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AVRO AIRCRAFT LIMITED

CF105

BROCHURE E-4

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ELECTRICAL SYSTEM

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AVRO AIRCRAFT LIMITED

MALTON ONTARIO

Please send to DADR/AFHQ

S/L J H Cooper

BROCHURE E4-A

DESCRIPTION OF

ELECTRICAL SYSTEM

FOR

CF-105

SUPERSONIC ALL WEATHER FIGHTER

Classification cancelled/changed to..................

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

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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.

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 Altitude

Sea level to 60,000 ft.

1.2.2 Temperature

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.

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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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FORM 1749 A

INDEX OF SERVICES

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.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.

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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

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.

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

Down 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.

What is this + why will it go on

OK

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FORM 1749 A

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

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.

Is this not locked in position

yes opened

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.

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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".

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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

(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.

If this was not normally at taxiing speeds

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.

TABLE "A"

BOOT NO	CYCLING SEQUENCE	SHEDDING AREA	LOAD V.A.	PARTING STRIP	LOAD V.A.
1	1	1	1752	1	480
2	1	2	2088	2	120
2	2	3	3840	3	120
2	3	4	3840	4	121
2	4	5	3840	5	122
2	5	6	3840	6	123
2	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

(4) Fuel Differential Pressure

What is this

(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.

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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.

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.

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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.

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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

UNCLASSIFIED

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.

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.

Extracts From CAP 479 (Cont'd)

70.27 SWITCHES (cont'd)

(4) Ignition switches shall be installed so that they are readily accessible and not liable to be inadvertently operated. In metallic aircraft the ignition switches shall be grounded through the mounting, and in non-metallic aircraft the ignition switches shall be grounded by a wire, or wires, to the engine mount(s).

70.28 RELAYS

Hermetically sealed relays should be used if available. Other relays shall be mounted so that contacts can be inspected without having to remove them or other equipment. Relays having exposed contacts shall not be located in compartments where fuel or combustible vapor is likely to be present, or where the armature might inadvertently be disturbed.

70.29 RESISTORS

Resistors shall be installed so that the heat dissipated will not harm or interfere with the operation of other equipment.

INTERIOR LIGHTING

70.30 FLOOD LIGHTING

Red flood lighting shall be installed in all aircrew compartments of all aircraft. In certain aircraft white flood lighting may also be required in the lower part of the pilot's compartment for use by day at high altitudes.

70.31 ANCILLARY LIGHTING

Ancillary amber lighting shall be installed in the pilot's(s') and navigator's compartments for map reading. Desks and tables at all aircrew stations shall be lighted by one or more flexibly mounted light. The number of ancillary lights shall be kept to the minimum necessary for adequate illumination.

70.32 INSTALLATION

The intensity of all lights in aircrew compartments shall be controlled by conveniently located rheostat switches. Interior lights shall be arranged so as to avoid direct and reflected glare.

Extracts From CAP 479 (Cont'd)

70.33 INSTRUMENT LIGHTING

Instruments shall be illuminated by shielded red light. The intensity of illumination shall be controlled by conveniently located rheostat switches.

70.34 CONSOLE LIGHTING

Consoles shall be illuminated by plastic plate type red lighting.

EXTERIOR LIGHTING

70.41 EXTERIOR LIGHT ASSEMBLIES

(1) Aeroplanes shall be provided with the following exterior light assemblies:

(a) navigation lights;

(b) fuselage lights;

(c) taxi lights (when specified);

(d) landing lights; and

(e) other exterior lights required by the RCAF Aircraft Specification.

(2) The installation of additional exterior light assemblies for any specific purpose shall be subject to approval by the RCAF.

70.42 GLARE

Exterior lights shall be installed or shielded as to prevent direct or reflected glare.

70.43 CONTROLS

(1) All exterior light controls shall be operable by the pilot.

(2) Separate controls shall be provided for each exterior light assembly, except the navigation and fuselage light assemblies.

(3) Controls for navigation and fuselage lights shall provide for selection of either steady or flashing operation. The control shall be a single switch having three positions, viz: "Flash" - "Off" - "Steady".

Extracts From CAP 479 (Cont'd)

70.44 NAVIGATION LIGHTS

(1) Colour. Navigation lights shall consist of:

(a) red and green wing tip lights; and

(b) red and white tail lights.

(2) Installation. Navigation lights shall be installed as follows:

(a) one red light on the port wing tip;

(b) one green light on the starboard wing tip; and

(c) one red and one white light at the extreme tail. Where practicable, the tail lights shall be installed in a vertical plane, the red light above the white light. If this is not practicable, they shall be installed in a horizontal plane, the red light to port of the white light. The distance between the center of the tail lights shall not exceed 6 inches.

(3) Intensity. The intensity of navigation lights shall be as follows:

(a) wing tip lights shall be 24 candle power; and

(b) tail lights shall each be 21 candle power.

(4) Visibility. Each wing tip navigation light shall be visible from all positions within an angle of 90 degrees above and below the horizontal, and from directly forward through an angle of 110 degrees measured horizontally outward from a line parallel to the longitudinal axis of the aeroplane. Where practicable, provisions should be made for visually checking operation of the wing tip lights. Tail lights shall be visible from all positions within an angle of 90 degrees above and below the horizontal, from directly astern through an angle of 90 degrees to port and starboard.

(5) Operation. The navigation and fuselage lights shall normally operate on a flashing sequence, in two circuits, alternately operating:

(a) wing tip lights and white tail light on one circuit; and

(b) fuselage lights and red tail light on the other circuit.

Extracts From CAP 479 (Cont'd)

70.44 NAVIGATION LIGHTS (cont'd)

(5) (cont'd)

One circuit shall be ON while the other is OFF. Each circuit shall flash at a rate of approximately 40 cycles per minute. Provision shall also be made for continuous operation of both circuits together when selected by the pilot.

70.46 TAXI LIGHTS

(1) Colour. Taxi lights shall be white, with clear cover glass.

(2) Installation. Taxi lights shall be installed if required by the RCAF Aircraft Specification. Taxi lights may be mounted on the fixed portion of the landing gear or on the wing. On aeroplanes equipped with nosewheels, the taxi light(s) shall, if practicable, be installed on the nose wheel assembly, so that the light will turn with the wheels. The width of the taxi light beam shall be approximately 30 degrees and the depth of the beam shall be approximately 5 degrees.

(3) Power. Taxi lights shall be 150 watts.

(4) Visibility. The taxi light(s) shall be installed so as to afford the pilot the maximum visibility for manoeuvering on the ground. Unless otherwise specified, the taxi lights shall be installed so that, with the aeroplane on level ground in normal taxiing position, the center of the beam strikes the ground at a point 250 feet ahead of the pilot's windscreen. This point shall be the point nearest the longitudinal axis of the aeroplane that is visible to the pilot at that distance.

70.47 LANDING LIGHTS

(1) Colour. Landing lights shall be white, with clear cover glass.

(2) Installation. Where practicable, landing lights shall be installed on the wing.

(3) Power. If one light only is provided it shall be 600 watts. If two lights are provided they shall each be 250 watts.

(4) Visibility. The landing light(s) shall be installed so as to afford the pilot the maximum visibility for landing.

Extracts from CAP 479 (Cont'd)

ACCESSORIES

70.51 TRANSFORMERS

Wherever practicable, transformers shall not be installed in locations having high ambient temperatures. Sufficient space shall be provided around transformers to permit free circulation of air for cooling. Transformers shall not be installed below lines containing inflammable fluids.

70.53 MOTORS

(1) Wherever practicable, motors shall not be installed in locations having high ambient temperatures. Sufficient space shall be provided around motors for inspection and servicing, and to permit free circulation of cooling air. Motors shall be installed and provided to prevent the entrance of water or other foreign matter and so that a minimum hazard exists due to leakage of combustible vapours or fluids. Motors should not be installed below lines containing inflammable fluids.

(2) A good electrical contact between the motor frame and the aircraft structure, through the motor mounting, shall be ensured for internally grounded motors. If the motors frame is not mounted on a structure that will provide a good ground, suitable bonding jumpers shall be installed.

(3) Care shall be exercised to mount the motor in line with the driven equipment. The mounting shall be sufficiently rigid to prevent relative motion between the motor and the driven equipment.

INSTALLATIONS

70.61 MOUNTING

All electrical equipment shall be mounted so as not to be inadvertently actuated by inertia forces or other causes.

70.62 CLEARANCE

Clearance shall be provided in accordance with the installation clearance drawing for the equipment. If an installation clearance drawing is not available, the space provided shall be not less than the dimensions shown on the maximum envelope drawing for the equipment together with adequate additional clearance to allow for maintenance and proper ventilation. Adequate spacing shall be provided for shock mounted units.

Extracts From CAP 479 (Cont'd)

70.63 MAINTENANCE

Adequate provision shall be made for ease of maintenance, including installation, removal, and adjustment and inspection in place.

70.64 PROTECTION

Protection shall be provided against moisture, water leakage, condensation, excessive heat, combustible vapours and fluids, and physical damage.

70.65 COMPASS DEVIATIONS

Electrical wiring and equipment shall be installed so as not to cause excessive compass deviations.

70.66 VIBRATION

Electrical equipment shall be installed so that it will not be subject to vibrations exceeding the limits specified in the applicable equipment specification.

70.67 MOUNTING SCREWS

Machine screws or bolts shall be used for mounting electrical equipment. Self-tapping screws shall not be used for mounting electrical equipment or for making electrical connections.

70.68 DAMAGE

Electrical shall be installed so as not to cause damage to or be damaged by other equipment, wiring, or plumbing.

70.69 SAFETYING

Electrical equipment shall be properly secured by means of safety wire, cotter pins (split pins), locknuts, or other approved means.

70.691 VENTILATION

Adequate ventilation shall be provided. Electrical equipment shall be installed in such a manner that the operating temperature range of the equipment will not be exceeded under normal operating conditions.

70.692 FIRE PREVENTION

Particular care shall be exercised to insure that electrical equipment is not installed in such a manner as to constitute a fire hazard.

Extracts From CAP 479 (Cont'd)

70.693 GROUND RETURN

Equipment that incorporates a ground terminal shall be grounded by the shortest practicable lead to the nearest suitable metallic ground. Equipment that does not incorporate a ground terminal and that is internally grounded shall be grounded by the shortest practicable lead if adequate contact to metallic ground cannot be made or maintained without corrosion occurring at the mounts of the equipment.

70.694 Static Ground

Provision shall be made for the discharge of accumulated charges of static electricity by automatically bringing the aircraft to ground potential on landing.

70.695 FUEL NOZZLE GROUNDING RECEPTACLES

Fuel nozzle grounding receptacles shall be installed adjacent to all fuel tank inlets.

EXTRACTS FROM USAF - ARDCM 80-1

6. BODY

6.104 GENERAL ILLUMINATION

All crew compartments and crew passageways shall be generally illuminated on walls, ceilings, and floor. Electrical installations shall be in accordance with paragraph 10.1, Electrical Systems; Reference paragraph 23.16, Warning Sign.

6.17 LIGHTNING PROTECTION

6.170 GENERAL

Lightning protection is required on aircraft so that a lightning discharge current may be carried between any two points on the aircraft without risk of damaging flight controls, destroying part of the aircraft structure, such as plastic antenna housing, canopies, or electrically isolated conductors, and without injuring personnel.

All metal aircraft are inherently well protected from lightning damage, and rarely, if ever, suffer disabling damage from lightning strokes. Bonding of control surfaces, flaps, etc., in accordance with Specification MIL-B-5087 will provide adequate protection for the airframe.

6.171 STROKE GUIDANCE DEVICES

All external electrically isolated conducting objects, except antennas, which protrude above the aircraft surface, must have a bonding jumper or suitable stroke guidance device to the aircraft skin or structure to prevent serious structural damage to the aircraft in the event of a lightning strike.

6.512 SIGNAL LIGHTS AND ALARM BELLS

Signal lights and an alarm bell shall be installed in accordance with Specification MIL-L-6503.

6A.01 AIRCREW CONTROLS - GENERAL

6A-011 DIRECTION OF MOTION

All controls shall be so designed that the actuation thereof forward, upward or clockwise shall result in increased performance of the component or the aircraft. All controls shall be so designed that the actuation

Extracts From USAF - ARDCM 80-1 (Cont'd)

6A-011 DIRECTION OF MOTION (Cont'd)

thereof aft, downward or counterclockwise shall decrease the performance of the component or the aircraft.

On overhead switch panels, where the panel is divided into two or more flat sections, having different angles with respect to the horizontal, the average angle of the combined panels shall determine the direction of motion for all switches as follows:

(a) If this average angle is greater than 45 degrees with respect to the horizontal, all switches shall actuate upward for increased performance.

(b) If the average angle is less than 45 degrees from the horizontal, all switches shall actuate forward for increased performance.

All controls of a variable nature induced by a rotary motion shall move clockwise from the "OFF" Position, through the "LOW" or "DIM" to "HIGH" or "BRIGHT". This direction of control motion shall be established with reference to the operator, not the aircraft.

6A.13 EMERGENCY CONTROLS

6A.131 FIRE EXTINGUISHING CONTROLS AND WARNING INDICATORS

(a) Location and Actuation - The fire fighting controls shall be located on an emergency panel accessible to the pilot's throttle hand.

The controls shall consist of a single fire emergency control for each engine for critical area, and one agent discharge switch. The control may be either electrical or a combination of electrical and mechanical and shall include a handle incorporating the fire warning light. The control motion, which shall be pulled for "ON" shall perform all necessary fire extinguishing operations except the discharge of the agent. Actuation of the agent discharge switch shall release agent to the area selected by the fire emergency control.

Extracts From USAF - ARDCM 80-1 (Cont'd)

6A.131 FIRE EXTINGUISHING CONTROLS AND WARNING INDICATORS (Cont'd)

(b) Control Knob Shape and Fire and Overheat Warning Lights - The fire extinguishing controls shall be "T" shaped and shall be fabricated of clear translucent plastic and shall incorporate a red warning light. If overheat warning lights are to be installed, they shall be incorporated into the fire warning light by the use of an interrupted circuit. The indication for an overheat condition will be an interrupted signal from the fire warning light. The indication for fire shall be a steady signal. The light colour shall be red and one light shall be used for each power plant or critical area. When no fire extinguishing controls are incorporated, as in single engine aircraft, the fire warning light shall be located in the upper right hand portion of the instrument panel. Aircraft which have the controls at the flight engineer's position shall have a master fire warning light on the pilot's panel.

6A.16 ELECTRICAL CONTROLS

6A.161 LIGHT AND MISCELLANEOUS SWITCHES

(a) Light Switches - (General Requirements) - In single or tandem pilot aircraft, all light switches except landing light switches shall be located on the right side of the cockpit in a light panel. The grouping of the switches and rheostat within the panel shall be as functional as possible. In side by side pilot aircraft, the lighting panel shall be on the overhead switch panel in number 4 position on Figure 6A-2.

The actuating motion of the light switches shall comply with the requirements set forth in paragraph 6A.011, Direction of Motion.

(b) Landing Light Switches - The landing light switch(s) shall be adjacent to the power control or landing gear control lever. The actuation of landing light switches shall be up or forward for "ON" and down or aft for "OFF".

(c) Miscellaneous Switches - All switches other than light and landing light switches shall be of the "centre-off" position type for easy determination of switch position.

Extracts From USAF - ARDCM 80-1 (Cont'd)

6A.162 ELECTRICAL POWER CONTROLS

This panel will include the battery, generator, primer(s), starter(s), inverter(s) and oil dilution controls.

In single or tandem pilot installation the subject switches shall be grouped together on the right side forward. In side by side pilot aircraft the subject switches shall be grouped on the overhead panel, the number 3 position in Figure 6A-2.

6A.163 CIRCUIT BREAKERS

In single or tandem pilot aircraft, the circuit breakers shall be located forward on the inboard face of the right console. In side by side pilot aircraft, the circuit breakers shall be on the overhead panel, the number 5 position in Figure 6A-2. The circuit breakers will be grouped functionally with the most critical circuit breakers in the most accessible position. In aircraft where available cockpit space for circuit breakers is limited, only those circuit breakers essential to safety of flight will be located in a position accessible and readable 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.

6A.17 INSTRUMENT ARRANGEMENT AND LIGHTING

6A.171 LIGHTING

Except for instrument lighting the cockpit lighting for all aircraft shall be in accordance with Specification MIL-L-6503. Instrument lighting shall be designed and installed in accordance with Specification MIL-L-5667. All switch, radio, and auxiliary control panels shall be lighted by plastic plate edge lighted panels as specified in Specification MIL-P-7788.

The cockpit lighting controls which shall be located as specified in paragraph 6A.161, Light and Miscellaneous Switches, shall consist of the following:

(a) A single rheostat for control of the light intensity within each basic flight instrument panel.

(b) A single rheostat for control of the light intensity within the engine - instrument and auxiliary control panels (radio, switch panels etc.).

Extracts From USAF - ARDCM 80-1 (Cont'd)

6A.171 LIGHTING (Cont'd)

(c) A single rheostat for control of all type A-11 red flood lights at each station.

(d) A single dimming control and a selector switch for red and white cockpit utility lighting if applicable.

6A.172 WARNING LIGHTS, CAUTION INDICATORS AND ON - OFF INDICATORS

(a) Warning Lights - Warning lights are defined as those lights which serve as an attention-getting device to alert the operator to some existing dangerous condition requiring immediate action. They will be red in colour. Normally the only lights falling into this category will be the oxygen and emergency fire warning lights. Warning lights should not be used for denoting minor or routine deviations from normal operations, nor should they be used for indicating malfunctions which require attention but not necessarily immediate action. Warning lights should indicate immediately the nature of the hazardous condition, to facilitate necessary action. An identification label or placard shall appear immediately adjacent to the warning light except when word warning panels are provided. In either case the identification shall be plainly legible under day or night conditions. The fire warning lights shall be located on an emergency panel accessible to the pilot's throttle hand. The oxygen warning lights will be located in the upper right hand portion of the instrument panel in single and tandem pilot aircraft.

(b) Caution Indicators - Caution indicators are defined as those indicators which serve to alert the operator to an impending dangerous condition requiring attention, but not necessarily immediate action. Caution indicators should not be used to denote minor or routine deviations from normal operating conditions. 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 labeled or placarded "Master Caution", the legend to be clearly legible under day or night conditions. It shall be located high

Extracts From USAF - ARDCM 80-1 (Cont'd)

6A.172 WARNING LIGHTS, CAUTION INDICATORS AND ON - OFF INDICATORS (Cont'd)

(b) (Cont'd)

on the right hand side of the instrument panel in single or tandem pilot aircraft, and high in the centre of the instrument panel in side-by-side pilot aircraft.

The caution indicator panel will consist of appropriate words or abbreviations, to signify the conditions of caution, provided on a plastic panel in accordance with MIL-P-7788. Each condition specified on the panel will be provided with a suitable lighting fixture to identify the legend in red at night. The same or an auxiliary fixture(s) in the centre of and/or adjacent to the legend shall provide a suitable visual indication, red in colour, in daylight. An example of a suitable daylight indication is a direct view of the lamp filament through a clear identification red filter. The caution panel shall be located on the right side forward in single and tandem pilot aircraft, and on the centre console immediately aft of the power quadrant in side-by-side pilot aircraft. The actuation of the caution system shall be as follows:

(1) When a malfunction occurs, the master light and the appropriate light(s) on the caution panel will be activated.

(2) The master light will be manually reset to indicate another malfunction by the actuation of a switch located on or adjacent to the caution indicating panel.

(3) The appropriate light(s) on the caution panel will continue to indicate a caution condition until the malfunction is corrected.

(4) A test circuit shall be incorporated for testing the lamp filaments.

(c) On-Off Indicators - On-off indicators are defined as those indicators in which the "ON" position denotes a satisfactory or normal operating condition. In general, the absence of an indication

Extracts From USAF - ARDCM 80-1 (Cont'd)

6A.172 WARNING LIGHTS, CAUTION INDICATORS AND ON - OFF INDICATORS (Cont'd)

(c) (Cont'd)

serves the same purpose, and this principle should be adhered to if at all practicable. In such cases where it is necessary to denote a normal or satisfactory condition by an indication, a mechanical or electro mechanical indicator should be used. This type of indicator should be located as near to the display or control to which it is related as is possible.

(d) General - The number of warning lights, caution, and on-off indicators shall be kept to an absolute minimum. In no case shall warning or indicator lights be located directly within or above the basic flight instrument group. A dimming circuit shall be provided for all warning lights and caution lights. This dimming circuit will provide for manual selection of either a bright or a dim level of light intensity wherever the flight instrument light circuit is energized; the dimming circuit shall be "reset to bright" automatically wherever the flight instrument light circuit is de-energized. The method of dimming shall be subject to the approval of Wright Air Development Centre.

7.6 LANDING GEAR RETRACTING MECHANISMS

7.61 DESIGN REQUIREMENTS

7.611-K SAFETY SWITCHES

Leakproof, sealed units using Air Force approved low travel switches (Specification MIL-S-6743) shall be installed on all main landing gear in such a manner that the initial compression motion of the shock absorber actuates the switch. The purpose of these switches is to prevent retraction of the landing gear while the airplane is on the ground. They shall be wired in series with each other and shall be capable of being adjusted to operate at 1/2 inch plus 1/4 inch minus 0 inch from the shock strut fully extended position. These shall serve to operate a continuous duty-type solenoid for unlocking the landing gear control lever.

Extracts From USAF - ARDCM 80-1 (Cont'd)

7.61 DESIGN REQUIREMENTS (Cont'd)

7.630-B SEQUENCE VALVES

If the sequence valves are operated electrically, the control switches shall be ruggedly constructed and be of a type which will not freeze because of condensed moisture or water. They shall be mounted on rigid supports and not subjected to malfunctioning caused by normal structure deflections or jamming caused by foreign matter. It shall be possible to extend the gear by the emergency means in the event of failure of the electrical circuit.

9.41 LANDING GEAR CONTROLS

9.410 MOVEMENT

The landing gear control lever shall be installed with a solenoid released locking device which functions to preclude motion of the control lever from "down" to the "up" position when on the ground. The solenoid will be operated by the closing of safety switches, Ref. para. 7.611-K, Safety Switches.

For hydraulic or pneumatic retracting systems where the landing gear lever operates selector valves electrically by means of a conventional control circuit, a separate "contact" (multiple-pole switch) will be closed for each selector valve energized. For electrical retracting systems where the landing gear control lever operates electric motors by means of a conventional control circuit a separate "contact" (Multiple-pole switch) will be closed for each electric motor energized.

The solenoid shall be of the "continuous duty" type.

9.522 IGNITION CONTROL SYSTEMS

In turbojet and turboprop aircraft, the master switch shall be located adjacent to its respective power control.

Since the ignition system is designed for intermittent duty, it must be limited to a maximum of 3 minutes operation each attempted start. On single

Extracts From USAF - ARDCM 80-1 (Cont'd)

9.522 IGNITION CONTROL SYSTEMS (Cont'd)

or twin engine aircraft this shall be accomplished by using an approved time delay switch or by some automatic means such as a tie-in with other systems. On aircraft with more than two engines a momentary on-switch is acceptable if it can be operated conveniently by flight personnel.

It shall be possible to crank the engine without energizing the ignition system until a safe cranking speed is obtained. A safe cranking speed is defined as that speed at which no damaging flashback may occur.

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. This arrangement is not mandatory on aircraft having more than two engines.

10.1 ELECTRICAL SYSTEM

The types and characteristics of aircraft electrical systems shall conform to Specification MIL-E-7894. In selecting one or more of these systems for an airplane, careful consideration shall be given to such factors as reliability of operation, simplicity, light weight, vulnerability, and ease of installation, operation and maintenance. The designer shall submit to Headquarters, Wright Air Development Center, sufficient data on the chosen system or systems to permit an evaluation relative to the applicability to that particular airplane. The general requirements for the installation of electrical equipment in aircraft, as set forth in Specification MIL-E-7080, MIL-E-7563 and MIL-E-7614, shall be complied with where applicable. Certain important requirements called for in the above three specifications are listed in the following paragraphs for the convenience of the designer, together with other requirements.

All electrical equipment and components subject to high acceleration forces will be mounted so as not to be inadvertently actuated by the application of acceleration forces.

The power requirements for all Air Force airborne equipment may be found in Air Force Specification Bulletin No. 89.

4.3.14.2.1 (Continued)

A centrifugal switch on the left engine is used to determine which ice detector is in use. When the left engine is shut down, the switch is in the closed position and the circuit to the R/H ice detector is made; when the left engine is started and reaches a speed of 3020 R.P.M., the switch is actuated and the circuit to the right hand ice detector is broken and the circuit to the left hand detector made.

4.3.14.2.2 THE ICE DETECTOR

The ice detector is a unit containing a pressure switch, four relays and three resistors and mounting two electrically heated probes which protrude into the air stream. One probe is continuously heated during flight and provides a reference static pressure. The other probe has a number of small holes exposed to the air stream, on the forward side, and under normal conditions of flight will record a positive pressure in respect to the reference pressure recorded in the reference pressure probe. When icing is encountered ice will form over the holes in the detector probe and reduce the pressure below that of the reference probe. This condition is sensed by the pressure switch which operates to send a signal to the de-icing controller and supply power to the detector probe heater. The heater rapidly clears the probe of ice and re-establishes a positive pressure balance. If icing conditions continue, the icing and de-icing cycle of the detector probe is repeated continuously, thereby transmitting pulses to the de-icing controller at a rate directly proportional to the rate of icing.

Extracts From USAF - ARDCM 80-1 (Cont'd)

10.10 POWER SOURCES

10.100 GENERATORS

Generators are identified by Air Force type designations and are listed together with their power ratings in the U.S. Air Force Status of Equipment Book. Generators shall be installed in accordance with the applicable requirements of Specification MIL-E-7080, MIL-E-7563 and MIL-E-7614.

10.101 STORAGE BATTERIES

Batteries shall conform to applicable specifications referenced in United States Air Force Specification Bulletin No. 59. Battery installations shall conform to the requirements of Specification MIL-E-7080 and shall not be located so as to be affected by excessive heat.

Batteries shall be installed at a point which will not impose hazardous conditions to maintenance personnel, aircraft and batteries, when servicing, testing, and replacement operations are performed. The installation shall be easily accessible and will not require the use of specially designed ground handling equipment for servicing, testing, and/or battery replacement. They shall be securely anchored in place. If a nonshielded battery is specified, a container made of acid-resistant material shall be provided. Battery access doors, located on the outside of the aircraft, shall be plainly marked "BATTERY" in red letters no less than 3/4 inch high.

10.102 ACCESSORY POWER PLANTS

Accessory power plants shall be selected from those listed in the USAF Specification Bulletin No. 59.

10.103 EXTERNAL POWER RECEPTACLE

An external power receptacle shall be installed in the aircraft in accordance with the requirements of Specifications MIL-E-7080, MIL-E-7563 and MIL-E-7614. The installation shall, in addition, conform to the following:

Extracts From USAF - ARDCM 80-1 (Cont'd)

10.103 EXTERNAL POWER RECEPTACLE (Cont'd)

(a) The receptacle shall not be located in the propeller blast or adjacent to the jet blast. A spring-loaded hinged access door shall be installed over the external power receptacle(s) so that the door will open against the slip stream and will close unaided after withdrawal of the power cable.

(c) The receptacle(s) shall be located and connected to the bus system so that the voltage drop between the receptacle(s) and the starter terminal will not exceed 2 volts.

(d) The receptacle(s) shall be located to permit the maximum practical spacing between the external power plant(s) and any fuel system servicing openings or vents, from which fuel spillage may occur during refueling, defueling and engine starting etc.

10.11 POWER APPLICATIONS

10.110 MOTORS

Direct current motors shall conform to Specification AN-M-40 and alternating current motors shall conform to Specification 32590. Motors shall be installed in accordance with Specifications MIL-E-7080, MIL-E-7563, and MIL-E-7614.

10.111 STARTERS

Starters shall be as specified in the aircraft type or model specification.

10.112 LIGHTING

Lighting equipment shall be installed in accordance with Specification MIL-L-6503.

Landing lights shall not be located in front of wing fuel cells, unless no other suitable location is available. This will minimize the fire hazard during crash landings.

Extracts From USAF - ARDCM 80-1 (Cont'd)

10.113 HEATERS

Design and installation data pertaining to electrical heater units shall be submitted to Headquarters, Wright Air Development Center, for approval.

10.114 ACTUATORS

Electro-mechanical actuators shall conform to specification MIL-A-8064.

10.12 CIRCUIT ACCESSORIES

Requirements relating to switches, rheostats, voltage regulators, relays, circuit breakers, fuses, and other circuit accessories, are set forth in Specifications MIL-E-7080, MIL-E-7563 and MIL-E-7614. Fuses when used, shall be quickly accessible, and spare fuses shall be provided.

10.13 WIRING

The wire sizes shall conform to existing Air Force-Navy requirements. Wiring shall be installed in accordance with Specification MIL-W-5088.

Electrical connections and wiring shall be adequately protected from damage or short circuit caused by movement of personnel, cargo, or equipment. Extreme care shall be taken to insure that all bare conductors, connections, terminals, or other exposed current carrying parts are adequately protected against short circuits caused by loose or foreign objects. This protection may be provided by means of suitable covering, by installing in junction boxes, by locating in such manner that additional protection is not required, or by other means acceptable to WADC. Installations having dielectric strength equivalent to that described in Specification MIL-C-3162 will be acceptable.

Lightly insulated high tension or radio wires shall clear all parts normally at ground potential by at least 3/4 inch.

10.14 BONDING

All metallic parts of the aircraft and its equipment and accessories shall be bonded in compliance with Specification MIL-B-5087.

Extracts From USAF - ARDCM 80-1 (Cont'd)

10.15 DUPLICATE POWER SOURCES

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

10.16 ELECTRICAL EQUIPMENT

Electrical equipment shall comply with the requirements of Specification 32466.

11.2 FIRE ELIMINATION SYSTEMS

11.20 FIRE EXTINGUISHER SYSTEM

Fire-extinguishing systems shall be installed on all multi engine aircraft and may be specified for experimental single-engine aircraft where the complexity of the engine installations, the inclusion of new and untried features or the general importance of the project warrants the additional protection for the aircraft.

Fire-extinguishing systems shall be installed and tested in accordance with Specification MIL-E-5352.

11.21 FIRE-DETECTION SYSTEM

Fire-detection systems shall be installed in all multi engine aircraft and in single-engine aircraft that have the engine behind the pilot or submerged in such a way that the pilot would not be readily aware of engine fire. The fire detection systems shall normally be in accordance with Specification MIL-D-7253. In instances where an overheat - detector system is used in lieu of the fire-detection system, it shall be in accordance with Specification MIL-D-8114.

11.22 OVERHEAT-DETECTOR SYSTEM

An overheat-detector system shall be installed in all turbo-jet, turbo-prop, and rocket-type aircraft in accordance with Specification MIL-D-8114. The overheat-detector system shall be installed unless it can be demonstrated that primary structure cannot be overheated by either normal or abnormal engine operation.

UNCLASSIFIED

Extracts From USAF - ARDCM 80-1 (Cont'd)

11.3 ICE-ELIMINATION SYSTEMS

All Air Force Aircraft with the exception of gliders, trainers, liaison and rotary wing aircraft will be equipped with complete anti-icing systems.

13.615 Isolation of Electrical and Oxygen Equipment from Combustible Materials and Fuel Tanks. 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 breaking, or normal arcing.

Oxygen equipment should be isolated from hydraulic fluid, oil, and fuel lines and tanks to reduce the chances of severe explosion of fire due to battle damage. Sparks formed as the metal is ruptured by the flak or shell fragment, and the hot fragments themselves have caused fires when the damage allowed combustible fluid to escape. It is highly desirable to isolate electrical and oxygen equipment from combustible fluids and reasonable weight penalties to accomplish this are justified. An example of isolating the fuel and electrical equipment from each other is an electric fuel valve installation in which the valve is flange mounted on a bulkhead with all electrical equipment on one side of the bulkhead and the fuel valve and fuel lines on the opposite side of the bulkhead. 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 effect of gravity, air flow, or battle damage, and hydraulic lines will be routed below electrical equipment and wires wherever they cross paths, pursuant to Specification MIL-E-7563.

Most electrical equipment, such as standard switches, relays, and motors (with the exception of inverters, electric motor-driven hydraulic pumps, and open relays), are explosion proof. Inverters, electric motor driven hydraulic pumps, and open type relays or other non-explosion-proof equipment should be located away from any vicinity likely to be contaminated with fuel vapor as a result of a fuel leak due to poor maintenance or battle dammage. If this equipment is located in such an area it shall be isolated

Extracts From USAF - ARDCM 80-1 (Cont'd)

13.615 (Cont'd)

from that area by a metal inclosure which renders it explosion proof. This inclosure should be vented to the atmosphere in such a manner that fuel vapor from the interior of the aircraft will not come in contact with the brushes, commutator, open contacts, etc., at any time including when the electrical equipment is not operating or when the aircraft is at rest on the ground. Cooling air from an inverter or electric motor should not be exhausted to the interior of the aircraft. If a fuel leak forms an explosive mixture both inside the aircraft and near the commutator, while the aircraft is at rest, and if any sparking occurs when the equipment is first started, it is highly probable that an explosion will occur which will flash back into the aircraft. As a further precaution to prevent explosions, a small explosion proof blower should be installed in these inclosures. These blowers should be started before the main electrical equipment to remove all explosive vapors from the inclosures. If ram pressure is used to ventilate the inclosures in flight, the blowers could be turned off.

Explosion proof equipment is equipment which will not explode a combustible atmosphere in which the equipment is operating under all environmental conditions as defined in its model specification.

13.618 FIRE DETECTING AND EXTINGUISHING

The general findings of the various fire tests show that a serious nacelle fire can reach maximum intensity in 30 seconds for a two-zone cowl and 15 seconds for a single-zone cowl. To be of any value, a fire detector should be capable of detecting a fire in 3 seconds or less.

It is seen from the above paragraph that a very few seconds elapse between the time that the fire-warning light indicates an engine fire and the time that the fire drill must be completed. It is evident that the cockpit, controls, and fire-extinguishing equipment must be arranged so that the drill can be performed automatically and at great speed.

Extracts From USAF - ARDCM 80-1 (Cont'd)

13.717 MISCELLANEOUS DESIGN REQUIREMENTS

(d) Electrical Bonding - In all vibration - isolating and vibration absorbing mountings, adequate provision shall be made for compliance with the electrical bonding requirements of Specification MIL-B-5087.

15.200 POWER PLANT ELECTRICAL EQUIPMENT

Electrical equipment used throughout the power plant and throughout the aircraft in connection with the power plant shall be designed in accordance with Specification 32466. All equipment in the power plant installation or in compartments containing combustible fluids or combustible vapors shall be explosion - proof.

15.52 GENERAL DESIGN REQUIREMENTS

15.524 ELECTRICAL WIRING

All wiring in the aircraft constituting part of of the control system shall conform to Specification MIL-W-5088. When required, control equipment shall be positively grounded with an external metal strap in accordance with Specification MIL-B-5087.

16.623 ICE PROTECTION

The entrance to the air induction system shall be protected against ice formation and build-up. This is considered necessary because of the severe power losses associated with relatively small disturbances to engine air flow.

In addition, all airframe parts in the air induction system, such as engine accessory covers or air duct valves, subject to collection of ice shall be protected. The meteorological design conditions shall conform to Specification MIL-E-5007.

When an ice accretion meter is not furnished with the engine, an ice detecting device shall be installed in the air inlet duct and connected to a suitable indicator light. Cyclic operation of the indicator light will furnish the pilot with an indication of the rate of ice formation.

Extracts From USAF - ARDCM 80-1 (Cont'd)

19.4 MISCELLANEOUS INSTRUMENTS

19.411 CAPACITANCE

This type fuel quantity gage consists of one or more tank units (electro static capacitors) all capacitance measuring circuit and equipment incident thereto, and an indicator. The tank units are located in the fuel tank so that, as fuel replaces air as their dielectric, their total capacitance will vary according to the quantity of fuel in the tank. The tank units are characterized such that all linear relationship exists between the electrical capacitance of the tank units and the volume of fuel sensed. The tank unit capacitance is measured by the circuit and is registered by the indicator. The indicator is calibrated in pounds since a correlation exists between the dielectric constant and density of aircraft fuels. The electrical and physical properties of fuels conforming to Specification MIL-F-5624 which are normally used in jet engine aircraft, vary such that excessive errors are introduced in the gage. Therefore, the gage when used in jet aircraft must be compensated to minimize the errors thus introduced. For jet aircraft applications the gage shall incorporate dielectric constant compensation and shall conform to Specification MIL-G-7818.

19.412 Installation and Calibration - The installation and calibration of any fuel quantity gage shall be engineered so that the inherent accuracy of the gage system shall be utilized to the best possible advantage. The installation and calibration of fuel-quantity gages shall be in accordance with Specification MIL-G-7940. The fuel quantity gages shall be calibrated in pound units.

21.365 TRANSIENT VOLTAGES

Circuits for all electrically primed ordnance such as rockets, electrically primed ammunition, seat ejection charges, explosive volts, etc., shall be adequately protected as required to prevent inadvertent ignition of the primers by transient induced EMF. Protection may include the use of shielded wire (or coaxial cable), firing leads properly routed to achieve separation from other cabling and/or grounded through a safety relay located as close to the ordnance item as possible.

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UNIT	MANUFACTURER	MFG. DWG. OR PART NO.	AVROCAN SPEC.	SPEC. OR SPEC. DWG.	CAN. STD. DWG.
Relay 50 AMP SPDT Class A	Can. Diaphlex	650 - 4063	-	MIL-R-6106A	CS-R-128
External Supply Receipt & Cover	Albert & J.M. Anderson Mfg. (Powerlite)	-	E-345	-	CS-R-127
Relay - Time Delay	Rogers-Majestic	M6412	E-285	-	CS-R-126
Switch 3 Pos. Slide	M-H	X11618	-	MIL-S-6743 MIL-S-6744	CS-S-153
Plug - External Power Lanyard Release	Albert & J.M. Anderson Mfg.	R67 with Springs	E-345	-	CS-P-123
Switch (Mom Contact) Jettison	Hetherington (Leonard Elect)	-	-	MIL-S-6743	CS-S-156
Indicator Light (Miniature)	Dialite	-	E-318	-	CS-I-107
Connector, Panel - Dual	Cannon	-	E-394		CS-C-145 CS-C-146
Connector, Panel	Cannon	-	E-394		CS-C-143 CS-C-144
Panel - Landing Gear Control	M.J. Johnson	-	E-279	7-1152-8	
Indicator - Warning - Dual	Korry	204 BPRL 204 BPAL	-	MIL-I-3661	CS-I-109
Alternator 208/120V A.C. 400 cycle	Lucas Rotax	60601	E-202	7-1125-11	
Control Unit & Transformer Rectifier	Lucas Rotax	60349	E-202	7-1156-12	
Relay - Differential Current Protection	Lucas Rotax	-	E-360		CS-R-123
Light - Indicator Case Grounded	Hetherington (Leonard Elect)	L2000G	E-268		CS-L-132

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APPENDIX 3

A REVIEW OF THE ENGINEERING DECISIONS ON BATTERY SELECTION FOR THE CF-105 AIRCRAFT AND THE EFFECT OF A POSSIBLE MODIFICATION

A REVIEW OF THE ENGINEERING DECISIONS ON BATTERY SELECTION FOR THE CF-105 AIRCRAFT AND THE EFFECT OF A POSSIBLE MODIFICATION

1. INTRODUCTION

A nickel cadmium battery was selected for the CF-105 aircraft. This selection was largely based on the reduction of weight and the ease of maintenance that a nickel cadmium battery would provide over the more commonly used lead acid battery. Recently, the selection of this type of battery has been queried by the RCAF in letters Ref. No. S-1038-105-8 (ACE-1) and 1038-CF105-80 (AMTS/DIE ENG). The objections raised to the present installation appear to be:

(a) that the battery has been incompletely evaluated and qualified by the U.S. Navy, and

(b) that the batteries of this type are prone to self destruction at aircraft operating temperatures.

It was also pointed out that a quick disconnect type of battery connector was desired. Accordingly, this review of the present situation is presented so that the full details of the present installation may be known, and further, that the details of those changes that would be necessary should a decision be made to adopt a lead acid alternative battery.

2. THE QUALIFICATION AND RELIABILITY OF NICKEL CADMIUM BATTERIES

2.1 QUALIFICATION STATUS

A nickel cadmium battery of this type has been electrically qualified as indicated in a letter dated 7th, December 1955 Ref. AER-EL-501/67, Department of the Navy Bureau of Aeronautics, Washington 25, D.C. A copy of this letter has been sent to Mr. D.M. Fraser, 1070 Birchmount Rd., Toronto 16, Ontario, and its contents state that the battery meets the electrical qualification test requirements of MIL-B-8565. However, full qualification is contingent upon the completion of the specified life test and the elimination of two difficulties which are:

(a) a mounting structure failure, and

(b) a connector malfunction.

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2.2 SERVICE USE

This type of battery has given 200 hours of trouble free operation in a CF-100 MK.4 aircraft and it is understood that aircraft companies in France, England, and in the United States have reported good performance and reliability of similar units.

2.2.1 SERVICE DIFFICULTIES

There are no known records of the so called "vicious cycle" that have not been associated with overcharging of the battery. However, it may be noted that this problem also exists with the lead acid type of battery but in that case the temperature rise on overcharge is not as high as in the nickel cadmium type. The higher temperature rise of the latter is due to the low internal resistance which makes possible its superior high discharge rate characteristics.

3. BATTERY SELECTION

A 15 ampere-hour battery is required for the CF-105 aircraft. The nearest standard size is the AN 3151, a 24 A.H. lead acid type battery for which a dimensionally equivalent nickel cadmium type of 22 ampere hours capacity is available. However, the use of the 22 ampere hour battery would entail a weight penalty of 10 to 12 lbs. over the 15 A.H. nickel cadmium battery presently selected, and a still greater weight penalty would have to be taken should the lead acid type be selected.

4. NICKEL CADMIUM BATTERY TEMPERATURES

4.1 GENERAL

The early model of the nickel cadmium battery required a wiring connection to the generator controls to control the voltage at the battery terminals. Later, this arrangement was modified and a thermocouple was incorporated in an outer battery cell to control a relay which isolated the battery from its charging circuits when overcharging caused the outer reference cell to reach 135°F, which is equivalent to an inner cell temperature of 160°F.

4.2 CHARACTERISTICS OF THE BATTERY TEMPERATURE RISE

Battery temperature does not start to rise until the battery reaches 100% of full charge as is shown by a temperature rise versus % overcharge curve (Fig. 1) furnished by the battery manufacturer. This curve indicates a rapid temperature rise if a high charging rate is maintained after the battery comes up to full charge.

However, this temperature rise may be prevented by:

(a) completing the final charging of the battery at a low rate, or

(b) by charging at a voltage which is less than the full charge voltage of the battery.

5. ANTICIPATED TEMPERATURES OF THE NICKEL CADMIUM BATTERY AS INSTALLED IN THE CF-105

5.1 ELECTRICALLY INDUCED TEMPERATURE

As the no-load full charge battery voltage is 29 volts and the aircraft charging voltage is 27.5 ± 1 volt, there appears to be no danger of overcharging and consequently overheating the battery.

5.2 AMBIENT TEMPERATURE

The battery is installed in a compartment whose ambient temperature is controlled by the air conditioning system to 160°F. However, to take full advantage of the nickel cadmium batteries' characteristics it was decided to provide a special cooling facility to maintain the battery itself at 80°F.

6. TERMINALS

Nut type terminals were selected because no quick disconnect type existed for the 15 a.h. battery and maintenance requirements did not appear to demand its development. In this regard it was noted that the maintenance problems connected with the use of nickel cadmium batteries appeared to be far less than

6. TERMINALS (Continued)

those associated with the lead acid type, and for this reason the nickel cadmium battery did not appear to require removal as frequently as the other type. In particular, the lead acid battery was frequently removed for servicing for any one of the following conditions:

(a) exposure of the aircraft to low temperatures often required replacement of a cold battery with a warm one, especially when an electrical starter was used.

(b) routine inspecting involving cleaning and corrosion prevention due to spillage, leakage and spraying of the electrolyte.

(c) routine specific gravity adjustment of the electrolyte, entailing bench checking.

(d) on some aircraft installations, the battery must be disconnected to disable the canopy ejection mechanism when the aircraft is hangared.

The nickel cadmium battery installation as proposed for the CF-105 aircraft will not require battery removal for any of the above reasons. This aid to ease of maintenance of the aircraft is brought about through the following characteristics of the nickel cadmium battery and its installation.

(a) the low temperature characteristics of the nickel cadmium battery are superior to those of the lead acid type and in addition, the fact that it is not used for starting completely eliminates the necessity of replacing it in cold weather. (The a.h. efficiency of the nickel cadmium battery is approximately 30% at -40°F, while that of the lead acid battery is only 17% at -20°F. (Reference - Defence Research Chemical Laboratories Report No. 180 of project No. D52-54-80-08).

(b) the battery is a sealed unit and so requires no electrolyte maintenance.

(c) the canopy and seat ejection mechanisms of the CF-105 aircraft are not electrically operated.

7. THE EFFECT OF MODIFYING THE PRESENT INSTALLATION TO RECEIVE A LEAD ACID TYPE BATTERY

Provisions could be made to install either an AN 3151 battery or its dimensionally equivalent 22 a.h. nickel cadmium battery at the expense of the following changes:

(a) a larger sealed compartment would have to be provided,

(b) the aircraft's primary structure would have to be modified to accommodate the larger compartment,

(c) corrosion protection would have to be provided for the compartment and all areas within 12 inches of the battery,

(d) equipment and wiring would have to be moved from the battery area into other already congested equipment space,

(e) space allowance would have to be provided to cater for the maintenance of the lead acid type battery, i.e., measuring and refilling electrolyte, removing caps, etc.

(f) the battery compartment would have to be vented according to AND 10441.

8. THE INCORPORATION OF A QUICK DISCONNECT CONNECTOR

The battery mounting is such that a quick disconnect connector could be installed (with alterations to the existing battery) at a weight penalty of over one pound. However, in view of the small amount of maintenance required on this battery, it would appear that it would not be economical to take the necessary weight penalty for the small convenience for which it would provide.

9. RECOMMENDATION

It is recommended that the present installation remain unchanged, as from the foregoing paragraphs it would appear that the present installation provides the lightest and most compact installation, and in addition provides performance and maintenance advantages over other installations investigated.

SAFT BATTERY - VOLTABLOC VO.15 EFFICIENCY STUDY

1. EMERGENCY D.C. BUS FLIGHT LOADING

SERVICE	AMPS	
External Tank Jettison	1.35	Momentary
U/C Indication	.09	Cont.
Fire Detection	1.00	Cont.
Canopy Seal	1.00	Cont.
Speed Brake Act.	.60	Cont.
Master Warning Box	1.35	Cont.
Emergency Cockpit Lts.	.60	Cont.
Turn & Bank Ind.	.20	Cont.
Ignition (Relight)	7.00	30 Secs. Max.
A.R.C. 34 U.H.F.	14.00	Cont.
A.I.C. -10 Intercomm.	2.90	Cont.
	31.09	

Momentary & Short Duration 8.35

Continuous 21.74

2. EFFECT OF TEMPERATURE ON TIME OF OPERATION IN FLIGHT

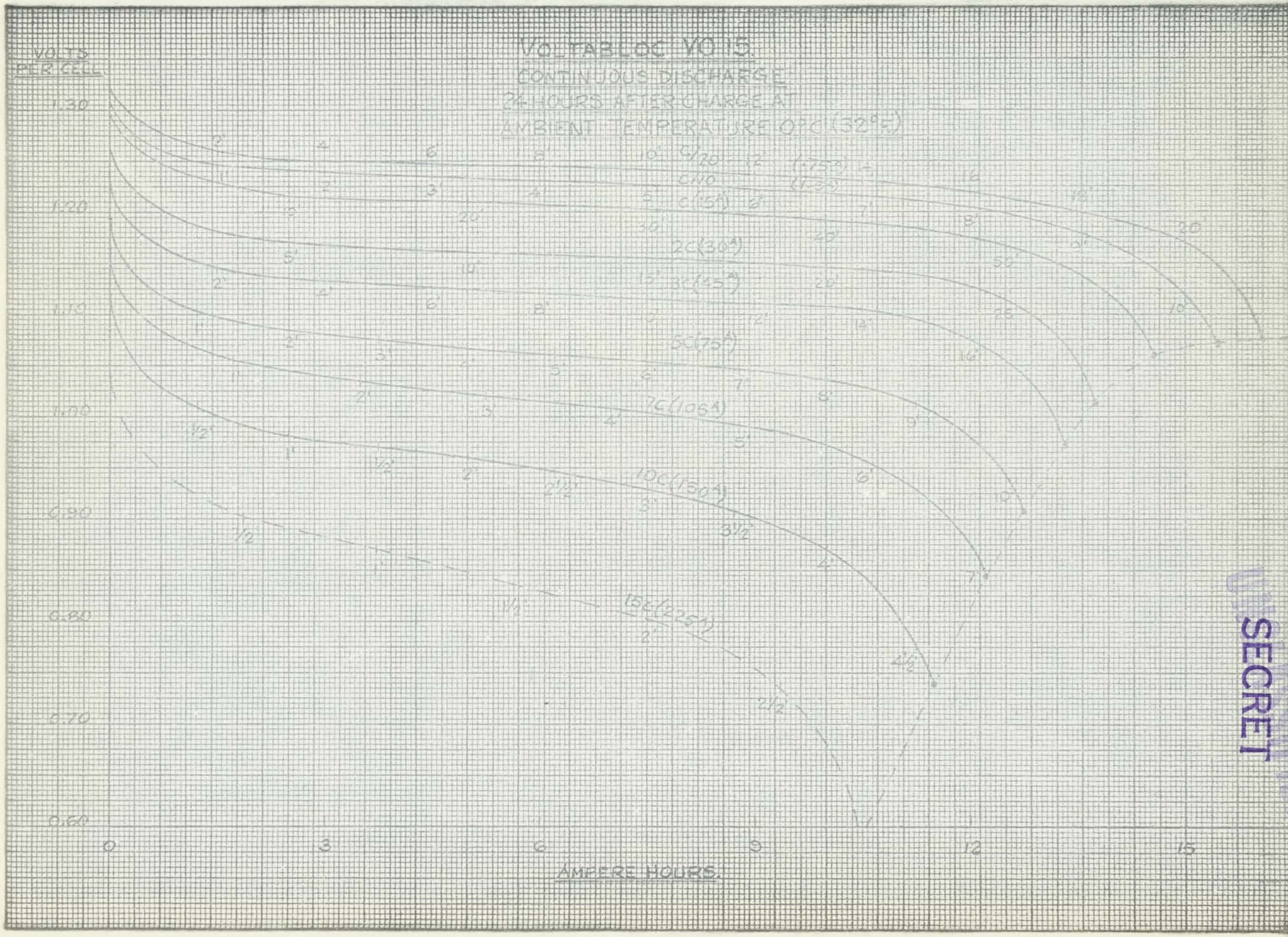
Calculating on the basis of a 22 Amp continuous Load.

AMPS	TEMPERATURE	MINUTES OF OPERATION
22	+68°F	48
22	+32°F	38½
22	- 4°F	29-3/4
22	-40°F	18½

3. MISCELLANEOUS LOADINGS

SERVICE	AMPS	
Canopy Act & Control	16.8	8 Sec.
Fire Extinguishing	8.0	MOM
L/P Cocks (Close (2)	12.8	1 Sec.
I.F.F. Crash	40.0	MOM

The above services are not normally used in flight emergency.



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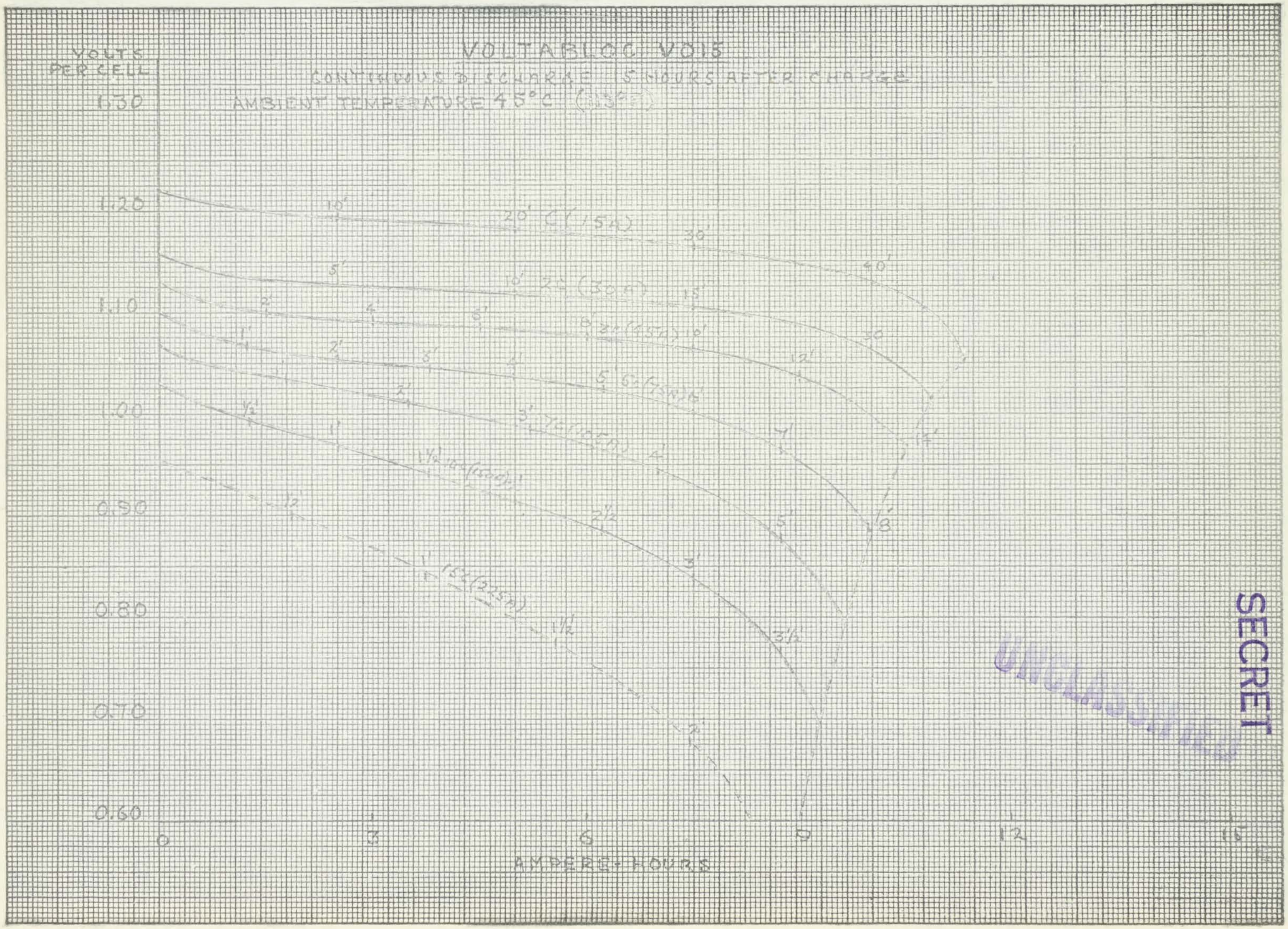
VOLTABLOC VO15
CONTINUOUS DISCHARGE 5 HOURS AFTER CHARGE
AMBIENT TEMPERATURE 45°C
VOLTS PER CELL
1.30
1.20
1.10
1.00
0.90
0.80
0.70
0.60
0
3
6
9
12
15
AMPERE-HOURS
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INDEX OF ILLUSTRATIONS

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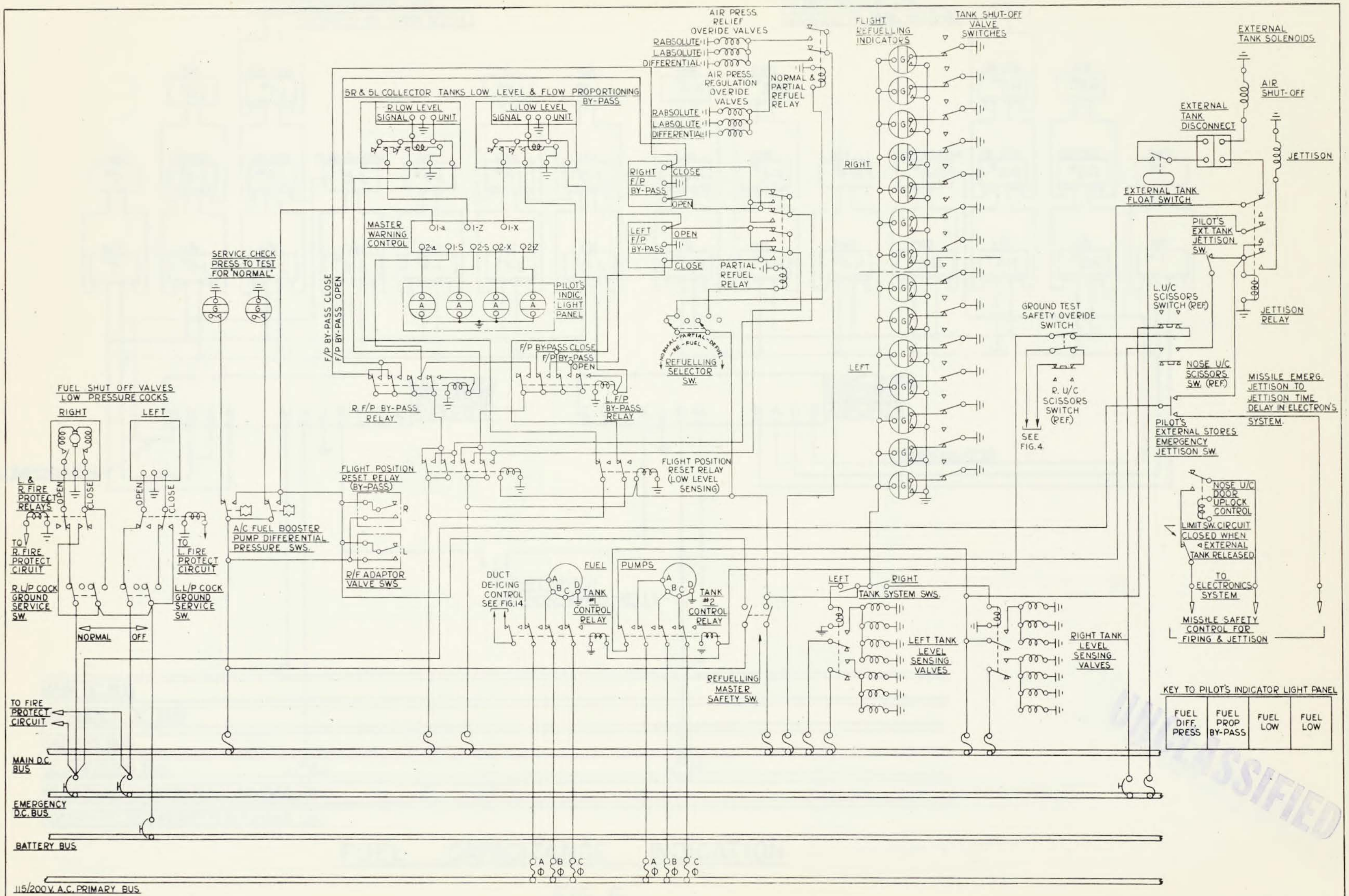

FIG. 5 FUEL SYSTEM

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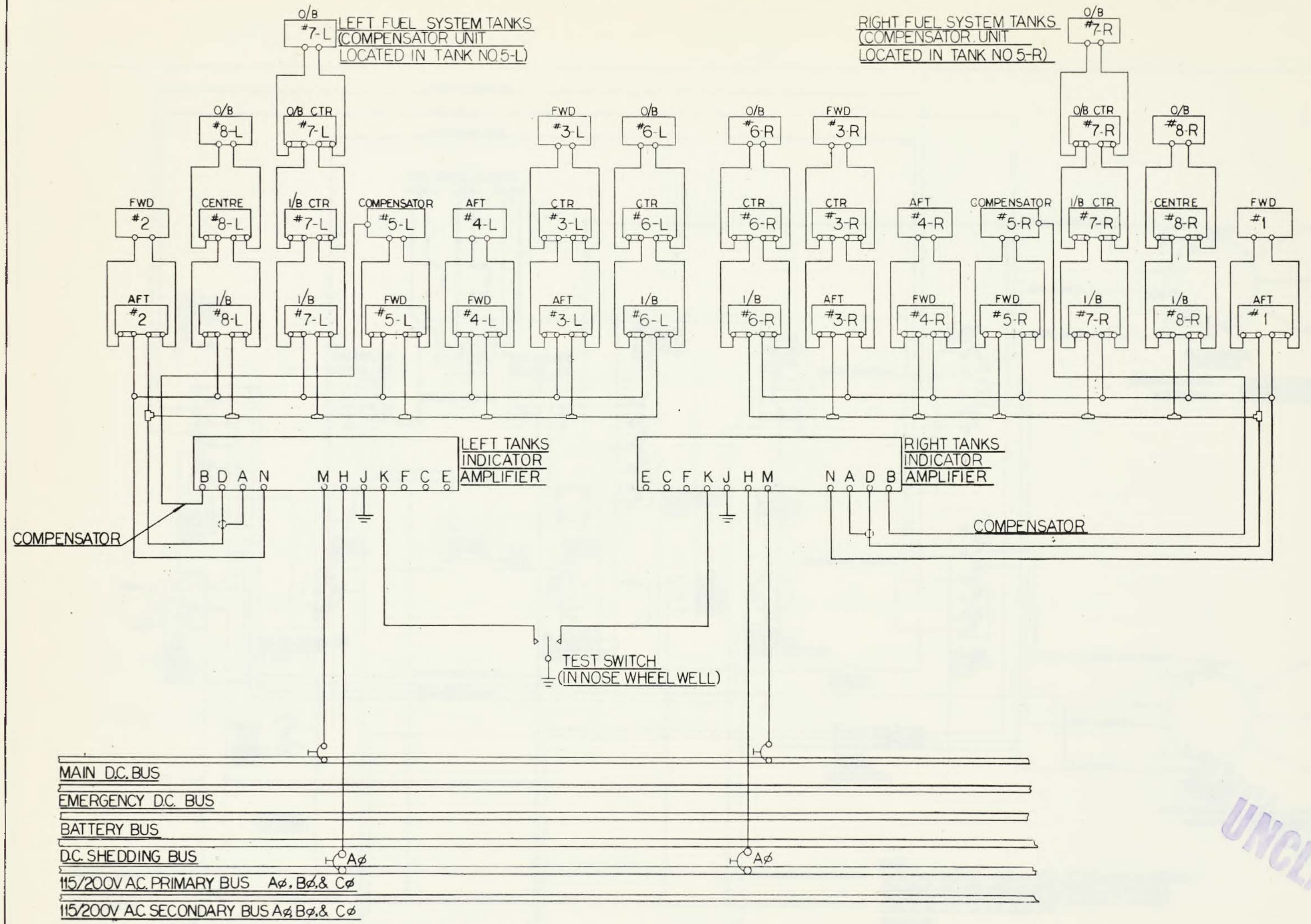

FUEL CAPACITANCE INDICATION

FIG. 6

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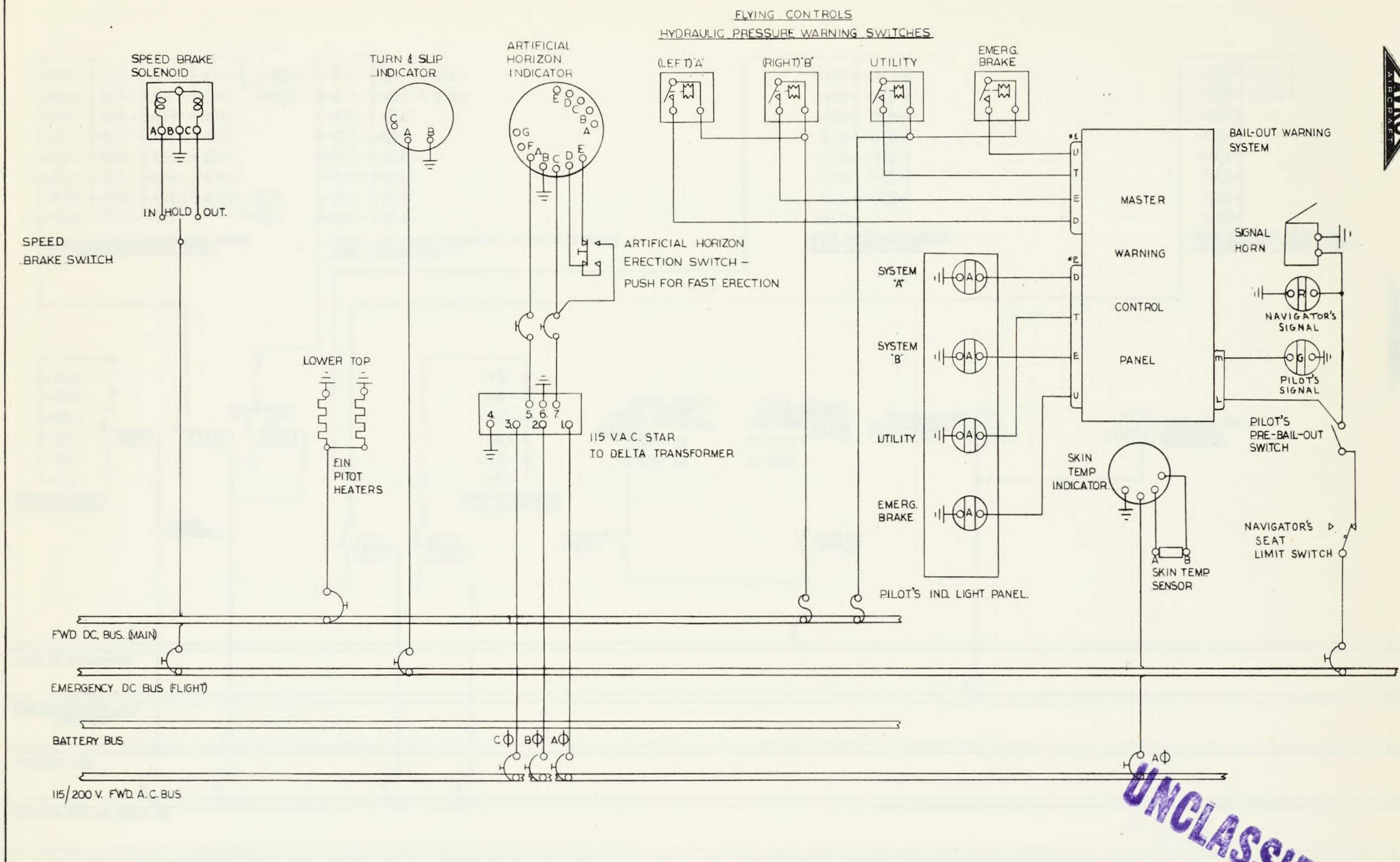

SPEED BRAKE SOLENOID   TURN & BANK INDICATOR   ARTIFICIAL HORIZON   HYDRAULIC PRESSURE WARNING.

FIG. 11.

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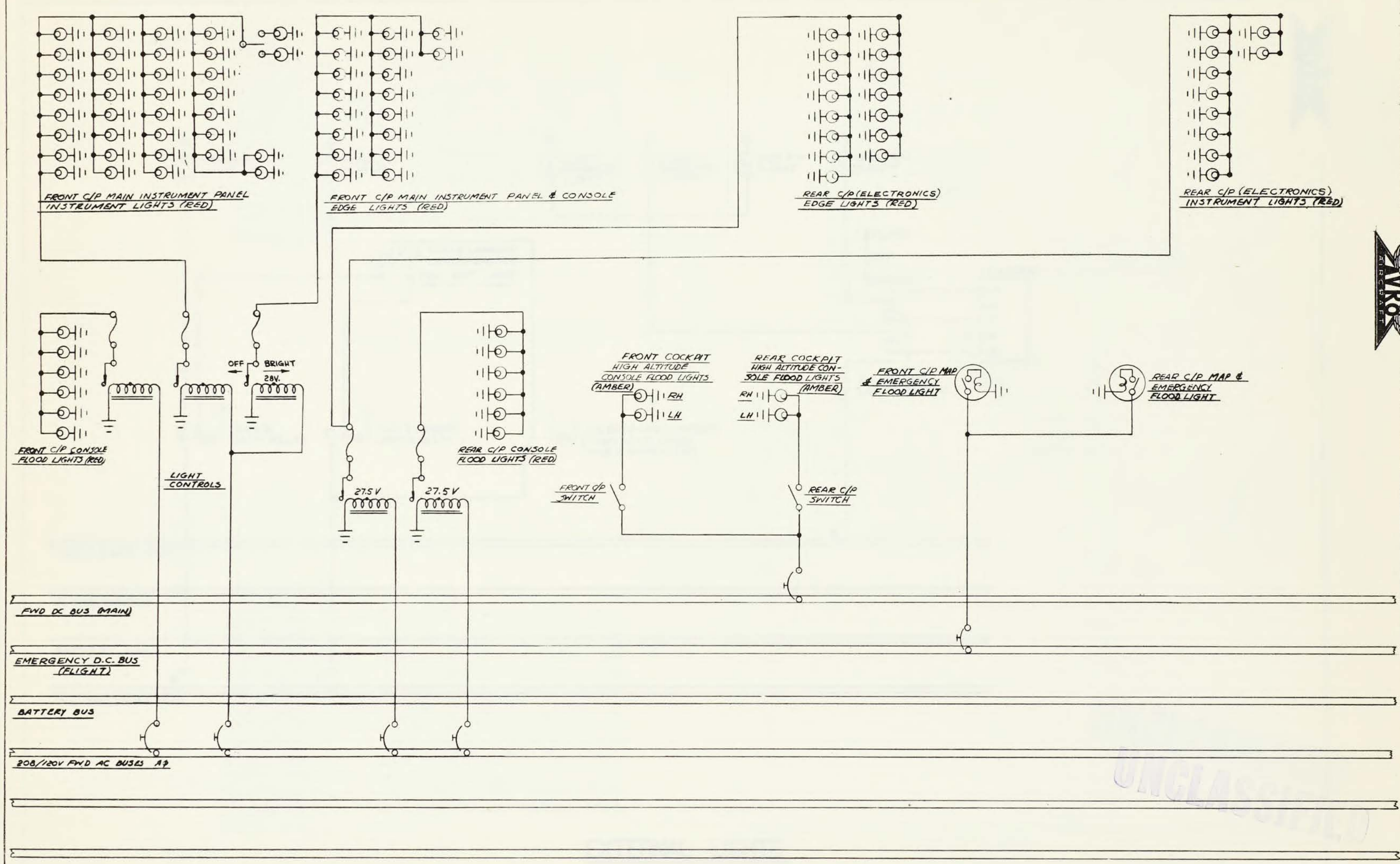

AVRO AIRCRAFT

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COCKPIT LIGHTS

FIG. 12

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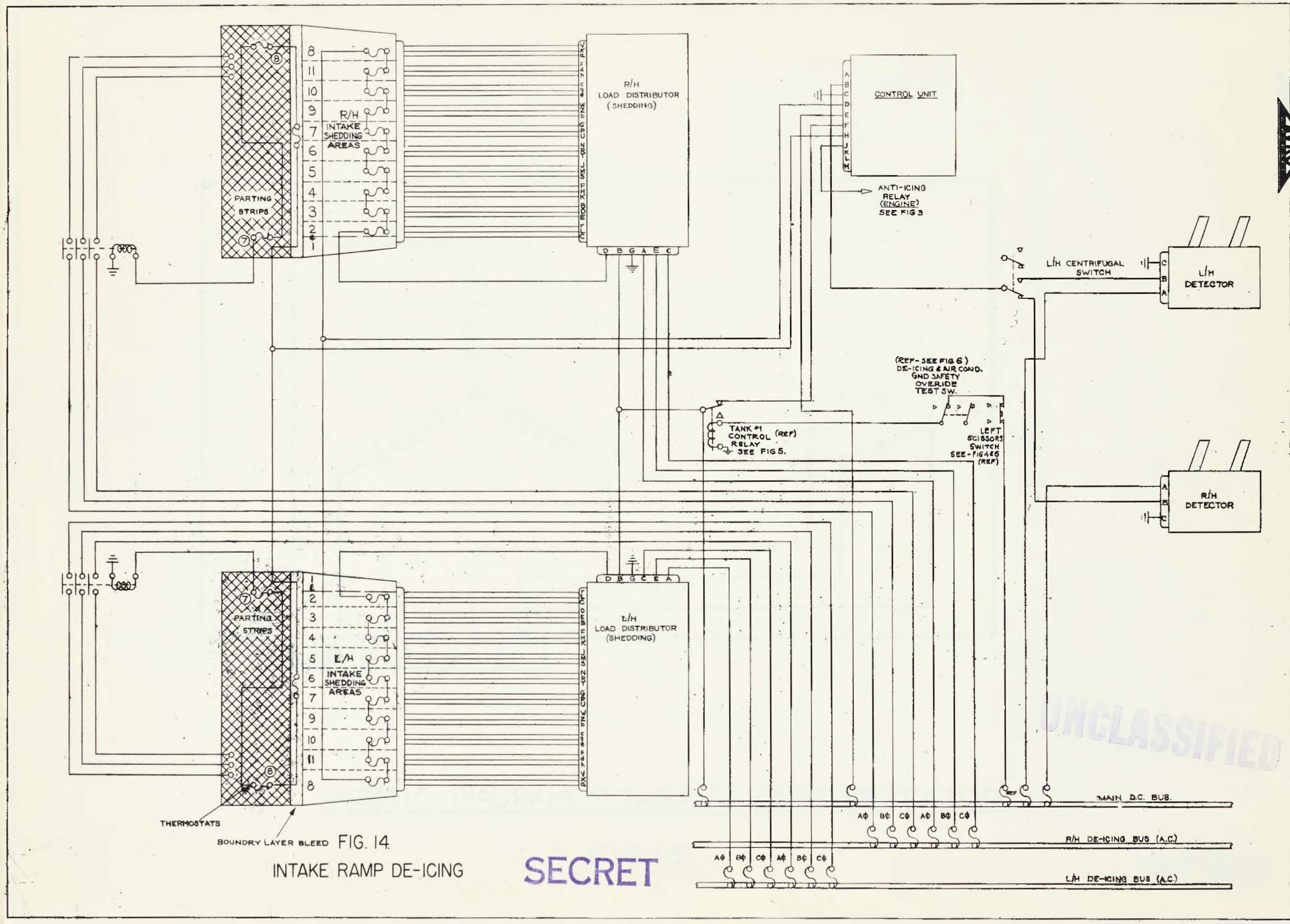

FIG. 14
INTAKE RAMP DE-ICING

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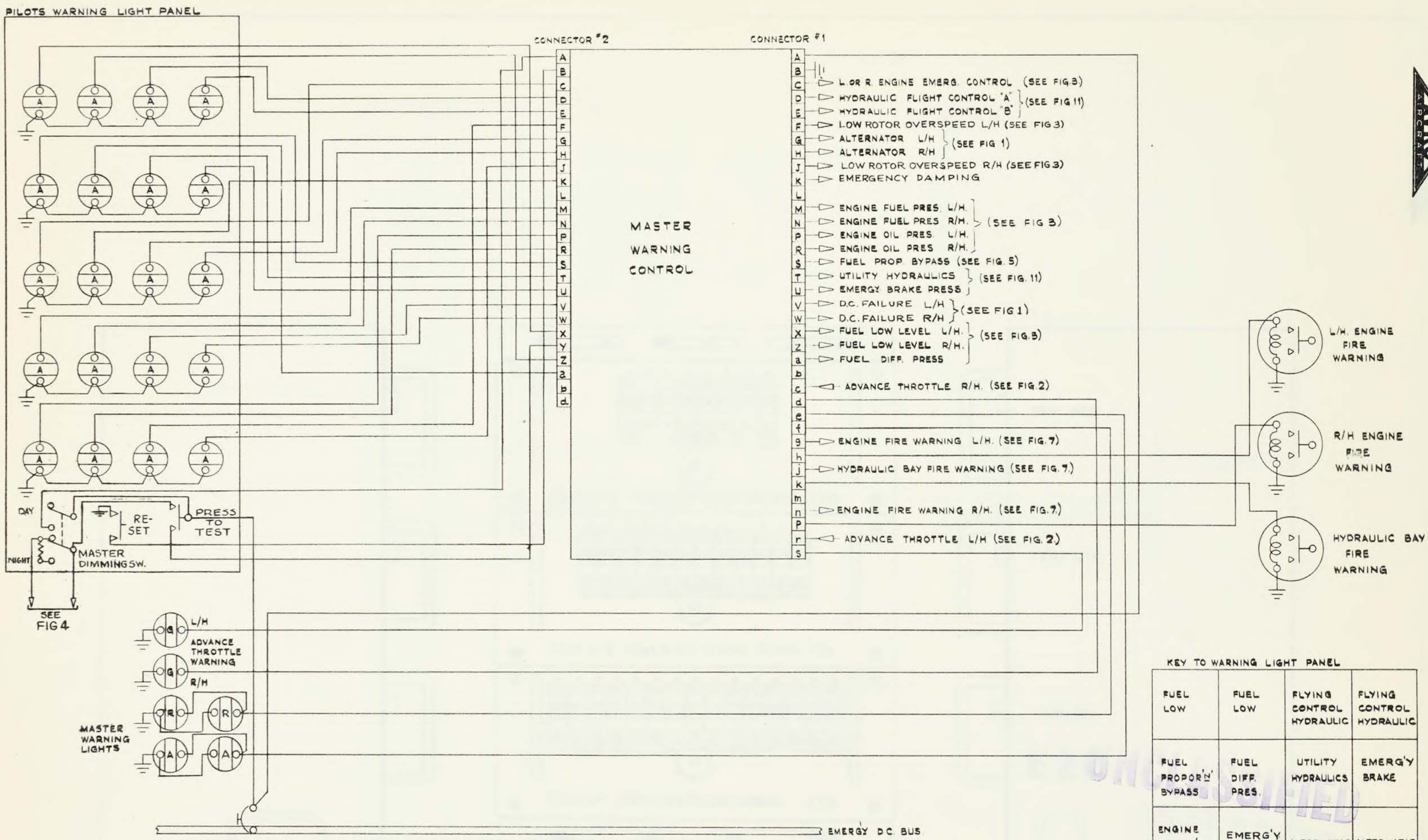

KEY TO WARNING LIGHT PANEL

FUEL LOW	FUEL LOW	FLYING CONTROL HYDRAULIC	FLYING CONTROL HYDRAULIC
FUEL PROPOR'N' BYPASS	FUEL DIFF. PRES.	UTILITY HYDRAULICS	EMERG'Y BRAKE
ENGINE EMERG'Y FUEL	EMERG'Y DAMPING	ALTERNATOR	ALTERNATOR
ENGINE FUEL PRES.	ENGINE FUEL PRES.	D.C. FAILURE	D.C. FAILURE
OIL PRES	OIL PRES.	LOW ROTOR OVERSPEED	LOW ROTOR OVERSPEED

MASTER WARNING CONTROL SYSTEM

FIG. 16

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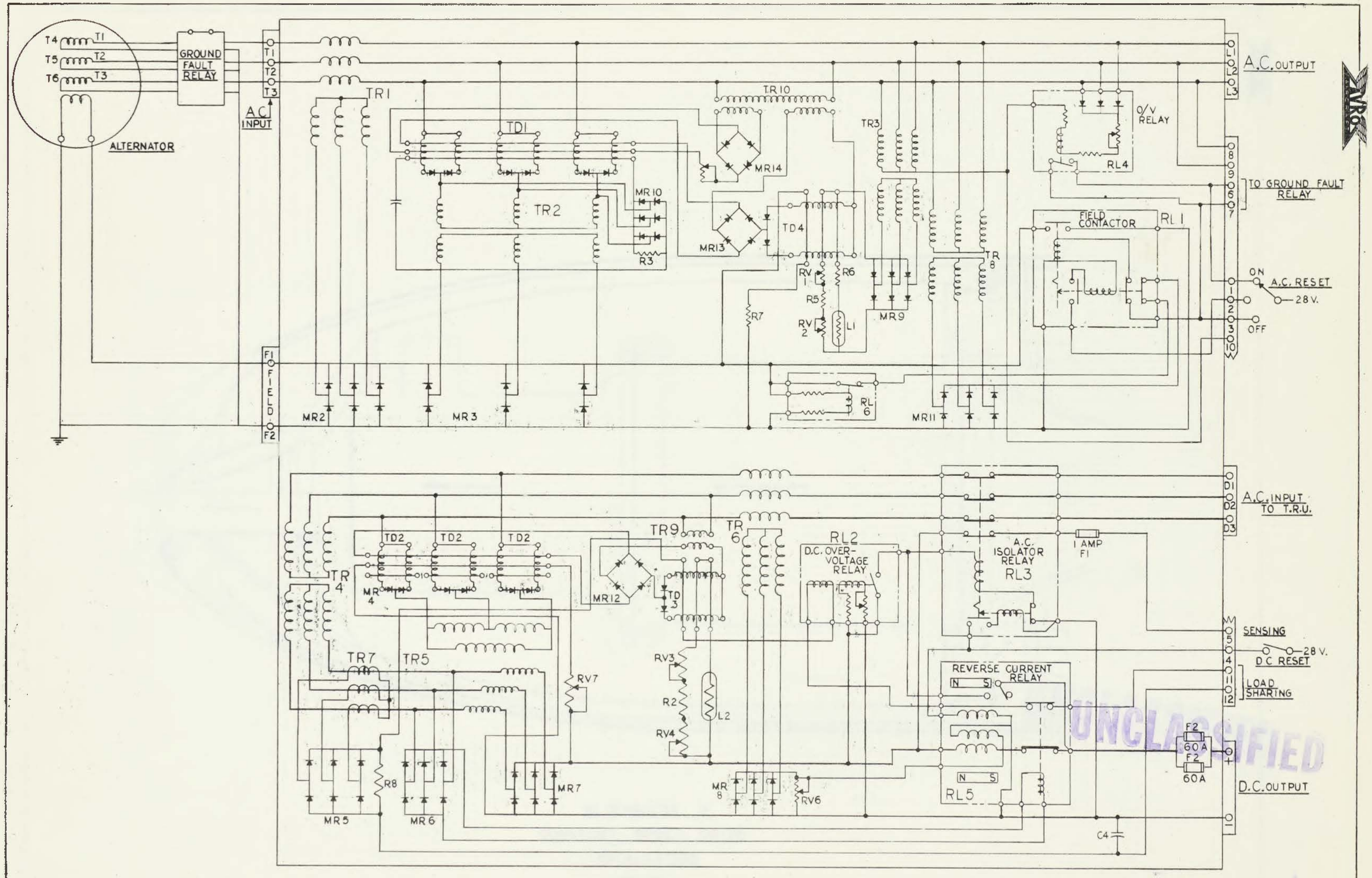
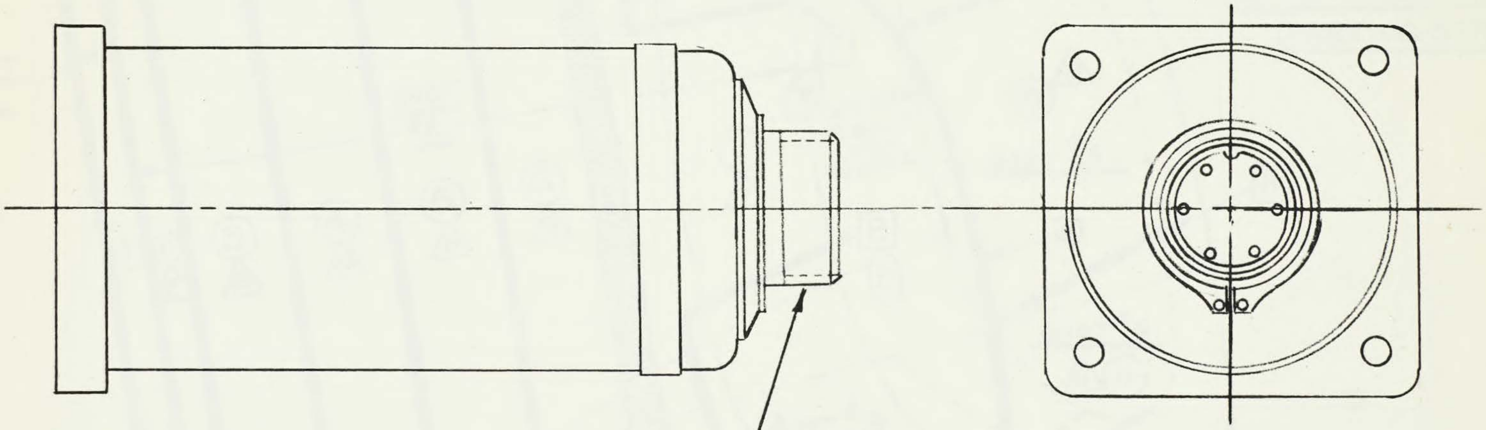
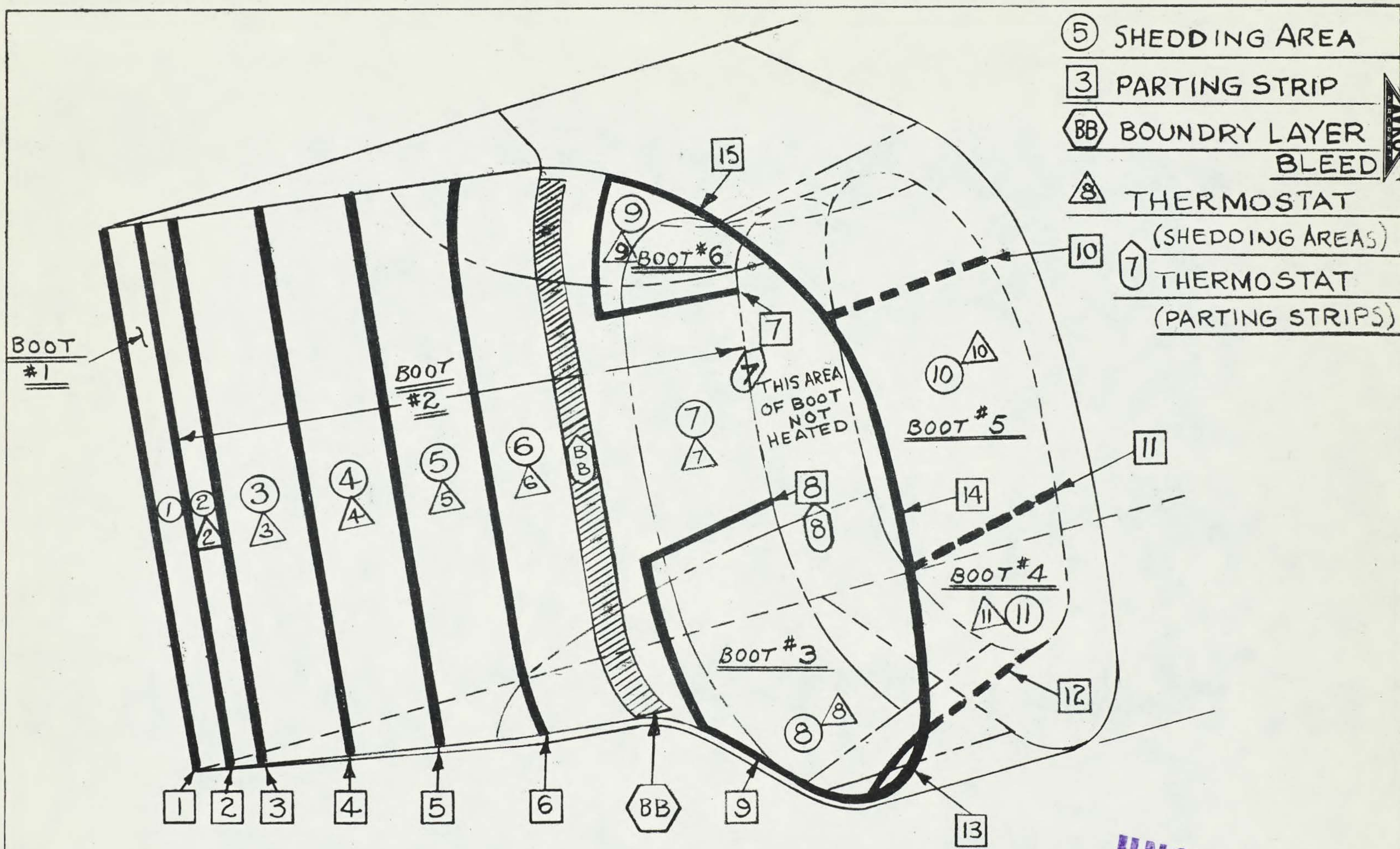


WIRING DIAGRAM A.C. CONTROL AND TRANSFORMER RECTIFIER UNIT. FIG. 21.



POWER FAILURE DETECTOR

FIG. 23



DUCT DE-ICING BOOTS

FIG. 24

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