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MALTON ONTARIO

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BROCHURE E4-A

S/L J.H. Cooper

DESCRIPTION OF

ELECTRICAL SYSTEM

FOR

CF-105

SUPERSONIC ALL WEATHER FIGHTER

Classification cancelled/changed to ....

by authority of ....(date)..........

Signature ....Rank F/L

CONSISTS OF 130 PAGES

COMPILED BY		APPROVED BY	
DATE	Feb. 9, 1956	DATE	Feb. 9, 1956
CHECKED BY	Stanley H. Brown	RELEASED BY	
DATE	Feb. 9, 1956	DATE	Feb. 9, 1956

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CF-105 ELECTRICAL SYSTEM

1. INTRODUCTION

1.1 An A.C. power generating system, utilizing static transformer rectifiers for D.C. requirements, has been selected for the CF-105 to avoid the high altitude and high temperature brush wear problems usually associated with the use of D.C. generators and inverters.

Because of the frequency sensitive nature of the electronic loads to be supplied by the electrical system, the alternators are driven through mechanical-hydraulic constant speed units. This choice of drive over that of an air turbine was made in view of the high overall operating efficiencies to be gained, and the consequent saving in weight and space.

A ram air cooled alternator has been selected because of the significant advantages to be gained with respect to development weight and simplicity over the liquid cooled type for all altitude and speed conditions within the aircraft flight envelope.

The alternator has therefore been mounted on the engine nose pad to take advantage of the readily available ram air flow for cooling purposes, thereby avoiding the weight and space penalties associated with a more elaborate ducting system which would be required for cooling purposes if it were mounted elsewhere.

A twin alternator installation, one being mounted on each engine, is required in order to provide a system with reliability compatible with that of a twin engine aircraft. In line with this design approach, all equipment and circuits between the alternators and the electrical power utilizers have been laid out to provide the same degree of reliability.

In addition, the complete aircraft electrical system has been divided into two separate and self contained systems - one on each alternator - to insure maximum availability of power in the event of failure of one alternator. Simplification to the circuitry is also achieved.

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1.1 (Continued)

Although full advantage has been taken of the latest design features of control equipment for alternators and rectifiers to ensure long life, and reliability, the alternator and rectifier controls, and the transformer-rectifiers have been installed so that they may be easily removed from the aircraft for ease of servicing and maintenance.

The high performance of the aircraft places additional demands on the crew and dictates an attempt to achieve to the greatest extent a completely automatic system. In this respect the system is designed so that in the case of failure of one system all essential loads are automatically switched to the operating system to ensure continuity of supply. This action will be indicated to the pilot so that he may change flight plan and/or report the defect to the maintenance crew.

Consistent with this approach, all circuit breakers are located outside of the cockpit where they are readily accessible for ground inspection.

The electrical circuits have been laid out to achieve, to the greatest extent, invulnerability to all anticipated types of aircraft damage consistent with the requirement of minimum overall weight.

1.2 The electrical system is designed to meet the following operational requirements:-

1.2.1 <u>Altitude</u>

Sea level to 60,000 ft.

1.2.2 <u>Temperature</u>

1.2.2.1 Units located in areas which are not supplied with conditioned air flow, are designed for operation throughout the range -65°F to +248°F, the latter temperature being a result of the aerodynamic kinetic heating effects at M = 2.0 (For flight limitations at this temperature see paragraph 1.2.4 below.)

1.2.2.2 Units located in air conditioned areas will not be exposed to temperatures exceeding + 160°F thus permitting the use of equipment already proved and qualified for operation throughout the - 65°F to + 160°F range.

1.2.3 Flight Attitudes

All operatonal attitudes including inverted flight, with limitations as shown in paragraph 1.2.4.

1.2.4 Flight Limitations

1.2.4.1 Inverted flight duration of 15 seconds.

1.2.4.2 Endurance at maximum speed:

10 minutes at $M = 2$, based on 15 minutes cruise out at $M = 1.5$ and fuel temperature limitations at the engine inlet of 160°F.

2. DESIGN OBJECTIVES

2.1 To provide power for the aircraft electrical services.

2.2 To provide power for the aircraft electronic services when necessary.

2.3 To provide power, as required, for the weapons, prior to launch.

2.4 To provide power for the aircraft anti-icing and de-icing services.

2.5 To provide 100% reserve power generation capacity over the known requirements of the electrical services to date to allow for growth with future aircraft development.

2.6 To provide adequate power to permit safe flight in the event of failure of any one generating system.

2.7 To meet the above requirements with a system having a minimum installed weight.

2.8 To provide a system which will power all essential services during a twin-engine flame out for a period long enough to permit the relighting procedure to be carried out.

2. (Continued)

2.9 To meet the design requirements of ARDCM-80-1 and CAP-479, which includes the following specifications:-

MIL-E-7080	Electrical Equipment:	Installation of Aircraft General
MIL-E-7563	Electrical Equipment:	Aircraft - Installation of General Specification for
MIL-E-7614	Electrical Equipment - Alternating Current Aircraft Installation of General Specification for	
MIL-E-7894	Electric Power, Aircraft, Characteristics of	

2.10 To meet the general design requirements of Specification AIR-7-4 and, in particular, those of para. 9 "Electrical Equipment".

3. GENERAL DESCRIPTION OF SYSTEM

3.1 The CF.105 airplane is equipped with two completely independent and self contained electrical systems, one being powered by each aircraft engine. This primary mechanical power is converted by the alternators into 120/208 Volt, three phase, 400 C.P.S. electrical power. A portion of the generated A.C. is rectified to supply the aircraft D.C. loads at 27.5 Volts D.C.

The generation, rectification and control equipment for each system is identical and symmetrical up to the main A.C. and D.C. buses. The distribution of A.C. power within the aircraft is arranged into two separate systems with provision of automatic switching of essential loads to the operating system in the event of failure of one system. D.C. distribution is carried out from a common D.C. bus supplied concurrently by both systems.

The load centres for each electrical system of the aircraft are located as follows:-

Station 129-147 Forward Circuit Breaker Panel housing the following:-

Forward D.C. Bus
Shedding D.C. Bus
Emergency D.C. Bus
Battery Bus
Forward A.C. Bus

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3. <u>GENERAL DESCRIPTION OF SYSTEM</u> (Cont'd)

3.1 (cont'd)

Station 490	Aft limiter panel
Station 485	D.C. bus

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4.0 GENERAL CIRCUITS

4.1 Deleted
4.2 Deleted

INDEX OF SERVICES

.4.3.1 POWER SYSTEM (See Fig. 1)

4.3.1.1. GENERAL

The electrical power supply is provided by two engine mounted, ram air cooled, 30 K.V.A., 120/208 volt, three phase, 400 cycle alternators. Each alternator is driven independently by its respective aircraft engine through a mechanical-hydraulic constant-speed drive unit which maintains the output frequency constant within ± 5% from idling to maximum rated input speed.

D.C. is provided by two 3KW transformer rectifier units each of which is supplied independently from its respective system 30 KVA alternator. A single cabinet contains the two system transformer-rectifier and alternator control boxes, and cooling is provided to each box from the aircraft air conditioning system.

4.3.1.2 A.C. SYSTEM

The output of each alternator is taken from alternator terminals T1, T2, and T3 which are fed into the respective control panel through terminals T1, T2 and T3. From the control panel the A.C. supply is fed through a line relay, which is controlled by the ON/OFF switch in the cockpit, and thence to the main A.C. bus bars.

The A.C. loads are divided into two groups and each group is supplied by one 30 KVA alternator through its own main bus bars, thus forming two completely independent and self contained electrical systems which are designated - the right hand system and the left hand system.

Primary or essential A.C. loads in the aircraft are fed from the Primary A.C. Bus, which, under normal operating conditions, is supplied from the Right Main A.C. Bus through a transfer relay. In the event of failure of the Right hand Alternator two transfer relays will operate, one to disconnect the Right Main A.C. Bus and the other to connect the Primary A.C. Bus to the Left Main A.C. Bus, thus supplying the essential loads from the Left Alternator. Transfer is effected automatically when the Right Power Failure Detector Unit senses the failure and actuates an integral single pole double throw switch to cause the transfer relays to operate.

4.3.1.2 A.C. SYSTEM (Continued)

Secondary, or less essential A.C. loads are connected to the A.C. Shedding Bus and fed from the Left Main A.C. Bus through a three pole single throw relay which is held in the closed position under normal operating conditions. In the event of failure of the Right Hand Alternator this relay will be de-energized to shed the secondary loads and enable the Left Alternator to accomodate the essential loads on the Primary A.C. Bus. The relay will be operated automatically by the Right Power Failure Detector, upon sensing failure.

Two amber indicator lights, one for the Right Hand Alternator, and one for the Left Hand Alternator are located on the warning light panel in the pilots' cockpit, to indicate when an alternator failure occurs.

Should a fault cause the controls to take an alternator off the line, it can be reapplied by first operating the appropriate alternator "RESET" switch in the pilots cockpit to the "RESET" position and then selecting the "ON" position. If the fault has not cleared the system will again trip out.

4.3.1.3 D.C. SYSTEM

The transformer rectifier units are fed from their respective main A.C. buses through terminals D1, D2 and D3 and the output from each is fed to the main D.C. bus.

The main D.C. bus supplies power to (a) the forward bus from which all essential services are supplied including the battery and emergency buses and (b) the shedding bus to which all unessential services are connected.

The shedding bus is connected to the main D.C. bus by a single pole single throw relay which is held in the closed position when the systems are operating normally. The coil of this relay is connected to terminal 5 of the right hand transformer rectifier unit through the contacts of a slave relay, the coil of which is connected to terminal 5 of the

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FORM 1749 A

4.3.1.3 D.C. SYSTEM (Continued)

left hand transformer rectifier unit. Since terminals 5 supply 28 volts D.C. when the systems are normal, should either transformer rectifier system fail, the associated relay will open and de-energize the D.C. shedding bus, and energize the appropriate D.C. failure indicator light on the pilots warning light panel.

The battery bus is connected to the D.C. forward bus through a single pole single throw relay, the coil of which is connected to 28 volts DC through two small slave relays in parallel, whose coils are energized by terminal 5 of the R/H and L/H transformer rectifier units respectively.

Should one unit fail, only the slave relay concerned will open and the battery bus will remain connected to the forward bus.

The emergency bus is connected to the battery bus through the contacts of a relay, the coil of which is energized when the Master Electrical switch is operated to the "ON" position.

Should both units fail, both slave relays will open, causing the S.P.S.T. relay to open and isolate the emergency bus and battery, which compose the flight emergency system.

A single pole double throw, momentary - on with centre "off", switch is provided in the pilots cockpit for D.C. reset, one side is idented right and the other left. If a fault should cause one of the transformer rectifier units to be removed from the line it will be reset by pulsing the reset switch to the appropriate side, provided the fault has cleared itself.

4.3.1.4 EXTERNAL SUPPLY SYSTEM

External A.C. power is utilized by plugging the supply into the aircraft receptacle and positioning the cockpit "Master Switch" to the "On" position. 28 volts D.C. from the external supply Pin "E" will then close the two three-phase line relays and put power on the aircraft right and left hand A.C. buses; and at the same time open a slave relay to prevent the main A.C. line relays from closing which prevents feed-back to the alternators.

4.3.1.4 EXTERNAL SUPPLY SYSTEM (Continued)

D.C. power is obtained from the transformer rectifier units as before, with the external A.C. power supplying the A.C. requirements to the transformer rectifier units.

4.3.2 STARTING AND IGNITION (Ref. Fig. 2)

4.3.2.1 The system consists essentially of an air turbine starter and an ignition system for each engine. The system is divided into two parts:

(a) that part of the system within the airframe consisting of the starter control switches and relays, the engine relight switches, and the ignition system.

(b) the ground starter cart which contains a compressed air supply controlled by two air shut-off valves, one for each engine, and a 28 Volt D.C. supply.

The ground starter cart is connected to the aircraft through two air hoses and an electrical connector, these connections are automatically withdrawn from the aircraft by means of lanyard releases when it starts to taxi.

The electrical connector provides two control leads (one for each air valve), one ground lead, a 28 Volt D.C. power lead, a 28 Volt D.C. control lead, and intercommunication leads.

4.3.2.2 STARTING

Engines may be started individually or simultaneously. The starting sequence is as follows: The ground cart, air hoses and electrical connector are plugged into the aircraft and the master switch selected to the "ON" position. The starting power relay functions, isolating the aircraft battery from the bus and replaces it with the external D.C. supply, and the emergency bus relay closes to connect the battery bus to the emergency D.C. bus. This prevents drain of the aircraft battery due to the normal and emergency services connected to the battery and emergency buses, while the starting cart is plugged in.

To start the engine, the desired starting switch is set momentarily to the "START" position, this locks in the starting relay thereby opening the external air control valve to supply air to the air turbine starter.

4.3.2.2 STARTING

When the engine attains a speed of 700 rpm a centrifugal switch on the starter closes which energizes the ignition circuit and illuminates the advance throttle indicator light. The applicable throttle lever is then advanced to the "IDLE" setting. When the starter reaches a speed of 3020 rpm a second centrifugal switch opens which de-energizes the starting relay thus completing the starting cycle.

If it should be necessary to interrupt the starting cycle; the starting switch is momentarily set to the "RE-SET" position which de-energizes the "locked" starting relay, thus resetting the system preparatory to a further start.

4.3.2.3 RELIGHT

To relight the engines after a flame out while airborne, the push button relight switch is actuated to feed 28 volts D.C. directly to the ignition system.

4.3.3 ENGINE SERVICES (Ref. Fig. 3)

4.3.3.1 OIL PRESSURE WARNING

When the oil pressure falls below 25 p.s.i., an amber, cockpit warning light is lit by the actuation of a pressure switch.

4.3.3.2 FUEL PRESSURE WARNING

When the fuel pressure falls below 18 p.s.i. absolute, an amber, cockpit warning light is lit by the actuation of a pressure switch.

4.3.3.3 TURBINE DISCHARGE TEMPERATURE

A thermocouple system is provided using four thermocouples located on the turbine discharge shroud ring of each engine and an indicator for each engine is located in the pilot's cockpit.

4.3.3.4 PRESSURE RATIO INDICATING

This system, details of which are not yet finalized, will include a pressure transmitter and indicator for each engine, indicating percentage of thrust available.

The indicator will be located in the cockpit and will be calibrated to indicate the thrust output from the engine as a percentage of maximum available thrust.

4.3.3.5 AFTER BURNER ACTUATION

The after burners are actuated by limit switches in the throttle box. When the right-hand and/or left-hand throttle lever is pushed forward into the after burner region and depressed, the associated limit switch is closed, actuating the appropriate after burner relay which selects the after burner valve to "OPEN". The heat exchanger oil cooler valve which is controlled by the same relay is also activated to the "OPEN" position to allow engine oil to pass through the heat exchanger, instead of by-passing the heat exchanger as in normal operation.

VM

What is the emergency fuel system

4.3.3.6 ENGINE FUEL CONTROL

A switch for each engine, marked "Normal/Emergency" is located in the front cockpit, which enables the pilot to isolate the main engine fuel system in the event of malfunction and select the emergency system to take over.

4.3.3.7 ZONE #1 EJECTOR

Zone #1 ejector valve is utilized to prevent air containing fuel vapour flowing from Zone #1 into Zone #2 of the engine.

It is operated by a differential pressure switch, which senses the difference between atmospheric and the pressure in Zone #1 of the engine to control the operation of the Zone #1 Ejector Valve which vents this zone to the atmosphere. The ejector valve is maintained open whenever the pressure in Zone #1 is less (by approx. 2 inches of water) than the atmospheric pressure.

4.3.4 LANDING GEAR SYSTEM (Ref. Fig. 4)

This system is comprised of landing gear actuation, landing gear indication, nose wheel steering, anti-skid control and automatic wheel braking to prevent wheel spin after "UP" selection.

4.3.4.1 LANDING GEAR ACTUATION

4.3.4.1.1 NORMAL OPERATION

Normal landing gear actuation is controlled by a special four pole double throw switch, with no "OFF" position, which has a solenoid interlock incorporated to prevent accidental retraction while the aircraft is on the ground. The interlocking is accomplished by limit switches, actuated by the main U/C scissors which are in series with the interlock solenoid. This locks the actuation lever until the aircraft is airborne.

Actuation of the complete landing gear is controlled by one electrically operated hydraulic valve. The solenoids of the hydraulic valve are energized by selecting "UP" or "DOWN" on the landing gear actuating switch.

When the landing gear is up and locked, the "UP" solenoid is de-energized by the three door up-lock limit switches wired in series; with landing gear down, the "DOWN" solenoid is energized until the master switch is shut off.

4.3.4.1.2 EMERGENCY OPERATION

For emergency "UP" actuation, when the aircraft is on the runway, a push button switch is provided to override the interlocking of the actuation lever and the lever must be selected "UP" while the push button switch is depressed. Power is provided from the battery when the master electrical switch is "ON".

Emergency down selection is accomplished by pulling the landing gear actuation switch lever down through a gate and

Also amber warning light when u/c up for landing

4.3.4.1.2 EMERGENCY OPERATION (Continued)

actuating a pneumatic valve mechanically.

4.3.4.2 LANDING GEAR INDICATION

The position of landing gear is shown by three AN type indicators which indicate "UP" "DOWN" and neutral positions of each of the three landing gear components, and are actuated by limit switches on the "UP" and "DOWN" locks of the three landing gear components.

In addition, these limit switches control the power to a warning light in the handle of the landing gear selection lever on either an up or down selection until all three landing gear components are locked in the selected position.

4.3.4.3 NOSE WHEEL STEERING

The nose wheel steering system is selected by a push button switch on the control column. This switch, wired in series with the nose scissors switch, actuates the coil of the nose wheel steering hydraulic valve, permitting flow of hydraulic fluid to the steering jack.

4.3.4.4 ANTI SKID SYSTEM

The anti-skid brake control system is designed to prevent too rapid deceleration of the aircraft braked landing wheels during any phase of the landing roll. It also prevents landing the aircraft with the wheels locked, in the event brake pressure is metered prior to touchdown or during a bounce.

When landing gear "DOWN" is selected the door "UP" relay No.2 is de-energized by the opening of the door "UP" limit switches. The closing of contacts of the door "UP" relay No.2 energizes the anti-skid system.

The control of this system is effected through the use of the following equipment:

(a) The skid detector unit at each braked landing gear wheel which sends signals to anti-skid control unit if excessive wheel deceleration or lack of wheel rotation occurs. This is

4.3.4.4 ANTI-SKID SYSTEM (Continued)

(a) (continued)
called the wheel unit.

(b) Solenoid operated hydraulic valve in brake line to each wheel which releases or re-applies pressure at wheel when operated by anti-skid control unit.

(c) Anti-skid control unit which interprets signals from wheel unit and causes brakes to be released or re-applied on appropriate wheels as necessary.

(d) Switch on main landing gear scissors which sends a signal to the anti-skid control unit when the weight of the aircraft is not on the wheels.

4.3.4.5 WHEEL BRAKING

A solenoid in each main wheel brake control valve is automatically energized during normal "UP" actuation of the landing gear, and admits hydraulic pressure to the wheel brakes to stop wheel spin before the wheels are fully retracted.

Scissor switches on the landing gear prevents these solenoids from being energized during emergency retraction.

4.3.5 FUEL SYSTEM

4.3.5.1 FUEL VALVE CONTROL

The left and right low pressure fuel valves are supplied with 28v DC power from the emergency bus through left and right fire extinguisher relays respectively and left and right ground service switches respectively. In the "NORMAL" switch position power is supplied to maintain the valve open.

When either the open or closed actuation is completed, limit switches in the valves will cut the power from that side of the valve. (In the intermediate valve positions both limit switches are closed).

Actuation of an extinguisher button in the cockpit will energize the fire extinguisher relay and supply power from the battery bus to close the appropriate valve.

4.3.5.2 LOW LEVEL FUEL SENSING AND FLOW PROPORTIONER VALVE CONTROL

A low level fuel sensing system is provided to warn the pilot of the existance of low level fuel conditions and to provide a means of control of the flow proportioner by-pass valve. The functioning of this system is described below.

One left and one right by-pass valve in the respective flow proportioner unit is controlled automatically by low level signals from the collector tanks (#5L & #5R). If a signal from either the left or right tank is received, supply from the main D.C. bus will actuate a relay to open a by-pass valve in the respective flow proportioner unit and lock it in this position for the remainder of the flight. Simultaneously, a signal is sent to the Master Warning Control Unit and the fuel proportioner amber warning light in the front cockpit will be locked "ON". The low level amber light for the affected tank will also go on.

4.3.5.2 LOW LEVEL FUEL SENSING AND FLOW PROPORTIONER VALVE CONTROL (Continued)

Is this not locked in position

yes opened

When the by-pass valve is opened the fuel level will rise in the respective tank and the low level light will go out. Should a low level condition occur again after this, the pilot will receive a second warning on the low level fuel light for that tank.

The lock-on relays will automatically reset when power is shut off after the aircraft has landed but the by-pass valve of the flow proportioner unit affected will remain in the open position until power is subsequently turned on.

For ground checking, visual indicators on the proportioner by-pass actuators will show that they are open. These indicators will stay in the "by passed" position until electrical power is turned on once again. When either the open or closed actuation is completed, the power is cut from that side of the valve.

4.3.5.3 AIRCRAFT FUEL BOOSTER PUMP PRESSURE WARNING

A signal will be supplied to the Master Warning Control Unit and the applicable cockpit warning light will be illuminated through either the left or right fuel differential pressure warning switch when a low differential pressure occurs across the booster pump.

For purposes of ground checking with the engines running, one left and one right press-to-test green light is located on the refuelling and test panel E21 on the left side bottom skin between the speed brakes hinges. The "press-to-test" indicates normal pressure by means of the same pressure switches that are used to operate cockpit warning lights.

4.3.5.4 EXTERNAL TANK SYSTEM

4.3.5.4.1 FUEL TRANSFER

When the external tank is fitted, a float switch in the tank is connected to a solenoid operated shut off valve in the air pressurizing line to the tank. The shut-off valve is supplied with 28V D.C. from the main bus through U/C limit switch control preventing transfer when the A/C is on the ground. The shut off valve is opened when the A/C is airborne by the presence of fuel actuating the float switch thus transfering fuel automatically until the tank is empty when the valve is closed by the float switch.

4.3.5.4.2 NORMAL TANK JETTISON

When the jettison switch in the pilot's cockpit is closed, emergency D.C. power actuates a relay to energize the external tank jettison solenoid and simultaneously removes power from the external connector and the air shut off valve.

As a safety measure the power supply to the external tank connector and to the normal jettison switch will not be available until the main landing gear scissors limit switches have been actuated, i.e. when the aircraft is airborne.

4.3.5.4.3 EMERGENCY TANK JETTISON

An emergency jettison switch is provided to override the landing gear scissors switch, (Ref. 4.3.5.1.2) and supply emergency D.C. power to the jettison system.

4.3.5.4.4 SAFETY PROVISION

A missile safety control limit switch, connected in series with another safety control limit switch on the nose landing gear door uplock, is included on the tank fitting to prevent missile release when the external tank is connected to the aircraft.

4.3.5.5 GROUND REFUELLING SYSTEM

The following electrical equipment is mounted on the ground refuelling panel located on the fuselage between the hinges of the left hand dive brake:

(a) One master refuelling switch supplied from the main D.C. bus. A mechanical interlock prevents this switch being left in the "ON" position when the panel is shut.

(b) One selector switch with three "ON" positions (Normal Refuel, Partial Refuel and Defuel). The switch is supplied from the main D.C. bus when the master refuel switch is "ON".

In addition, two panels (one in the left and one in the right main U/C bay) each contain seven green indicator lights and one refuelling control switch. The indicator lights are wired to switches on the seven tank shut off valves.

The refuelling control switch supplies D.C. power to the solenoids on the seven fuel level sensing valves.

When the master refuelling switch is turned on, main D.C. power will "arm" the fourteen tank indicator lights and automatically actuate the flight position relays to break the main D.C. supply normally furnished to the left and right Flow Proportioner Unit during flight.

4.3.5.5.1 Main D.C. through the selector switch will operate the appropriate relays to actuate the valves according to selections as described below:

(a) Normal Refuelling

VALVES	POSITION
By-Pass-Flow Proportioner	Open
Air Press. Relief	Open
Press. Regulating Valves	Open

4.3.5.5.1 (Continued)

(b) Partial Refuelling

VALVES	POSITION
By-Pass-Flow Proportioner	Close
Air Press. Relief	Open
Press. Regulation Valves	Open

(c) Defuelling

VALVES	POSITION
By-Pass-Flow Proportioner	Open
Air Press. Relief	Close
Press. Regulating Valves	Close

4.3.5.6 RE-FUELLING

4.3.5.6.1 NORMAL REFUELLING

When the fuel nozzle is connected, and after the master refuelling switch is turned on and selector switch turned to the required position, all tank indicator lights should be "ON" before closing the refuelling control switches indicating the fuel shut-off valves are closed. If a precheck is required in the initial stages of fuelling, the Refuelling Control switches are turned on until the valves open and indicator lights go out. Then both refuelling control switches are shut off and all tank indicator lights reappear indicating the valves re-close.

4.3.5.6.1 NORMAL REFUELLING (Continued)

Refuelling may then proceed by turning both refuelling control switches on again. As the tanks commence to fill the indicator lights go out and then will come "ON" again when the tanks are full. The refuelling control switches for both tank systems are shut off. Then the master switch is turned off and the main D.C. supply through the flight position relays (which have returned to normal) will close both by-pass valves in the flow proportioner units.

Should main D.C. power be shut off before the refuelling master switch is shut off all the valves will remain in the last selected positions until main D.C. power is turned "ON" once more. All the valves will then return to their normal flight positions.

4.3.5.6.2 PARTIAL REFUEL

Partial refuelling is carried out as described under normal refuelling in para. 4.3.5.6.1 except that the refuelling control switches are shut off when the desired fuel quantity is shown by the capacity indicators in the cockpit. Communication is required between the cockpit and the control panel for this operation and is catered for by provisions in the aircraft inter-communication system.

4.3.6 FUEL CONTENTS INDICATION (Ref. Fig. 6)

The fuel quantity indicating system comprises two independent and symmetrical systems of the capacitor type, one right hand and one left hand, and each indicating the total weight of fuel remaining in the tanks for their respective sides.

The right indicator shows the total contents of the seven tanks 1, 3R, 4R, 5R, 6R, 7R & 8R, and the left indicator shows the total contents of the seven tanks 2, 3L, 4L, 5L, 6L, 7L & 8L.

Each tank contains a number of capacitor units wired in parallel and they in turn are wired in parallel with all the other units in the same system. The number of units in each tank is sufficient to ensure accurate indication over a wide range of flight attitudes. Collector tanks 5L & 5R also each contain a compensator unit which is connected to its respective amplifier/indicator unit to compensate for changes in dielectric properties and densisites of the fuel.

4.3.7 FIRE PROTECTION SYSTEM (Ref. Fig. 7)

This system consists of a Fire Detection System and a Fire Extinguisher System, which provides protection for each engine and the hydraulic equipment bay.

4.3.7.1 FIRE DETECTOR SYSTEM

The system consists of a detector cable, warning light and control unit, and it functions as described below.

The insulation resistance of the detector cable falls on exposure to localized heating, and this characteristic is utilized to light a warning lamp in the pilot's cockpit when the resistance drops to a preset value. The value of insulation resistance which causes the light to glow is set by a resistor connected across terminals 1 & 4 of the control unit. This also permits interchangeability of the control unit between systems having detector cables of different lengths by merely changing the resistors. For test purposes a switch (located on the refuelling & test panel E 21) is provided which opens the continuous loop of the detector cable and grounds one end when actuated, thus causing a warning signal. The fire detector system is supplied from the emergency bus.

4.3.7.2 FIRE EXTINGUISHER SYSTEM

4.3.7.2.1 MANUAL OPERATION

The extinguisher system is actuated by a push button switch for each zone (which is combined in one unit with the appropriate fire detector indicator light). Two fire extinguisher bottles, containing Freon 12B2, are provided, each having three outlets and each outlet is independently controlled by an electrically actuated valve. Each one of the three outlets from each bottle is piped to one of the three fire zones.

To operate the system, the appropriate switch is pushed and through a system of sequencing relays the associated fire bottle is triggered thus discharging the extinguishing medium into the fire zone.

4.3.7.2.1 MANUAL OPERATION (Continued)

If the fire is not then extinguished the pilot may, by closing the "SECOND SHOT" switch, discharge the contents of the second bottle into the same zone, or alternatively, should a fire occur in either of the other two zones the contents of the remaining bottle may be discharged by actuating the appropriate push button switch.

In the case of fire in an engine zone, provision is made through the use of shut off relays and time delays for closing the low pressure fuel cocks automatically before the extinguishers are actuated.

Power for the first shot for each area is supplied from the battery bus, therefore, in the event that the master switch is "OFF", power from the battery bus will be available to close the low pressure cocks and discharge the fire extinguisher bottles.

4.3.7.2.2 AUTOMATIC OPERATION

Under crash conditions, the crash or intertia switch will actuate the system, discharging the contents of one extinguisher bottle into both right and left engines, and the contents of the other extinguisher bottle into the hydraulic bay. Power for this operation is also supplied from the battery bus.

4.3.8 NORMAL CANOPY ACTUATION (Ref. Fig. 8)

Actuation of each canopy is effected by the operation of an electrical linear actuator containing load limiting switches. These switches ensure full travel of the canopy before they operate to remove the D.C. supply to the actuator. In addition a control to energize the canopy seal pressurizing valve is incorporated.

Each canopy is controlled by a pair of double pole double throw switches (with centre "OFF") in parallel, one of each pair is located in each cockpit and the other on the canopy arch for ground actuation.

These ground service switches are for normal service and will not override the safety latches in the cockpit. However, a mechanical emergency release is incorporated which may be externally or internally operated.

Interlocking relays are provided in the circuit to prevent the ground switches being actuated in opposition to those in the cockpit.

Both front and rear cockpit systems have limit switches in series with the D.C. supply to prevent energizing the actuators with the canopies locked, the limit switches are actuated when the latch is fully open.

The canopy seal pressurizing valve is actuated automatically when both front and rear canopy latches are in the fully closed position by means of limit switches in the locks which are wired in series. Power is taken from the emergency D.C. bus.

Power for canopy actuation is supplied from battery bus so that the canopies can be opened or closed after the master switch has been turned "OFF".

The canopies should only be opened once and closed once on the aircraft battery during a single flight cycle, but once the master switch is turned on, power can be obtained from the starting cart or ground energizer, if the engines have not been started.

4.3.9 RADOME DE-ICING & WINDSCREEN ANTI-ICING

4.3.9.1 RADOME ICE PROTECTION

A solenoid operated shut-off valve is controlled automatically by a signal from the forward ice detector, through a "hold-on" relay. This detector is additional to the two detectors supplied in the Engine Anti-icing and Duct De-icing systems (Ref. Para. 4.3.14) but its operation is identical.

When an icing signal occurs, the solenoid valve will be opened for 1-5 seconds supplying de-icing fluid to the radome. The actuation of the valve is controlled by the "hold-on" relay and when the valve closes the "hold-on" relay prevents further signals operating the valve for 15 seconds.

Power to operate the valve is supplied from the main D.C. bus through normally closed contacts of a relay which are opened with A/C on the ground, thus preventing waste of de-icing fluid.

4.3.9.2 WINDSCREEN & CANOPY ANTI-ICING

Anti-icing of the pilot's windscreen and canopy is accomplished throughout by electrical heating. An electrically conductive transparent coating is incorporated on the inner surface of the outer glass lamination of the windscreen panels and sensing elements are built into the panels to control the temperature.

The panels will be supplied from the primary A.C. bus and except for periods when the sensing elements opend the circuit, the panels will be energized continuously while the Master Switch is "ON".

4.3.10 AIR CONDITIONING SYSTEM (REF. FIG. 10)

4.3.10.1 COCKPIT TEMPERATURE CONTROL

The cockpit temperature is controlled by regulating the mixture of hot air with cooling air which is supplied to the cockpit. The temperature may be regulated to any temperature selected by the pilot within the range of 40 to 80°F except that under 20,000 ft. altitude the minimum temperature is limited to 55°F.

This is achieved by a control valve located in the hot air supply duct which is controlled by a temperature sensing system. The sensing system is a balanced bridge circuit with sensing units located in both the inlet and outlet ducts.

4.3.10.2 DEFOGGING

A de-fog switch is fitted in the pilot's cockpit, to override the temperature sensing system and provide the inlet air at a temperature of 95°F to disperse fogging.

4.3.10.3 EMERGENCY OPERATION

Failure of cooling air flow to the cockpit at high speeds will result in critically high cockpit temperatures due to kinetic thermal effects. To lower the temperature, speed must be reduced immediately, and the pilot's air supply switch must be switched from "NORMAL" to either the "OFF" or the "EMERGENCY" position. The "OFF" position will close the cockpit air inlet control valve thus shutting off all air supply to the cockpit and will also supply a signal to shut off the electronic equipment in the nose radar compartment which would be damaged by the rise in temperature.

Selection of the "EMERGENCY" position will initiate the same operations described for the "OFF" position, but in addition a ram air inlet control valve would be opened to introduce ram air for cooling the essential compartments and the inlet control valve of the nose radar will be closed.

4.3.10.4 In the case of failure of either the Cockpit or the Radar Temperature Controller, power to the Radar Temperature Control Valve or the Cockpit Temperature Control Valve (as applicable) will be cut, permitting the valve to return to the closed position.

4.3.10.5 GROUND OPERATION

If this was not [illegible]

(1) Satisfactory ground operation is automatically provided for by the operation of a scissors switch on the right main undercarriage which opens when the weight of the aircraft is on the undercarriage.

(2) This operates a Fan Inlet Relief Valve on each engine to supply air, bled from the engine intake ducts, to the air conditioning system which maintains stable operating conditions when the aircraft is not in flight.

4.3.10.6 RAIN REPELLANT

Rain is removed from the windscreen by hot, high pressure air directed at the windscreen; operation is selected by the pilot through a switch which controls the rain repellant air control valve.

A thermostat prevents operation of the valve if the air supply temperature is over 250°F.

This does not aid in cockpit conditioning

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4.3.11 MISC SERVICES (Ref. Fig. 11)

4.3.11.1 SPEED BRAKES

The control for the speed brakes, which are hydraulically actuated, is provided by a double acting electrically operated control valve. A single pole double throw switch, located in the right hand throttle handle in the pilot's cockpit, is used to control this valve. The mid position of the switch is marked "HOLD" and the positions, to either side, are identified "IN" and "OUT" respectively.

When this switch is moved to the "IN" or the "OUT" position, the corresponding coil of the valve is energized from the emergency D.C. bus, resulting in movement of the speed brake to the extreme position in the selected direction. Any intermediate position between fully extended or fully retracted can be obtained by moving the switch to "HOLD" when the speed brake reaches the desired position.

4.3.11.2 TURN & SLIP INDICATOR

This is a conventional circuit powered from the emergency D.C. bus.

4.3.11.3 ARTIFICIAL HORIZON

This is a conventional instrument with a fast erection switch and is powered from the instrument transformer 115V 3 phase delta connected.

4.3.11.4 BAIL OUT SIGNAL

A switch, for use in emergency, is located in the pilots cockpit. When the switch is operated a red warning lamp and a signal horn will be energized to warn the radar operator to bail out. A green indicator lamp in the pilots cockpit will also light up when the bail-out switch is operated. Power to the lights and signal horn is fed from the emergency bus through the normally closed contacts of a limit switch mounted on the ejection seat in the rear cockpit.

4.3.11.4 BAIL OUT SIGNAL (Continued)

The extinguishing of the green indicator light in the front cockpit indicates to the pilot that the radar operator has ejected.

4.3.11.5 HYDRAULIC PRESSURE WARNING

Four warning lights located in the Pilots cockpit are as follows:

(a) Flying Control System A

(b) Flying Control System B

(c) Utility System

(d) Emergency Brake

In flying control systems A and B and the utility system, the lights are energized by means of pressure switches when the hydraulic pressure drops below 1000 p.s.i. and are de-energized when the pressure subsequently rises to 3000 psi.

In the emergency brake system, the light is energized by means of a pressure switch when the hydraulic pressure drops below 1,600 p.s.i. and are de-energized when the pressure subsequently rises to 3,000 p.s.i.

4.3.11.6 SKIN TEMP INDICATOR

Full details of the skin temperature indicating system are not yet available but the equipment will include a sensing unit and a combined amplifier/indicator instrument mounted on the pilots instrument panel. Power supply for the system will be single phase A.C.

4.3.11.7 PITOT HEAT

Two pitot heads with conventional heaters are fitted on the fin tip, and supplied with 28v D.C. from the main bus.

4.3.12 COCKPIT LIGHTS (REF. FIG. 12)

Five lighting circuits for each cockpit provide power to:

(a) Main Instrument Panel Instrument lights (RED)

(b) Main Instrument Panel & Console Edge Lights (RED)

(c) Console Flood Lights (RED)

(d) Main Instrument Panel High Altitude console Flood Lights (Amber Flood)

(e) Emergency Flood and Map Lights

The circuits may be turned "ON" or "OFF" independently. Circuits "a", "b" and "c" have individual dimming control through variable transformers. Each emergency flood light has a dimming feature incorporated. Circuits "a", "b" and "c" are fed from the 115 volt main A.C. bus, circuit "d" from the 28 volt D.C. main bus and circuit "e" from the 28V D.C. Emergency bus.

4.3.13 EXTERNAL LIGHTS (REF. FIG. 13)

4.3.13.1 LANDING AND TAXIING LIGHTS

Two AN3129-4523 lamps are fitted on the nose U/C. One is fitted on the steering portion for "Taxi" purposes and the other on the fixed portion of the leg.

The landing and taxiing lights are controlled by a double pole double throw switch with a centre "OFF" position. The control is through the undercarriage door "UP" relay to prevent the landing andtaxiing lights being turned on with the undercarriage up.

One side of the control switch is connected to a single pole single throw relay which controls the landing light, the other side controls the taxi light through a similar relay.

The switch is marked "TAXI", "OFF" and "LANDING". With the undercarriage down and "LANDING" selected,both taxi and landing lights will be illuminated; when "TAXI" is selected, only the taxi light will be on. Power is supplied from the 28 volts D.C. shedding bus.

4.3.13.2 NAVIGATION LIGHTS

The navigation lights, consisting of the right and left wing tip lights and the two fin tip lights, are controlled by a flasher unit through a four pole double throw switch with a centre "OFF" position. Power is supplied from the main D.C. bus.

The control switch has two "ON" positions marked "STEADY" and "FLASH". When the "STEADY" position is selected all lights are turned on, and when the "FLASH" position is selected the two wing tip lights and the white fin tip light will be on together and will flash alternately with the red fin tip light, each being on .75 seconds and off .75 seconds alternately.

4.3.14 ENGINE ANTI-ICING AND DUCT DE-ICING SYSTEMS (REF. FIG. 14)

4.3.14.1 GENERAL

The anti-icing and de-icing installation is divided into three parts:

(a) Ice detecting
(b) Duct intake ramps and lips de-icing
(c) Engine anti-icing

The power supplies to the above systems are as follows:

(a) 28 Volt D.C. supply from the Main D.C. bus to the ice detectors and controls for both engine anti-icing and duct de-icing systems. The supply for the duct de-icing is fed through the U/C scissors switch to prevent heating of the parting strips and shedding areas while the A/C is on the ground.

(b) 115/200 volt three phase supply from the left and right de-icing buses to the respective distributor for heating the parting strips and shedding areas in the duct de-icing system.

4.3.14.2 ICE DETECTING

4.3.14.2.1 GENERAL

The detection of ice formation is accomplished by two electrically heated ice detectors. The units are identical and one is mounted on the lip of each engine intake duct.

Normally, the detector on the left hand engine intake lips is used for ice detection; the detector on the right hand intake lips is used when the right engine only is running, or when some malfunction of the left engine causes the speed to drop below 3020 R.P.M.

4.3.14.3 DUCT DE-ICING SYSTEM

De-icing of the engine ducts is accomplished by electrically heated rubber ice protectors which are automatically controlled. The units which make up this system and their operation are described below.

4.3.14.3.1 CONTROLLER

The de-icing controller is supplied with signals from the ice detector as described in para. 4.3.14.2 and interprets them to control both the duct de-icing and engine anti-icing systems.

On receipt of the first ice detector signal, the controller:

(a) turns on the engine anti-icing system.

(b) energizes the parting strips on the ice protectors.

(c) commences a count of icing signal.

When the count of icing signals reaches a pre-determined number, which may be preset in the controller between the limits of 4 to 12 signals, the controller:

(a) starts the shedding cycle of the duct ice protectors.

(b) recommences the count of icing signals.

Upon completion of the shedding cycle, the controller:

(a) starts a new shedding cycle if the pre-set number of signals (or more) have been received since the commencing of the shedding cycle.

or (b) continues the signal count until the pre-set number of signals has been received and then starts a new shedding cycle.

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T A B L E "A"

BOOT NO	CYCLING SEQUENCE	SHEDDING AREA	LOAD V.A.	PARTING STRIP	LOAD V.A.
1	1	1	1752	1	480
2		2	2088	2	120
	2	3	3840	3	120
	3	4	3840	4	121
	4	5	3840	5	122
	5	6	3840	6	123
	6	7	3840	Bleed Holes	3090
3	7	8	3840	8 9 12	60 245 38
4	10	11	3840	11 13	38 245
5	9	10	3840	10 14	38 245
6	8	9	3840	7 15	60 245
				TOTAL	5390

NOTE: Watt dissipation to arrive at loads in table are as follows:

20 watts per square inch for parting strips.
12 watts per square inch for shedding areas.

Total load for one intake 9320 V.A.
Should it be necessary to increase the watt dissipation to the shedding areas to 15 watts per square inch, the total load to one intaks would then be 10.190 V.A.

or (c) after a pre-set time (variable between 40 to 160 seconds) elapses from (i) the recipt of the last icing signal or (ii) the commencement of the last shedding cycle, will initiate a final shedding cycle at the conclusion of which both the engine anti-icing and the duct de-icing systems are shut off.

Also incorporated in the controller is a provision for adjusting the shedding time for each shedding area between 4 and 12 seconds.

4.3.14.3.2 DISTRIBUTOR

The distributor supplies 115V A.C. power to the heated areas of the rubber ice protectors in the proper sequence and as directed by signals from the controller.

4.3.14.3.3 ICE PROTECTORS

The ice protectors are made of rubber and have wires embedded below the outer surface to carry the heating current. They are divided into two types of heated areas as appropriate to their location as follows:

(a) parting strips, which are continuously heated under icing conditions and thus kept free of ice at all times.

and (b) shedding areas which are intermittently cycled according to the icing conditions encountered.

The protectors are prevented from overheating by thermostats provided on the installation which interrupt the heating current through the distributor for the shedding areas and via a relay for the parting strips.

4.3.14.4 ENGINE ANTI-ICING SYSTEM

Engine anti-icing is accomplished by the use of engine bleed air. Operation of the system is started automatically by the controller through the actuation of an air supply valve and the system functions continuously during icing conditions.

4.3.15 MASTER WARNING SYSTEM (REF. FIG. 16)

4.3.15.1 The complete system consists of the pilot's warning light panel, the master warning control unit, two master warning lights, two advance throttle warning lights and the fire extinguisher warning lights.

4.3.15.2 Two master warning lights are provided on the pilot's main instrument panel. These lights are used in conjunction with the individual system warning lights on the warning light panel and provide a master indication of any trouble existing in the systems. The master warning light may be turned off by pulsing a reset switch on the warning light panel, however, the system warning light will remain on until the fault is cleared.

4.3.15.3 The pilot's warning light panel consists of 20 amber warning lights, a master warning light reset switch, a push to test switch and a dimmer switch. Each light is identified on the lens as to the system it indicates and a light is supplied for each of the following services.

(1) R/H Fuel Low

(2) L/H Fuel Low

(3) Fuel Proportioning Bypass

What is this

(4) Fuel Differential Pressure

(5) R/H Engine Fuel Pressure

(6) L/H Engine Fuel Pressure

(7) R/H Oil Pressure

(8) L/H Oil Pressure

(9) Utility Hydraulics

(10) Emergency Brake

(11) Flying Control Hydraulics System A

(12) Flying Control Hydraulics System B

(13) R/H Alternator Failure

4.3.15.3 (Continued)

(14) L/H Alternator Failure

(15) R/H D.C. Failure

(16) L/H D.C. Failure

(17) Engine Emergency Fuel

(18) Emergency Damping

(19) R/H Low Rotor Overspeed

(20) L/H Low Rotor Overspeed

4.3.15.4 The master warning control unit consists of a series of sequencing relays which operate the warning lights in the following manner:

(a) On receipt of a fire signal, the appropriate fire warning light will go on together with the red master warning light.

(b) On receipt of a signal from either Flying Control Hydraulics systems A or B, the associated amber light on the pilot's warning light panel and the master amber warning light are activated. Should systems A and B fail simultaneously or consecutively, both lights on the pilot's warning light panel will light and also both the red and amber master warning lights.

(c) When the pilot switches on Engine Emergency Fuel, the light on the pilot's warning light panel is activated, a reminder to the pilot that Emergency Fuel has been selected. No master warning light is lit.

(d) Any of the remaining lights as listed in 4.3.15.3 will be energized simultaneously with the amber master warning light on receipt of a warning signal.

4.3.15.5 The master red and amber warning lights are dual units having 2 filaments each in parallel for reliability.

4.3.15.6 The bail out light and U/C warning light are also connected to the test and dimming circuits.

5.0 DESCRIPTION OF EQUIPMENT

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5. DESCRIPTION OF EQUIPMENT

The following pages cover Descriptions of items of equipment which are part of the Electrical system.

5.1 Alternator, Controls & Transformer Rectifier.

5.2 Power Failure Detector.

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5.1 ALTERNATORS

The alternators are designed to give their rated output at .75 to unity P.F., 208 volts 400 cycles when driven at 8,000 r.p.m. The alternators are of the revolving field pattern D.C. excited through slip rings. No separate exciter is fitted thus possible troubles associated with the operation of commutator machines at high altitude is avoided. High remanence materials are used to ensure that the system is self-exciting under all conditions of operation.

The stator winding is a symmetrical 3 phase winding 'Y' connected, and all phases are brought out to terminals. Line and phase voltages are symmetrical and within ± 1% of their mean values. In the line voltage no individual harmonic exceeds 2% and the R.M.S. value of the total harmonic residue is less than 5% of the fundamental value when loaded at any value between zero and full load on any balanced resistive load.

The alternator is capable of being loaded by unbalanced loads such that the line currents may vary by ± 10% of rated value.

5.2 ALTERNATOR CONTROLS AND T.R.U.'S

5.2.1 These units are designed to house the following equipment:

ALTERNATOR CONTROLS

Compounding Transformer and Rectifier
Magnetic Amplifier & Shunt Exciter
Transformer Rectifier
Voltage Sensing Control Unit
Voltage Trimmer Resistors
Tickler Transformer Rectifier
Rotor Field Relay and Thermal Delay Unit
O/V Relay
Tickler Isolation Relay

D.C. SUPPLY

Main Transformer
Magnetic Amplifier
Boost Transformer
Main Rectifier
Voltage Sensing Unit
Trimmer for Volt. Sense Unit
Load Sharing Trans. Rect.
Reverse Current Relay
A rect. to operate hold on coil to Cont. D.C.
Current Transformer-Rectifier for Fault Current Isolation

5.2 ALTERNATOR CONTROLS AND T.R.U.'S (Continued)

5.2.1 (Continued)

The control boxes are mounted in a cabinet. Inlet air at a temperature 80°F is provided which enters at the top of each box, the boxes being interchangeable. The boxes are fastened in the cabinet by quick release fasteners, and either may be removed independently for Servicing.

Terminals are provided in front of each box for the main A.C. and D.C. inputs, outputs and auxiliary connections. The box is of aluminum construction and is braced to withstand accelerations of 10G ultimate in any plane.

5.2.2 PROTECTIVE DEVICES

The number of units used in protecting this system has been kept to a minimum consistent with safety. Protection is afforded against the following faults on the A.C. supply system.

ALTERNATOR CONTROL SYSTEM

GROUND FAULT ON ALTERNATOR OUTPUT LINES

This causes a signal to be received from the O/V or ground fault relay depending on the nature of the fault. The operative relay then closes the rotor relay and shuts down the system.

O/V ON A.C. OUTPUT

The O/V relay operates and provides a signal which closes the rotor relay.

GROUND FAULTS ALTERNATOR WINDINGS

In the event of a fault in the windings of the alternator, a signal could be received from either the ground fault relay or O/V relay: which in turn would close the rotor relay.

CONTROL CIRCUIT FAULTS

Faults within the alternator control circuit would cause the O/V relay to operate, which, in turn, would close the rotor relay, thus shutting down the system.

5.2.3 D.C. SUPPLY SYSTEM

O/V ON D.C. OUTPUT

The O/V relay operates, which isolates the A.C. input to the T.R.U. from the busbar, which then causes the D.C. contactor to drop out.

SHORT CIRCUIT OF MAIN RECTIFIER

This causes a reverse current which operates the polarized relay which provides a tripping pulse to the A.C. feed to the T.R.U., upon which the output from the hold on coil rectifier ceases, and the rectifiers become disconnected from the D.C. bus.

5.2.4 INSTALLATION

ALTERNATOR

This unit incorporates a flange and shaft to AND 10266 and operates at a speed of 8,000 r.p.m. Cooling air is provided.

TRANSFORMER RECTIFIER UNIT

These units are base mounted and will withstand an acceleration of 10G ultimate and both units will fit into a space 13" x 19½" x 27".
18 lbs. of air per minute at a temperature of 80°F are supplied for cooling both units.

WEIGHTS

Alternators	52 lbs. each
Two Alternator Control and TRU Units in Cabinet	168 lbs.
Weight Per Unit	78 lbs.

5.3 POWER FAILURE DETECTOR

The Power Failure Detector is used to monitor the electrical power supply to the Primary A.C. Bus to maintain power to priority loads by detecting low voltage conditions and improper phase sequence.

The Power Failure Detector contains a small induction motor which actuates a single pole double throw micro switch.

In the event of power failure in one or more of the phases, the motor will actuate the switch which in turn is used to actuate relays, and to give a visual indication of alternator failure by means of a warning light.

Weight 20 ozs.

6.0 DEVIATIONS

DEVIATIONS

6.1 ACCESSIBILITY OF ALTERNATORS & DRIVES

6.1.1 Specification MIL-E-7614, para. 3.5.2 and 3.6.1 require that "the generator shall be accessible for inspection of all brushes, commutators and slip rings while installed.

The constant speed drive and flexible shaft shall be accessible for inspection and servicing while installed and for removal for servicing without requiring the removal of other accessories except the generator."

6.1.2 The above requirements are complied with except for removal of the constant speed drive. Removal of the alternator and constant speed drive requires the partial removal of the engine.

6.2 SWITCHES - SPACE PROVISIONS

6.2.1 Specification MIL-E-7080, para. 3.4.1.4 require "space shall be provided on each switch panel containing four or more switches, for subsequent installation of one spare switch conforming to Drawing AN3022 and one switch conforming to Drawing AN 3023."

6.2.2 Space limitations on switch panels prevent installation of additional switches.

6.3 PRESSURIZED CONNECTORS

6.3.1 Specification MIL-W-5088A, para. 3.6.6.6 requires that "Pressurized connectors shall be installed with the flange on the high pressure side."

6.3.2 In some cases pressurized connectors have been installed with the flange mounted on the low pressure side. This occurs in locations where the wiring installation is such that compliance with the above requirement could only have been achieved by the use of an additional connector.

6.4 CABLE ROUTING

6.4.1 Specification MIL-W-5088A, para. 3.7.3.5 requires that "cables to each equipment which must operate to maintain flight of the aircraft under normal

6.4.1 (Continued)

or emergency conditions shall be separately routed from other cables."

6.4.2 Space limitations prevent separate routing of cables essential to maintain flight.

6.5 CABLE GROUPING

6.5.1 Specification MIL-W-5088A, para. 3.7.3.4 requires that "unprotected wires and cables of the primary electrical power system shall not be bundled or grouped with distribution circuit wires and cables."

6.5.2 Power source cables are bundled with distribution cables in some instances where space limitation prevents segregation.

6.6 BATTERY DISCONNECT

6.6.1 Specification CAP 479, para. 70.05(3) requires "Disconnect - a quick disconnect device shall be provided at the battery terminals to disconnect the battery from the electrical system."

6.6.2 Nut type terminals are used with a cover to provide terminal insulation, corona barrier and high creepage protection.

Weight saving factor and reliability of a hermetically sealed nickel-cadmium battery preclude the necessity for quick disconnect devices.

6.7 WARNING LIGHTS

6.7.1 Specification ARDCM 80-1, para. 6A.172(b) requires that "the caution indicator system shall consist of a master indicator light and an indicator panel. The master light shall be red in colour and shall be labelled "Master Caution" --------- The caution indicator panel -------- shall provide a suitable visual indication, red in colour ---------"

6.7.2 One master warning light and all caution indicators are amber in colour.

The warning light system proposed by Avro was approved by the RCAF, Reference letter S1038-105-4 (ACE-1) dated 23 August, 1955.

6.8 CIRCUIT BREAKERS

6.8.1 Specification CAP 479, para. 21.62(1) requires that "In single or tandem pilot aircraft, the circuit breakers shall be located forward of the inboard face of the right console."

6.8.2 Limitation of space precludes the installation of circuit breakers in the cockpit. Circuit breakers are used for protection only and not as combination protection and switch. Trip free breakers are used which cannot be closed when a fault in the circuit exists. The circuit breakers will be located on a circuit breaker panel in the nose wheel bay.

6.9 ISOLATION OF ELECTRICAL EQUIPMENT

6.9.1 Specification ARDCM 80-1, para. 13.615 requires that "Electrical equipment and fuel should be isolated to prevent ignition of the fuel by arcing of broken electrical and fuel lines resulting from battle damage, accidental breakage or normal arcing.

-----Fuel, oil and hydraulic lines and equipment shall never be located in a position where leaking fluid will come in contact with electrical equipment through either the effects of gravity, air flow or battle damage, and hydraulic lines will be routed below electrical equipment and wires whenever they cross paths, pursuant to Specification MIL-E-7563."

6.9.2 Electrically operated fuel control valves and associated electrical cables are located inside the fuel tanks. Fuel and hydraulic lines and electrical cables are located in close proximity in the fuselage under the wing and aft of station 485.

The fuel tanks in the wing sections are integral with the wing structure, and space limitations in other sections of the airplane preclude possibilities for wider separation of electrical components and cables from fuel and hydraulic lines. Where necessary, adequate insulation and explosion proof type components and connectors are installed to avoid possible arcing and fire hazards.

6.10 IGNITION CIRCUIT

6.10.1 Specification ARDCM 80-1, para. 9.522 requires that "single and twin engine aircraft shall utilize ignition systems with dual circuitry, each circuit being separately fused. The dual circuitry shall extend back to the power source."

6.10.2 Single wire circuitry is installed for the ignition system as that part of the system supplied on the engine has only a single ignition circuit.

Engine relight in the air is accommodated by means of a separate circuit which is connected to the common ignition point on each engine.

6.11 REVERSE CURRENT CUT-OUTS ACCESSIBILITY

6.11.1 Specification CAP 479, para. 70.26(1) requires that "the reverse current cut-out(s) shall be accessible for unhampered inspection and maintenance while the engines are running with the aircraft on the ground."

6.11.2 The reverse current cut-outs are not accessible for unhampered inspection and maintenance when installed as these devices require an air conditioned location and are therefore installed in the transformer rectifier unit and alternator controls box.

These protection devices are accessible only when the transformer rectifier unit and alternator controls box is removed from the aircraft. (Resetting of these units is accomplished, on observation of D.C. failure warning light, by means of a switch located in the pilot's cockpit).

6.12 OVERHEAT DETECTION - TURBOJET ENGINE INSTALLATION

6.12.1 Specification CAP 479, para. 23.61 requires that "an overheat detection system of approved type shall be installed in all turbojet ----- propelled aircraft."

6.13.2 No specific overheat detection system is installed, the fire warning system is based on overheat temperature and additional overheat protection would, therefore, be duplication.

APPENDIX I

TERMS OF REFERENCE

Extracts from specifications: ———— AIR 7-4
ARDCM 80-1
CAP 479

TERMS OF REFERENCE

EXTRACTS FROM SPECIFICATION AIR 7-4

4.3 LANDING GEAR

4.3.1 A fully retractable, power operated landing gear with the nose wheel steerable, shall be fitted. A device shall be fitted to prevent inadvertent retraction of the undercarriage while the wheels are on the ground.

4.3.2 EMERGENCY SYSTEMS

Emergency operation of the landing gear shall be provided as follows:

4.3.2.1 Wheels Down - An emergency means of extending and locking the landing gear in the event of failure of the landing gear primary power system shall be installed.

4.3.2.2 Wheels Up - Override Retraction: A control shall be installed to override the device for prevention of inadvertent retraction, thus allowing emergency retraction of the undercarriage when the wheels are on the ground.

4.4 SPEED BRAKES

4.4.1 The controls shall be arranged such that the pilot can set the brakes at the fully opened, intermediate, or fully closed position. -------

4.5 ANTI-ICING PROTECTION

4.5.1 GENERAL

Adequate anti-icing protection shall be incorporated so that the aircraft can be operated for its maximum endurance under all weather conditions, including heavy rain or snow and icing conditions.

4.5.2 Particular attention shall be given to protecting the following:

4.5.2.1 Engine

4.5.2.2 Windscreen

4.5.2.3 Radome

4.5.2.4 Engine Air Intakes

4.5.2.5 Control Surfaces, Flaps and Doors

Extracts From Specification AIR 7-4 (Cont'd)

4.6 PROTECTION FROM ENEMY FIRE

4.6.1 During the design of the aircraft special consideration shall be given to the building-in of invulnerability to enemy fire by the maximum use of such inherent types of protection as positioning of components, fuel and hydraulic lines, and like items.

4.7 MAINTENANCE AND READINESS

4.7.2 TURN AROUND

4.7.2.1 Turn around is a first line maintenance operation required to return the aircraft to an operationally serviceable condition after landing and shall include replenishment of all consumable stores and liquids as well as a between flight inspection. -----

4.7.2.2 Turn around time shall not exceed five minutes.

4.7.3 READINESS

4.7.3.1 The aircraft shall be designed such that, once having been certified operationally serviceable, it shall be capable of remaining serviceable at the highest state of readiness for a period of at least 24 hours.

4.7.4 AUTOMATIC DISCONNECT COUPLINGS

All external connections from the aircraft to ground equipment shall be made through automatic disconnect couplings preferably located in an adjacent area. The automatic couplers shall be positioned such that they will be disconnected by the aircraft taxiing straight away.

6 POWER PLANT INSTALLATION

6.1.2 Variations of thrust with the afterburners operating shall be provided, and shall be controlled by a means agreed between the Department and the Contractor, in conjunction with the engine manufacturer. Such variation shall be effected by means of the throttle control for each engine.

Extracts From Specification AIR 7-4 (Cont'd)

6.2 ACCESSORIES

6.2.1 The hydraulic pump and electric generator shall be duplicated or arranged so that there is an adequate supply of power to operate essential hydraulic and electrical services and allow a limited combat capability after the failure of any one engine.

6.3 ENGINE INSTALLATION

6.3.3 All engine controls, fuel lines, and electrical leads shall incorporate quickly detachable connectors to facilitate engine installation and removal.

6.3.4 Special attention shall be given to the engine and accessory installations so that maintenance and inspection will be facilitated to the utmost.

6.6 FIRE PREVENTION

6.6.1 A continuous wire type or a rate of change of temperature type fire detection system shall be installed.

6.6.2 A fire extinguishing system shall be installed in accordance with CAP 479.

6.7 ENGINE STARTING

6.7.1 The engine starting shall be under the control of the occupant of the front cockpit. -------

8 INSTRUMENTS

8.2 INSTALLATION

8.2.4 All air lines and electrical leads shall be flexible and fitted with quick-disconnects and shall be of sufficient length to allow easy instrument removal.

8.3 ENGINE INSTRUMENTS

8.3.1 All engine instruments shall be of the electrical remote single indicating type and 2 inch case size in accordance with U.S. Drawing AND10412 and shall be clamp mounted in accordance with MIL-C-6818.

Extracts From Specification AIR 7-4 (Cont'd)

8.3.2 The tachometers shall conform to the requirements of OR1/2-5.

8.3.3 A capacitance type fuel contents system shall be installed.

9 ELECTRICAL EQUIPMENT

9.1 GENERAL

9.1.1 All electrical equipment shall be installed in accordance with Specificiations MIL-E-7614 and MIL-E-7080, MIL-G-6099 and the requirements of U.S.A.F. Drawings 53D6792 and 53D6793 applied to the rating of the alternators except as otherwise detailed in this specification. The requirements of U.S.A.F. Drawings 53D6792 and 53D6793 have precedence over MIL-G-6099. Approved hermetically sealed components shall be installed where available.

9.2 A nickel-cadmium type battery or equivalent shall be installed.

9.3 An external alternating current power receptacle in accordance with the outline of U.S. Standard Drawing AN3114 shall be provided.

9.4 INTERIOR LIGHTING

9.4.1 The cockpit lighting system shall be in accordance with CAP 479.

9.4.2 Where possible all cockpit control panels shall be provided with red console lighting in accordance with USAF Specification MIL-P-7788. The intensity of illumination shall be controllable.

9.5 EXTERIOR LIGHTING

9.5.1 External lights shall be provided in accordance with CAP 479 except that the landing light shall be installed on the nose gear structure, and upper and lower fuselage lights are not required.

9.5.2 A taxi light shall be fitted to the nose undercarriage assembly such that it will follow the direction ofthe nosewheel steering.

EXTRACTS FROM CAP 479

ENGINEERING DATA

2.04 REQUIREMENTS FOR ENGINEERING DATA

(7) Electrical Load Analysis - The electrical load analysis shall present in tabular form complete and accurate information on the electrical loads for both AC and DC power, that may occur during the following operating conditions:

(a) taxiing
(b) take off and climb
(c) cruise
(d) maximum continuous level speed
(e) combat and
(f) landing

The analysis shall form the basis for selecting the necessary power supply equipment, for designing the electrical distribution system and for determining the safe reserve capacity of the electrical system. During the life of the aircraft the analysis will be used to determine the feasibility of installing additional items of electrically operated equipment without jeopardizing the safety of the aircraft.

(16) Reproducible Schematic Diagrams - Schematic diagrams shall be submitted for the hydraulic, electrical, pneumatic, communications, armament, fuel and lubrication systems.

CONTROLS

21.02 Clearance and Ease of Operation - Controls shall be designed and located so that the operator can readily move each control throughout its entire range of travel, without moving any other control, while wearing heavy gloves and flying equipment, and with shoulder harness in place, but not necessarily locked.

21.03 Direction of Motion

(1) Except for three position switches, where the center position shall be "OFF", all controls shall be so designed that their movement in a predominantly forward, upward, or clockwise direction shall result in increased performance of the component or the aircraft and, conversely, their movement in a backward, downward or counter-clockwise direction shall result in decreased performance of the component or the aircraft.

Extracts From CAP 479 (Cont'd)

(2) All variable controls operated by a rotary motion shall move clockwise from the "OFF" position through "LOW" or "DIM" to "HIGH" or "BRIGHT".

(3) The direction of motion of controls is described with reference to the operator, and not to the aircraft.

21.04 SHAPE AND LOCATION OF CONTROL KNOBS

To assist identification without visual reference control knobs shall be of distinctive shape. All controls of a like function should be grouped together, with normal operating and emergency controls having prefered position.

FLIGHT CONTROLS

21.25 SPEED BRAKE CONTROL

The speed brake (air brake) control shall be located on the power control or adjacent to the power quadrant in stick-controlled aircraft. The control motion shall be aft for the speed brake operative and forward for the brake inoperative, and the control shall be marked "OUT" and "IN" respectively. Where the speed brake control is of a type which can be switched off, the control shall be marked: "OUT" - "OFF" - "IN".

21.30 LANDING GEAR CONTROL AND INDICATORS

(2) Emergency System - Emergency operation of the landing gear shall be provided for in the landing gear control as follows:

(a) Wheels Down - The landing gear control lever shall be moved through a gate, past the normal "Wheels Down" position, to effect emergency operation.

(b) Wheels Up-Override Retraction - A control, adjacent to the landing gear control lever, shall be operated momentarily to enable the movement of the landing gear control lever from "Wheels Down", to "Wheels Up", thus overriding the safety device fitted to prevent inadvertent retraction.

(3) Warning Light - The landing gear control lever shall incorporate a red warning light which shall be illuminated when any gear is not exactly in the position selected, as when any power control is placed below the minimum cruise position with the landing gear not safely down and locked.

Extracts From CAP 479 (Cont'd)

(4) Landing Gear Position Indicators - Landing gear position indicators which shall indicate the position of each wheel at all times shall be provided on the instrument panel or adjacent to the landing gear control lever.

21.33 GROUND STEERING CONTROLS

(1) Nose Wheel Type Aircraft - The nose wheel lock or steering control shall be in accordance with the RCAF Aircraft Specification.

21.40 EMERGENCY CONTROLS

General - All emergency controls shall be coloured with black and yellow strips.

21.41 FIRE EXTINGUISHING CONTROLS AND INDICATOR

(1) The fire fighting controls shall be located on an emergency panel accessible to the pilot's throttle hand.

(2) Fire and Overheat Warning Indicators - A red fire warning light shall be incorporated in or installed adjacent to each fire fighting control in such a manner as to indicate the location of the fire. If overheat warning lights are installed, they shall be incorporated in the fire warning light by means of an interrupted circuit. The indication for overheating shall be an interrupted signal, and the indication for fire shall be a steady signal from the fire warning light.

21.56 FUEL SYSTEM CONTROLS

(1) General - The fuel system controls shall be grouped as far as practicable according to function on a fuel system panel.

(3) Emergency Fuel Pump - When a separate emergency fuel pump control is provided this control shall be located adjacent to the appropriate fuel selector. The control motion shall be predominantly forward or upward for "ON".

21.59 IGNITION SWITCHES

(2) Turbojet and turbo-propeller Engines - The master switch for turbojet and turbopropeller engines shall be located adjacent to its respective power control.

Extracts From CAP 479 (Cont'd)

ELECTRIC CONTROLS

21.60 Master Electrical Switches - In single or tandem pilot aircraft, the master electrical switches shall be located together, forward on the right hand console.

21.61 ELECTRICAL POWER CONTROLS - The electric power controls include the battery, generator, primer, starter, booster coil, inverter, and oil dilution controls. In single or tandem pilot aircraft, these switches shall be mounted on a panel on the right hand side of the cockpit, forward of the light switch panel. In all aircraft the booster coil, primer, and starter switches for each engine shall be mounted side by side, so that they can be operated together by one hand.

21.62 CIRCUIT BREAKERS

(1) In single or tandem pilot aircraft, the circuit breakers shall be located forward on the inboard face of the right hand console.

(2) The circuit breakers should be grouped according to function, with the most critical circuit breakers in the most accessible position. In aircraft where cockpit space is limited, only those circuit breakers essential to safety of flight should be located in a position accessible to the pilot. Circuit breakers not essential to the safety of flight may be placed in the cockpit in such positions as not to occupy space considered more essential for operating controls.

21.63 LIGHT SWITCHES

(1) General - In single or tandem pilot aircraft all light switches except landing light switches shall be located on the right hand console to the rear of the master switches, and functionally grouped together.

(2) Landing Light Switches - The landing light switches shall be adjacent to the landing gear control lever or the power control lever.

MISCELLANEOUS CONTROLS

21.73 Defrosting, De-icing, Air Conditioning, and Cabin Pressurization Controls.

(2) In single or tandem pilot aircraft, these controls shall be grouped on a panel on the right hand side of the cockpit.

Extracts From CAP 479 (Cont'd)

MATERIALS

23.53 Electrical Equipment - Electrical equipment located in potential fire zones shall be explosion-resistant.

WARNING SYSTEMS

23.60 Fire Warning System - A fire warning system of approved type shall be installed in all aircraft, to indicate fires in potential fire zones.

23.61 Overheat Detection System - An overheat detection system of approved type shall be installed in all turbojet, turbopropeller, and rocket propelled aircraft.

FIRE EXTINGUISHING SYSTEM

23.74 Operation: The fire extinguishing system shall be operated:

(a) manually, by a control readily accessible to the pilot, or to the flight engineer, if applicable and

(b) automatically, by a switch which will be actuated by a a crash landing.

The automatic system shall provide one discharge of agent to each power plant, and one discharge of agent to each potential fire zone other than cargo compartments.

EXTREME CLIMATIC CONDITIONS

24.60 Batteries

(1) The output of electric storage batteries falls progressively with decreasing temperatures and they must be removed frequently for re-charging and placing in heated storage during cold weather. It is therefore essential that the battery storage be easily accessible to personnel wearing heavy clothing and mitts.

(2) Dry batteries are seriously affected by low temperatures, their output falling until the chemical action ceases.

Extracts From CAP 479 (Cont'd)

24.61 INSULATION

(1) Most types of synthetic rubber insulation on electric cables are unsatisfactory at low temperatures as they loose their flexibility and crack when bent. This defect applies particularly to the wiring of landing gears and wing circuits on folding-wing aircraft. Natural rubber or silicone rubber insulation is recommended for use at such locations.

(2) Sharp bends in electric cables should be avoided as much as practicable.

24.62 SWITCHES

(1) Switches shall be sufficiently large and widely spaced to permit operation while wearing heavy gloves.

(2) Spring return tumbler switches shall be used for oil dilution, engine starting, booster coil, and priming circuits. Push button switches should be avoided.

24.63 PLUGS AND SOCKETS

Plugs and sockets shall be easily accessible. Connectors should be installed at unavoidable sharp bends in cable runs where necessary to permit easy replacement.

24.64 OIL SEALS

Oil seals on starter motors and generator shall be made of a material that will withstand extreme low temperatures and that is not affected by diluted engine oil.

ANTI-ICING AND DE-ICING REQUIREMENTS

26.05 AIR INTAKES

Turbojet and Turbo-propeller Engines - The entrance to the air induction system of turbojet and turbo-propeller engines shall be protected against ice formation. This is necessary because of the severe power losses which may be cause by relatively small disturbance to the air flow. In addition, all airframe parts in the air induction system, such as engine accessory covers or air duct valves, which may be subject to ice accretion, shall be protected.

Extracts From CAP 479 (Cont'd)

26.06 WINDSCREENS

Approved means shall be provided for preventing the formation of ice, or the accumulation of snow, on the pilot's, second pilot's, and bomb aimer's windscreens. Means shall also be provided for preventing the fogging and frosting of all transparent areas provided for the use of the crew.

ELECTRICAL SYSTEM

70.01 GENERAL

The electrical power supply shall be either:

(a) a direct current, nominal 28 volts, single wire, negative grounded system; or

(b) an alternating current, 115/200 volts, nominal 400 cycles per second, three phase, four wire, star connected, neutral grounded system; or

(c) an alternating current, 115/200 volts variable frequency, three phase, four wire, star connected, neutral grounded system.

70.02 CHARACTERISTICS

Characteristics of the electrical power supply shall conform to the technical requirements of US Military Specification MIL-E-7894, Electric Power, Aircraft, Characteristics of.

70.03 DUPLICATE POWER SOURCES

Any combination of engine that will maintain the aircraft in flight shall drive generators of sufficient capacity to operate the electrically actuated services that are normally necessary to permit continued flight and landing.

70.04 GENERATORS

(1) Accessibility-Generators shall be installed so that:

(a) they may be readily removed and replaced, and the security of the mounting may be checked visually;

(b) the terminals are readily accessible for maintenance and inspection;

(c) the brushes, commutators, and slip rings may be easily inspected without removel of the generator.

Extracts From CAP 479 (Cont'd)

70.04 GENERATORS (cont'd)

(2) Mounting - An approved quick attach/detach device may be utilized for mounting.

(3) Cooling - Where required by the generator design, means shall be provided for adequate cooling. The cooling air duct shall have sufficient flexibility to allow for the maximum possible relative movement between generator and duct, shall not impose stresses on the generator, and shall be designed to minimize the possibility of water or other foreign material entering the generator.

(4) Wiring - Generator control wires and power wires shall be routed separately and in such a manner as to minimize the possibility of short circuits between them. The wiring shall have sufficient slack so as not to place stresses on the generator terminal block, under all conditions of engine vibration. The terminals shall be provided with insulating nipples. The negative cables of DC generators shall be grounded to the nearest main structure. The negative cable of AC generators shall be grounded to the nearest main structure after passing through the differential current transformer.

(5) Rating - In aircraft provided with only one generator, the rated continuous current capacity of the primary generator system should exceed the average continous electrical load by 33 per cent. In aircraft provided with more than one generator, the rated continuous current capacity of the primary generator system should be at least twice the average continuous electrical load.

(6) Frequency - Variable frequency alternating current generators shall deliver power within the frequency range of 380 to 1200 cycles per second. Constant frequency alternating current generators shall deliver power within the frequency range of 380 to 420 cycles per second.

(7) Constant Speed Drives - Where a constant speed drive is used it shall be accessible for inspection and servicing in place, and shall be capable of being removed and replaced without requiring the removal of other accessories except the generator. It shall be installed so that the center line of the input shaft of the constant speed drive coincides with the center line of the power plant accessory drive. Adequate cooling of the constant speed drive shall be provided.

Extracts From CAP 479 (Cont'd)

70.04 GENERATORS (cont'd)

(8) Controls - Adequate controls shall be provided to maintain the characteristics of the power supply within the specified limits.

70.05 BATTERIES

(1) Type - Storage batteries shall conform to approved type specifications.

(2) Installation - Batteries shall be installed in accordance with approved specifications and drawings and shall be provided with retaining devices of approved type. The installation shall not require the use of specially designed ground handling equipment for servicing, testing or replacement of batteries, and shall not impose hazardous conditions on maintenance personnel, the aircraft, or the batteries, during servicing, testing or replacement. Precautions shall be taken to prevent damage to equipment that might be harmed by battery acid spillage, spray, or fumes.

(3) Disconnect - A quick diconnect device shall be provided at the battery terminals to disconnect the battery from the electrical system.

70.06 EXTERNAL POWER RECEPTACLES

(1) Unless otherwise specified, external power receptacles shall be provided in all aircraft except gliders. Receptacles shall conform to approved standards, specifications, and drawings.

(2) Receptacles shall be accessible from the ground. A spring loaded access door shall be installed over the receptacle(s) so that the door will close automatically after the power cable is withdrawn. Receptacles shall not be installed in wheel wells or bomb bays.

(3) The receptacle(s) shall be located and connected to the electrical system so that the voltage drop between the receptacle(s) and the starter terminals does not exceed two volts.

(4) If the external power receptacles are installed side by side, there shall be a distance of at least 2½ inches separating them, center to center.

Extracts From CAP 479 (Cont'd)

70.06 EXTERNAL POWER RECEPTACLES (cont'd)

(5) Receptacles shall be far enough removed from propellers, jet intakes and exhaust outlets to avoid hazards to personnel. They shall be located so as to provide adequate spacing between the external power plant and any openings from which fuel might spill during refuelling, defuelling, or engine starting.

(6) For interceptor fighters the installation of the receptacle(s) shall be such that the external power cables will be automatically released upon movement of the aircraft.

DISTRIBUTION

70.20 WIRING

The installation of electrical wiring in aircraft shall conform to the technical requirements of U.S. Military Specification MIL-W-5088. Wiring, Aircraft, Installation of, and applicable related specifications, drawings and other publications, except that preference shall be given to equivalent specifications etc., issued by the Department of National Defence.

70.21 BONDING

All metallic parts of the aircraft and its equipment and accessories shall be bonded in accordance with the technical requirements of U.S. Military Specification MIL-B-5087, Bonding; Electrical (for Aircraft), and applicable related specifications, drawings, and other publications, except that preference shall be given to equivalent specifications etc., issued by the Department of National Defence.

70.22 RADIO INTERFERENCE

All equipment shall be installed so as to insure freedom from radio interference in accordance with RCAF Specifications RAD 81-2, Radio Interference Suppression in Aircraft.

70.23 FILTERS, CAPACITORS (Radio Interference)

(1) Filters shall be installed only where necessary to insure satisfactory freedom from radio interference. Filters shall be of an internally grounded type. Capacitor type filters should be used except where this type is not adequate.

UNCLASSIFIED

SECRET

Extracts From CAP 479 (Cont'd)

70.23 FILTERS, CAPACITORS (cont'd)

(2) The filters should be located as near as practicable to the source of interference. Filters shall be installed so that:

(a) they are properly grounded through their mountings; and

(b) the connecting wire to a capacitor type filter shall be as short as practicable and not longer than 4 inches.

70.24 CIRCUIT PROTECTIVE DEVICES

(1) Circuit breakers shall be used for the protection of circuits. Fuses, fusible links, current limiters, and other circuit protective devices shall only be used where specifically authorized by the RCAF. Remote control and automatic reset type circuit breakers may be used when practicable.

(2) Unless otherwise specified, circuit breakers shall be installed for the protection of all circuits at the point of power take off, or not more than 3 feet from the bus. Vital circuits such as indicators, warning, lighting, and armament circuits, shall be individually protected; non-vital circuits may be connected in multiple to a single circuit breaker.

(3) Switch circuit breakers shall be used only where it is desired to combine circuit protection and control in one unit.

(4) Push-pull circuit breakers shall be used where it is desired to combine circuit protection and emergency control.

(5) Automatic-reset circuit breakers shall be preceded by switches, control relays, or other means of opening the circuit.

(6) Space shall be provided on the circuit breaker panels for the installation of at least one additiona circuit breaker for each group of six breakers.

(7) The capacity of the circuit protective device selected shall be such as to ensure opening of the circuit before damage will occur to the circuit wiring under short circuit or sustained overload conditions.

Extracts From CAP 479 (Cont'd)

70.24 CIRCUIT PROTECTIVE DEVICES (cont'd)

(8) Guards that do not interfere with operation of the unit may be installed on switch circuit breakers, subject to the approval of the RCAF.

70.25 VOLTAGE REGULATORS

(1) Voltage regulators should be located so that they are accessible for maintenance and adjustment while the engines are running with the aircraft on the ground. In twin and multi-engine aircraft, except fighters, the voltage regulators should be accessible in flight. Regulators shall be adequately ventilated.

70.26 REVERSE CURRENT CUT-OUTS

(1) The reverse current cut-out(s) shall be accessible for unhampered inspection and maintenance while the engines are running with the aircraft on the ground. In twin and multi-engine aircraft, except fighters, the reverse current cut-out(s) should be accessible in flight and shall be located close to the voltage regulators serving the same generator.

(2) The reverse current cut-out(s) shall be remotely controlled by controls installed on the flight engineer's electrical switch panel, if installed, or on the pilot's switch panel.

70.27 SWITCHES

(1) Toggle switches used in flight shall be accessible and shall be located and grouped so as to reduce the hazard of improper selection and inadvertent operation. Switch guards may be installed, subject to approval of the RCAF.

(2) Space shall be provided on each switch panel containing four or more switches for the installation of at least two additional switches.

(3) Limit switches shall be installed and protected so that foreign matter will not interfere with their operation. Where limit switches are used in landing gear control and warning systems, hermetically sealed limit switches and switch actuators shall be used, if available. Toggle switches and push button switches shall not be used as limit switches.