder is moved by a jack-type PFCU which incorporates twin piston assemblies in tandem. One of the pistons is operated by the No. 1 controls hydraulic system and the other by the No. 2 controls system. Movement of the rudder bar displaces the control valve of the PFCU which in turn moves the rudder surface. Follow-up movement of the valve input linkage centralises the control valve in a similar manner to that of the ailerons. Failure of the hydraulic system to one piston does not affect the rate of operation.

4 Tailplane

The slab tailplane is moved by a twin screw-jack PFCU powered by two independent reversible hydraulic motors through a common gearbox. The starboard hydraulic motor is supplied from the No. 1 controls hydraulic system and the port from the No. 2 controls system. Fore and aft movement of the control column displaces the control valves of the PFCU which in turn move the control surfaces simultaneously. The control valves are centred when the tailplane angle is equal to the demanded angle in a similar manner to that of the ailerons and rudder. Failure of the hydraulic supply to one of the motors will reduce the rate of operation but this will not cause any embarrassment at 2G or below, to which manoeuvring is restricted following such a failure.

5 Artificial feel system

(*a*) *Aileron feel*

Artificial feel on the aileron control is provided by a torsion bar connecting the control column and the control run. Feel force is directly proportional to control column movement and does not vary with speed.

(*b*) *Tailplane and rudder feel*

(i) Artificial feel on the rudder and tailplane control is provided primarily by a hydraulic feel unit linked in each control run. When the control column or rudder bar is moved, a piston in the feel unit is displaced against hydraulic pressure. Relaxing the force on the control column allows the piston to move back to neutral, returning the control column or rudder bar to its trimmed position. Additionally, spring feel is provided to assist the hydraulic feel and to act as a stand-by system in the event of failure of the hydraulic feel system.

(ii) The hydraulic feel units are designed to give a linear increase in feel force with rudder or control column movement. They also give an increasing feel force with increase in pitot/static differential pressure controlled by a feel simulator control unit which meters the hydraulic pressure to the feel units. If the pitot/static pressure is increased, e.g. by increasing airspeed, the feel simulator control increases the hydraulic pressure, reacting against the piston in the feel units. At speeds above 0·9M, however, feel is maintained at a constant value for a given altitude, per degree of control movement, irrespective of any further increase in dynamic pressure. Differences in static pressure will continue to affect feel force, i.e. decrease in altitude will increase the feel force.

AL 9 (iii) To reduce foot load in the circuit, hydraulic feel to the rudder is cancelled when the undercarriage is selected down and is restored when an UP selection is made.

(iv) Hydraulic feel may be cancelled manually by a switch on the port console.

(v) Two accumulators are in the hydraulic feel system, the smaller to act as a damper between the feel units to prevent rapid operation of the controls causing interaction between the rudder and tailplane units and to prevent reaction on the simulator valve which would otherwise cause the flight instruments to fluctuate. The larger accumulator prevents changes in feel occurring when the general services are being operated. It also provides for the continued operation of the feel units for a limited period after failure of the services hydraulic system.

6 Trimming

Trimming is effected by an electric motor in each system which is connected to the hydraulic feel unit or, in the case of the ailerons, to the torsion bar. Movement of the motor displaces the feel unit or aileron torsion bar which in turn ◀displaces the control column or rudder bar. ▶

Controls and Indicators

7 Controls

(*a*) The controls are operated by a pistol grip control column and a rudder bar. The control column carries on it an aileron/tailplane trimming switch, G.90 camera operating button, nose-wheel steering button (inoperative), armament firing trigger and safety catch, FCS engage switch, press-to-transmit switch and a wheel brake lever and parking catch.

(*b*) The rudder bar may be adjusted for leg reach by a PULL TO ADJUST RUDDER BAR handle situated on the starboard instrument panel. When the handle is pulled out a toothed plunger is withdrawn from the rudder rack allowing the rudder bar to move aft under the action of a compression spring, or to be pushed forward against the pressure of the spring. When the desired position is reached, releasing the handle allows the toothed plunger to re-engage the rack.

8 Trim switches and indicators

(*a*) *Rudder trim*

Twin, 3 position, ganged switches marked RUDDER TRIM, spring loaded to the central off position, on the port console, control the rudder trim in the natural sense. Both switches must be selected together before the trimmer will operate.

(*b*) *Aileron/tailplane trim*

A four-way, spring loaded to off, ganged dual switch on the control column, when operated in the natural sense, controls the aileron and tailplane trim. To test the two switches individually the ganging bar can be raised and pivoted to one side by pulling to the right the spring-loaded pin at the top of the ganging bar.

(*c*) *Trim indicators*

A combined TRIM INDICATOR on the port instrument panel shows the trim position on three scales for the RUDder, TAILplane, and AILeron. A fourth scale on the indicator shows the position of the airbrakes.

(*d*) *Feel selector*

A two-position FEEL, ON/OFF switch, guarded to ON, is fitted on the port console. When the switch is at ON the hydraulic feel units are brought into operation. In the OFF position, hydraulic feel is cancelled and only spring feel remains.

9 Power control failure warnings

No 1 or No 2 controls system hydraulic failure warning is given by the illumination of the HYD 1 or HYD 2 warnings respectively on the AWP. Failure of both systems is indicated by a HYD warning appearing on the SWP in addition to the two warnings on the AWP.

Normal use of the System

10 Pre-flight checks

(*a*) Test the rudder trim for a 'live' circuit by operating each of the two rudder switches separately. If any movement of the rudder trim actuator occurs the aircraft must not be flown. Test the rudder trim over its full range by operating both switches together and then set to neutral. Replace the ganging bar.

(*b*) Test the aileron and tailplane trim for a 'live' circuit by raising the ganging bar and operating each switch separately. If any movement of either aileron or tailplane trim occurs the aircraft must not be flown. Replace the ganging bar and check that it is locked by attempting to lift it. Test the aileron and tailplane trim over the full range and then set the ailerons to neutral and the tailplane to T.O.

(*c*) Prior to take-off, operate the controls over the full range for freedom of movement. If excessive rate of movement of the tailplane is made it is possible that the control valve in the PFCU will bottom. This will be noticed by an increase in feel force. No mechanical damage should occur, however, due to this condition.

(*d*) Check that the HYD 1 and HYD 2 warnings on the AWP and HYD warning on the SWP are out prior to take-off.

11 In-flight

Limitations are imposed on the use of the ailerons (see Part 2, Chap 1).

Malfunctioning of the System

12 Power control failure

(*a*) Failure of one hydraulic controls system is indicated by a HYD 1 or HYD 2 warning appearing on the AWP. A slight reduction in the maximum operating rate of the ailerons and tailplane will occur.

(*b*) Failure of both hydraulic controls systems is indicated by a HYD warning appearing on the SWP in addition to the two warnings on the AWP. The accumulators in the systems will provide for limited operation for a short period before the controls become immovable. If the failure is caused by double engine flame-out the accumulator supply may be supplemented from the pumps by wind-milling the engines at a speed of 250 knots or above.

◀(*c*) The most likely cause of a double flying controls systems warning, not associated with a double flame-out, is aeration of the system followed by pump cavitation usually after negative G flight. A period of straight and level flight is likely to restore hydraulic power. The drill contained in the Flight Reference Cards recommends engagement of the auto-pilot (if serviceable) for the following reasons:

(i) The auto-pilot detects changes of direction quickly, preventing large diversions from straight and level flight.▶

(ii) It is stressed that the controls will ultimately go stiff following a double warning. It is essential that the aircraft is in stabilised flight; the auto-pilot, if engaged, would ensure this and a stabilised aircraft should stay in level flight sufficiently long enough for the controls to recover. This has been known to take up to eight minutes.

(*d*) Emergency actions for control system failures are given in the Flight Reference Cards.

13 Trimmer malfunction

(*a*) If the aileron trimmer fails, the control forces can be held even if the trimmer has failed at the extreme end of its travel. Prolonged flight in this condition is tiring. The aircraft should be landed as soon as practicable.

(*b*) In the event of tailplane trimmer runaway, switch OFF the autopilot master switch.

(*c*) If the rudder or tailplane trimmers runaway or fail to function, control forces may be reduced by switching off the hydraulic feel. If the tailplane trimmer has seized in a nose-down trim position, the hydraulic feel should be switched off during the approach and landing.

WARNING: When flying with the feel switched off, the control forces are light and care must be taken not to exceed the airframe limitations; speed should be restricted to below 400 knots.

PART 1

Chapter 7—OTHER AIRCRAFT CONTROLS

Contents

Description

1 Undercarriage controls and indicator

(*a*) The hydraulically operated undercarriage comprises two main wheels retracting rearward and outward into the wings and a nose wheel retracting forward into a well in the fuselage nose.

(*b*) An undercarriage lever on the port instrument panel controls the undercarriage hydraulic selector. The lever carries a spring-loaded cup which enters a gate in both the UP and DOWN positions, thus preventing inadvertent operation. A pitot switch prevents an UP selection being made at airspeeds below 165 knots. ~~(pre mod 4398 150 knots).~~ In an emergency this switch can be overridden by turning the undercarriage lever clockwise through 45° and selecting UP.

X AL 14

(*c*) An undercarriage position indicator is fitted on the lower port instrument panel.

(*d*) The emergency retracting facility must not be used as a means of stopping the aircraft in a high speed ground emergency. There is a strong likelihood of asymmetric retraction with the consequent danger of the aircraft cart-wheeling. It is recommended that the use of the facility is restricted to normal taxying speeds.

(*e*) The limiting speed for undercarriage selection is 250 knots.

2 Undercarriage emergency lowering control

(*a*) The UC emergency lowering control is a yellow and black striped T-handle on the port console. The button on top of the handle must be pushed in before the handle can be pulled. The handle must be pulled to its full extent and just prior to full travel it will fall away as it leaves its guide.

(*b*) When the T-handle is operated, No 1 controls system hydraulic pressure is directed to:

(i) Close a valve in a protection unit to isolate services pressure from the normal selector.

(ii) Open a valve in the protection unit to connect all the up lines to return irrespective of the setting of the normal selector.

(iii) Operate shuttle valves and relay valves to direct the No 1 controls system pressure to lower the undercarriage.

(*c*) Once the emergency system has been used it is not possible to retract the undercarriage until the system has been ground serviced.

(*d*) Emergency lowering may be achieved even with the No 1 engine windmilling. The rate of lowering is slower than normal and selection should be made early. Flying control demands should be limited to the minimum necessary during lowering. About 70 seconds will be required for locking down with no control demands; aerodynamic assistance may be necessary if controls are used. After an emergency selection ensure that the normal selector lever is in the down position to render the armament circuits safe.

3 Flap control and indicator

(*a*) The flaps, which are utilised as fuel tanks, are hydraulically operated from the services hydraulic system and electrically controlled by a two-position, FLAPS fully UP or fully DOWN, switch on the port instrument panel. A pitot-pressure switch, set to close at 250 knots, automatically selects the flaps to UP if they have been inadvertently left in the DOWN position and speed increased to 250 knots; when speed is reduced below 250 knots, the flaps will take up the position as selected by the flap switch. However, whenever automatic retraction occurs, the flap switch should be set to UP.

(*b*) Asymmetric operation of the flaps in hydraulic failure conditions is prevented by hydraulic locking, so that if the flaps are fully extended both flaps are held in this position or, if the failure occurs during lowering, they are held at the position reached when failure occurred. However, transient asymmetry can occur with certain flap system defects. The flap jacks incorporate a mechanical lock in the UP position which automatically disengages when DOWN is selected.

(*c*) No provision is made for in-flight operation of the flaps in the event of services system hydraulic failure or DC electrical failure.

(*d*) A flap position indicator is on the port instrument panel. Each of the two needles on the instrument shows the position of its respective flap.

(*e*) The limiting speed for operation of the flaps is 250 knots.

4 Airbrakes Control and Indicator

(*a*) The airbrakes are hydraulically operated and are electrically controlled by a 3-position IN/off/OUT switch on the No 2 engine throttle lever; the switch is spring-loaded to the off position. Synchronous operation of the airbrake doors is achieved by a valve in the hydraulic lines to the airbrakes which throttles the supply to either door should it tend to lead in operation. A mach switch, set to operate at 1·25 to 1·3M, causes the airbrakes to be automatically closed if this speed is exceeded. When speed is reduced below the setting of the mach switch, the airbrakes will remain closed until the next out selection is made. The airbrakes can only be selected to fully out and fully in.

(*b*) A combined trim and airbrake indicator is on the port instrument panel. One of the four scales on the indicator shows the position of the airbrakes and has a special provision for the locked-in position.

(*c*) The limiting speed for operation of the airbrakes is 650 kts/1·3M.

5 Wheelbrakes Control

(*a*) The hydraulically operated disc-type brakes are fitted with maxaret anti-skid units and are controlled by a lever on the front of the control column. The brake lever is connected to a differential control valve which is mechanically linked to the rudder bar to give differential braking in the conventional manner. Full differential pressure at the brakes is achieved at half-pedal displacement.

(*b*) The accumulator pressure available for brake operation is shown on a gauge on the port instrument panel. The normal reading is 3000 $\pm$ 100 PSI, but in flight the pressure may build up above this figure to as much as 3600 PSI. Should the brake pressure during flight reach 3600 PSI, operate the wheelbrakes to relieve the pressure.

6 Braking Parachute

(*a*) A ribbon-type braking parachute is housed in a compartment on the underside of the rear fuselage. The parachute is streamed by pulling to the fullest extent a handle marked TAIL CHUTE—PULL on the port instrument panel. This action mechanically operates a selector to hydraulically open the parachute doors from No 1 controls system pressure: as the doors open, the parachute streams.

(*b*) Depressing a CHUTE JETT push-button, on the port coaming panel, operates an electro-magnetic release unit which opens to jettison the parachute. The release unit can be operated only after the parachute has been streamed.

(*c*) If the parachute is inadvertently streamed at excessive speed it will become detached from the aircraft through the rupturing of a shear-pin in the parachute cable.

(*d*) The TAIL CHUTE—PULL lever, if reset by the pilot, should be held and returned slowly, otherwise damage to the cable will result.

(*e*) The limiting speeds for operation of the braking parachute are:

Normal operation	...	150 knots
Emergency operation	...	170 knots

Note: Whenever the parachute is streamed at speeds above 150 knots, the fact must be reported after flight so that the parachute can be assessed for re-use.

(*f*) If the No 1 engine is shut-down, but is windmilling, the hydraulic supply should be sufficient to operate the parachute doors. However, a no parachute landing must be anticipated.

PART 1

Chapter 8 — FLIGHT INSTRUMENTS

Contents

Description

1 Integrated flight instrument system — General

(*a*) The Integrated Flight Instrument System (IFIS) derives its information from the following sources:

(i) The dynamic reference system.
(ii) The air data system.
(iii) Tacan, ILS and UHF coupling units.

(*b*) The information is presented on an attitude indicator, a navigation display, a speed display and a height and rate of climb display.

(*c*) To cover failure of the dynamic reference system, a standby artificial horizon and direction indicator are fitted. A conventional ASI and a conventional altimeter are fitted to meet the case of failure of the air data system.

(*d*) 28 volt DC and 115 volt, 3 phase, 400 Hz AC power supplies, controlled by the INST MASTER switch, are required for the operation of the IFIS.

2 Dynamic reference system

(*a*) The dynamic reference system uses a master reference gyro to supply continuous flight attitude and heading information. This information is supplied to the following equipment:

(i) Attitude indicator.
(ii) Gyro magnetic compass.
(iii) Flight control system.
(iv) AI23B.

(*b*) *Master reference gyro* (*MRG*) *Mk* 2

(i) The MRG consists of a gyro reference platform and an electronics unit mounted in the equipment bay. The system uses 115 volt, 3 phase, 400 Hz AC and 28 volt DC power supplies via the INST MASTER switch. Control of the system is by an MRG toggle switch on the port instrument panel. The switch has a locking device to prevent inadvertent operation from the ON position: to switch off, the toggle must be pulled to clear the lock before selection. In AC failure conditions, the system is operated by the standby inverter supply provided the MRG is already erected. However the 100A inverter may not function normally when fed by battery voltage only, thus precipitating MRG failure; this may be noticed after landing when the main AC supplies go off line.

(ii) To start the MRG, power must be supplied from an external AC source or from the alternator, as an electrical interlock prevents start-up from the standby inverter supply. Additionally, the INVERTER switch must be set to NORMAL; the electrical interlock is ineffective with the switch at STANDBY and damage would occur to the inverter if connected to the MRG on start-up. Correct sequence of switching on is important and is as follows:

MRG switch	Off (down)
AC power (external or from alternator)	On
INVERTER switch	NORMAL
INST MASTER switch	ON
STANDBY INVERTER indicator	Black
MRG switch	ON (up), locked

(iii) When the MRG is switched on, the roller blind of the attitude indicator may or may not rotate and change to all black, depending upon the standard of the instrument (see sub-para (*c*)); full erection is complete after 2 to 3 minutes approximately, shown by the orange disc disappearing from the attitude indicator. The gyros normally take about 25 minutes to run down after switching off; during this period the MRG can be switched on again but, if a break of more than 105 seconds (30 seconds Post Mod MRG 207) has occurred between switching off and switching on, the erection cycle could take up to 5 minutes.

(iv) Vertical errors in the system can be removed at the rate of 15°/min approx by depressing and holding the

* MRG 207 is FE

spring-loaded MRG FAST ERECT button on the left instrument panel. The selection should be made in straight and level unaccelerated flight and only when gross errors are obvious and good external reference for levelling the aircraft is available.

(v) Compass monitoring is cut off whenever DG is selected or when flight accelerations and attitudes would cause errors during compass detection. In the compass mode, no indication of compass monitoring cut-off is given and the annunciator is free to move as the aircraft moves in azimuth; no attempt should be made, therefore, to use the manual synchronising facility during accelerating, decelerating, banking or turning flight.

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(vi) If, whilst operating on the standby inverter supplies, the MRG is switched off, the interlock circuit functions making it impossible to restart the MRG unless main AC supplies can be made available.

(*c*) *Attitude Indicator F4C*

(i) The attitude indicator, which is operated by signals from the MRG, gives a continuous indication of pitch and roll by a roller blind presentation.

(ii) A translucent orange disc, bearing two arrows, indicates power failure to the attitude indicator and the navigation display. It is normally covered by a black disc which lifts up to show the orange disc if power is lacking. One arrow points to the attitude indicator and the other to the navigation display, for which no separate warning device is fitted.

(iii) Depending upon modification state, partial failure of signals from the MRG is shown by the roller blind either changing to all black or rotating continuously at a steady rate, indicating elevation signal failure and bank signal failure respectively.

(iv) A flight director bead, for use in conjunction with the flight control system, is incorporated in the instrument (see Part 1, Chapter 14).

(v) A slip indicator is fitted above the face of the instrument.

3 Navigation Display

The instrument combines the function of either a gyro magnetic compass or a directional gyro together with any of the following displays:

ILS
Violet Picture
Direct Tacan
Offset Tacan
Air-to-Air Tacan
Data Link (inoperative)

The modes are selected by a COMP/ILS/TAC/DL mode selector on the instrument.

(*a*) *Compass Mode*

(i) With COMP selected at the mode switch, the display shows only the compass card. A COMP/DG button to the left of the display selects either magnetic compass monitoring or directional gyro. If DG is selected, the window beneath the button shows DG; with COMP selected, it remains blank.

(ii) A compass monitoring annunciator window is on the face of the instrument. With compass selected and synchronised, a dot/cross annunciator slowly oscillates in the window. If DG is selected the annunciator is rigid in the de-energised central position. Fast synchronisation is achieved by the use of a SYN knob at the bottom right of the instrument. The knob must be pressed and turned. If resistance is felt, it indicates turning in the wrong direction.

(iii) At the bottom left of the instrument is a HDG knob which, when pressed and turned, moves a heading selection pointer on the instrument. Any divergence between the aircraft heading and the selected heading is transmitted to the flight control computer for use with the flight director or automatic control.

(iv) With the MRG switch off, the compass card does not rotate in sympathy with aircraft heading changes. To allow for correct ILS and Tacan display orientation, however, it is possible to synchronise the compass manually by use of the SYN knob, provided that AC supply is available to the navigation display. If resistance to turning is felt, select DG until the heading is approximately the same as the aircraft heading. Final synchronisation can be made with the COMP/DG button selected to COMP.

(*b*) *ILS Mode*

With the mode selector set at ILS the ILS presentation appears, framed by the compass card, as a pair of parallel lines representing the runway or localiser beam. When all appropriate ILS selections have been made and signal strength is sufficient for reliability, the BEAM and GLIDE amber lights in the windows at the top right of the instrument are covered by black shutters. The parallel lines move in sympathy with heading changes or displacement from the beam. A localiser datum marker is visible through an aperture in the display and represents the centre of the beam. The datum marker can be set to the beam QDM by pulling *out* and turning the HDG knob. The glide path indicator is represented by a horizontal bar across the display, moving up or down relative to the centre of the display. A blue flashing ILS MARKERS light (activation of which is difficult to detect in daylight) is at the bottom of the display.

(*c*) *Violet Picture*

The ILS localiser indicator may be used for UHF homing with ILS set on the mode selector and the VP/ILS IND switch set to VP. (See Part 1, Chap 15).

(*d*) *Tacan*

The Tacan display is a series of concentric arcs, each representing 20 NM distance from the homing point. Distance to the homing point is read off at the centre of the display and is also repeated in a RANGE NM window at the top left of the display. A line bisecting the range arcs indicates the magnetic bearing of the homing point when read against the compass card. Three Tacan modes may be obtained by use of TAC or DL mode selections on the navigation display, provided Tacan is switched ON, the appropriate channel selected and the air-to-air/air-to-ground switch set as required. The modes are:

(i) *Offset*

With TAC selected, the display indicates the aircraft's range and magnetic bearing from a selected homing point from the Tacan beacon. The selected homing point is set in by an offset computer located on the UHF control panel. The computer has two controls and two veeder counters by which the range and bearing of the homing point from the Tacan beacon are selected.

(ii) *Direct*
With DL selected, (Data Link not fitted), direct range and bearing from the Tacan beacon is presented.

(iii) *Air-to-Air*
With DL selected and the appropriate channel and A/A set on the control unit, air-to-air range is presented in the RANGE NM window. Range cannot be read off the display as bearing lock-on cannot be achieved in this mode and the roller blind rotates at about 3 RPM.

(*e*) *Data Link* — inoperative.

4 Air Data System

(*a*) *General*

(i) The air data system measures pitot/static and static pressure signals, converts them into electrical signals by transducers and passes them to an air data computer. The computer transforms the signals into suitable output for the speed display, the height and rate of climb display, the AI23B/C and the Flight Control System computer.

(ii) The system is powered by 115 volt, 3-phase AC controlled by the INST MASTER switch. There is no standby supply if AC failure occurs.

(*b*) *Speed Display*

The speed display consists of a white strip moving horizontally across a fixed IAS scale above the strip and a moving mach number scale below the strip. The white strip is not visible below 80 knots. A white spot ahead of the strip should be located between two white reference bars on the left of the display when the aircraft is at rest with the INST MASTER switch on. A transparent tape behind the strip carries an indicator to show Data Link Command mach number. This mach number is repeated on a counter at the bottom left of the display. Both the transparent tape and the repeater are inoperative. An orange disc appearing in the centre of the IAS scale denotes power failure to the display.

(*c*) *Height and Rate of Climb Display*

(i) The height and rate of climb instruments present normal indications to the pilot. When power supplies to the display are lacking an orange disc appears to replace the O scale mark on the height dial.

(ii) Two windows at the top of the display, labelled COMMAND and TARGET show Data Link command height and target height respectively (inoperative).

5 Standby Artificial Horizon and Direction Indicator System

(*a*) The standby artificial horizon and direction indicator are fitted to meet the case of failure of the MRG. The system is controlled by the INST MASTER switch and is powered by 115 volt AC or, on AC failure, by the DC busbar (also via the INST MASTER switch), the change-over being automatic. Should both AC and 28 volt DC fail, the control unit automatically selects and uses power from the aircraft main battery supply (via the main DC busbar). After reversion to the battery supply has taken place the battery continues to supply the direction indicator and artificial horizon via the DC busbar, even when the standby generator is switched on. However, if the actions detailing load shedding listed in the FRC are correctly performed, the standby generator continues to keep the battery fully charged. ▶◀ The system cannot be started by DC power and no attempt should be made to test the system on the ground unless power is supplied from an AC source.

(*b*) The Mk 6H standby artificial horizon, on the left instrument panel, incorporates a fast erection button and an orange and black striped off flag. The off flag disappears about 10 seconds after switch-on and the instrument is ready for use about 50 seconds after switching on. To restore the gyro axis, press the FAST ERECTion button keeping it pressed until fast erection is complete. FAST ERECTion should only be used in straight and level unaccelerated flight.

◀ Note: It is possible for the FAST ERECTion button to stick in intermittently against the return spring; ensure that it returns out after being depressed. ▶

(*c*) A direction indicator is fitted on the windscreen left member. The synchronising button of this instrument is also a fast erection push switch. When used for fast erection, a blue light on the indicator comes on. On releasing the button, if the blue light goes out, the instrument is ready for use. If the blue light remains, auto fast erection takes place. The blue light goes out when the instrument is ready for use.

6 Standby Pitot/Static System

A separate pressure head and static source supply a conventional ASI on the left coaming panel and a conventional altimeter on the left instrument panel. These instruments are provided to meet the case of failure of their equivalent IFIS instruments. It should be noted that these standby instruments incur large errors at supersonic speeds.

7 Emergency Compass

An E2B compass is on the windscreen right frame member.

8 Pitot Heaters

A PITOT HEATER, NORMAL / OFF / STANDBY switch is on the starter panel. When NORMAL is selected, 28 volt single phase AC supplies the heaters of both pressure heads. If AC supply fails selecting STANDBY connects the DC supply to the heaters.

9 Miscellaneous Instruments

(*a*) An accelerometer is fitted to the right coaming panel, and a cabin altimeter is on the right console.

(*b*) Command mod/Lightning/0227/STC introduces an additional accelerometer as a temporary fitment to be used during aerobatic displays. It is mounted on the left coaming above the air-to-air refuelling lights. When fitted, the accelerometer obscures the standby DI.

(*c*) Post-mod 4150, a ram air temperature gauge is fitted on the windscreen right member and is supplied from the DC busbar. The gauge is graduated 0 to 150°C and is connected to a thermal sensing probe mounted on the rear spine. Use of the gauge facilitates compliance with the equipment ram temperature/time limitations given in Part 2, Chapter 1.

(*d*) Mod 4808/4664 repositions the accelerometer to the windscreen right member and the ram air temperature gauge (post-mod 4150) to the starboard coaming panel.

10 Monte Carlo Stop Watch FE

Post-mod 4635 a Monte Carlo type stop watch is mounted on the left canopy arch adjacent to the high intensity lamp.

PART I

Chapter 9—GENERAL EQUIPMENT AND CONTROLS

Contents

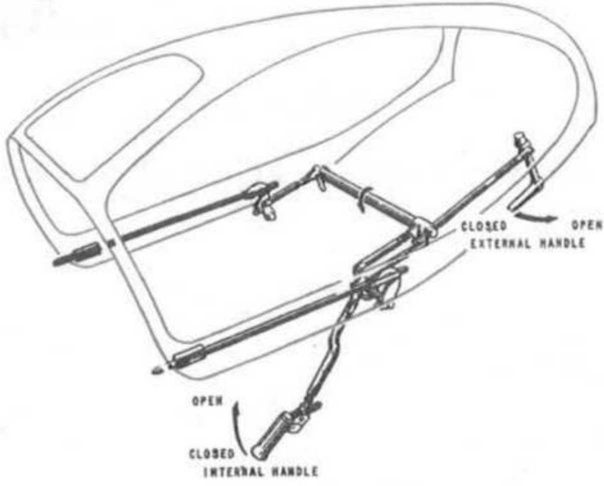
Description

1 Canopy control and operation

(*a*) The jettisonable clam-shell canopy is hydraulically operated and electrically controlled. The canopy is opened or closed from inside the cockpit by a CANOPY handle and toggle switch at the left of the seat pan. When the handle is pulled up, the canopy is unlocked and a toggle switch is exposed in the head of the handle. When the switch is moved to OPEN, the canopy hydraulic jack operates to open the canopy. To close the canopy, set the toggle switch to CLOSED and when the canopy has closed push the handle down to lock the canopy. The toggle switch is spring-loaded to a central off position allowing intermediate positions between fully open and fully closed to be selected.

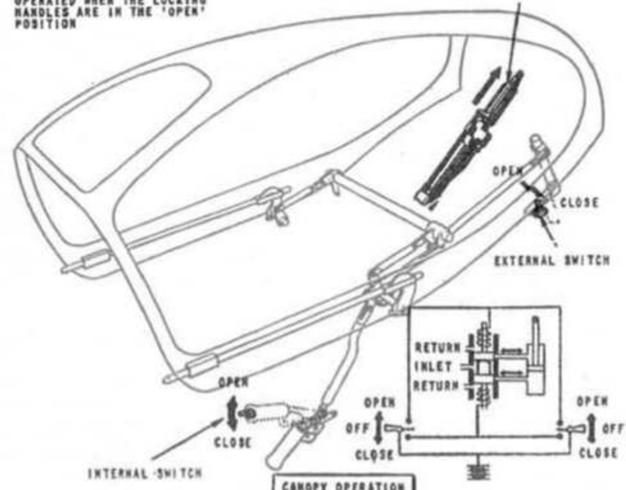
(*b*) For external canopy operation a CANOPY EXTERNAL RELEASE HANDLE is behind a panel on the port ◀side of the fuselage spine. The canopy is opened by rotating the handle and operating a CLOSE/off/OPEN switch to OPEN: the reverse procedure is used to close the canopy.▶

(*c*) When closing and locking the canopy, the last few degrees of downward movement of the CANOPY handle causes the canopy seal to be inflated (see Part I, Chap. 10, para. 3).



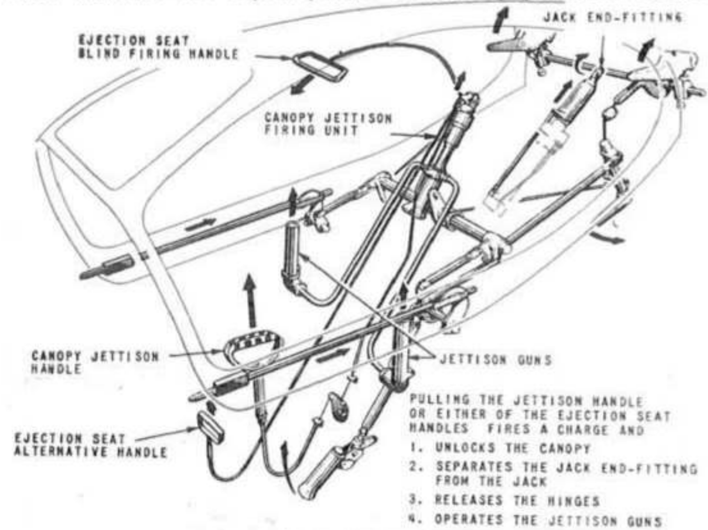
CANOPY LOCKING/UNLOCKING

THE SWITCHES CAN ONLY BE OPERATED WHEN THE LOCKING HANDLES ARE IN THE 'OPEN' POSITION

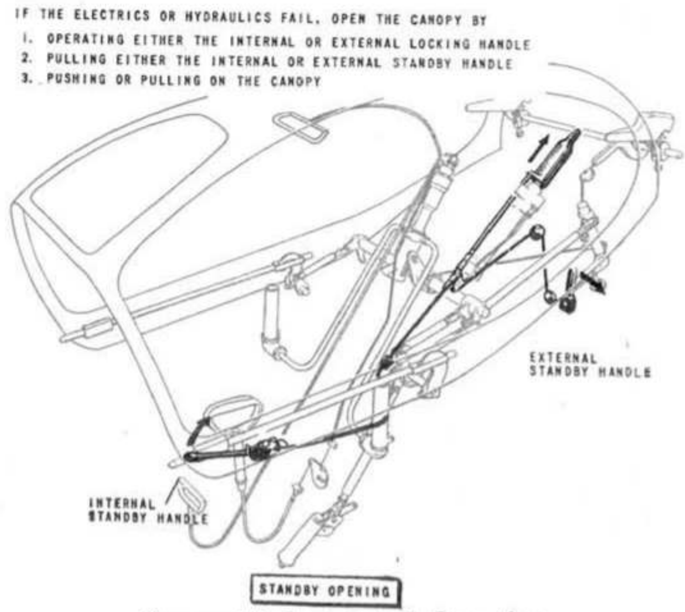


CANOPY OPERATION

CANOPY MECHANISM AND OPERATION



CANOPY JETTISON



STANDBY OPENING

Canopy Mechanism and Operation

(*d*) An electric buzzer on the starboard side of the canopy bulkhead gives audible warning whenever the canopy is being operated.

(*e*) A CANOPY warning is on the AWP. It illuminates whenever the canopy is unlocked, provided the DC bus-bar is energised.

(*f*) Two mechanical indicators, marked CANOPY FREE—LOCKED, are on each inside face of the lower canopy frame to show the position of the canopy shoot-bolts.

2 Canopy emergency operation

(*a*) An EMERGENCY CANOPY JACK RELEASE lever, located on the cockpit port wall, may be used to open the canopy on the ground if the normal system fails. The lever mechanism incorporates a ratchet which holds the lever out after use and thus prevents re-locking of the jack end. A ratchet release button is fitted in the end of the lever: it is only used to reset the jack end release. The lever should be pulled inboard to free the jack-head, the normal CANOPY handle pulled up and the canopy opened (physically from outside as it is almost impossible to raise the canopy while sitting in the seat). A similar release lever for external use is adjacent to the CANOPY EXTERNAL RELEASE HANDLE in the port spine.

(*b*) If, in hydraulic or electrical failure conditions, the canopy opens only slightly after a normal selection, it may not be possible to free the jack head either by use of the EMERGENCY CANOPY JACK RELEASE or by the canopy jettison system. Proceed as follows:—

(i) Close the canopy by selecting and holding CLOSED on the OPEN—CLOSED switch (manual assistance may be necessary to close the canopy).

(ii) Push the CANOPY handle down to lock.

(iii) Operate the EMERGENCY CANOPY JACK RELEASE.

(iv) Pull the CANOPY handle upwards taking care not to operate the OPEN—CLOSED switch.

(v) With external aid, physically lift the canopy.

(*c*) If it is vital to vacate the aircraft quickly, do not attempt to unlock the canopy. Use the CANOPY JETTISON handle, but see para. 3(~~*b*~~). (c)

3 Canopy jettison

(*a*) The canopy may be jettisoned by pulling a yellow and black striped CANOPY JETTISON handle at the left of the seat pan or, on ejection, by pulling the face screen or seatpan firing handles of the ejection seat. An external jettison handle, labelled EMERGENCY CANOPY JETTISON is fitted flush with the skin on the port side of the fuselage below the cockpit. Operating any one of these controls withdraws the sear of the jettison firing unit which fires a charge to unlock the canopy, free the hinges and the hydraulic jack-end fitting, jettisoning the canopy by means of two jettison jacks.

◀(*b*) During an ejection attempt, if the canopy fails to jettison after both seat firing handles have been pulled it is likely that the canopy jettison system is faulty; in this event use the normal opening handle to remove the canopy.▶

(*c*) If the nosewheel has collapsed and the canopy is jettisoned, there is a probability of the canopy falling back on to the cockpit.

4 Internal lighting

(*a*) The normal internal lighting is controlled by dimmer switches on the starboard console and starboard wall. The switches are as follows:

(i) Two switches, labelled PORT and STBD, control the pillar lamps and flood lamps for the port and starboard consoles respectively.

(ii) A FWD PORT switch controls the pillar lamps and flood lamps for the port instrument and port coaming panels and the hydraulic pressure gauge.

(iii) A FWD STBD switch controls the pillar lamps and flood lamps for the starboard instrument panel, the starter panel, the UHF controller panel, the voltmeter and the accelerometer. This switch also dims the attention light whenever cockpit lighting is on.

(iv) A CENTRE switch controls the internally mounted 4 volt lighting for the IFIS display.

(v) A STBY DI, E2B switch controls the integral lamps in the standby direction indicator and E2B compass.

(*b*) A FREQ CARD LIGHT control is above the AWP. A push-pull switch controls the lamp.

(*c*) A floodlamp on a swinging arm is below the AWP. The lamp, which can be used to illuminate the pilot's knee pad, is controlled by a toggle switch under the arm.

5 Emergency internal lighting

(*a*) If the normal cockpit lighting fails, cockpit emergency lighting may be selected by switching ON the EMERGENCY LIGHTS switch on the port coaming panel to illuminate the instrument panel and consoles by a distribution of amber floodlamps. The electrical supply is taken from the emergency battery. With the switch at ON, the E2B compass and standby direction indicator lighting supply is transferred from the DC bus-bar to the emergency battery; brilliance is still controlled by the STBY DI, E2B switch which must remain at the ON position.

(*b*) Two high intensity white anti-dazzle lamps are fitted to the windscreen arch, port and starboard, and are controlled by a BRIGHT/OFF/DIM switch on the starboard instrument panel.

6 External lighting

(*a*) The external lighting is taken from the DC supply and is controlled by the NAV LTS and TAXI LTS switches on the starboard console. The NAV LTS switch is a 3-position ALL/ON/OFF/TIPS switch which, when set to ALL ON, lights the port, starboard and tail navigation lights and flashes the anti-collision lights. With TIPS selected the navigation lights only are lit. The TAXI LTS switch is two-position ON/OFF.

(*b*) When the air-to-air refuelling probe is fitted, it can be illuminated by use of the PROBE LIGHT switch on the port coaming panel. There is no electrical supply to the switch unless the NAV LTS switch is set to ALL ON or TIPS.

PART I

Chapter 10—PRESSURISATION AND AIR CONDITIONING

Contents

Description

1 Cabin air system

(*a*) Air tapped from the engine compressors is used to pressurise the cockpit and regulate the cockpit temperature. An electrically-operated shut-off valve controls the supply of air. With the shut-off valve selected open, temperature control is effected by passing the bulk of the air flow through a refrigeration system and routing the remainder to by-pass the refrigeration system, the latter flow being controlled by a variable-position electrically-operated valve. The regulated hot air from the by-pass is then mixed with the cold air from the refrigeration side of the system and delivered to the cockpit via a water extractor.

(*b*) The refrigeration system, comprises a pre-cooler using ram air as a cooling medium, a water boiler and a cold air unit.

(*c*) At the cockpit, the conditioned air is delivered through two perforated spray nozzles and through two ducts on either side of the control column pedestal.

(*d*) A ram air valve is on the cockpit port wall; when the valve is selected open it allows air at atmospheric pressure and temperature to enter the cockpit.

(*e*) Partially cooled air from the same supply as that used for pressurisation and air conditioning is used for the anti-G and AVS systems and to pressurise the AI23B radar unit compartment (see relevant chapters).

2 Pressure controller

(*a*) A pressure controller on the forward pressure bulkhead controls the cockpit pressure automatically by regulating the action of an air discharge valve. At about 7,500 ft. the unit commences to control the discharge of air and as height is increased the cockpit pressure builds up until the full differential of 4 PSI is reached at about ◀32,000 ft., above which it is maintained constant.

(*b*) Severe over-pressurisation is normally prevented by relief valves built into the air discharge valve, which open if cabin pressure exceeds approximately 4·5 PSI. If, over-pressurisation occurs, the following action must be taken:

(i) At cabin differential pressures above 4·0 PSI but below 6 PSI, restrict speed to 1·3M/500 knots.

(ii) At cabin differential pressures of 6 PSI or above, reduce speed to below 1·3M/500 knots and return to base and land.

(*c*) Corresponding values of aircraft altitude and cabin altitude are as follows:—

Aircraft altitude	*Cabin altitude at 4 PSI*	*Cabin altitude at 6 PSI*
35,000 ft.	17,750 ft.	11,500 ft.
45,000 ft.	22,500 ft.	15,500 ft.
55,000 ft.	26,500 ft.	18,000 ft. ▶

3 Canopy seal

(*a*) When the canopy is closed, final movement of either the internal or external canopy locking handles operates a pneumatic valve to allow air to inflate the canopy seal. Similarly, initial movement of either handle when unlocking the canopy will cause the seal to deflate. A storage cylinder in the system allows the seal to be inflated when the engines are stopped.

(*b*) If the canopy handle is not fully home during locking, the canopy seal may not fully inflate and pressurisation may be lost.

Controls and Indicators

4 Cabin air controls

(*a*) Air conditioning of the cockpit is controlled by a CABIN AIR switch on the starter panel. With the switch selected ON the electrically-operated shut-off valve is opened to allow air to pass to the cockpit.

(*b*) A RAM AIR—OPEN/CLOSED control is on the port console.

(*c*) A Mk. 21 cabin altimeter is on the starboard console. It indicates the cockpit pressure in terms of altitude.

5 Cockpit temperature control

(*a*) A CABIN AIR TEMP controller is located on the starboard console. The controller is divided into a black AUTO sector and a red MANUAL sector. COOL and WARM are marked at opposite ends of each sector and a central FIXED position is marked on the MANUAL sector.

(*b*) When the temperature control selector is in the MANUAL sector it is self-centring to the FIXED position. The control must be held to either WARM or COOL until the desired temperature is attained. The valve takes about 20 seconds to move from full WARM to full COOL and a time delay of about 1 minute will occur before the full effect of a temperature change is felt.

(*c*) When the temperature control selector is in the AUTO sector it remains in the position selected and the temperature is automatically regulated by the temperature control valve.

6 Pressurisation failure indication

A serious reduction in cockpit pressure is indicated by the illumination of the CPR warning on the SWP.

Management of the System

7 Cabin air and temperature control

(*a*) The CABIN AIR switch is set to on before engine starting when the starter bar is lifted. The switch should normally be left in this position for the duration of the flight. Before take-off ensure that the ram air valve is closed.

(*b*) Set the CABIN AIR TEMP controller into the AUTO sector as required. Variations in the selected temperature can be expected with large changes in speed or altitude. If a new temperature is selected there will be a delay of about one minute before it is obtained.

(*c*) The MANUAL sector of the CABIN AIR TEMP controller is not normally used. If the AUTO facility fails, however, the selector should be moved into the MANUAL sector and held at either COOL or WARM as required. There will be a similar delay between selecting and obtaining a new temperature.

Malfunctioning of the System

8 Pressurisation failure

Pressurisation failure is indicated by the CPR warning appearing on the SWP and can be confirmed by the cabin altimeter. If the failure occurs at high altitude an immediate ◀descent must be made to 35,000 ft. cabin altitude and the oxygen mask toggle set to the down position. The regulator should be set to 100% and EMERGENCY and the▶ DEMIST lever set to ALL ON. The descent must then be continued to a cabin altitude of 25,000 ft. or below for decompression sickness reasons and a return to base made at the lowest practicable altitude. Ram air may be used as necessary for ventilation.

9 Temperature control failures

If overheating or overcooling occurs, select MANUAL on the temperature controller and hold at COOL or WARM as necessary for 10–15 seconds. If the temperature responds, leave the controller in the MANUAL sector making manual adjustments as required. If the overheating or overcooling persists, however, proceed as for a pressurisation failure and select CABIN AIR to off.

10 Emergency decompression

If it is necessary to decompress the cockpit, set the CABIN AIR switch to off and open the ram air valve.

11 Smoke or mist in the cockpit

(*a*) If smoke occurs in the cockpit and appears to be coming from the air diffusers, select the CABIN AIR switch to off until the smoke disappears. If the smoke is coming from any other source open the ram air valve. In both cases cockpit pressure will be lost whilst the CABIN AIR switch is off or the ram air valve open and the pilot will therefore be required to proceed as for pressurisation failure. The ◀oxygen regulator should be selected to 100% and EMERGENCY whilst smoke persists. ▶

(*b*) If misting occurs select the DEMIST control to ALL ON. When misting disappears reselect to OFF or TOP ON (*see* Chapter 11, para. 2(*b*)).

RESTRICTED

PART 1

Chapter 11—WINDSCREEN DEMISTING AND RAIN DISPERSAL

Contents

Description

1 Windscreen heating

(*a*) The front windscreen is heated electrically by inter-laminar elements fed from the AC supply system. With the W/SCR FRONT switch on the starter panel selected on, the heating to the windscreen is automatically controlled and regulated. The electrical supply to the elements is controlled by the 165 knot (pre mod 4398 150 knot) pressure switch associated with the undercarriage circuit, so that only half heat is applied below this speed and full heat above it. FE

(*b*) The port windscreen panel is electrically heated by inter-laminar elements fed from the AC system and controlled by the W/SCR SIDE switch on the starter panel. A temperature control unit in the circuit automatically regulates the heating of the panel.

2 Hot air demisting

(*a*) The port and starboard side windscreen panels and the canopy top panel are fitted with hot-air sprays. Hot air from the engine compressors is passed by two pipes, one for the side windscreen sprays and one for the canopy spray. Each pipe has a stop valve and the canopy hot air passes through a pressure reducing valve. A 3-position DEMIST control handle is on the port console. The selected positions are:

ALL ON — Full flow to the canopy and to the side windscreen

TOP ON — Full flow to the canopy and partial flow to the side windscreens.

OFF — Heating supply shut off.

(*b*) Hot-air demisting of the side windscreens should only be used if the electrical demisting is inadequate or fails. As soon as demisting is completed, select the DEMIST control to OFF or TOP ON.

3 Canopy interspace demisting

Air in the canopy interspace is dried by passing it through chemical air driers. The air is circulated by a blower motor controlled by the W/SCR SIDE switch on the starter panel.

4 Rain dispersal

(*a*) The system improves forward visibility by completely clearing a centre strip of the windscreen from top to bottom in light continuous rain, or by completely clearing the lower area of the strip and partially clearing the upper area in heavy rain. Air, tapped from both engine compressors, is directed to a nozzle assembly located in the fuselage skin upstream of the windscreen. This assembly has two angled nozzles away from the windscreen and a wiper nozzle at the base of the windscreen; the efflux from the angled nozzles creates high turbulence of the airflow, breaking up large rain droplets, whilst the jet of hot air from the wiper nozzle keeps an area of the windscreen free from moisture.

(*b*) The system is controlled by the DE-ICE/OFF/RAIN DISPL switch on the port console, and by two pressure-operated switches which automatically limit operation to below 10000 ft altitude and 350 knots airspeed. A magnetic indicator on the throttle box shows white/R with the system in operation.

(*c*) The system should be exercised before each flight during the "Checks after starting" thus reducing the possibility of valve seizure due to corrosion.

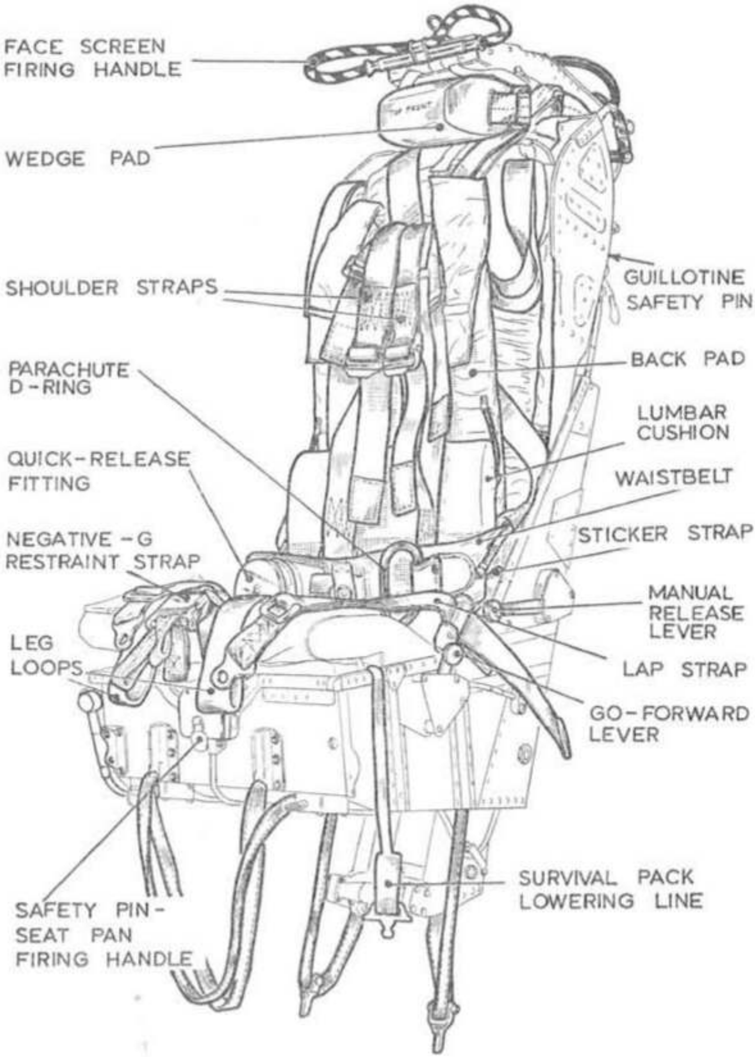
(*d*) Limitations on the use of rain dispersal are given at Part 2 Chap 1.

PART 1

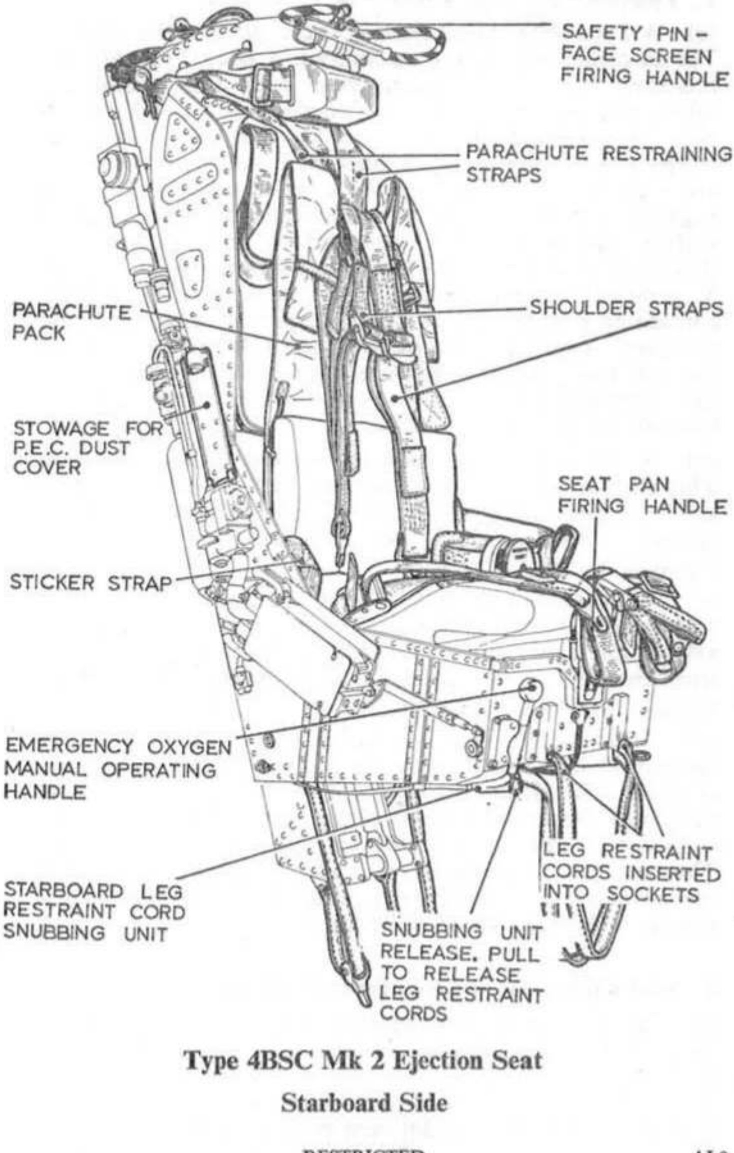
CHAPTER 12 — AIRCREW EQUIPMENT ASSEMBLY AND ASSOCIATED SYSTEMS

Contents

WARNING: Whenever the aircraft is on the ground, the ejection seat must be 'safe for parking', ie the safety pins must be fitted to the face-screen firing handle, to the seat-pan firing handle and in the guillotine unit sear.



Type 4BSC Mk 2 Ejection Seat
Port Side



Type 4BSC Mk 2 Ejection Seat

Starboard Side

Ejection Seat

1 Ejection Seat Type 4 BSC Mk 2—General

(*a*) A Type 4BSC Mk 2 ejection seat is fitted in the aircraft. The associated parachute assembly is a Back Type Mk 45; this embodies a horsehoe parachute pack and a combined safety and parachute harness fastened by a quick release box, the pack being held in position by two restraining straps at the upper end. A back pad and lumbar cushion are included in the harness for comfort and support. A negative–G restraint strap is fitted to restrain the pilot against vertical movement when subjected to negative-G forces; this strap should be tensioned during strapping-in. A Type V personal survival pack (PSP), complete with cushion, containing a dinghy and survival equipment, is housed in the seat pan. Two leg restraint cords are fitted to the front of the seat pan. An emergency oxygen bottle is mounted on the starboard seat beam. A guillotine unit is incorporated to assist in manual separation.

(*b*) A pack restraint spreader system is incorporated. This consists of two arms mounted just beneath the parachute support arch which swivel forward giving quicker release of the parachute pack restraining straps during separation after an ejection.

(*c*) At the rear of the seat is the 80 ft/sec ejection gun. The seat has two firing handles, both of which are provided with safety pins. One, which has an integral face screen, projects from the front of the drogue container; the other is positioned centrally on the front face of the seat pan.

(*d*) A drogue gun is mounted on the port seat beam. For ejection at high speed below 10000 ft a G-controller switch is fitted. This delays operation of the barostat time-release unit until the forward speed of the seat and pilot has been sufficiently reduced to ensure safe parachute deployment. The seat has a ground-level ejection capability provided the aircraft's flight path is parallel to the ground with a minimum forward speed of 90 knots.

2 Seat adjustment and lean-forward release

(*a*) The seat height may be adjusted by an electric motor controlled by a switch unit mounted just above and forward of the personal equipment connector (PEC) on the right of the seat. The seat pan moves relative to the headrest and can therefore accommodate different body lengths, at the same time ensuring that the occupant's head is correctly located whatever the position of the seat pan. The switch is spring-loaded to the central (off) position and operates in the natural sense, i.e. a downward movement of the switch lowers the seat and vice-versa.

(*b*) An inertia-reel go-forward system is fitted. It is controlled by a three-position spring-loaded lever on the port side of the seat pan. When the lever is moved fully forward and then allowed to return to the central position the pilot can lean forwards or backwards at will. An automatic inertia device locks the harness during an ejection or if more than 2½G in the forward plane is experienced. Movement of the lever to the rear position brings into action the snubbing unit in the top harness lock thus preventing further forward movement: as the wearer leans back the harness is locked in the rearward position.

3 Personal equipment connector (PEC)

The PEC is fitted to the starboard side of the seat pan and provides R/T, intercomm, oxygen and supplies to both the air ventilated and anti-G suits. It consists of three components; a seat portion which is fitted permanently to the starboard side of the seat pan; a man portion which is fitted to the flying clothing; and an aircraft portion which disconnects automatically on ejection. The system allows the services to be either connected or disconnected in one action. On ejection the services are automatically disconnected. Emergency oxygen is supplied through the PEC but is only provided on ejection or when selected. The leg restraint system is also interlinked; disconnecting the man portion of the PEC releases the restraint cords.

4 Combined harness quick-release box

The quick-release box, when fastened, secures the combined restraint and parachute harness and therefore secures the pilot to the seat. *On no account must this box be operated when carrying out manual separation in the air since this will free the pilot from both seat and parachute.*

5 Manual separation

(*a*) Manual separation of the parachute from the seat is effected by a guillotine unit. The unit is positioned on the port side of the drogue container and incorporates a sear safety pin for use during servicing. The parachute withdrawal line passes through the guillotine unit, the sear of which is connected to the parachute pack by a static line. The parachute withdrawal line is severed by the guillotine unit firing as a result of forward movement of the pilot in the seat during manual separation.

(*b*) A manual separation lever is at the port side of the seat pan. When pulled out of its gate and upwards, the combined harness is released from its three attachment points and the parachute restraining straps are also released; at the same time the man portion of the PEC and the leg restraint cords are released. As the pilot leaves the seat, the guillotine unit operates.

NOTE: If the lever is inadvertently operated, the seat must subsequently be serviced by suitably qualified maintenance personnel.

6 Rip-cord D-ring

The rip-cord D-ring is provided for manual opening of the parachute should the ejection seat fail to fire or the automatic opening of the parachute fail after ejection.

7 Leg restraint cords and adjusting controls

(*a*) The cords ensure that the pilot's legs are drawn back automatically and restrained close to the seat pan during ejection thus providing clearance and preventing the legs being blown apart after ejection. The cords are attached to the aircraft structure at their lower ends and pass through the snubbing units beneath the front of the seat pan, which allow the cords to pass freely down through the units but prevent them passing upwards. An adjusting ring on each snubbing unit, when pulled forward, allows the pilot to adjust the cords to give sufficient leg movement for application of full rudder. The ends of the cords plug into units on the forward face of the seat pan.

(*b*) After ejection, the legs are held in position until auto or manual separation occurs; the restraint cords are then free to pull through the garter D-rings.

8 Normal operation of the seat

(*a*) When either firing handle is operated the canopy is jettisoned immediately: this in turn removes an interdictor from the time-delay firing unit (TDFU). Continued pressure on the firing handle is necessary however, to remove the sear on the TDFU after the canopy has gone: after a delay of 0·6 seconds the seat is ejected. The drogue gun, which is operated by a static rod, fires 0·5 seconds later. The 'double pull' effect is unlikely to be noticed if the system operates normally. The main purpose of this system is to ensure that the seat is not in a 'live but unfired' condition if the canopy fails to jettison. If the seat pan firing handle is used, press the head firmly back against the head rest cushion to minimise the risk of spinal injury on ejection. Leave the feet on the rudder pedals; drawing the feet back could lead to injury.

Note: The seat cannot eject if the canopy fails to jettison.

(*b*) On ejection the aircraft portion of the PEC is disconnected from the seat portion: on separation the man portion of the PEC disconnects from the seat.

(*c*) ▶◀ Provided the IFF/SSR transponder is operating, the emergency coding signal is automatically triggered on pilot ejection.

(*d*) As the seat is ejected a static rod readies the barostatic time-release mechanism. When the seat has descended on the drogue to 10,000 feet (or 5000 metres, depending upon modification state) or at once if ejection has taken place below that height and provided the g stop has not operated, the barostat removes an obstruction to the gear train of the mechanism allowing it to operate. After 1·25 seconds the combined harness is released from the seat, the seat-stabilising drogues are freed and, by a drogue link line, the face screen is disconnected and the parachute deployed. The pilot may momentarily be prevented from leaving the seat by two restraining straps until deployment of the parachute lifts him clear of the seat.

ANTI-G AND AVS SYSTEMS

9 Anti-G System

(*a*) Partially cooled air from the cabin air conditioning system is fed via a stop valve to the barometric/anti-g valve. When the latter operates, air at controlled pressure is admitted via the PEC to the pilot's anti-g suit. On this aircraft the anti-g suit is inflated when positive accelerations are applied or if the cabin altitude exceeds 35,000 feet.

(*b*) The stop valve is controlled by a handle below the right console, and is provided with a spring-loaded catch retaining it in the OFF (forward) or ON (aft) positions.

The stop valve is for emergency use only and should normally be in the ON position. In the event of failure of the barometric/anti-g valve and a build-up of pressure in the anti-g suit the stop valve should be placed in the OFF position when the suit will deflate. If the stop valve is inadvertently left at OFF there will be no protection to the lower part of the body in the event of loss of cabin pressure at high altitude.

(*c*) (i) Under g conditions the barometric/anti-g valve applies a pressure to the anti-g suit according to the severity of the g force applied. A selector valve below the right console, when set to H, admits pressure to the suit at 1·25 PSI per g applied (but not below approximately 1·9g applied) and, when set to L, at 1·05 PSI per g applied (but not below approximately 2·25g applied). The setting of the valve is determined by individual requirements; it should not be necessary to change from one setting to the other in flight.

(ii) If loss of cabin pressure occurs at high altitude, the barometric/anti-g valve applies pressure to the anti-g suit at a value of approximately 0·5 PSI greater than the oxygen system applies pressure to the lungs and pressure jerkin.

(*d*) A test button on the selector valve permits the system to be tested provided an engine is running, the PEC is connected and the stop valve switched ON. Pressure should be felt in the suit.

10 Air Ventilated Suit System

(*a*) Partially cooled air tapped from the cabin air conditioning system is used to provide the air ventilated suit supply. A flow control selector is on the right console. The rotary control is turned clockwise to OPEN. The temperature of the air supply to the suit is controlled by the CABIN AIR TEMP controller.

(*b*) An AVS ground air supply connection is on the right side of the fuselage.

OXYGEN SYSTEM

11 General Description

A liquid oxygen system is installed. A 3·5 litre oxygen container and associated evaporator and stabilising system are housed in the equipment bay. The system provides

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gaseous oxygen at a suitable temperature and pressure to the Mk 17F (pre Mod 4763 Mk 21B) demand regulator positioned on the starboard console.

12 Contents indication

(*a*) Liquid oxygen is stored within the inner shell of the double-walled container. The inner shell and a perforated gauging shell form the two plates of a variable capacitor in a capacitance bridge to measure the level of the liquid. The capacitance is converted by the gauge control unit to give a contents reading on the contents gauge.

(*b*) The contents indicator on the starboard console is calibrated in fractions of tank capacity from 0 to F (full). The dial has two red sectors known as 'failure arcs'. When the indicator is energized the pointer should read 0 or above according to the contents of the container. If the pointer remains in the failure arc below zero, a power failure is indicated; with a capacitance gauging failure the pointer will move toward, or into, the top arc.

13 Pressure demand regulators

(*a*) *Mk 17F Regulator*. The Mk 17F regulator provides normal protection against hypoxia at cabin altitudes up to 42000 feet and affords the user 'get down' protection from a maximum altitude (after depressurisation) of 50000 feet without the aid of pressure garments.

The regulator controls and indicators are as follows:

(i) An OXYGEN SUPPLY—ON/OFF cock, wired to ON, controls the oxygen entering the regulator.

(ii) A pressure gauge indicates the input pressure. This should read between 150-215 PSI when the OXYGEN SUPPLY cock is ON, but may increase up to 270 PSI following a high demand.

(iii) An OXYGEN FLOW INDICATOR shows a white vertical bar when oxygen is being drawn from the regulator on breathing in and black when no flow is taken on expiration or if there is no electrical supply to the regulator. A repeater indicator is on the port coaming panel.

(iv) An air dilution switch marked NORMAL OXYGEN/100% OXYGEN (see (*c*) below).

(v) A 3-position button marked EMERGENCY—PRESS TO TEST MASK (see (*d*) following).

(*b*) (i) The Mk 17F regulator must only be used with the types of oxygen mask and associated clothing as listed at paras 18 and 19. Other types of oxygen mask are not compatible, even at low altitudes. Oxygen is supplied to the pilot on demand under all operating conditions. At cabin altitudes below 10000-12000 feet, oxygen is delivered at ambient pressure and above this height, up to 40000 feet, it is delivered under a slight "safety pressure" to ensure that any mask leakage, perhaps due to a badly fitting or wrongly adjusted mask, is outward.

(ii) Above about 40000 feet, and up to 50000 feet, a pressure of oxygen is maintained in the lungs adequate to prevent hypoxia so that an emergency descent may be performed.

(iii) Following loss of cabin pressurisation at high altitude, the harness toggle of the oxygen mask must be moved to the down position and an immediate descent at maximum rate made to a cabin altitude of below 40000 feet. Continue a rapid descent to below 25000 feet (cabin altitude). The limits of protection afforded by the regulator and type P or Q oxygen mask are given at para (*g*).

(*c*) (i) With the air dilution switch at NORMAL OXYGEN, an air/oxygen mixture of appropriate proportions is delivered up to a cabin altitude of about 32000 feet. Above this altitude only 100% oxygen is supplied. This conserves the oxygen supply but makes recognition of low flow delivery from the regulator difficult.

(ii) With the air dilution switch at 100% OXYGEN, neat oxygen only is delivered at all altitudes. If the cabin becomes filled with smoke or noxious fumes, 100% OXYGEN should always be selected.

(*d*) (i) The regulator characteristics described at (*c*) are obtained with the 3-position EMERGENCY button in the central (normal) position. Moving the emergency button to right or left delivers oxygen at a slightly greater pressure than normal, ie, approximately safety pressure up to 10-12000 ft cabin altitude and approximately double safety pressure above this height.

(ii) The mask may be tested before flight by deflecting the emergency button sideways and holding breath. If the OXYGEN FLOW INDICATOR on the regulator remains black there are no leaks around the mask; if the indicator shows a white vertical bar, the mask requires to be tightened.

(*e*) *Mk 21B Regulator*. The Mk 21B regulator provides normal protection against hypoxia at cabin altitudes up to 45000 feet; combined with a pressure jerkin it affords 'get down' protection from a maximum altitude (after depressurisation) of 56000 feet. The limits of protection afforded by the various combinations of regulator and personal equipment are given at para. (g). Although basically similar to the Mk 17F, the Mk 21B has the following differences:

(i) A lever marked NORMAL/EMERGENCY/ MASK TEST/JERKIN TEST in lieu of the 3-position emergency button (see para (*f*) below).

(ii) Oxygen is used to provide inflation of the pressure jerkin when cabin altitude exceeds 40000 feet approximately.

(iii) A pressure of oxygen, adequate to prevent hypoxia, is maintained in the lungs above about 40000 feet up to 56000 feet to enable an emergency descent.

(*f*) (i) The regulator characteristics described at para (*c*) are obtained when the test lever is at NORMAL. When this lever is set to EMERGENCY, oxygen is delivered at a slightly greater pressure than normal, ie, approximately safety pressure up to 10-12000 ft cabin altitude and approximately double safety pressure above this height.

(ii) The MASK TEST and JERKIN TEST positions of the test lever are intended only for ground checks prior to take-off. To select either of these positions the knurled knob on the end of the lever must first be pulled out. The JERKIN TEST position must not be used unless a pressure jerkin is worn.

◀(*g*) The limits of protection afforded by the regulators are:

Marks 17F and 21B regulators

Personal Equipment	*Limiting altitude*		*Total time from loss of cabin pressure to 40000 ft cabin altitude*	*Initiate descent in:*
	Mk 17F	Mk 21B		
P or Q mask ...	50000 ft	45000 ft	2 min	½ min
P or Q mask with sleeveless jerkin and anti-G trousers ...	—	56000 ft	2 min	½ min

NOTE: Aerodynamic suck will cause the cabin altitude to exceed aircraft altitude by up to 8000 ft if the canopy is lost.

14 Emergency oxygen

(*a*) The emergency oxygen set is located on the back of the ejection seat and feeds its supply into the rear of the seat portion of the PEC and thence into the oxygen mask tube. If failure of the main oxygen system occurs emergency oxygen may be manually selected by pulling up the yellow and black striped knob on the front of the ejection seat. On ejection, the operation is entirely automatic, the movement of the seat up the rails initiating the supply. When the pilot separates from the seat after ejection the separation of the man and seat portion of the PEC breaks the emergency oxygen supply and allows air to be breathed. An emergency oxygen demand regulator is fitted between the emergency oxygen set and the PEC.

(*b*) The function of the demand regulator is to control the supply from the emergency oxygen cylinder and to regulate the delivery in accordance with the physical requirements of the user at altitude. A pressure/contents gauge is fitted at the starboard rear of the ejection seat; the reading on the gauge should indicate in the white sector before flight. ▶

15 Normal management of the oxygen system

With the regulator cock set to ON, oxygen is supplied to the pilot on demand once the man portion of the PEC is connected to the seat portion.

16 Pre-flight checks

NOTE: Comprehensive pre-flight checks of all personal equipment should be carried out on a Godfrey Test Cabinet with the assistance of a Safety Equipment Worker. The following checks are necessary to check aircraft equipment:

(*a*) Check that the oxygen contents gauge shows a minimum of 1·7 litres or ½ full.

(*b*) Check that the main ON/OFF cock on the regulator is wired to ON.

(*c*) Check that the input pressure gauge indicates a supply pressure of 150-215 PSI.

(*d*) Set the air dilution switch marked NORMAL/100% OXYGEN to NORMAL.

(*e*) Check that both OXYGEN FLOW INDICATORS annunciate during breathing.

(*f*) *On the Mk 17F regulator,* using the 3-position button marked EMERGENCY—PRESS TO TEST MASK check as follows:

(i) Deflecting the emergency button sideways check that an increase in safety pressure is felt. Check the flow indicators show BLACK when the breath is held.

(ii) Having moved the harness toggle on the oxygen mask into the down position select the press-to-test facility. Oxygen will now be fed under pressure into the mask. Holding breath check that the flow indicators show BLACK. Return both toggle and button to normal positions.

(*g*) *On the Mk 21B regulator,* using the lever marked NORMAL / EMERGENCY / MASK TEST / JERKIN TEST check as follows:

(i) Setting the lever to EMERGENCY check that an increase in safety pressure is felt. Check the flow indicators show BLACK when the breath is held.

(ii) Having moved the harness toggle on the oxygen mask into the down position select MASK TEST. Oxygen will now be fed under pressure into the mask. Holding breath check that the flow indicators show BLACK. Return both toggle and lever to normal positions.

17 Oxygen system malfunctions

In the event of a system malfunction proceed as follows:

(*a*) *Suspected hypoxia*

Check oxygen connections and mask fit

Operate emergency oxygen

Descend to a cabin altitude of 10000 feet or below

(*b*) *Difficulty in breathing in*
Oxygen Flow Indicator malfunction
Low pressure on regulator gauge

Check oxygen connections, mask hose and mask fit
Select 100% OXYGEN

If able to breathe normally	If unable to breathe normally or white oxygen flow indicator persists
Continue sortie with frequent checks of oxygen contents and pressure	Operate emergency oxygen Descend to a cabin altitude of 10,000 feet or below

(*c*) *Difficulty in breathing out*

Check cabin altitude

If cabin altitude below 40,000 feet	If cabin altitude above 40,000 feet
Mask off face momentarily Deflect EMERGENCY button sideways. (Mk 21B set regulator lever to MASK TEST)	Mask toggle down Descend at maximum rate through 40,000 feet Continue a rapid rate descent to a cabin altitude of 25,000 feet

If difficulty persists	No further difficulty Continue with sortie
East mask off face Descend to a cabin altitude of 25,000 feet	

(*d*) *Toxic fumes*

Select 100% OXYGEN
Deflect EMERGENCY button sideways. (Mk 21B set regulator test lever to EMERGENCY).

Note: If the emergency oxygen set is used and exhausted, inhalation will become difficult; this can be relieved by setting the dilution lever to NORMAL OXYGEN or by disconnecting either the mask or helmet hose.

FLIGHT PROCEDURES

18 The **Safe for Parking** and **Ejection Seat Checks** are given in the FRC. ▶◀

19 Strapping-In Procedures

(*a*) Remove the dust cover from the seat component of the PEC and fit into the stowage on the right-hand of the seat pan. Connect the man portion of the PEC to the seat portion ensuring that it is locked.

(*b*) Connect survival pack lanyard to life preserver or pressure jerkin ensuring that it passes outside the left leg.

(*c*) Pass left-hand leg restraint cord through right leg garter D-ring and plug into the socket above the left-hand snubbing unit. Pass right-hand leg restraint cord through left-hand garter D-ring and plug into socket above right-hand snubbing unit. Pull sharply on each cord to ensure that they are securely locked in their sockets, remembering that unless the man portion of the PEC is locked to the seat component, the leg restraint cords cannot be secured. It is not important which cord is secured first provided that they are not interlaced. If the cords are not long enough to provide the required range of leg movement pull the ring of each snubbing unit and ease the cord forward; slack should be taken up by pulling the cord backwards through the snubbing units.

(*d*) Pull up the parachute back pad and adjust the height of the lumbar cushion. Pass the looped ends of the blue 'Y' section of the negative-g strap over their respective lap strap lugs and connect the lugs to the harness quick-release box, then tighten the straps. Tighten the adjustable part of the negative-g strap.

Note: The lap straps must be as tight as possible.

(*e*) Pass the left leg loop upwards over the inside of the thigh and through the D-ring on the left strap (from the inside of the ring towards the outside of the leg). Bring the end of the leg loop over towards the quick-release box and pass the lug of the left shoulder strap through the leg loop (from the top downwards) and insert the lug into its appropriate slot in the quick-release box. Snug the loop over the lug. Repeat these operations with the right leg loop and shoulder strap.

(*f*) Adjust the sitting height to the desired position.

(*g*) Ensure that the shoulder straps pass under the lobes of the life preserver or pressure jerkin stole. Tighten the inner (blue) straps and then the outer (khaki) straps.

Note: It is undesirable to tighten these straps excessively since this action may arch the back and lead to spinal injury if ejection becomes necessary. The inner straps should not press down unduly on the shoulders but equally there should be no slack. The outer straps should be adjusted similarly to provide a comfortable fit.

(*h*) Operate the lean-forward lever and lean forward. Move the lever fully back, lean fully back and re-tighten the khaki straps. Failure of the harness to re-engage and lock back indicates that the harness is not secure.

(*j*) Put on helmet and connect oxygen supply and Mic/Tel lead.

(*k*) Check that the face screen firing handle is resting on top of the protective helmet. Check that the firing handle can be reached with both hands.

(*l*) Have the face screen and seat pan firing handle safety pins removed and placed in their stowages.

20 Leaving the Seat After Landing

(*a*) Have the ejection seat made Safe for Parking. ▶◀

(*b*) Unlock the harness quick-release fitting, free the straps and return the fitting to the locked position.

(*c*) Disconnect the man portion of the PEC and replace the dust cover on the seat portion.

(*d*) Free the leg-restraint cords from the garters.

(*e*) Disconnect the personal survival pack lanyard.

21 Abandoning the Aircraft in Flight Posture

(*a*) If possible, reduce speed to 250 knots and fly in straight and level or climbing flight.

(*b*) Pull the face screen firing handle fully down over the face ensuring that the elbows are kept well in and that the head and back are pressed firmly against the seat. If the face screen firing handle cannot be reached, pull up the seat pan firing handle.

(*c*) If automatic separation fails, operate the manual separation lever and fall clear of the seat. When clear, pull the rip-cord D-ring.

(*d*) *Seat Fails to Eject*

(i) If the seat fails to eject, pull the firing handle again. If this fails, pull the other firing handle; retain grasp on face screen firing handle.

(ii) If the seat still fails to eject, jettison the canopy using the normal operating handle, then pull either the face screen or seat pan firing handle again.

(iii) If the canopy has left the aircraft and the seat still fails to eject, operate the manual separation lever and abandon the aircraft. When clear, pull the rip-cord D-ring. The emergency oxygen system is disconnected when separation takes place.

(*e*) The method of initiating ejection must depend upon conditions. Use the seat pan handle *always* if conditions are, or are becoming critical — ie below 3,000 feet AMSL, or below 5,000 feet in a dive. Use the face screen handle if at high altitude and, if preferred, when time is *not* critical.

PART 1

Chapter 13—WARNING SYSTEMS

Contents

1 Standard warning system

(*a*) The standard warning system ensures that all important emergencies are brought to the notice of the pilot and groups the warnings on the standard warning panel above the port console. All warnings are operative whenever an AC and DC supply is available and the ENG MASTER and the INST MASTER switches are on. The fire warnings, however, are still operative with the ENG MASTER switch at off but they require the INST MASTER switch to be on.

(*b*) The windows in the standard warning panel (SWP) when illuminated, give warning of the following emergencies:—

FIRE 1	—	Fire in No 1 engine bay (zones 1 and 2)
FIRE 2	—	Fire in No 2 engine bay (zones 1 and 2)
RHT 1	—	Fire in No 1 jet pipe area (zone 3)
RHT 2	—	Fire in No 2 jet pipe area (zone 3)
GEN	—	Both generators off-line
HYD	—	Failure of No 1 and No 2 controls hydraulic systems
◀AP	—	Autopilot malfunction. See note below.▶
CPR	—	Cockpit pressurisation failure
OXY	—	Oxygen failure

◀NOTE: The AP caption illuminates whenever the autopilot G switch trips when engaged AP, if incorrect sequence of selection to AP is made, if failure of the Phase A or C AC supplies occurs or in the event of DC failure to the autopilot. ▶

(*c*) Also on the panel are three buttons, T (test), M (mute) and C (cancel). The M and C buttons contain integral lights. To the rear of the panel are two, F1 and F2, fire extinguisher indicator switch units.

(*d*) A red attention light, which flashes whenever a warning appears, is above the strip speed display. Additionally, an audio warning is heard on the R/T whenever a warning occurs. The audio warning may be attenuated by pulling out the AUDIO WARNING PULL MUTE switch just forward of the SWP.

(*e*) When a warning is received the following indications are given:—

(i) The attention light flashes
(ii) The audio warning sounds
(iii) The appropriate window on the SWP lights up
(iv) The C pushbutton light flashes
(v) If an engine fire has occurred, the appropriate fire extinguisher switch unit lights.

(*f*) The audio and flashing warnings can be cancelled by pressing the C button. All warnings cease except for the appropriate SWP caption and, if a FIRE 1 or FIRE 2 warning, the fire extinguisher switch unit light. If AC change-over now occurs and the warning is FIRE or RHT, the attention-getters will be re-activated. Also, if after pressing the C button another warning is received, the audio and attention light warnings operate to draw attention to the new warning.

(*g*) If a warning appears transiently but triggers the attention lights and audio warning, these will be removed automatically when the warning disappears.

(*h*) When the M button is pulled out (integral light on) all warnings are rendered inoperative except for the fire warnings. The M button must always be in during flight.

(*j*) The T button, when pressed causes all indicator lights and warnings to operate, providing the M button is in, AC and DC power is available, the UHF is switched on and the ENG MASTER and INST MASTER switches are on. When the T button is released all warnings will disappear except for the attention lights, the audio warning and any warning which would normally be energised due to the aircraft condition in which the testing was done.

(*k*) A DAY/NIGHT selector is at the aft end of the panel.

2 Auxiliary warning system

(*a*) The auxiliary warning panel (AWP) groups the less important warnings on a panel mounted above the starboard console. All warning circuits on this panel are DC operated.

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AP
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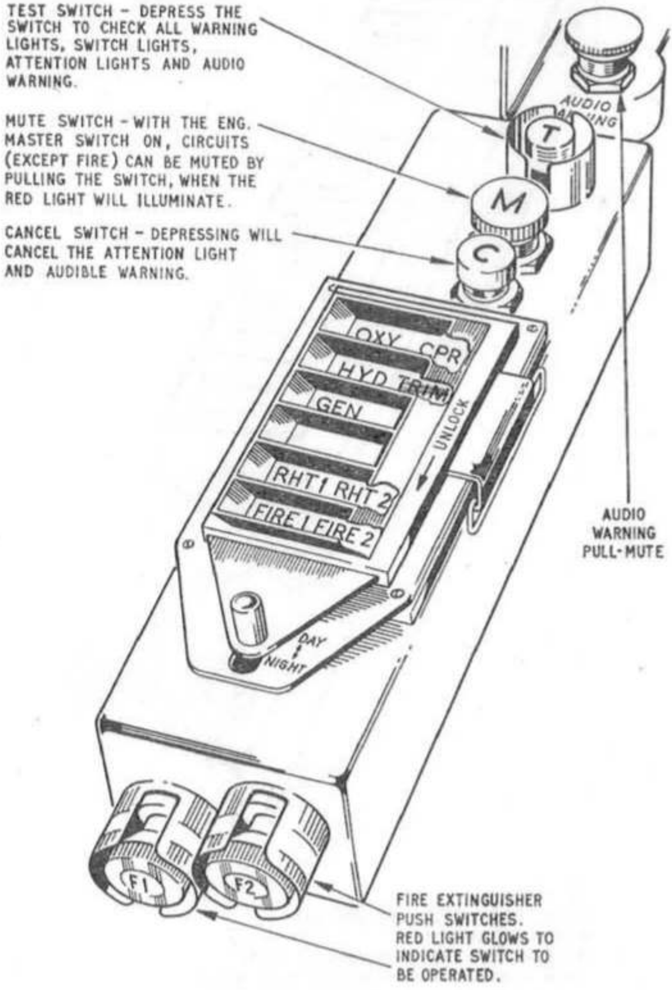
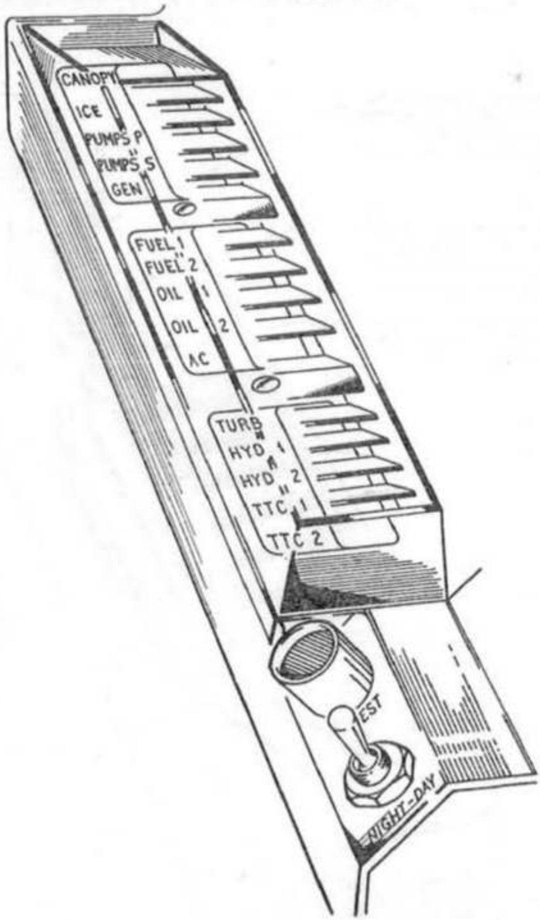


STANDARD WARNING PANEL



TO TEST :-
PUSH TEST SWITCH AND OPERATE
NIGHT/DAY SWITCH.
AT 'DAY', TWO LAMPS ASSOCIATED WITH
EACH WARNING CIRCUIT SHOULD GLOW UNDER
THE AMBER-COLOURED PLASTIC WINDOWS.
AT 'NIGHT', TWO LAMPS UNDER EACH CAPTION
LIGHT UP AND THE CAPTIONS APPEAR RED.

◀AUXILIARY WARNING PANEL▶

(*b*) The panel windows when illuminated, give warning of the following failures:

CANOPY —	Canopy unlocked warning	
ICE —	Icing conditions warning	
PUMPS P —	Loss of fuel transfer pressure to collector box (port)	
PUMPS S —	Loss of fuel transfer pressure to collector box (starboard)	
GEN —	Main generator failure	
FUEL 1 —	Booster pump pressure warning (port)	
FUEL 2 —	Booster pump pressure warning (starboard)	
OIL 1 —	Oil pressure warning (No 1 engine)	
OIL 2 —	Oil pressure warning (No 2 engine)	
AC —	Alternator supply failure warning	
TURB —	Air turbine stall warning	
HYD 1 —	Hydraulic failure warning (No 1 controls system)	
HYD 2 —	Hydraulic failure warning (No 2 controls system)	
TTC 1 —	Reheat trip operated (No 1 engine)	but see Part 1, Chap 3
TTC 2 —	Reheat trip operated (No 2 engine)	

(*c*) The warning indications can be tested by depressing the TEST pushbutton at the rear of the panel.

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(*d*) A DAY/NIGHT selector switch is at the aft end of the panel. When DAY is selected, the two lamps under each amber window are in circuit. When NIGHT is selected the two lamps under each caption are in circuit. When mod 4193 is embodied, the warning captions are duplicated on the amber 'day' windows.

(*e*) Whenever the HYD 1 and HYD 2 warnings appear together, the HYD warning on the SWP will light up and consequently the attention light will flash and the audio warning will operate.

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◀ (*f*) Post-mod 4795, a push/pull 'mute' switch is mounted on the forward end of the AWP to prevent overheating of the panel during long periods of servicing. With the switch pulled out, an integral lamp in the switch is illuminated and all warnings on the AWP are extinguished. A positive check that the switch is depressed and the lamp is extinguished must be made prior to start. ▶

PART 1

Chapter 14—FLIGHT CONTROL SYSTEM

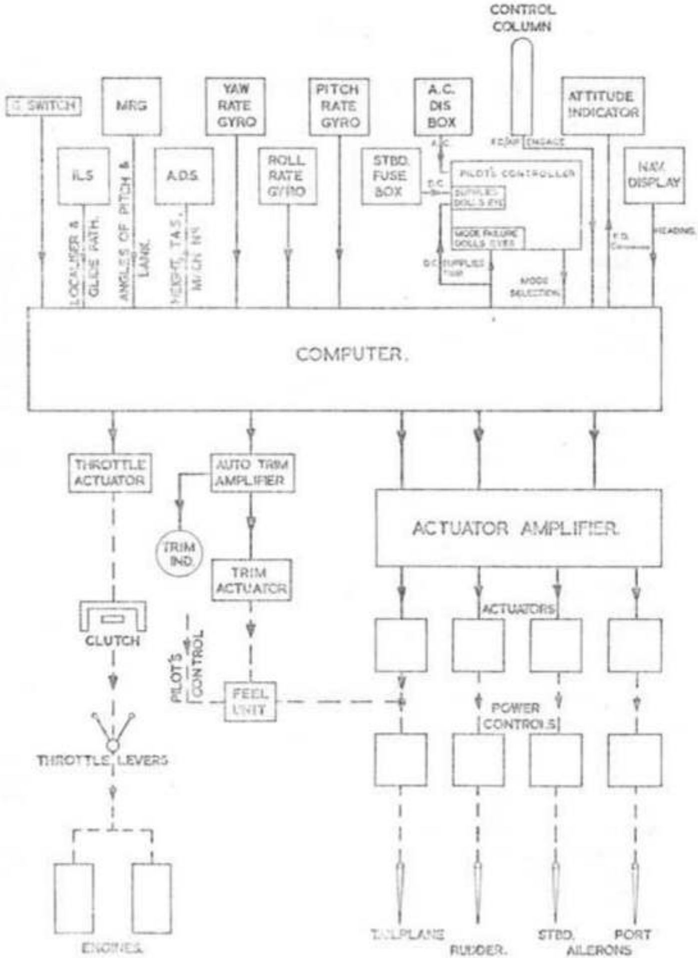
(Completely revised by AL9)

Contents

Description

1 Facilities provided

(*a*) The flight control system Mk 1 is a combined autopilot and flight director system developed specifically to meet the operational requirements of the Lightning aircraft. The autopilot provides automatic control facilities, whilst the flight director provides full flight information for use during



Flight Control System Mk 1

manual-controlled flight; the following facilities are provided:—

Three axes autostabilisation
Programmed climb (flight director mode only)
Height lock
Height and heading lock
Pitch and bank attitude hold
Auto/Flight Director ILS
Auto-throttle

(*b*) Limitations on the use of the FCS are given at Part 2 Chapter 1, and in the Release to the Service document.

2 Basic system

(*a*) The automatic flight control system is essentially a control surface position demand system of limited authority working through the autostabilisers. Each control surface in this system is operated by its own autostabiliser amplifier and electro-hydraulic actuator, and the system works to null any error between the actuator extension and the demanded position.

(*b*) The principal unit of the system is the flight control computer (FCC) which is primarily controlled by the switches and selector keys on the pilot's controller. Its function is to effect all data storage, computation, and switching required for the system. The computer receives signals from:—

(i) Three rate gyros detecting disturbances in pitch, roll, and yaw.

(ii) The MRG, providing pitch and bank attitude data.

(iii) The navigation display, providing heading error information.

(iv) The air data system, providing pressure height, rate of change of height, mach number, IAS, and switching signals for system gearing changes.

(v) The ILS receivers, giving angular displacement from the localiser and glide path beam centre lines.

(vi) The tailplane trim motor giving tailplane trim position.

(*c*) Demands from the FCC are transmitted to either the flight director display on the attitude indicator, or to both the flight director display and to the flying control surfaces via the autostabiliser actuators, depending upon whether flight director (FD) or auto (AP) is engaged. Demands are also transmitted to the auto-throttle actuator.

(*d*) The autostabiliser actuators move their respective control runs to act on the control valves of the PFCU's. Therefore, there is no feed-back to the pilot's controls in autostabilisation. However, in Attitude Hold and Height Lock, with AP engaged, an autotrim facility transfers any FCC demand on the tailplane greater than 0·5° to the tailplane trim motor, which moves to ensure that the autostabiliser actuator is kept roughly central. This allows larger changes of tailplane angle than would otherwise be possible with the limited authority of the autostabilisers. In these modes the control column will follow-up any autotrim movement, via the feel units.

(*e*) The full control surface authority of the aileron and rudder actuators is $\pm 3°$, and of the tailplane actuator $\pm 3° \; 18'$. Full authority of the tailplane actuator is only available, however, with Track engaged and with the undercarriage down. In all other modes the tailplane authority is limited to $\pm 1°$.

3 Safety devices

(*a*) To prevent structural damage to the aircraft in the event of an autostabiliser runaway, each actuator has a DC solenoid-operated stroke restrictor which must be energised to obtain full authority. The DC current is supplied via an AC relay. Therefore with power off, or failure of either AC or DC, or operation of the G trip, the actuator stroke is restricted. The aileron and rudder actuators centralise (restricted stroke zero), but the tailplane actuator may drift to one or other end of its restricted stroke of $\pm 1°$.

(*b*) In the event of hydraulic failure, the actuators will remain in the position at the time of failure. Some out-of-trim rudder force may have to be held, but aileron and tailplane forces can be fully trimmed out.

(*c*) If engaged AP, a G switch automatically trips out the system at normal accelerations outside the range of +3G to 0G. In addition to disengaging automatic control, the autostabiliser stroke restrictors are de-energised.

(*d*) Whenever the autopilot is disengaged by AC or DC failure or by G switch trip, a series of electrical interlocks prevent the pilot re-engaging unless the correct engaging sequence is followed.

(*e*) On a number of occasions when AP has been engaged with a mode apparently selected, due to wear on the mode selector keys no mode circuit has been energised. The failure has not been apparent to the pilot. With ~~Mods 0213/STC and 0214/STC~~ embodied, autopilot and flight director cannot be engaged when a 'no-mode' condition exists. If a failure occurs, the FD/AP engage switch should be set to OFF, the required mode reselected more deliberately and then re-engage.

mod 4556/ AL 13

(*f*) If engaged in AP the pilot can override the system in all three axes. The auto-throttle can also be overridden by using sufficient force on the throttles.

(*g*) An autopilot AP warning is given on the AWP when any one of the following faults occurs:

(i) A G trip occurs whilst engaged AP.

(ii) Incorrect engaging sequence is carried out.

(iii) Phase A or C autopilot AC failure

(iv) DC failure to the autopilot

4 Autostabilisation

The autostabilisers increase the aircraft's damping in pitch, roll, and yaw. A rate gyro in each axis senses changes in angular rates and feeds correcting signals to the appropriate flying control. To compensate for changes in control effectiveness with changes in altitude, IAS and mach number, the gearings of the pitch and yaw channels are automatically switched at pre-set values of height and IAS derived from the air data system. Autostabilisation in all three axes is selected by a single switch on the pilot's controller, and must be on before any auto-pilot mode of the flight control system (FCS) can be engaged.

5 Programmed climb

The programmed climb mode is a flight director mode only. It provides demands for the initial climb (about 10° nose-up pitch), a turn on to a selected heading, and the maintenance of the optimum subsonic climb schedule in maximum cold thrust in ISA conditions. Although satisfactory for reheat take-offs, the mode is not programmed for reheat climbs. The mode is designed for engagement before take-off; the demands will be incorrect if engaged after take-off.

6 Height or height and heading lock

(*a*) With Height Lock alone engaged, the aircraft flies height hold, and wings level. The height datum is that held at the moment of engagement and deviations from this datum result in corrective tailplane demands until the datum is restored. In the transonic region between 0·98M and 1·06M, the height hold is automatically replaced by pitch attitude hold. When accelerating or decelerating out of this speed band the height hold relocks automatically at the indicated pressure height.

(*b*) The heading datum is that indicated by the selected heading pointer on the navigation display. Errors from this datum result in corrective turn demands until the aircraft takes up the datum heading.

7 Attitude hold

With attitude hold engaged, the aircraft is locked to the pitch and bank datum signalled by the MRG to the FCC at engagement. Any error thereafter produces corrective control movements.

8 Auto/FD ILS modes (Track and Glide)

The Auto/FD ILS modes enable the FCS to be coupled to ILS localiser and glide-path signals.

9 Auto-throttle

With any mode selected, automatic throttle control can be clutch-coupled by the pilot to both engine controls. An actuator, operative in the speed range 166-188 knots (162-182 knots in Glide), moves the throttles in an attempt to maintain a constant approach speed. The actuator moves in response to airspeed alone with a mode selected but responds to both airspeed and pitch angle when the auto-pilot is engaged. The auto-throttle facility is inhibited above 275 knots.

Controls and Indicators

10 FCS engage switch

T5 differs.

A three-position, FD/OFF/AP, switch is fitted on the control column. Except for autostabilisation, no preselected mode of the FCS is operative either in auto or flight director until AP or FD is engaged.

11 Pilot's controller

The FCS pilot's controller, on the starboard console, comprises the following controls and indicators:—

(*a*) *Master switch*

The two-position MASTER/OFF switch when switched to MASTER, connects the AC and DC power supplies to the FCC and to the three rate gyros, and in addition centres the pitch autostabiliser actuator.

(*b*) *Supplies indicator*

The SUPPLIES magnetic indicator shows white/OFF whenever the MASTER switch is OFF, and shows black when DC is available and the INST MASTER and FCS MASTER switches are on. The indicator changes to white/OFF whenever the failure interlocks operate. Note that AC failure will not change the indicator to white/OFF unless the STAB switch is on.

(*c*) *Mode indicators*

MODE indicators, one on the pilot's controller and the other above the voltmeter, show OFF when no mode is engaged or if climb is selected and engaged AP. When AP or FD is engaged the mode indicators show AUTO or DIR respectively. The indicators change to white/OFF whenever the failure interlocks operate due to AC phase-failure, DC failure or G-switch trip.

(*d*) *Stab switch*

If the INST MASTER and FCS MASTER switches are on, selecting STAB on the two-position STAB/OFF switch removes the aileron and rudder autostabiliser actuator stroke restrictors. In addition, the autostabilisers are electrically connected to the FCC.

(*e*) *Mode selector keys*

The modes of the system are selected by the following six rocker type keys:—

CLIMB	—	Programmed climb
HEIGHT	—	Height lock
HDNG	—	Heading lock
ATTITUDE	—	Pitch and bank attitude hold
TRACK	—	A combined height and heading lock which can be coupled to ILS localiser signals.

GLIDE — Couples ILS glidepath signals to the FCS, provided TRACK is engaged.

The keys are mechanically interlocked in two ways:—

(i) With the FCS engaged FD or AP, the following direct changes of mode are possible:—

HEIGHT to HEIGHT and HDNG
HEIGHT and HDNG to HEIGHT (by pressing tail of HDNG key)
HEIGHT or HEIGHT and HDNG to TRACK
TRACK to TRACK and GLIDE

(ii) For all other changes of mode, the FCS must first be dis-engaged.

(*f*) *'Little stick' control*

The vertical speed control (VSC) and bank angle control (BAC) allow the pilot to make adjustments to the flight path when HEIGHT, HEIGHT and HDNG or ATTITUDE hold modes are engaged AP. VSC and BAC are override controls.

(*g*) On the Type F controller, a two-position switch is added. The switch must be in the rear position and the locking plate engraved Mk 3.

12 Heading selector (navigation display)

(*a*) When engaged in HEIGHT and HDNG, or TRACK modes, heading demands are fed to the FCC from the selected heading pointer on the navigation display.

(*b*) ~~In the S-band homer mode of AI23B the heading selector knob is used to position the marker on the CRT over the S-band return.~~ When AP selected and ALIGN is selected on the E/F Band homer the heading selector is free to turn without causing a heading change through the FCS.

13 Flight director

The flight director bead on the attitude display is controlled by the FCC when engaged AP or FD. When engaged FD, to follow the flight director, 'fly' the centre reference circle of the attitude display on to the bead. The bead parks in the 2 o'clock position when the FCS is disengaged with the INST MASTER switch at ON. With the FCS engaged, should either a G-switch trip or any failure of AC or DC within the FCS occur the bead will move to its parked position.

14 Autopilot trim indicator

(*a*) The autopilot trim indicator, on the port instrument panel, indicates any out-of-trim condition held by the tailplane autostabiliser actuator. The model aeroplane shows how the aircraft will pitch if AP is disengaged:—

Indicator down	—	(aircraft will pitch nose down on disengagement)—trim nose up
Indicator up	—	(aircraft will pitch nose up on disengagement)—trim nose down

(*b*) If an out-of-trim indication is sustained after selection of any mode except CLIMB, a latent malfunction of the FCC should be suspected and the mode not engaged; when CLIMB is selected, the indicator moves down. However, with HEIGHT and HDNG or TRACK mode selected it is normal for an out-of-trim indication to be present during a turn. When only autostabilisers are in use the indicator will fluctuate about the centre position. If the HEIGHT or ATTITUDE hold modes are engaged AP, the instrument is a visual indicator of the auto-trim. With the FCS MASTER switch OFF the indicator is off the scale.

15 Throttle servo control

A THROTTLE SERVO ENGAGE/DISENGAGE lever on the side of the port console engages auto-throttle control. The lever is spring-loaded to the DISENGAGE position. When the lever is moved to ENGAGE, a spring-loaded catch retains the selection; downward pressure on the catch releases the lever, which then returns to the DISENGAGE position.

Normal use of the System

16 Pre-flight checks

A limited pre-flight check is given in the Flight Reference Cards. If a comprehensive pre-flight check is necessary, it should be carried out as follows:—

(*a*) *Preliminary checks*

Throttle servo	DISENGAGED
Stick switch	OFF
MASTER switch	OFF
STAB switch	OFF
Mode keys	ATTITUDE (to cancel previous selection)

(*b*) *Switch-on checks*

AC and DC power	On-line
INST MASTER switch	ON: FD bead should park
MASTER switch	MASTER: SUPPLIES indicator black, autopilot trim indicator 'in-trim'
STAB switch	STAB: SUPPLIES indicator remains black.

(*c*) *CLIMB checks*

Mode keys	CLIMB: Check that autopilot trim indicator indicates hard nose-down
Stick switch	AP: MODE indicators remain white/OFF, FD bead remains parked
	FD: MODE indicators change to DIR, FD bead demands climb (10° approx.)
	OFF: MODE indicators change to white/OFF, FD bead parks.

(*d*) *ATTITUDE checks*

Mode keys	ATTITUDE
Stick switch	FD: MODE indicators change to DIR, FD bead moves to centre, autopilot trim indicator shows 'in-trim'. Operate BAC and VSC and check that FD bead moves in sympathy.
	AP: MODE indicators change to AUTO. Check as for FD.
	OFF: MODE indicators change to OFF, FD bead parks.

(*e*) *HEIGHT and HDNG checks*

Mode keys	HEIGHT and HDNG
Navigation display	Set HDG pointer to aircraft heading
Stick switch	FD: Check as for ATTITUDE checks
	Move HDG pointer and check that FD bead moves in sympathy.
	AP: MODE indicators change to AUTO. Check as for FD.
	OFF: MODE indicators change to OFF, FD bead parks

(*f*) *Cancellation checks*

MASTER switch	OFF: SUPPLIES indicator changes to white/OFF, trim indicator off-scale
STAB switch	OFF
Mode keys	ATTITUDE

17 Mode engagement sequence

Interlocks prevent engagement of the system unless the correct switching sequence is followed. To engage the FCS in flight, carry out the following drill:—

(*a*) Stick switch OFF.

(*b*) MASTER switch on, SUPPLIES indicator black.

(*c*) STAB switch on.

(*d*) VSC central.

(*e*) HDG pointer on navigation display set to required heading.

(*f*) Press required mode key(s).

(*g*) Trim the aircraft.

(*h*) Autopilot trim indicator moving sensibly about centre.

(*j*) Engage AP or FD.

NOTE 1: If MASTER is selected in flight, under the worst conditions a pitch increment of $\pm 2{\cdot}8$G can be experienced unless pilot action is taken within 2 seconds.

NOTE 2: In flight, one minute must elapse between selection of MASTER and STAB to allow the rate gyros to run-up.

NOTE 3: At least 6 seconds should elapse between mode selection and engagement to allow the FCC to collate the new data and compute the new demands.

18 Autostabilisation

(*a*) The autostabilisers operate when, with AC on line, the INST MASTER, FCS MASTER, and STAB switches are on.

(*b*) The autostabilisers may be engaged at any time during a sortie, subject to the limitations given in Part 2 Chap 1 para 13.

(*c*) When jury struts are fitted in place of autostabiliser actuators, the MASTER switch should be selected ON (to retain full tailplane) but the STAB switch is to be selected OFF.

19 Programmed climb

(*a*) CLIMB should be selected and FD engaged before take-off. Set the desired climb-out heading on the navigation display, not greater than 175° from the runway heading to ensure the turn is made in the required direction.

(*b*) No attempt should be made to follow the bead in pitch until 300 knots has been attained because the initial climb demand is too steep. Above 275 knots the bead will deflect to left or right, demanding a turn on to the pre-set heading. Turns are limited to 45° bank. Transition from constant rate of climb to the climbing mach number starts at about 0·85M at 9 000 feet. It is important to follow the bead very closely at this point to avoid overshooting the mach datum of 0·89M $\pm$ 0·01M, which will be reached at about 14 000 feet and held constant until 32 000 feet. Above this height, mach number will decrease slowly to about 0·85M but will recover if the climb is continued above 38 000 feet. The stick switch should be selected OFF when the desired height is reached and the aircraft levelled normally.

20 Height or height and heading lock

(*a*) *Use of mode keys*
Heading lock cannot be used without height lock and the keys are paired so that depressing HDNG also selects HEIGHT. HEIGHT can be selected independently, and HDNG can be de-selected separately without disengaging HEIGHT lock. With HEIGHT alone engaged AP, a new heading can be pre-selected and the turn initiated when required by operating the HDNG key.

(*b*) *Use of heading pointer*
When HEIGHT and HDNG is engaged AP, turning the heading pointer on the navigation display immediately banks the aircraft towards the new heading. Since the aircraft always turns the shorter way to the new heading demanded, the applied bank will be reversed if the pointer is turned through the reciprocal of the aircraft's heading: therefore, heading changes through more than 180° should be selected in sectors of 90° approximately. With the BAC centred, bank angle is limited to $\pm$28° below 400 knots/ 1·06M and $\pm$50° above 400 knots/1·06M.

(*c*) *Use of the 'little stick'*

(i) *BAC*

The rate of turn on to a new heading can be varied by use of the BAC which, in this mode, increases or decreases the autopilot bank demand by up to 12° in steps of 4°, defined by three click stops either side of central. The control must be centred manually after use to prevent a standing heading error of up to 2½°. The BAC is spring-loaded to the nearest click stop but automatically centres when AP is disengaged, failure interlocks operate or a G trip is experienced.

(ii) *VSC*

The VSC may be used in this mode as a climb rate demand system, having an authority of ±7 500 feet/minute. The control must first be centred before a rotation to UP or DOWN is effective. On either side of the central detent position is a small dead band. The control remains in any position selected; on centring the VSC after use, the system will revert to height lock and the new datum is the altitude at the moment of centring.

(*d*) *Quality of height control*

(i) *Subsonic performance*

In subsonic flight, the performance varies significantly with IAS and, to a lesser extent, with altitude. The flight conditions giving the most satisfactory control are in the speed band 300 kts to 350 kts at altitudes between 20 000 ft and 30 000 ft. At speeds above 350 knots, especially below 20 000 ft, control is oversensitive such that large or rapid throttle movement may produce pitch disturbances which, if severe, will lead to G switch trip. Conversely, as speed is reduced below 300 knots and/or altitude increased above 30 000 feet, height holding becomes increasingly less precise. During turns at heights near the tropopause, up to 500 feet may be lost if speed variation is allowed to occur.

(ii) *Supersonic performance*

In supersonic flight, control varies with altitude but is virtually independent of speed changes. The most satisfactory altitude band is between 20 000 ft and 40 000 ft. Above 40 000 ft, control deteriorates rapidly, especially in turns. Below 20 000 ft control becomes increasingly sensitive. Pitching oscillations may occur with a combination of high altitude (above 36 000 ft) and high bank angle (greater than 50°); this can be alleviated by reducing the bank angle.

(iii) *Transonic performance*

When accelerating in the transonic region, the aircraft will climb and, in the worst circumstances, acceleration may cease. During decelerations, there is also a tendency to climb, particularly when deceleration is rapid. Below 25 000 ft, during both acceleration and deceleration, the pitch disturbances induced by the change from height hold to pitch attitude hold, and vice versa, may trip the G switch; for this reason the mode should not be used during transonic accelerations and decelerations between 15 000 feet and 25 000 feet.

(*e*) *Considerations*

(i) Any change in flight condition requiring large and rapid trim changes will produce temporary height variations. These variations are normally greatest in the climb sense but during rapid deceleration from high speed/low level, the aircraft will tend to descend and this will prove significant at very low altitude.

(ii) Lateral and directional trim changes which occur and are not corrected manually by use of the aircraft trimmers will produce heading errors or, in height lock alone, bank errors. The trim changes will cause a partial loss of autopilot authority in roll, giving a low roll rate in response to a heading demand.

(*f*) (i) A variable height gearing system is incorporated to compensate for changes in control effectiveness with changes in altitude and mach number. The effect is to decrease the tailplane/height error ratio at high subsonic speeds at low altitude and increase the ratio at high altitudes at supersonic speeds.

(ii) Height losses are likely to occur during rapid decelerations at low altitude. Gains of height occur during rapid acceleration, particularly at low altitude. These height excursions are not likely to be as severe as those for fixed-geared autopilots.

(iii) G-switch trips are likely to occur in the 15–20 000 feet height band during transonic accelerations. They are also likely to occur if the VSC is used at speeds above 375 knots approximately below 20–25 000 feet.

(iv) Careful trimming is essential when carrying asymmetric loads (e.g. single missile or air-to-air refuelling probe).

21 Attitude hold

(*a*) The pitch and bank attitude hold is satisfactory throughout the flight envelope. However, pitch errors will occur if the mode is used in conditions requiring rapid trim changes, eg, high rate of climb. Bank errors will also occur if lateral trim changes are required.

(*b*) The 'little stick' may be used to vary the pitch and bank datum. The authority in this mode is $\pm 6{\cdot}6°$ in pitch, and 30° in bank (10° per click stop).

22 Auto/FD ILS modes (Track and Glide)

(*a*) The TRACK key selects a combined height and heading lock to which ILS localiser signals can be coupled. This enables the aircraft to be flown AP or FD to intercept the ILS beam.

(*b*) Operation of the GLIDE key demands a 3° nose down pitch and couples the ILS glide-beam to the FCC. GLIDE cannot be selected unless TRACK also is selected. On disengagement, the GLIDE key trips out leaving TRACK still selected. Auto-ILS procedure is covered in detail in Part 3 Chapter 3. The system allows fully automatic ILS approaches down to decision height. After disengagement the aircraft must be landed manually.

(*c*) The bank angle with TRACK selected is limited as in the height and heading lock mode ($\pm 28°$). With GLIDE selected, bank angle is limited to $\pm 15°$.

(*d*) The BAC and VSC are inoperative in this mode.

23 Track mode as a height and heading lock

In steady cruise conditions, the TRACK mode can be used as a height and heading lock. After checking that the ILS is off, engage the mode in the normal way. Due to variations between aircraft and production tolerances on equipment, some aircraft may perform a small pitching oscillation in this mode. Turns can be made by rotating the heading selector on the navigation display, but heading changes in excess of 40° are prohibited because there may be insufficient tailplane authority to maintain height in the turn. Height errors in turns can be reduced by applying stick or trim as required.

24 Auto-throttle

The auto-throttle can be used on visual approaches, GCA's, and manual and auto-ILS approaches. The auto-throttle should be set to stabilise the IAS at 175–180 knots. The stabilised speed can be changed by slipping the clutches and thus moving the auto-throttle datum.

25 FCS as a GCA aid

(*a*) The FCS, including auto-throttle, may be used as an aid during Ground Controlled Approaches.

(*b*) With the ILS receivers OFF and TRACK selected, AP engagement of the FCS provides height and heading lock for the initial phase of the GCA.

(*c*) To start the descent, provided the autopilot trim indicator shows 'in-trim', selection of GLIDE pitches the aircraft nose-down through 3° and the new attitude will be maintained to give a rate of descent of approximately 900 ft/min at 175–180 kts. The rate of descent can be adjusted manually by re-trimming. It should be noted, however, that changes in pitch attitude will take place slowly and only small trim adjustments should be made with a pause between each change.

Malfunctioning

26 FCS/autostabiliser defects

(*a*) With MASTER and STAB switches both OFF, there can be no malfunction in any channel, but the tailplane autostabiliser actuator is likely to drift to, and remain at either end of its restricted stroke ($\pm 1°$).

(*b*) With the MASTER switch on and the STAB switch OFF, malfunction is possible only in the tailplane channel within the restricted stroke of the autostabiliser actuator.

(*c*) With both MASTER and STAB switches selected on, with or without a FCS mode engaged, hard-over or oscillatory autostabiliser actuator malfunctions are possible. In the case of the tailplane this will be within the $\pm 1°$ limit, except when TRACK is engaged with the undercarriage down, when it becomes $\pm 3° 18'$.

27 Aircraft response to defects

(*a*) *Hard-over malfunctions*

In the worst cases, without immediate pilot action, the effects of a hard-over malfunction are as follows:—

(i) *Tailplane*

Mode	*Maximum normal acceleration increment*
Autostabilisation	±2·8G
Height and Heading, or Attitude	±4·5G

At subsonic speeds above 40 000 feet, pre-stall buffet may be felt, increasing in severity with increase in altitude. The autopilot will be disengaged automatically by the G-switch if accelerations less than 0G or greater than +3G are experienced during a malfunction. With certain malfunctions (with autotrim operative) should the pilot restrict the G and prevent a G-switch trip, the pull or push force will progressively increase until the autotrim reaches its limit stop or the FCS master switch is turned off. The G-switch is not effective in the autostabilisation mode alone.

(ii) *Ailerons*

90° bank can be applied in 2 seconds at high subsonic or high supersonic speeds.

(iii) *Rudder*

2° sideslip, 20°/sec peak roll rate can be experienced. Care should be taken during recovery to avoid large sideslip values due to abrupt rudder correction.

(iv) *All controls—Auto-ILS*

With TRACK engaged AP, and the undercarriage down, the peak increments are as follows:—

Tailplane	±0·85G
Aileron	15° bank in 2 seconds
Rudder	1·6° sideslip, 10°/sec peak roll rate.

(*b*) *Oscillatory malfunctions*

(i) Tailplane	±½G increment at 1 to 2 cycles/sec
(ii) Aileron	Negligible effect
(iii) Rudder	Large oscillatory sideslip

28 Action following malfunction

(*a*) Aircraft response to any malfunction should be corrected but an oscillatory malfunction should not be chased. The following cancellation checks must be carried out in this sequence:—

(i) MASTER switch	OFF (anticipate a change of trim in the tailplane channel)

(ii) STAB switch OFF

(iii) Stick switch OFF

Trim out any control forces.

(*b*) (i) After an FCS trip, the aircraft may or may not be in trim and it may not be obvious that a trip has occurred.

(ii) An FCS trip will be shown by the MODE and SUPPLIES indicators changing to white/OFF and the FD bead parking and could be due to:—

AC and/or DC supply failure

FCS fault

G-trip due to the aircraft exceeding the G-trip limits. If these indications are not accompanied by an obvious malfunction, an FD re-engagement may be attempted after cancellation. If satisfactory, AP may be re-engaged with caution.

(iii) The FCS should not be re-engaged after an obvious malfunction.

29 Instrument power supply change-over

Whenever the NORMAL/STANDBY INVERTER indicator changes to white/ON, whether accompanied by an aircraft oscillation or not, the FCS should be cancelled in the correct sequence (para 28(*a*)).

PART 1

CHAPTER 15 — RADIO AND RADAR

Contents

1 V/UHF — General

(*a*) The V/UHF installation provides normal RT communication.

(*b*) The transmitter/receiver is located in the fuselage spine and is connected to either one of two communication aerials, one in the fin and the other below the nose.

(*c*) Two sets are provided for RT, one for normal use and the other for standby use. The normal set allows any ◀one of 370 VHF or 3500 UHF channels to be selected;▶ the standby set has only two channel selections one normally pre-tuned to 243 MHz and the other to 243·8 MHz.

(*d*) The normal set may be used for homing in conjunction with the ILS localiser indicator. The standby set cannot be used in this way.

(*e*) Electrical power supplies required are 28 volt DC for RT and ADF.

2 V/UHF RT

The V/UHF control panel is on the centre pedestal. On it are the following controls:

(*a*) Three frequency selecting knobs with which any one of ▶◀ 370 VHF channels (117·5 MHz to 135·95 MHz) or 3500 UHF channels (225 MHz to 339·95 MHz) may be selected manually. To use a frequency selected in this manner, the CHAN selector must be set to M. The digits selected appear in windows adjacent to the knobs.

(*b*) A 20-position CHAN selector switch with which any one of 18 pre-tuned frequencies may be selected plus the guard frequency (243·0 MHz). The twentieth position, M, selects manual tuning.

(*c*) A VOLume control.

(*d*) A function switch which can be set to any one of seven selections:

(i) OFF — main V/UHF set off.

(ii) T/R or T/R+G — main set to transmit/receive. There is no separate guard monitoring facility.

(iii) ADF — selects UHF homing mode (see para 2 (*m*)).

(iv) DL — } When Data Link is not fitted the UHF
DL/T — } transmitter/receiver is inoperative.
TR/ON DL/OFF — Data Link is inoperative.

(*e*) A UHF SET, NORMAL/STANDBY switch selects the set required.

(*f*) A NORMAL/STANDBY, POWER switch is for use in conjunction with the standby set. When NORMAL is selected, the 28 volt DC supply to the standby set is reduced by a dropping resistor, to the 24 volts required to operate the set. Should both generators fail, STANDBY must be selected to bypass the resistor and allow the set to be run at battery voltage.

(*g*) An AERIAL, UPPER/LOWER switch selects the aerial required. The switch is for use with the main set only; the standby set is permanently connected to the lower aerial. When a VHF channel is selected, the upper aerial is automatically connected, irrespective of the position of the switch.

(*h*) A VP/ILS INDR switch selects the mode of operation of the ILS indicator.

(*j*) A VP SENS, MAX/MIN switch selects the sensitivity of the homing indication when VP is in use.

(*k*) On the left console is a UHF STANDBY—GUARD/ CHANNEL A switch, for use in selecting the required channel of the standby set. It should always be set to

GUARD in flight. The CHANNEL A setting is for testing purposes only.

(*l*) Two press-to-transmit switches are provided, one on the control column hand grip and the other on the No 2 engine throttle lever.

(*m*) *Violet Picture* (*Homing*)

The ILS localiser indicator on the navigation display may be used to provide homing directions. To operate the system, set the required frequency and select ADF at the function switch. Set the VP/ILS INDR switch to VP and select ILS on the mode selector of the navigation display. Set the ILS localiser index to the aircraft heading.

3 ▶◀ Not used.

4 External Intercom

Provision is made for external intercom via a socket in the right wheel well. When an external headset is plugged in to the socket, intercom between the pilot and the groundcrew is available on either the normal or the stand-by UHF set.

5 Telebriefing

A telebriefing installation can be connected to the aircraft at a socket in the right wheel bay. The light in the tele-briefing button on the lower left console comes on when the connection is made. The equipment is normally in the receive condition; to transmit, the button must be pressed.

6 ILS

(*a*) Standard ILS equipment is fitted, the presentation being shown on the navigation display of the IFIS. The ILS control unit is on the left console and the ILS MASTER switch and ILS VOL control on the UHF control panel. An ILS MARKERS lamp (with day/night screen) is below the navigation display and marker signals are heard provided UHF is switched on ▶◀. A VP/ILS INDR changeover is on the UHF control panel; ILS must be selected when ILS is required or false indications will be presented. The mode selector on the navigation display must also be set to ILS.

(*b*) The ILS presentation on the navigation display is covered at Part 1 Chapter 8, para 3 (*b*).

7 Tacan

(*a*) The Tacan navigational system provides distance and magnetic bearing information and presents it on the navigation display of the IFIS. The control unit is mounted below the left console. 28 volt DC and 115 volt single phase AC are required to power the system.

(*b*) The control unit has an ON/OFF switch, a VOLume control, an A/A-A/G switch, and four channel selector buttons.

(*c*) With Tacan switched to ON and A/G selected, and with the navigation display mode selector at TAC or DL, the Tacan display appears on the navigation display and its RANGE NM window blind clears. TAC selects offset Tacan and DL selects direct Tacan; in the latter case the UHF function switch must not be set to DL or DL/T.

(*d*) With Tacan switched to ON and A/A selected, direct Tacan air-to-air range can be obtained, readable only in the RANGE NM window. The Tacan channel to be selected always differs from that of the tanker aircraft by 63. Therefore, for tanker channels 1 to 63, select tanker channel plus 63 and for tanker channels above 63, select tanker channel minus 63.

(*e*) The Tacan offset computer on the UHF control panel enables a selected homing point from the beacon to be set in. When this is done and TAC is set on the navigation display mode selector, range and bearing from the selected homing point are presented on the navigation display.

(*f*) The Tacan presentation on the navigation display is covered in Part 1, Chapter 8, para 3 (*d*).

~~8 IFF~~

AL 14

(*a*) *IFF Mk 10*

The IFF controls are on the right console. They consist of:

(i) An IFF MASTER switch.

(ii) An OFF/STANDBY/LOW SENSITIVITY/ON control.

▶◀

8 IFF/SSR

(i) ▶◀ A Cossor 1520 IFF/SSR transponder allows the aircraft to be interrogated by civil or military ground radar, and make an automatic identifying response. The controller ▶◀ is on the starboard console and incorporates the controls detailed in the table below.

(ii) An IFF FAILURE light and a panel lighting switch are located adjacent to the controller. The failure light has a press-to-test facility which also tests the self-test light on the controller.

(iii) The operating modes and codes to be used are normally established before flight, but ground radar stations may request particular selections. Codes 7600 and 7700 are selected only in emergency to give a particular alarm as follows:

7600	...	Radio failure (normally loss of voice contact in a controlled airspace).
7700	...	Aircraft emergency.

Control	*Function*
4-position rotary switch with a fifth push to turn position OFF/SBY/ LOW/NORM/ EMGY PUSH	OFF: Equipment switched off. IFF FAILURE light on steady. When switch set from OFF to any other position, 50 seconds required for warm-up. SBY: Power to equipment. After warm-up transponder accepts interrogations on selected modes but cannot respond—IFF failure light flashes. LOW: Equipment functioning but with reduced sensitivity. Used at request of ground station to reduce clutter.

Control	*Function*
	NORM: Equipment functioning normally, accepting interrogations and responding on selected modes. EMGY PUSH: When switch pressed and turned to EMGY, transponder transmits immediate replies with emergency coding on modes 1, 2, 3A or B (see also CIVIL/MIL switch) irrespective of settings of mode switches.
Four on/off MODE switches (up for on): 1/2/C/D	MODE 1: Transponder replies to mode 1 interrogations using mode 1 code. MODE 2: Transponder replies to mode 2 interrogations using a preset code unique to aircraft. MODE C: Transponder replies to mode C interrogations transmitting coded altitude signals (encoding altimeter not yet fitted). MODE D: Not in use.
3-position rotary switch: 3A/OFF/B	OFF: Transponder isolated from mode 3A or B interrogations. 3A: Transponder replies to mode 3A interrogations using mode 3 code. B: Transponder replies to mode B interrogations using mode 3 code.
Code number selectors MODE 1—MODE 3/A/B	Four selectors and indicators for each. Indicators show 0000 to 7777, allowing 4096 codes to be selected.
2-position emergency coding switch: CIVIL/MIL	Used in conjunction with EMGY PUSH MIL: Normal setting. Response codes (Military modes 1, 2 or 3) modulated to emergency form. CIVIL: Code 7700 automatically selected for response to civil interrogation.
2-position switch (spring-loaded to off): I/P	Momentary operation triggers an identification pulse for 20 seconds, added to selected code.
Self-test push button incorporating a double filament green light: TEST	With equipment switched on, pressing button checks receiver sensitivity, transmitter power output and mode serviceability. Set NORM, press TEST: if check satisfactory TEST light comes on (IFF FAILURE light out). Unsatisfactory test indicated by IFF FAILURE light and no TEST light. Flashing IFF FAILURE light can occur if rotary switch at SBY. Steady light if switch to LOW.

Note: If the IFF FAILURE light illuminates during normal operation of the set, it is recommended that the equipment be switched off as there is a possibility that overheating and internal damage can occur.

9 AI 23B/C

(*a*) AI 23B/C equipment is installed. The hand controller is on the left console and the cathode ray tube display unit is above the instrument panel. In addition to its normal function, the AI 23B/C provides 'in range' signals to a light fighter sight (LFS) as an aid to visual attacks with all weapons.

(*b*) An E/F-band (2500 to 4100 MHz) homing sub-system can be switched in to give a display on the CRT to enable the aircraft to be homed on to an E/F-band jamming aircraft. The control unit is on the right console.

(*c*) Both 28 volt DC and 200 volt AC are necessary for the operation of AI 23B/C. The E/F-band homer requires 28 volt DC and 115 volt AC.

(*d*) An LFS SIGHT-LFS/CRT switch is to the left of the light fighter sight. With CRT set, an attack is carried out using the cathode ray tube alone. When set to LFS the attack is completed using the light fighter sight.

(*e*) Below the cathode ray tube is a COMPUTER switch with selections 1 to 6. This switch selects the different computer programmes appropriate to the weapons in use.

(*f*) A BRIGHTNESS control on the right of the CRT adjusts the brightness of the time-base; it has no effect on the brightness of the markers.

(*g*) If pressure failure occurs in the radar bullet, a pressure switch operates to switch off the AI transmitter and the ▶◀ failure is indicated by a warning on the CRT; additionally, an AI 23B PRESS — O RIDE/OFF switch, guarded at OFF, is at the right of the strip speed display. The switch, when set to O RIDE, allows the pressure switch to be overridden and the AI transmitter to continue to function, provided the AI transmitter switch is set to OFF and then ON again. A selection to O RIDE is permitted only if operational necessity demands.

(*h*) For full operating instructions see AP 101B-1003, 5 & 6-15B.

Intentionally Blank


847
ENTRANCE OTHER SIDE