Third Quarterly Progress Report

INVESTIGATION AND DEVELOPMENT OF NEW CONCEPTS FOR IMPROVEMENT OF AIRCRAFT ELECTRICAL POWER SYSTEMS

For

NASA Electronics Research Center
Cambridge, Mass.

Contract: NAS 12-659

68-4176(3)
May 13, 1969

(ACCESSION NUMBER) 155 (THRU) NONE
(PAGES) (CODE)
(NASA CR OR TMX OR AD NUMBER) #107329 (CATEGORY)
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I. INTRODUCTION AND SUMMARY

This is the third quarterly progress report on a study entitled "Investigation and Development of New Concepts for Improvement of Aircraft Electrical Power Systems," funded by NASA Electronics Research Center, Cambridge, Massachusetts, Contract No. NAS 12-659. Because of a modification of the study program, this quarterly progress report has been delayed.

I.1 OBJECTIVE AND METHOD OF APPROACH

The study objective is to formulate a philosophy for devising optimum electrical power systems for advanced aircraft. The philosophy recommended will consider improvements in reliability, safety, weight, size, and other factors that would result in increased revenue for transport aircraft; it will apply, modify, and extend advances in space power technology and related fields. The entire aircraft electric power system, including generation, conversion, distribution, and utilization equipment will be subject to a complete review in accordance with the above objective, independent of traditional 400-Hz aircraft power technology.

The study is oriented primarily toward the relative merits of various power system concepts. To accomplish this it is necessary to establish the approximate performance capabilities of the individual components so that they can be used collectively to indicate system performance. Consequently, the performance data in this report are representative rather than detailed component performance predictions.

The study information has been divided into three categories:

- Equipment in use whose performance primarily reflects technology existing at the time of its design (present large subsconic transports were designed in the late 1950's; the small subsonic transports in the early 1960's)
- Equipment that can be designed with current technology and has not yet been used in aircraft
- Equipment that can be predicted with technology anticipated for 1980 to 1985.

Meaningful presentation of the information from this study requires the application of anticipated component performance data to the various system configurations to arrive at relative system performance. A baseline system configuration (described in the first quarterly report) typical of existing large subsonic aircraft has been used as a reference point with which to compare the candidate systems.

Information presently available is primarily on the relative weights of the candidate systems; relative performance, reliability, maintainability, cost, etc., should be further defined later in the study.

In this quarterly report, the information presented is confined to that associated with equipment actually in use and equipment as represented by current technology.

The anticipated output of this study program will be the documentation of a philosophy for devising optimum aircraft electrical systems and the selection and description of candidate configurations for future aircraft electrical power systems. The performance (weight, reliability, size, maintainability, safety, etc.,) of the candidate systems will be compared. In addition, significant technological problems applicable to selection of the most promising candidate system will be discussed.

1.2 STUDY TASKS

To facilitate understanding the scope of the study, the major program tasks are listed below:

(a) Conduct an industry and literature survey on present day practice and state of the art in aircraft electric power generation, transformation, distribution, and utilization including present and projected electric loads on aircraft.

(b) Optimize the methods to supply electric power to the various loads to suit their inherent functional mechanism.

(c) Compile the necessary parametric information for the power system components on physical size, weight, and performance for the given power levels vs power system configuration, voltage, and frequency. This information will be based on the present state of the art of system component technology.

(d) Investigate application of new materials such as exotic magnetic materials, new insulation materials, unconventional conductor materials, etc. for critical electrical system components.

(e) Investigate new system component concepts for application to aircraft in the sense of the objectives of this program.

(f) Review and analyze the preferable power system heat transfer technique.

(g) Specify the techniques and displays required such as failure prediction, detection and compensation, efficient energy management, and data monitoring for the acquisition and characterization of the electric power system.

(h) Investigate power system component reliability and its tradeoff against component cost and weight.

(i) Investigate component maintainability and its tradeoff against component cost and weight.

(j) Survey and determine the static and dynamic power requirements for present hydraulic and pneumatic powered loads and establish the tradeoff criteria in the selection and application of electric or combined electric-hydraulic and electric-pneumatic power subsystems for that purpose.

(k) Determine the relative weight of factors that determine the "effectiveness" of the electric power system in the sense of program objectives.

(l) Formulate a philosophy of aircraft electric power system design consistent with the objectives stated above.

(m) Devise and analyze power system candidates with inclusion of system design and optimization. Provide complete power system diagrams and descriptions.

(n) Identify the significant research and technology problems associated with achieving the most promising candidate electric power systems.

1.3 PROGRESS TO DATE

1.3.1 First Quarter Work

During the first quarter, existing literature was surveyed to establish the state of the art and trends in system design. A bibliograph on aircraft electrical systems and components was compiled. Trips were made to major airframe companies to obtain data on the electric power systems of existing or planned aircraft. This data was needed to establish configuration and design requirements for typical current electrical power systems.

It has been established that the majority of the power system weight (typically about 11,000 lb for large existing transports) is in the utilization (typically 65 percent) and distribution (typically 27 percent) segments of a typical electrical power system. These are the areas in which weight reduction efforts will be most profitable.

The work on electrical load analysis, presented in greater detail in this progress report, consisted of a preliminary classification of the electrical loads, and a presentation of parametric data on both constant and wild frequency electric motors.

1.3.2 Second Quarter Work

In the second quarter, the electrical load analysis was continued. The parametric data of aircraft utilization components are divided into three groups: (a) components in existing aircraft, (6) current technology components to be applied to aircraft, and (c) components with future technology. The utilization weights were compared as a function of system frequency.

Possible methods of generating electric power were examined in the second quarter. The established existing generation system is the constant speed constant frequency (CSCF) system. Current technology systems under development are variable speed constant frequency (VSCF), rotating inverter (which uses two rotating machines to provide the required power), and high voltage dc systems. Analysis indicates that the more advanced generating concepts can offer performance advantages in comparison to the present CSCF systems while still remaining weight competitive. The cycloconverter VSCF system, tentatively selected for use on the SST, appears to be particularly attractive; another possibility is the Learverter system, although information on it is limited.

Power conversion techniques were considered and compared. Parametric weight data were presented for the various possible techniques. Selection of an optimum conversion system cannot be made without system optimization since the type of conversion equipment is dependent upon the load location and type, and the generating equipment output.

In the area of power distribution, design considerations on safety, corona, voltage and frequency levels were discussed. Data for voltage and frequency effect on wiring weight were also presented.

Because many aircraft loads are presently serviced by hydraulic or pneumatic power, the design criteria and performance capabilities of both the hydraulic and pneumatic systems have been surveyed. The survey indicates that the major hydraulic loads are flight controls, landing gear, brakes, and thrust reversers. It also indicates that for reasonable distances between the power source and the load, a hydraulic system can provide the load power for a lower weight than can the electric system.

1.3.3 Third Quarter Work

During the course of study in the first two quarters, it was discovered that the investigations and analyses of various tasks were restricted in both scope and depth to the limited effort allowed. Work performed so far on the program has indicated that there is considerable incentive for devoting more effort to deepen and broaden the study. With the approval of NASA Electronics Research Center, the present study is extended for one more year to July 1970. The program schedule is modified accordingly. The progress bar chart for the revised program from May 1969 to July 1970 is shown in Figure 1-1.

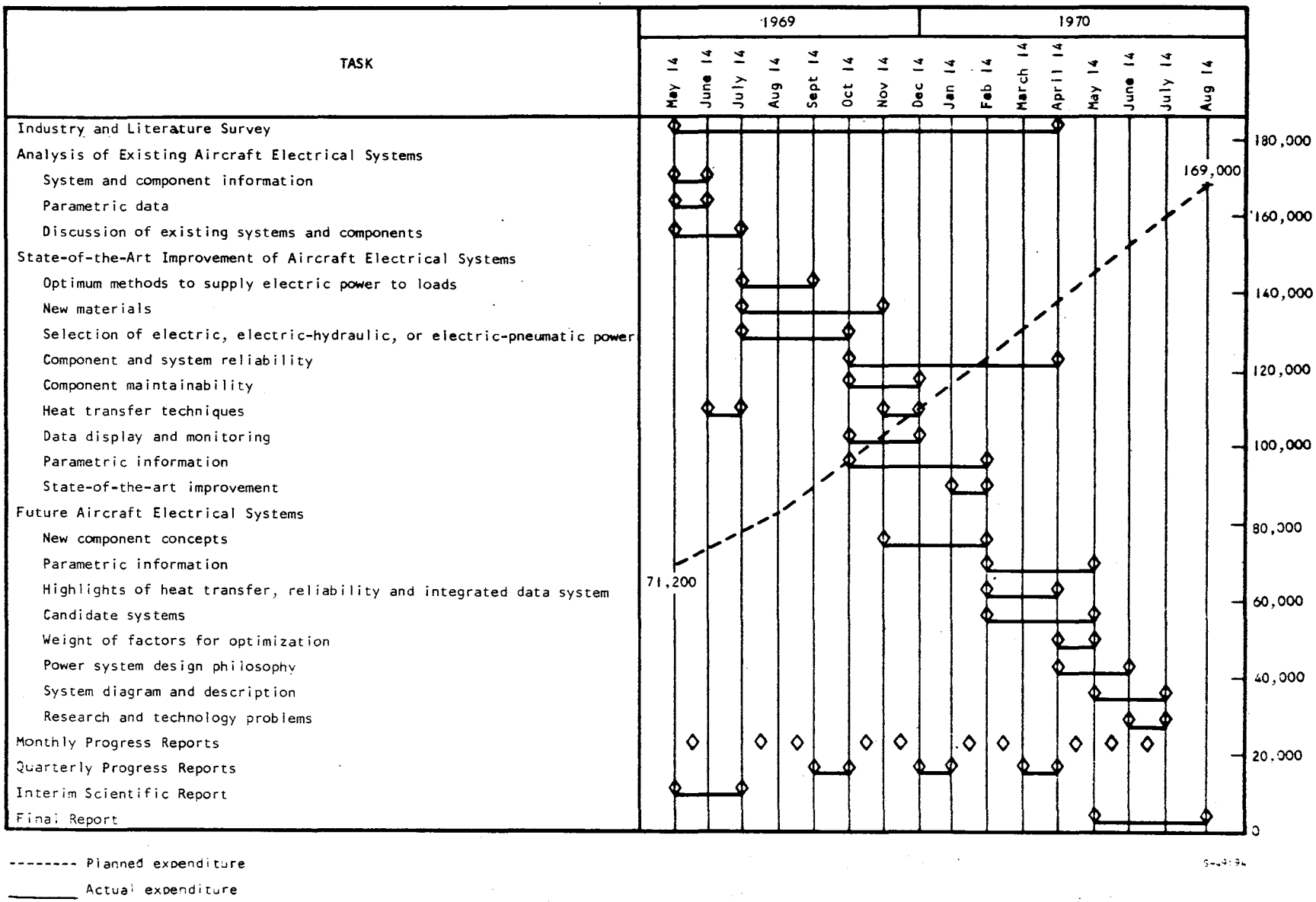


Figure I-1. Program Schedule

Because of the modification of the study program, this third quarterly progress report has been delayed. The activity of the study was also reduced in the last few months. Since more information is needed for electrical equipment currently installed in aircraft, this quarterly report is mainly devoted to this task.

Section 2 reviews the distribution system currently installed in large commercial transports. The characteristics and performance data of aircraft cable, circuit breakers, relays and connectors are also presented.

System control and protection methods are discussed in Section 3. Various protection schemes for single and parallel system operation are reveiwed.

Section 4 describes the emergency power sources in existing aircraft. These sources are storage batteries, gas turbine APU's, gasoline engine APU's and ram air turbines.

Section 5 outlines the current philosophy of displaying information on electrical power system in commercial transports.

Section 6 describes the present techniques of cooling aircraft electrical components and heat transfer systems. The possible improvements in heat transfer methods utilizing state-of-the-art technology are also included.

Section 7 discusses the objectives and criterion of aircraft electrical system reliability. The level of reliability currently achieved by various aircraft components and subsystems is also presented in this section.

1.4 WORK FOR THE NEXT REPORTING PERIOD

The work for the next reporting period will be devoted to the preparation of the interim scientific report. This interim report will summarize the work performed during the first year of this study program concerning the electrical components and system in existing commercial transports.

More precisely, the work in the next reporting period will consist of the following tasks:

(a) Summarize and describe the electrical power systems and components in existing commercial transports

(b) Rearrange all parametric information on current aircraft electrical equipment compiled during the past year. Develop additional data if necessary

(c) Evaluate and discuss the limitations of the existing aircraft electrical power systems and components

(d) Prepare the interim scientific report

2. POWER DISTRIBUTION

2.1 DISTRIBUTION SYSTEM

The function of the aircraft electrical power distribution system is to distribute the generated power to various electrically driven loads such as the fuel pumps, avionics, flight control systems, heating equipments, and lighting devices.

The distribution system is the system between the generator and the loads. It consists of two major areas:

(a) The transmission subsystem which is that part of the system between the generator and the main power shield

(b) The distribution subsystem which distributes the electric power from the main power shield to the loads

Since there are various types of aircraft, generating systems, and design requirements and considerations, many different types of distribution systems have been used in the aircraft electrical power system. In general, the choice of the system type depends on the following factors:

(a) Number of source buses

(b) Reliability of source buses

(c) Duplication of emergency load functions

(d) Weight, space, and complexity penalties involved in making more than one source bus available for the essential load buses

(e) Degree and type of selectivity available in the generator and bus protection system

2.1.1 Typical Distribution System in Existing Aircraft

Many different types of electric power generation systems and their associated distribution systems have been used or are being used in the various types of aircraft. It is almost impossible to investigate and discuss everyone of them. For this study program, it is intended that the discussion of the distribution system be limited to a system which is in widespread use today on large commercial aircraft. This system is of the constant speed constant frequency type (CSCF) with four generating channels.

2.1.1.1 System Capacity

For the four-engine commercial aircraft, four generators of the same capacity are usually installed, one for each engine. The generators are driven from the main engines via constant speed drives. The capacity of the generator selected is such that the total load demand of the aircraft can be met approximately with only three generators operating in parallel, although they usually require slight load monitoring with one generator out. When two generators are inoperative, the remaining generating capacity is normally large enough to supply all the essential loads.

One or more auxiliary power unit (APU) generators may be installed to supply the ground electrical loads when the main engines are shut down. The capacity of the APU generators is about the same as that of the main generators; however, in some cases, they can be rated at a higher output because of the extra cooling provided.

2.1.1.2 Type of Power

In the typical four-generator, constant speed, constant frequency system, the primary electrical power is 3-phase, 400 Hz ac power regulated at 115/200 v. During the flight, the power is supplied by the main engine driven generators, while on the ground this primary ac power can be supplied by either a ground power supply, or by the aircrafts own APU (if equipped).

In addition to the primary ac power, 28-vac and 28-vdc power are also provided on the aircraft to supply various lights, instruments, and other loads. The single-phase, 28-vac power is derived from single-phase autotransformers energized by the main ac system through various feeds. The 28-vdc power is supplied by transformer-rectifier (TR) units which receive input power from the main ac system. In order to ensure a reliable dc source, more than one transformer-rectifier unit is used. Normally, the TR units are operated in parallel.

In addition to the TR units, a battery is provided as a standby source. In some cases, this source permits starting the electrical system without an external source, supplies backup electrical system control power, and can supply power to minimum navigation, instrument, and communication systems and other 115-vac equipment needed for safe flight and landing. The 115-vac power, under such conditions where the battery power is the only electrical source, is supplied by the battery powered inverter.

2.1.1.3 Type of System

For the four-channel electrical power system, the typical distribution system is either an all parallel system or a split parallel system. In this system, each generator is connected to its own three-phase load busbar by a generator circuit breaker. All four generators may be operated in parallel or can be split into two parallel subsystems, one on each side of the aircraft. These arrangements can be accomplished by means of bus tie breakers and split system breakers.

The split parallel system has the advantage of complete independence of the two subsystems so that any fault or mismanagement in one subsystem cannot affect the other. This ensures a high reliability for the total generating system. The all parallel system, however, when designed to meet the one channel out case, gives a better utilization factor. A typical system diagram is shown in Figure 2-1.

2.1.2 Busbar Arrangement

2.1.2.1 Main AC Busbar

A three-phase main busbar is connected to each generator through a generator circuit breaker. The busbars are interconnected by bus tie breakers and split system breakers for parallel operation. Ground power, which can be derived either from the auxiliary power unit or from an external supply can be fed to a main ac busbar via a ground power breaker or to the synchronizing bus.

2.1.2.2 AC Essential Busbar

One or more three-phase essential busbars are utilized, normally supplied from the main ac busbars. In case the power can not be obtained from the main bus due to a fault at the main bus, the essential busbar will be supplied from an emergency ac generator driven by the aircraft hydraulic system. The changeover is usually automatic, but it may also be initiated manually, if required.

The ac essential loads supplied by the emergency generator may be divided into two categories:

(a) Essential Aircraft Loads--These loads are essential for handling the aircraft. This power supply will be maintained under all flight conditions until touchdown. When all ac power supplies are out, these loads are supplied by the battery-powered inverter.

(b) Essential Engine Loads--These are the loads required to maintain the engines. This supply will be discontinued in the event of all engine failure.

2.1.2.3 DC Busbars

The dc loads of the aircraft are supplied from the main dc busbar, one or more depending on the design requirements. During normal operation, this dc power is obtained from the transformer-rectifier (TR) units which are energized from the main ac buses. Normally, the TR units are operated in parallel.

In addition to the main dc busbar, the dc essential busbar is also used to supply the essential dc loads. This busbar can receive power from either the main dc busbar, the ac essential busbar through an essential TR unit, or from the standby battery.

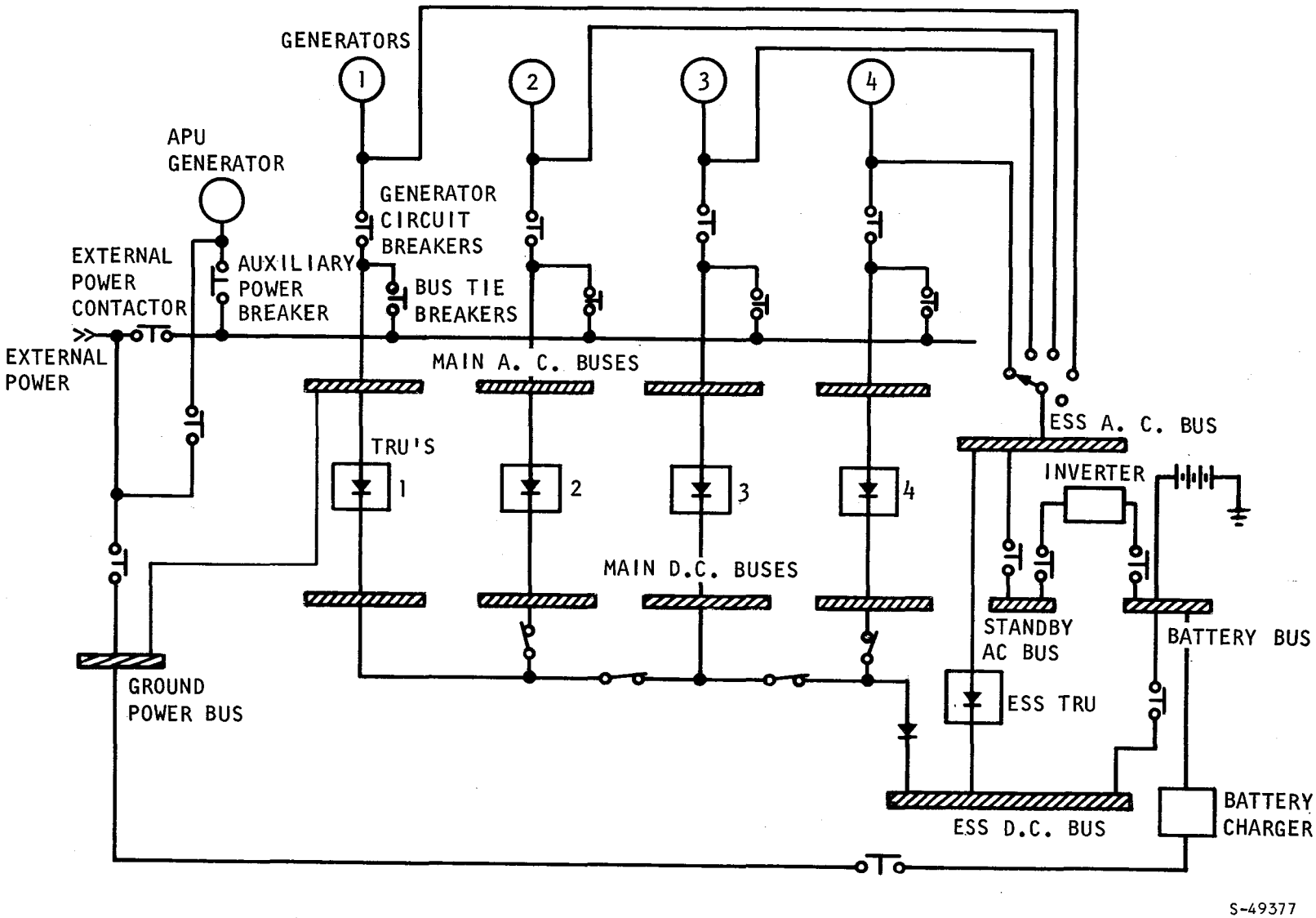


Figure 2-1. Typical System Diagram

2.1.2.4 28-VAC Busbar

Various lights and instruments require single-phase, 400-Hz, 28-vac power. The 28-vac busbar is supplied through an autotransformer from either the main ac busbar or the ac essential busbar.

2.1.2.5 Other Busbars

Depending upon the system configuration, other busbars may be used in the distribution system.

(a) Standby 115-VAC Bus--This bus supplies 115-vac to essential equipment needed for safe flight and landing. During normal operation, it is supplied by the essential ac bus. In the event of loss of essential ac power, the standby 115-vac bus is supplied by the standby battery through a dc-to-ac inverter.

(b) Ground Service Buses--The 115-vac and 28-vdc ground service buses provide power to the loads needed during ground operation. The power of the ac ground service bus may be obtained from the external power receptacle or from the APU generator through transfer relays. The dc ground service bus is supplied by the ac bus through a TR unit.

(c) Ground Handling Bus--The 115-vac and 28-vdc ground handling buses provide power to equipment used for cargo handling. The ac bus can receive power from either the external power receptacle or the APU generator through a transfer relay. It is not energized during the flight. The dc bus is supplied by the ac bus through a rectifier.

(d) Flight Instrument and Radio Buses--These buses providing power to flight instrument and radio equipment are supplied from the primary ac busbars (main ac busbar and ac essential busbar).

(e) Standby DC Bus and Battery Bus--These buses provide the standby dc power. They are energized by the essential dc bus during normal operation, but can be transferred to the battery source automatically in the event of loss of essential dc power.

2.1.3 Load Transfer

Two typical bus arrangements are shown in Figure 2-2. During normal operation, Scheme A would operate with all the circuit breakers closed. When one generating channel fails, that channel is disconnected from the system, and the rest continue to supply the required power. However, Scheme B, a split parallel system, would operate with the tie breaker open under normal conditions. Thus, the two subsystems are indepedent. The tie breaker would be closed only during ground operation or a complete failure in one subsystem.

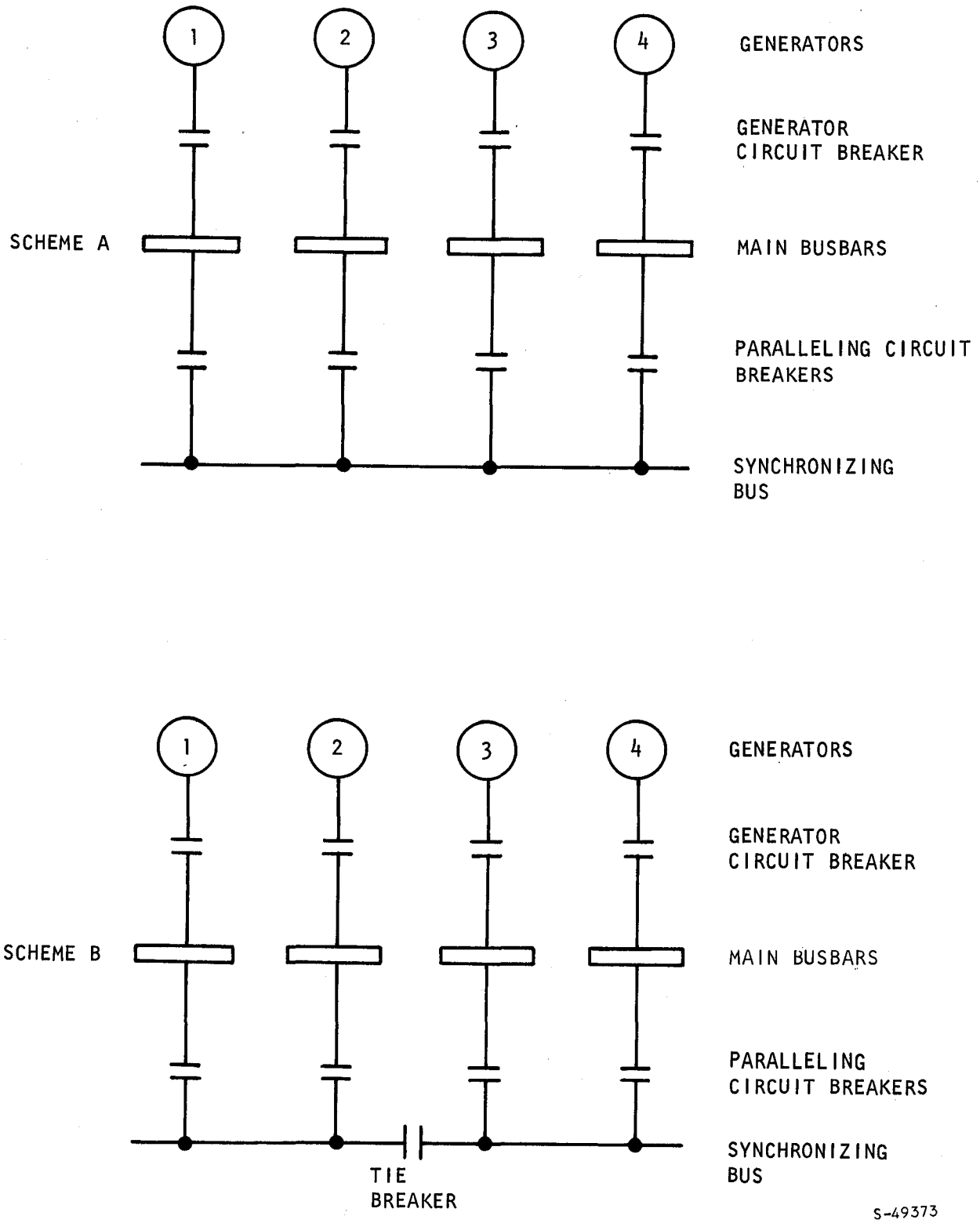
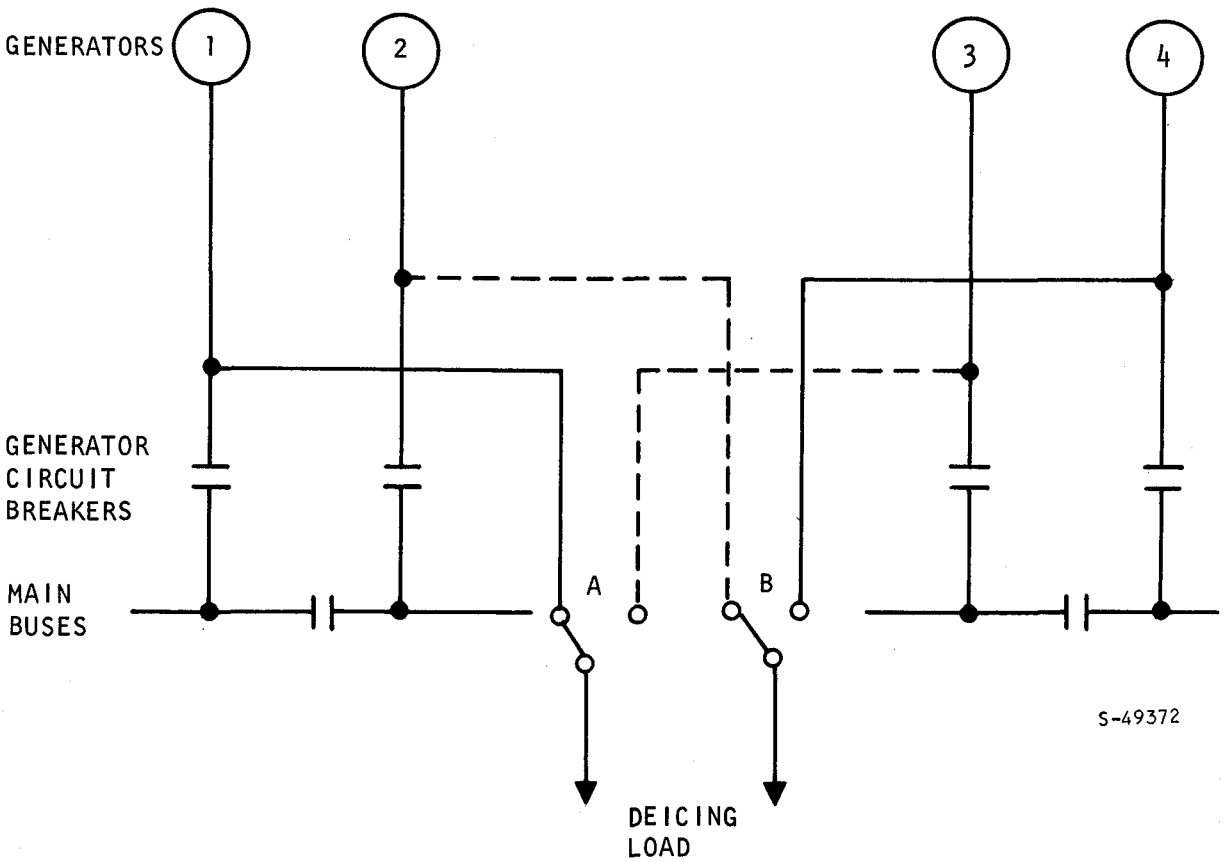


Figure 2-2. Typical Bus Arrangements



LOAD TRANSFER ARRANGEMENT

Figure 2-3. Load Transfer Arrangement

Scheme B has the advantage over Scheme A in that any fault or mismanagement in one subsystem cannot affect the other, thus reducing the possibility of losing the whole generating power due to a single fault. When one generating channel fails in Scheme B, the load is transferred to the healthy channel. When a whole subsystem fails, the tie breaker is closed, and the load in the failed subsystem is supplied by the operating subsystem.

When designed to meet the one channel out case, however, the advantage which Scheme A has over Scheme B is that the utilization factor is higher. In this four-channel system, Scheme A gives a generator utilization factor of 0.75, whereas Scheme B gives only 0.5.

There are alternatives for both Scheme A and Scheme B. In Scheme A, two paralleling circuit breakers can be left open under normal operating conditions; thus, there are three independent subsystems.

Scheme B, on the other hand, can operate in such a way that when Channel 1 fails, the generator circuit breaker of Channel 1 and the paralleling circuit breaker of Channel 2 open and the tie breaker closes. Thus, Generator 2 will supply its own load, and the load on Channel 1 will be supplied by Generators 3 and 4.

Surface deicing in the aircraft requires a great amount of power. Since this power is needed only occasionally, the load is usually split into two parts and supplied by the two subsystems, respectively, in a split parallel configuration. This limits the amount of power to be transferred, and uses the installed generating capacity more efficiently.

When one generating channel fails, the load transfer can be accomplished by an arrangement shown in Figure 2-3. Assuming Generator 4 fails, Contactor B can be switched from right to left, transferring the deicing load from Generator 4 to Generator 2. The general loads on Busbar 4 will be supplied by Generator 3.

There are many different types of load transfer arrangements, the choice of which depends on such factors as the reliability requirement, the generator utilization factor, and the characteristics of the loads.

2.2 AIRCRAFT CABLES

The wiring installed on large transport aircraft makes up for about 70 percent of the electrical system weight (generation and distribution only, not including load equipment). Wiring is the heaviest single item in the entire electrical system; it therefore attracts special attention in the continuous effort of weight reduction. In view of this, the importance of selecting the optimum wires (type and gage) for the various application is evident. Aside from the basic requirement of carrying current without excessive voltage drop and heating. A number of other factors must be considered. Starting once more with weight, these factors are discussed in the following paragraphs.

2.2.1 Weight

A considerable portion (60 to 70 percent) of the total installed wire weight is contributed by cables of size 20 or smaller. The ratio of the insulation weight to the conductor weight increase as the cable size decrease. The weight of copper is about equal to the weight of insulation for size 18 wire. For wires smaller than size 18, the weight reduction in insulation is, therefore, more meaningful than the weight reduction in conductor itself.

2.2.2 Flexibility

It is desirable that the cables remain flexible over the full range of ambient temperature, which may range from -50 to +100°C. In areas adjacent to the engine, the maximum temperature may reach 300°C. The cable insulation should not become too soft or too brittle within the temperature range to which it is exposed.

In present day aircraft there is usually a large number of wires in a harness and many wires are required to terminate at miniature or subminature high-density terminal blocks. All these, plus limited access to installation areas, tend to require more flexible cable.

2.2.3 Abrasion

During installation and maintenance, abrasion is almost unavoidable due to either installation design weaknesses or insufficient care by maintenance personnel. In flight abrasion may occur because of vibration.

In order to minimize this abrasion hazard every conceivable precaution should be considered. The cable should be sufficiently resistant to abrasion in the first place to prevent the wear and tear from causing a short circuit to ground. For comparing cables of different construction, an accurate method of determining the abrasion resistance of the cable is highly desirable. Unfortunately, this has been difficult as the repeatability of the standard abrasion test is very poor, unless the wide tolerances inherent in the test procedure are acceptable.

Aircraft and cable manufacturers have been seeking various forms of abrasion tests for evaluation purposes including:

(a) Scape - Simulates the action of a metal edge moving relative to the cable insulation

(b) Cut through - Simulates the action of pressure from a metal edge on the cable without relative motion

(c) Notch - Determines the tear strength of the cable; the result provides a measure of toughness

Successful means of determining cable abrasion resistance will be very valuable in improving cable characteristics.

2.2.4 Contamination

In many locations within the aircraft, wiring may come in contact with various fluids such as fuels, lubrication oils, coolants, anti-icing sprays and hydraulic fluids. In some cases, the contamination may be accompanied by elevated temperatures, enhancing the destructive effect. Therefore, it is desirable that all general purpose aircraft cables should have appropriate contamination resistance against all those fluids known to be used in aircraft.

Since synthetic lubricants, and kerosene fuels are widely used in contemporary aircraft, insulating materials such as nylon (which are proof against these fluids) should be used in outer sheaths.

2.2.5 Fire Resistance

It is known that deficiency in aircraft wiring is one of the major sources of fire hazards. The various causes which would induce fire in wiring are as follows:

(a) Circuit overload due to faulty electrical equipment

(b) Inadvertent shorted wires

(c) Wet wire fire

A number of fire incidents on aircraft were identified as "wet wire fires". A peculiarity of those incidents was that the cause of wire fires could not be attributed to circumstances such as circuit overloads due to faulty electrical equipment or inadvertent shorted wires. In some cases, circuit protection devices proved ineffective or did not respond until after smoke or fire had started.

This gave rise to closer investigation. Major airframe manufacturers conducted laboratory tests, simulating conditions which were believed to occur in the aircraft. Two conditions causing the hazard were determined to be damaged insulation and the presence of moisture. A third condition is that the wire bundle must be energized. Because of the necessity of moisture in the process, this hazard has been identified as "wet wire fire."

So far, wet wire fires were found to occur on general purpose aircraft wiring only. Since about 80 percent of all wiring in present day aircraft is made up of such wire, there is a need for an improved general purpose wire, to eliminate or minimize the potential danger of wet wire fires.

In the engine areas, some circuits are required to operate during and after a fire. Therefore, fire resistant cables which can operate satisfactorily after being subjected to a temperature of 1100°C for 5 minutes without creating any additional fire hazard must be used. In other areas, cables must at least be fire resistant, and not support combustion after the source of ignition has been removed.

2.2.6 Mechanical Strength

All aircraft wirings are liable to being pulled during service, such as through ducts, conduits, on terminal lugs or by accident during installation and maintenance. During service, cables are also subject to forces due to vibration. Without sacrificing other cable characteristics, the tensile strength of the cable must be as high as practical particularly for the smaller size calbes. Although copper is a good conductor, it has poor tensile strength and the ordinary type of insulation offers little assistance. For this reason, the smallest size of copper cable that was used in aircraft was limited to gage 22. However the new cadmium-chrome-copper alloy increased the tensile strength so much, that wires as small as gage 26 are presently installed in aircraft.

Other mechnical strength such as resistance to deformation and bending are also desirable. These resistances are primarily a function of the type of insulating material. The cushioning effect of glass braid insulation can enhance deformation resistance of cables.

2.2.7 Current Overload Performance

All cables in aircraft require protection against overload. The protective device, whether fuse or circuit breaker must be compatible with thermal characteristic of the cable. This prevents degradation of the cable from reducing its life below that of the aircraft. Some equipment has a very high inrush-current characteristic. To prevent this characteristics from tripping the protective device, a larger cable whose current carrying capacity exceeds the continuous load current must be used. Because the protective device has a manufacturing tolerance on its trip characteristic (for a typical circuit breaker the minimum and maximum trip values are 110 percent and 138 percent respectively of the nominal value) and is available only in a certain range, it is very difficult in practice to use the cable at its stated rating. If, however, the cable is allowed to carry a current in excess of the specification rating for limited duration, the restriction on its rating could be largely overcome.

Complete short-term ratings for standard aircraft cables cannot be found in published literature. Test procedures are being evaluated to provide the required information with which the circuit breaker characteristics could be determined to match the cable more accurately. Thus maximum savings in cable size and weight can be accomplished.

2.2.8 Conductor Materials

Copper and aluminum have both been used as conductor material in aircraft cables. The high conductivity of copper together with its ability to be soldered or crimped easily has made copper the preferred conductor material in the past. Its disadvantage is its high density. Aluminum has a volumetric resistivity of 1.6 times that of copper. However, the ratio of densities, aluminum to copper is only 0.3. Therefore, an aluminum wire weighs only about one half of that of a copper wire of the same resistance. Thus, considerable weight savings can still be achieved by using aluminum wires. But because of its low mechanical strength, the use of aluminum as conductor material is limited to larger size wires (size 8 or larger). Another serious drawback of the aluminum calbe is the termination problem. Better termination techniques must be developed, before extensive application of aluminum wires on aircraft can be realized.

As there are many circuits in aircraft that carry only a small amount of current, the conductivity of the conductor is not a limiting factor. A suitable conductor material, therefore, should have acceptable combination of both electrical and mechanical properties. Copper-dominated conductors have been used for aircraft cables for many years, including tinned copper, nickel-clad copper and silvered copper. Non-copper conductors used include aluminum conductors of larger sizes, stainless steel conductors, and conductors made of other alloys. Recent trends are towards copper alloys and aluminum alloys which give a satisfactory balance of mechanical and electrical properties. The binary alloy chromium-copper and the ternary alloy cadmium-chromium copper, for instance, have very attractive machanical properties.

Aluminum alloys such as aluminum-magnesium-silicon appear attractive. Their feasibility as conductors is being investigated. Aluminum alloys, however, may not be satisfactory for use in small wire sizes and their applicability depends largely on the development of successful termination techniques.

2.2.9 Cable Insulation

Most of the insulating materials used on aircraft cables are of the synthetic type. They are synthesized from basic materials by chemical processes of which polymerization is the main feature. This process is usually accomplished by applying heat, pressure, or a certain catalyst to the chemical reaction under strictly controlled conditions.

Synthetic cable insulations include PVC (polyvinyl chloride), PTFE (polytetrafluoroethylene), Teflon, nylon, kynar and polyaklene. The only natural insulating materials used are glass, asbestos, and rubber. Ordinarly, some modification process is required to change their natural state before they can be used. Cable insulation usually consists of several layers. From the conductor outward, the first layer may be a conductor wrapping made of inorganic material. The next layer is the main insulation, which is usually made of PVC or teflon. Glass braid is then the next layer, if used, and the outermost covering could be either made of nylon or some other composite material.

Table 2-I shows the properties of the most commonly used insulation materials.

TABLE 2-1

PROPERTIES OF AIRCRAFT CABLE INSULATIONS

Properties	Polytetrafluoroethylene	Polyethylene	Cellular Polyethylene
Specific gravity	2.1 to 2.3	0.92	0.50
Volume resistivity, ohm-cm (50 percent relative humidity and 23°C)	10^{15}	$>10^{15}$	-
Dielectric strength, short-time 1/8 in. thickness, volts per mil	480	460	220
Dielectric strength, step-by-step 1/8 in. thickness, volt per mil	430	420	-
Dielectric constant, 10^{6} cycles	2.0	2.26	1.5
Dissipation (power) factor 10^{6} cycles	0.002 to 0.005	<0.0005	0.0004
Effect of strong acids	None	Attacked by oxidizing agents	-
Effect of strong alkalies	None	Resistant	-

2.2.10 Current Carrying Capacity

The current carrying capacity of the cable at given ambient conditions is determined by the maximum allowable temperature of the cable, which is in turn dependent on the type of cable insulation and its heat dissipation capability. For continuous loads, the cable temperature must remain within certain limits to avoid excessive aging discoloration and deterioration of the insulation. Therefore, for each type of cable, a maximum allowable current at a given ambient temperature is specified for each cable size. Table 2-2 shows the current carryint capacity together with other data for aircraft wires and cables for the present 115/200 v, 400-Hz system in accordance with MIL-W-5086 and MIL-W-5088. It shows two current values for each size of cable, one for single conductors in free air and the other for bundled wires, which is based on bundles of 15 or more wires.

The current ratings for the bundled case are lower because the heat dissipation capability of the cable is reduced. For smaller bundles, the allowable percentage of total current may be increased as the bundle approaches the single wire condition,

The current ratings shown in Table 2-2 apply to general purpose wires for maximum wire temperature of 105°C with a maximum ambient temperature of 60°C. They also apply to wires, in accordance with MIL-W-7139, Class 1, for a maximum conductor temperature of 200°C with a maximum ambient of 155°C, and to wires in accordance with MIL-W-7139, Class 2 for a maximum conductor temperature of 260°C with a maximum ambient temperature of 215°C.

When the ambient temperature is above the maximum specified ambient temperature, the current carrying capacity is reduced because firstly the high ambient reduces the permissible temperature rise and secondly it increases the wire resistance and, therefore, the losses. Altitude also affects the current carry capacity. At high altitude, owing to the reduced air density, the heat dissipation capability is reduced and hence the current carrying capacity is degraded. Naval Research Laboratory has conducted experiments and has established that for wire sizes from No. 20 through 1/0. The relative values necessary to produce the same conductor temperature rise are as shown in the Table 2-3.

2.2.11 Voltage Drop

Voltage drop is another factor that governs the selection of the cable size for a given application. According to MIL-W-5088, the continuous value of the maximum allowable voltage drop of the 115 v system is 4 volts. That means the total impedance of the cable and the ground return in the circuit should not drop the voltage more than 4 volts between the point of voltage regulation and the load. This requirement limits the length of wire of any particular size. Figure 2-4 shows the maximum lengths of cable which can be used for each size for two conditions, the single conductor and the bundled. In present day large commercial aircraft, cable lengths of over 100 ft are quite common, and voltage drop considerations are usually the governing factor in the selection of size of smaller cables.

TABLE 2-2

PROPERTIES OF AIRCRAFT CABLE (COPPER)
(MIL-W-5086)

AN Size	20°C ohms per 1000 ft	Max Amp 60°C Ambient		Cir MIL Area	Max Dia Cable (in.)	Max Dia Conductor (in.)	Weight Finished Cable lb/1000 ft
		Bundled	Single Condition in Free Air				
22	15.52	5	-	755	0.075 ±0.005	0.033	4.7
20	9.7	7.5	11	1,200	0.085 0.005	0.041	6.8
18	6.08	10	16	1,909	0.095 ±0.005	0.052	9.5
16	4.76	13	22	2,409	0.105 ±0.005	0.061	11.9
14	2.99	17	32	3,830	0.125 ±0.007	0.076	18.3
12	1.88	23	41	6,088	0.143 ±0.007	0.096	26.0
10	1.16	33	55	9,880	0.189 ±0.007	0.128	44.0
8	0.70	46	73	16,864	0.240 ±0.007	0.176	70.0
6	0.436	60	101	26,813	0.293 ±0.007	0.218	110.0
4	0.274	80	135	42,613	0.355 ±0.010	0.272	165.0
2	0.179	100	181	66,832	0.425 ±0.010	01345	250.0
1	0.144	125	211	82,108	0.470 ±0.010	0.384	305.0
0	0.114	150	245	105,022	0.525 ±0.015	0.432	400.0
00	0.090	175	283	133,665	0.590 ±0.015	0.490	500.0
000	0.072	200	328	167,332	0.650 ±0.015	0.548	620.0
0000	0.057	225	380	211,954	0.720 ±0.015	0.615	785.0

TABLE 2-3

ALTITUDE DERATING OF CURRENT-CARRYING CAPACITY

Altitude, ft	Relative Current, Percent
0	100
20,000	93 ±2
60,000	82 ±2

2.2.12 Installation Requirements

Installations may vary according to the size and the type of the aircraft in which the amount of space available and the configuration of the electrical power system are the governing factors. Installations may also vary between different manufacturers whose policies deviate from each other. However, some basic guiding points should be generally observed. They are as follows:

(a) The risk of mechanical damage to the cable must be reduced to a minimum by proper and careful routing. The cable should be kept away from all moving parts such as control cables, control surfaces and linkages, etc. Areas which can be easily contaminated should be avoided whenever possible.

(b) Cables should be properly supported to prevent chafing and rubbing, beating under vibration, axial movement, mechanical strain and an possible physical interference with other equipment.

(c) When leaving or entering pressurized areas, cables should be effectively sealed against air leakage. It is preferable that the cables do not have to be cut to pass through the pressure wall.

(d) It has been found that some cables require more mechanical protection than others. This additional protection has been provided by using ducting, conduit or sleeving; which, however, adds considerably to the weight and cost of electrical installation.

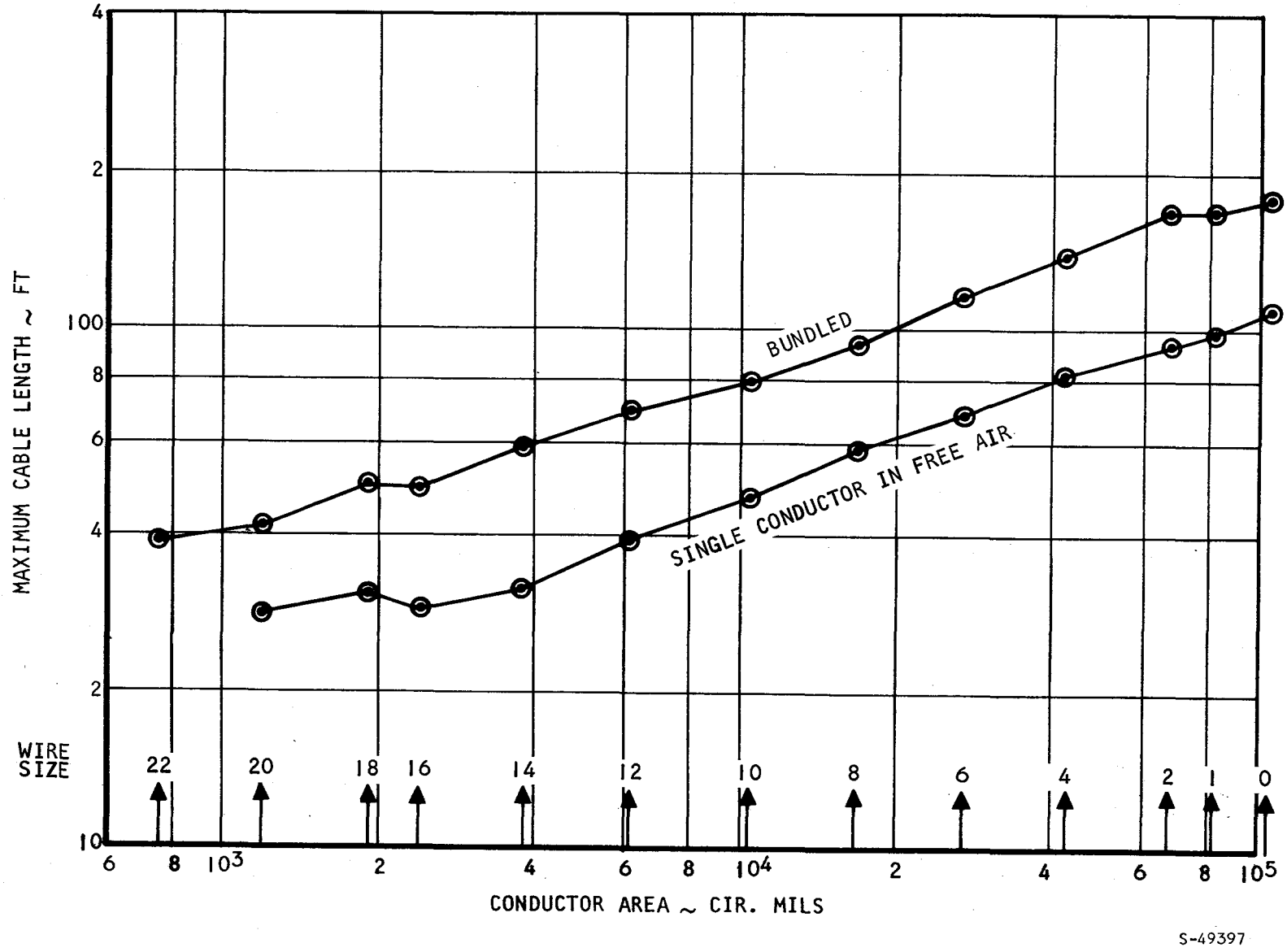


Figure 2-4. Conductor Cross-Sectional Area vs Maximum Cable Length for an Allowable Voltage Drop of 4 Volts

2.3 CIRCUIT BREAKERS

2.3.1 Introduction

Two basic types of circuit breakers are used in aircraft: thermal breakers and magnetic breakers. Thermal tripping is effected by a bi-metallic strip heated either directly by the current it carries or indirectly by a heating element in series with the load. The thermal circuit breaker is affected by the ambient temperature, although this is not so much of a disadvantage as it first appears because some kind of temperature compensation can be provided if necessary. One of the advantages of this type of circuit breaker is that it has an inherent time delay. Magnetic tripping is actuated by a solenoid which is sensitive to excessive current or voltage. This type of circuit breaker is unaffected by the ambient temperature; it has almost instantaneous operation and hence any required time delay has to be achieved by using a mechanical device such as a dash-pot, air-brake, etc. The magnetic type breaker, however, is scarcely used because of its sensitivity to shock and vibration, which results in inconsistent trip characteristics.

In selecting a circuit breaker for a load circuit, several factors must be considered before a proper rating can be determined to match the capacity of the wire. These factors are:

(a) The steady state conditions of the load versus the steady state capacity of the wire

(b) Start up and transient overloads together with their durations

(c) The ambient operating temperature and the means of heat dissipation

(d) The environmental conditions under which the circuit breaker must operate.

(e) The possible voltage and frequency variation of the circuit

(f) The allowable voltage drop

(g) The required trip-time delay

(h) Coordination with other breakers connected in series on the same line

For most applications in order to avoid nuisance tripping the circuit breakers are required to pass the start up and transient overloads which are of the short-duration type, as well as a continuous overload slightly higher than normal. Consequently their actual trip characteristics are higher than their rated current capacity. The thermal circuit breakers are designed with both a maximum and a minimum ultimate trip. The maximum ultimate trip is the percent overload under which the breaker must trip within one hour, while the minimum ultimate trip is the percent overload under which the same breaker must remain closed for at least one hour before tripping. Although 138 and 115 percent overloads are the typical maximum and minimum ultimate trip respectively, maximum limits as high as 180 percent and minimum limits as low as 100 percent may be required for specialized applications.

To ensure that the circuit breaker can provide optimum protection under faulty conditions while preventing nuisance tripping under transient conditions the circuit breaker must be matched to both the short duration overload characteristics and the capacity of the wire. Figure 2-5 shows approximately the characteristics of multi-conductor wire based on wires in bundles. Since each wire has a maximum allowable operating temperature, the current carring capacity of the wire is a function of time.

2.3.2 Temperature Effects

Since tripping of the thermal circuit breaker is initiated by the temperature rise in the heating element, any environmental conditions which affects the temperature rise would affect the performance of the circuit breaker. These include the externally induced heat, the internal heat dissipation, etc.

When the circuit breaker is subjected to extremes of temperature environment, serious derating of the circuit occures. In general, degradation by temperature can occur from the following three basic sources.

(a) Temperature variation

If the circuit breaker is installed in such an environment where the temperature varies in a broad range, the trip capacity of the breaker will be seriously affected. A circuit breaker rated at 115/138 percent at a nominal temperature of 25°C would have a significantly higher trip capacity at low temperature range and a very low trip capacity (in some cases, even below the normal rating) at high temperature. The amount of change in trip capacity is directly related to the temperature deviation When the trip capacity is raised due to low ambient temperature, the circuit breaker would not provide proper protection to both the equipment and the wire. On the other hand, when the trip capacity is lowered due to high ambient temperature, premature tripping would occur.

(b) Preloading

When the circuit breaker carries a load at or near full load for a relatively long period, the circuit breaker is derated due to the sustained heating. If the breaker is subjected to a short duration transient load, premature tripping may occur. Figure 2-6 illustrates the effect of preloading on trip times of a bi-metallic thermal breaker. The dotted line shows the load vs trip time with no preload, while the solid curve is for 100 percent preload. Since the effect is considerably linear, tripping characteristic curves with partial preload can be determined by interpolation between the curves.

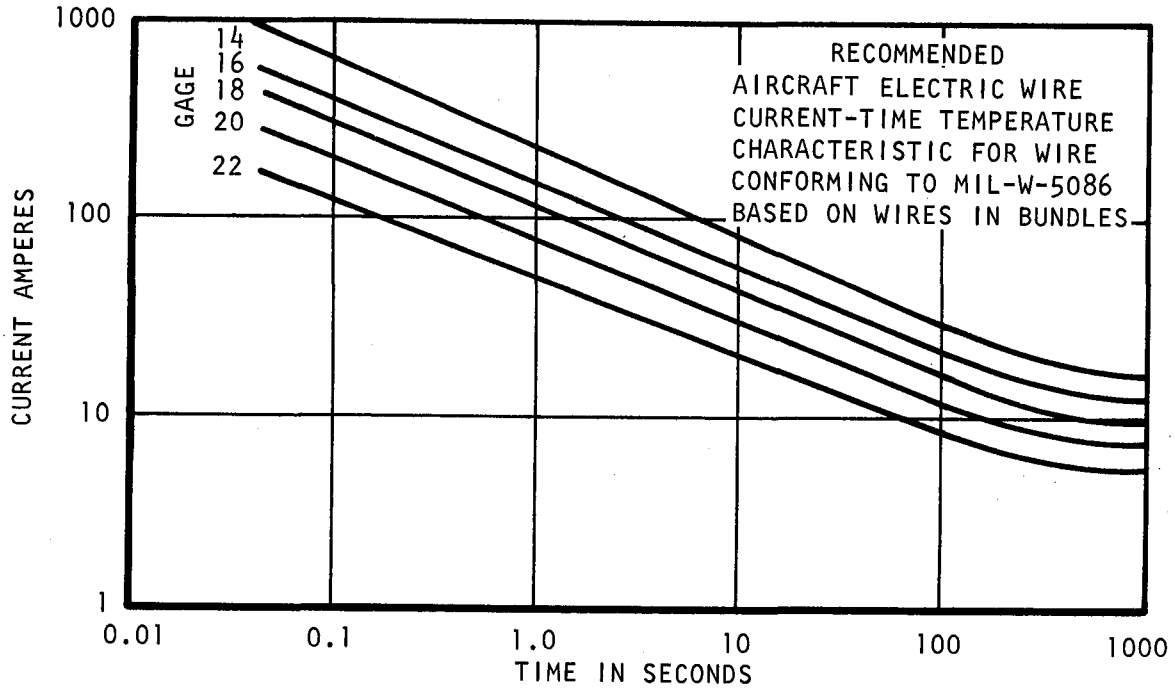


Figure 2-5. Current Amperes vs Time

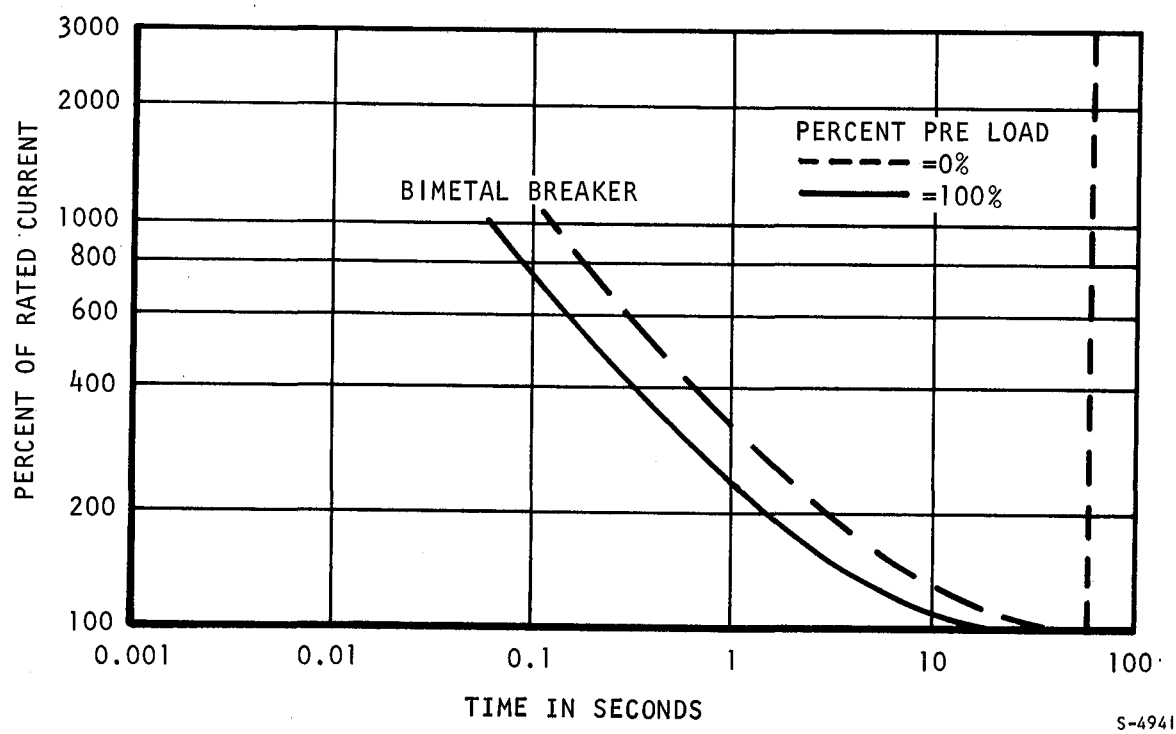


Figure 2-6. Percent of Rate Current vs Time

It should be mentioned that magnetic circuit breakers are also subjected to degradation due to preloading, although they are not actuated by temperature change. The effect, however, is nonlinear and more change in trip time is observed as full preload is approached.

(c) Heat dissipation

The heat dissipation capability of the breaker is closely related to the size and material of the case, wire terminals, mounting, and the surrounding atmosphere. These factors which affect the temperature rise are taken into consideration by the designer.

When a great number of breakers are mounted close together on a single panel, adequate heat transfer may be hampered and overheating results. As a consequence the circuit breakers have to be derated. Proper ventilation, therefore, should be provided to improve heat dissipation especially when the air surrounding the breaker is rather confined.

Altitude has a direct effect on the heat dissipation capability of the circuit breaker, simply because at high altitude, the air density decreases, and therefore the ability of transferring heat by convection is reduced. A circuit breaker having a maximum ultimate trip of 138 percent at sea level would be derated to approximately 130 percent at 50,000 ft and 120 percent at 100,000 ft.

It should be noted that in some applications, the wiring of a certain load may pass through a heat source such as the engine zone, the aircraft heating system, etc. while the circuit breaker is not exposed to it. Under this condition, the wire is subjected to a preload, and a slight overload might cause the wire to fail before the breaker is actuated. The maximum ultimate trip of the breaker, therefore, must be lowered in order to protect the wire properly; it may be necessary to re-route the wire if the breaker trip limit cannot be reduced due to the peculiar load characteristics.

2.3.3 Interrupting Capacity

Interrupting capacity of the breaker is the maximum current which the breaker is able to interrupt. This interrupting capacity is a function of many factors, among which the most significant ones are the size and geometry of the breaker, the line voltage and the current rating. In general high interrupting capacity is desirable; this requirement, however, is incompatible with the requirements of small size, light weight, etc. Low-rating breakers have smaller interrupting capacity than high-rating breakers because of the smaller physical size.

It had been a strict rule, in the design of aircraft electrical circuit protection, that a circuit breaker should not be used in a circuit in which the short circuit current exceeds the interrupting capacity of the breaker.

This is to prevent the breaker from being destroyed by the short circuit current and thereby leaving the circuit either unprotected or disconnected. Recently, however, a modification of this design philosophy has come under consideration.

There are many nonessential loads which are not critical to flight safety of the aircraft. If a circuit breaker having a lower interrupting capacity than the short circuit current is utilized in a nonessential circuit; and if the breaker can guarantee an open circuit after the destruction of the breaker, the most serious situation that can be encountered is the loss of the circuit after a destructive fault. Experience, however, has revealed that this type of fault during a flight is extremely rare. Therefore, if this can be tolerated by some of the nonessential circuits, many advantages can be realized. The most obvious ones would be the savings in volume and weight since smaller breakers are selected.

Nowadays, circuit breakers which guarantee an open circuit following the destruction of the breaker are available. This type of breaker is referred to as "fail safe."

2.3.4 Voltage Drop

Some utilization equipments are very sensitive to the terminal voltage. In calculating the total voltage drop of the circuit, therefore, the circuit breaker in series with the load must be taken into consideration if the expected performance of the load equipment is to be realized. Bi-metallic thermal breakers have low voltage drops (0.5 v) and usually do not create any problems. However, the voltage drop should not be ignored in making a selection.

Thermal breakers with rating below 5 amperes are usually of the indirectly-heated type; i.e., the trip device is heated by a wire-wound element. This type design is not desirable since it introduces an excessive voltage drop in the circuit.

2.3.5 Types of Thermal Breaker

(a) Switch-Type Breaker--These breakers are available in a variety of sizes and actions. Some of them are miniaturized and a single unit can frequently replace the toggle switch and the fuse.

(b) Manual Reset Breaker--This type of breaker is most widely used in aircraft. The breaker has one button on its face. Manual reset is done by pushing the button which will come out when it is tripped on overload. Most models can be tripped manually by pulling out the button.

The current rating of these breakers for aircraft use range from less than one amp to up to 50 amp for the ac system. A typical miniaturized breaker in this range weighs about 1.5 oz.

For general safety all breakers used on aircraft must be of the trip-free type. That means that the breaker is so designed that it will not maintain the circuit closed when carrying overload current regardless of the restraint placed on the actuator.

There are multi-pole circuit breakers constructed by combining several single-pole circuit breakers. It is so designed that whenever one pole is tripped the remaining pole will trip simultaneously. A three-pole breaker can be used for three-phase protection while a two-pole breaker can be used for a two-wire circuit or can be used as an interlock between two circuits serving the same equipment.

3.2.6 Tripping Time

Curves showing trip time-versus-load are usually provided for each circuit breaker to serve as a guide for application purposes. Figure 2-7 shows a typical trip time characteristic at 25°C. A band rather than curves displays manufacturer's or specification tolerances for a certain breaker rating. There are three regions in the diagram. In region 1, below the shaded area, the circuit breaker should not be tripped. In region 2, the shaded area, tripping may occur; in region 3, above the shaded area, the circuit breaker should definitely be tripped.

The temperature effect on trip time characteristics is shown in Figure 2-8. It is understood that temperature above 25°C tends to move the curve downwards while ambient temperature below 25°C tends to move the curve upwards. If the ambient temperature varies between -54°C to +71°C the shaded area of Figure 2-7 will be enlarged as shown in Figure 2-8. Temperature effects can be eliminated or reduced by employing a temperature compensating unit to control the thermal element.

2.4 RELAYS

Relays are used for various power and control functions throughout the aircraft. They come in ratings such as used for dry circuit applications and up to the power handling capabilities of over 100 amperes. Relays may be installed anywhere in the aircraft and may therefore be subjected to a wide varying range of environmental conditions. Environmental requirements include resistance to ambient temperature variations, pressure variations, shock, vibration and corrosion. Typical environmental conditions for a present day commercial jet aircraft are:

(a) Temperature range from -65°C to 125°C

(b) Pressure down to 1.3 in. Hg (corresponding to 70,000 ft altitude)

(c) Shock pulses of 15 to 50 g acceleration with minimum frequency of 2000 Hz near the engine

(d) 10 g acceleration with frequencies up to 2000 Hz remote from the engines.

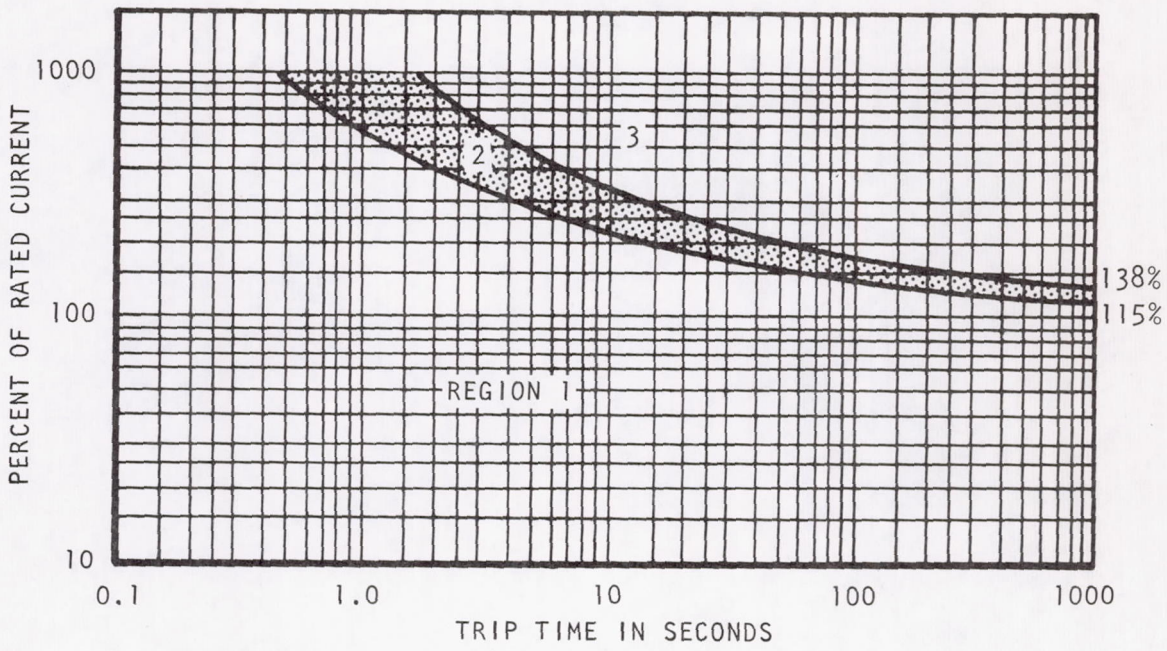


Figure 2-7. Tripping Time Characteristic of Circuit Breaker

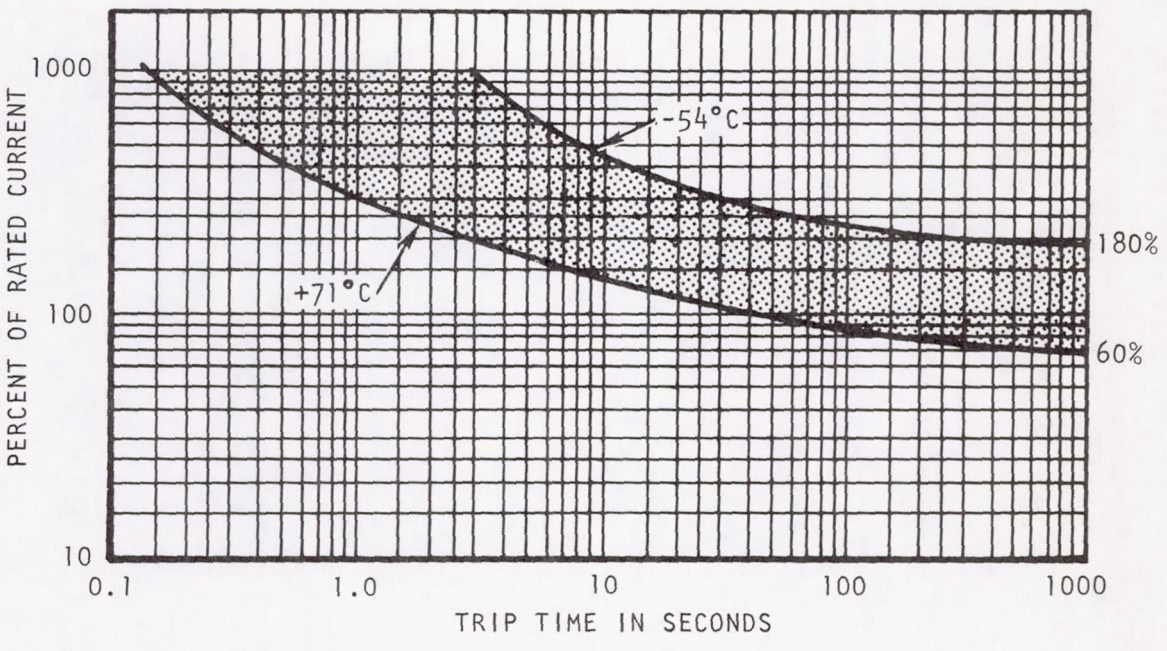


Figure 2-8. Temperature Effect on Tripping Time of Circuit Breaker

2.4.1 Relay Selection

In general aircraft relays should provide the following characteristics:

(a) Minimum weight consistent with reliable operation

(b) Sufficient closing and opening forces to ensure proper operation during acceleration or vibration

(c) Satisfactory operation in every respect between 75 and 125 percent of nominal system voltage

(d) No effect upon its rating with respect to change of position

(e) Nonwelding alloy contacts or hammer action to prevent the contacts from welding for heavy-current relays

(f) Satisfactory life for the contacts at the operating environment

(g) Low contact resistance in low-current contacts

(h) Resistance to shock, explosive vapor, corrosion, radiation, etc.

2.4.2 Relay Classification

The various relay designs, constructions and applications are so numerous that many classifications are needed to categorize them. One of the classifications is according to their basic form or construction, as follows:

(a) Hinged armature type

(b) Solenoid-operated type

(c) Rotary action

(d) Polarized

Each group can be divided into subgroups according to their current rating, supply voltage, order of making and breaking, whether normally open or closed, sealed or unsealed, etc.

2.4.2.1 Hinged Armature Type

This name applies to relays in which the armature is located on its pivots adjacent to the magnetic system. The relay contacts are either mounted on the armature or are actuated by the movement of the armature or an extension of it. A leaf or a spring is usually employed to hold the contacts in the "off" position when it is not energized.

2.4.2.2 Solenoid-Operated Type

In this type relay the armature is in the form of a core within a wound coil or solenoid. When the relay is energized, the core is driven to the center of the coil where the attraction is zero.

Solenoid actuation of relay contacts is used where relatively large movement of the contacts and considerable contact pressure are required. Similar to the hinged armature type, solenoid operated relays are sensitive to acceleration forces. Although the armature can be of a balanced type to reduce the acceleration effect, the additional weight and size necessary for proper balancing is rarely justified.

2.4.2.3 Rotary-Action Relays

The construction of the rotary-action relay is similar to that of a motor. The armature is supported on a rotating shaft and is enclosed within a magnetic system. The armature may be of the wound coil type while the field is a permanent magnet or vice versa; or both armature and field may be wound. When the relay is energized, the shaft is driven by a circumferential pull against the torsion of the return spring. The contacts are usually operated by a cam or lever on the shaft.

This type of relay has inherent stability against acceleration forces. It is manufactured in a wide variety of sizes and shapes and has been used in mobile ground equipment, aircraft, and missiles.

2.4.2.4 Polarized Relay

Polarized relays are those in which the direction of the movement of the armature is dependent on the direction of the current flow in the field coil. The relay may be designed such that the contacts are closed only when the current flows in one direction. Movement in the other direction is limited by a mechanical stop, or it can be designed in such a way that different sets of contacts are closed according to the direction of the current flow.

Polarized relays are suitable only for dc operation. They are very useful in paralleled dc generator systems. It should be noted that the armature of the polarized relays should be centered in the field very accurately since it is the most important factor that affects the relay performance.

2.4.3 Contact Materials

For various applications, the voltage across the contacts, the operating frequency, the type of load and environmental conditions vary within broad ranges. In order that the relay can operate successfully to satisfy a certain requirement the choice of proper contact material becomes very important, particularly for relays designed to switch low current and low voltage circuits (dry circuits). The most commonly used contact materials are silver, tungsten, platinum and silver alloys for which the advantages and disadvantages are listed in the following table.

TABLE 2-4

MATERIALS AND APPLICATIONS, ADVANTAGES AND DISADVANTAGES

Material	Application	Advantages	Disadvantages
Silver	General application	Relatively inexpensive; high thermal and electrical conductivity; requires low contact pressure	Produces silver sulphide coating (the resistance becomes important at low voltages)
Rhodium Plated Silver	Similar to that of silver (if arcing does not break down the plated film)	Same as silver; prevents formation of sulphide film	Expensive
Platinum	Low current (less than 0.5 amp) low voltage; super sensitive relays	Requires very low contact pressure; very good resistance to corrosion	Very expensive
Tungsten	Low current or nonsensitive low voltage circuits	High melting point; relatively low cost; not affected by arcing, inductive currents, or corrosion	Requires high contact pressure
Sintered Nickel-Silver	Heavy inductive current circuits	Requires moderate contact pressure; good electrical conductivity; less sensitive to sulphur	Contact resistance may increase with use

In addition to the materials listed in Table 2-4, other materials are being evaluated for relay contact applications. These include palladium, gold, palladium silver, iridium-platinum and sintered silver graphite.

2.4.4 Contact Pressures

For certain contact materials, the contact resistance depends primarily on the contact pressure; and the relation is consistent over many cycles. Figure 2-9 shows a typical curve relating contact resistance and contact pressure for switch contacts with copper plane surfaces. It is seen that the

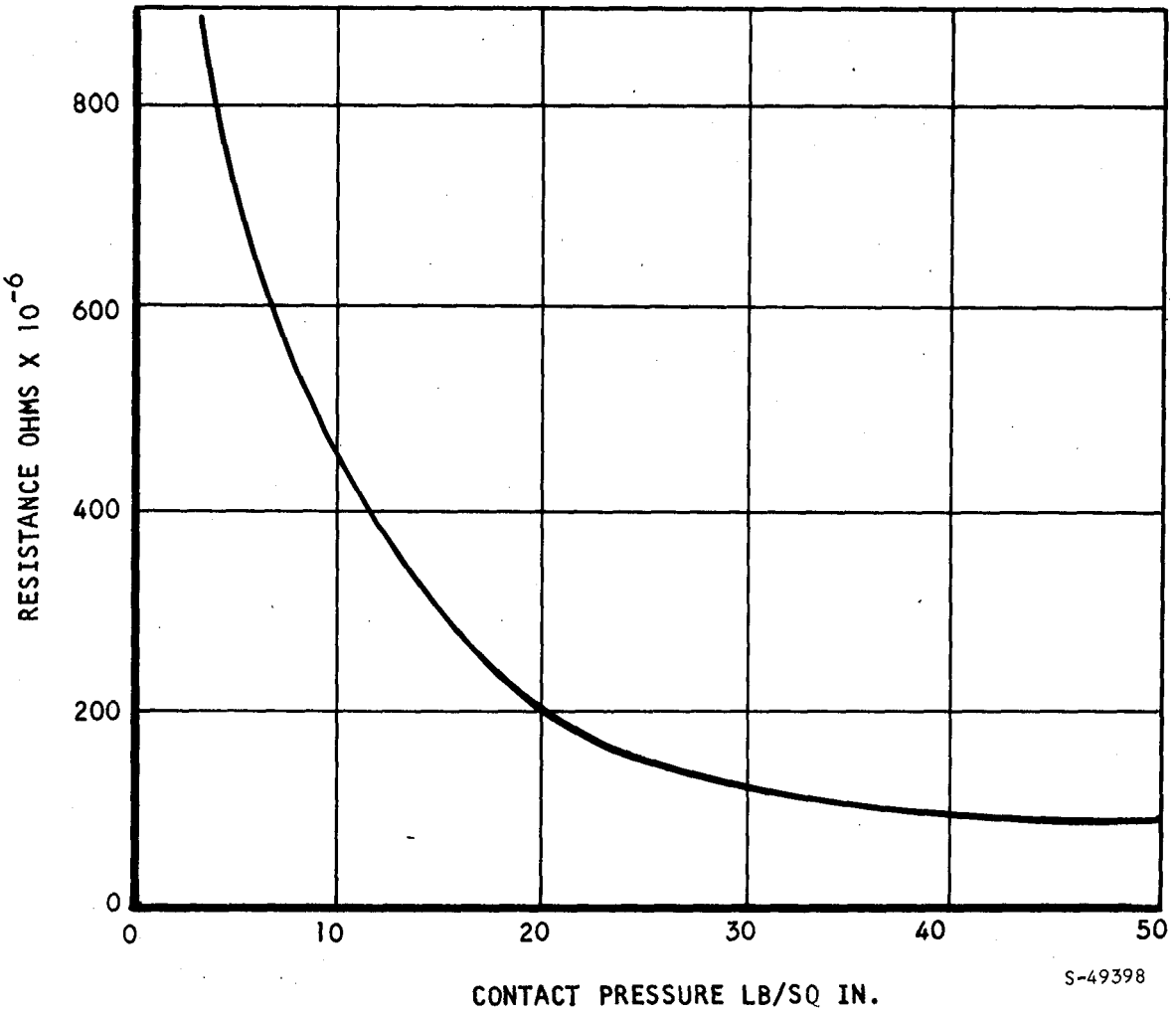


Figure 2-9. Contact Resistance vs Contact Pressure for Switch Contacts with Copper Plane Surfaces

contact resistance is inversely proportional to the contact pressure and when the pressure drops below a certain value, the contact resistance increases rapidly. Therefore, in order that the contacts give a fixed contact resistance, a certain minimum pressure is required. This requirement is one of the most important design considerations, since it directly affects relay performance.

2.5 CONNECTORS

Service experience has shown connectors to be one of the more vulnerable components to cause malfunction of the electrical system. Numerous incidents of open circuits, intermittent contacts and high circuit resistance due to corrosion are reported repeatedly. The need for reliability improvements in circuit interconnections is more eminent than for any other component in the system. The ideal case, as far as reliability is concerned, is to have continuous conductors throughout the entire circuit without using connectors at all. However this usually is not the case. Interconnections often are added to facilitate assembly, maintenance, future modification, inspection and trouble shooting.

At circuit design levels the principle considerations of interconnection are related to the selection, rating, and derating of individual conductors. (See Sec. 2.2 Aircraft Cables.) As the design of electrical/electronic equipment nears the stage of packaging and installation all the individual conductors begin to be placed in more or less orderly channels or multi-conductor cables or other mechanical combining modes. More often and frequently at great cost of time and money, the stage at which this channeling is considered is far too late. Mechanical restrictions have already been built into the packaging which minimize the number and size of cables and bundles that may be incorporated. Worse yet, it is often found that the packaged equipment is perfect in all respects except that no room has been left for adequate wiring. The first phase in the design of any cabling or multiconductor interconnection mode should be at the block diagram level. Here some planning must start about ultimate requirements. As the equipment progresses through design and other refining steps to mock-up and manufacture, the interconnection system must be developed along with it.

2.5.1 Interconnection Requirements

The following list contains a summary of various items a circuit designer should consider when specifying interconnections. The list should be thought of as being flexible--items do not appear in order of importance, nor do all of them apply at every stage of design.

Classes

- Power
- Signal
- RF
- Shielding

Thermocouple

Estimated resistance

Voltage drop-for long lines

Environment free (classify each requirement)

Non-environment free

High voltage

Hermetic

Groups

In conduit

Cabled

Harnesses

Single wire

Growth Factor

Utilization Factor

Number of Connector Contacts

Types of Contacts

Open entry

Closed entry

Contact material needed for conductivity rating and environmental considerations

Contact Size

All wire gages

Connector Sizes

Microminiature

Subminiature

Miniature

Standard

Heavy duty

Current Rating

Voltage Rating

Shells

Method of Installation

Pendant, both halves

Pendant, one half

Rack and panel rectangular

Rack and panel cylindrical

Chassis mount board

Chassis mount tape

Standard pendant, rectangular

Standard pendant, cylindrical

Couplings

Quick disconnect, no lock

Quick disconnect, lock

Push-pull

Pull-pull

Bayonet

Thread assist

Guide pin and lead screw

Shell Finishes

Insert Materials

As per Environment

Polarization

2.5.2 Contact Rating

The electrical rating of contacts is a very important consideration in the selection of a connector during the circuit design stage. Contacts are designed to carry about the same amount of current as the corresponding wire size. And the wire entry portion is normally large enough to accept this wire size. The most frequently used contacts are the cylindrical pin- and socket-type made of copper alloy. The superior conductivity of copper permits smaller diameter pins and sockets than other metals. Table 2-5 lists various metals used for making contacts, however many of them are impractical for all but a few specialized uses because of their high relative resistance.

TABLE 2-5

CONNECTOR MATERIALS

Materials	Relative Resistance	Temp. Coeff. of Resistivity	Specific Gravity	Coefficient of Thermal Cond.	Coefficient of Thermal Exp. ($\times 10^{-6}$)	Melting Point, Deg C
Aluminum	1.64	0.0039	2.7	2.03	28.7	660
Brass	3.9	0.002	8.47	1.2	20.2	920
Cadmium	4.4	0.0038	8.64	0.92	31.6	321
Constantan	28.45	0.0002	8.9	0.218	14.8	1210
Copper, Annealed	1.00	0.00393	8.89	3.88	16.1	1083
Copper, Hard Drawn	1.03	0.00382	8.94	---	---	1083
Copper, Beryllium	2.8-3.4	---	8.24	---	---	---
Germ, Silver	16.9	0.00027	8.7	0.32	18.4	1110
Gold	1.416		19.32	0.296	14.3	1063
Iron (pure)	5.6	0.0052-0.0062	7.86	0.67	12.1	1535
Magnesium	2.67	0.004	1.74	1.58	29.8	651
Monel	27.8	0.002	8.8	0.25	16.3	1300-1350
Nickel	5.05	0.0047	8.9	0.6	15.5	1455
Nickel Silver	16.0	0.00026	6.72	0.33		1110

(Table continues to next page)

TABLE 2-5 (Continued)

Materials	Relative Resistance	Temp. Coeff. of Resistivity	Specific Gravity	Coefficient of Thermal Cond.	Coefficient of Thermal Exp. ($\times 10^{-6}$)	Melting Point, Deg C
Phosphor Bronze	5.45	0.003	8.9	0.82	16.8	1050
Platinum	6.16	0.003	21.4	0.695	9.0	1774
Rhodium	2.6					
Silver	0.95	0.0038	10.5	4.19	18.8	960.5
Steel, SAE 1045	7.6-12.7		7.8	0.59	15.0	1480
Steel, Stainless	52.8		7.9	0.163	19.1	1410
Tin	6.7	0.0042	7.3	0.64	26.9	231.9
Zinc	3.4	0.0037	7.14	1.12	26.3	419

Cylindrical contacts, the most common, generally will not accept a wire larger than the gage size of the contact. However special types of contacts are available into which larger wires can be attached. These special connectors are useful where the resistance of smaller wire would be too great for the proper performance of the circuit, or where the smaller wire would not be physically able to take the abuse expected. Individual contacts and wires are capable of carrying considerable amounts of current when operating in free air. However this is an ideal situation. Most circuit designs require wires to be placed in close proximity to each other or often they are grouped together in a harness. When assembled tightly together they interfere with each other's ability to reject generated heat. Under these conditions their current carrying capacity is derated. Derating tables are available for wires in bundles versus single wires. (See Section 2.2 Aircraft Cables.)

2.5.3 Connector Criteria Checklist

The following list of connector requirements was formulated by component manufacturer and users throughout the aircraft industry. The list of criteria was then established and it should serve to obtain design goals and performance conditions for developing miniature-environmental connectors.

Criteria No. 1: Quick disconnect?

Criteria No. 2: Positive lock without safety wiring?

Criteria No. 3: Connector seal before lock?

Criteria No. 4: Visually inspectable for correct assembly and installation?

Criteria No. 5: Moisture seal?

Criteria No. 6: Vibration dampener?

Criteria No. 7: Corrosion resistant?

Criteria No. 8: Operation to 250^{0}F?

Criteria No. 9: Unaffected by altitude pressure variations?

Criteria No. 10: Wire comb?

Criteria No. 11: Wire support?

Criteria No. 12: Continuous dielectric separation--no voids?

Criteria No. 13: Closed entry contacts to accommodate an wire 22 through 18?

Criteria No. 14: No wet process involved?

Criteria No. 15: Good serviceability?

Criteria No. 16: Elimination of soldered connections?

Criteria No. 17: All of the components necessary for complete assembly delivered in a clear, sealed, dated package?

Criteria No. 18: Non-conducting exterior surface?

Criteria No. 19: Operating force always in direction of plug travel?

Criteria No. 20: Coupling and/or contact seal before electrical contact?

Criteria No. 21: Multiple insert "clocking"?

3. CONTROL AND PROTECTION

3.1 SYSTEM DESCRIPTION

3.1.1 Generating System

This section applies to a typical present-day electrical generating system, which is of the constant-speed, constant frequency (CSCF) type, and may consist of either three or four generating channels operating in parallel. Figure 3-1 illustrates a basic ac system in block diagram form. Each generator is driven by a hydraulic constant speed drive (CSD), which in turn receives its power from the accessory drive pad of one of the main jet engines. Driving torque to the generator is supplied at a constant speed of 6000, 8000, or 12,000 rpm, providing a constant frequency generator output of 400 Hz, with an 8-pole, 6-pole, or 4-pole generator, respectively.

CSD input speed variation is typically 2 to 1. The three-phase (four-wire grounded neutral) power output from the generator is carried over a typical 60- to 120-ft transmission line to the main power shield, located near the flight engineer's station in the control cabin. The power shield is the collection point for the feeders of all generating channels. It contains the individual generator circuit breaker's (GCB), the bus tie breakers (BTB), the generator load buses, and the synchronizing bus. The power from the generator load buses is distributed via thermal breakers to the various local distribution buses, and from there to the various loads throughout the airplane. Also located in the power shield and connected to the synchronizing bus is the external power contactor (EPC). During ground servicing operation, power can be supplied to the synchronizing bus, and hence to the load buses from an auxiliary ground power unit. Some electrical systems incorporate an airplane-installed auxiliary power unit (APU), which might also be electrically connected to the synchronizing bus (not shown in Figure 3-1).

The main ac power shield serves as the point of regulation for each generating system where nominal system voltage of 115/200 v is maintained. Design criteria for the performance, control, and protected of the system shall include the following:

(a) Continuous supply to load buses (During normal and most abnormal operation of the system, power shall be maintained to the load buses.)

(b) No single fault shall cause loss of power to an unfaulted load bus

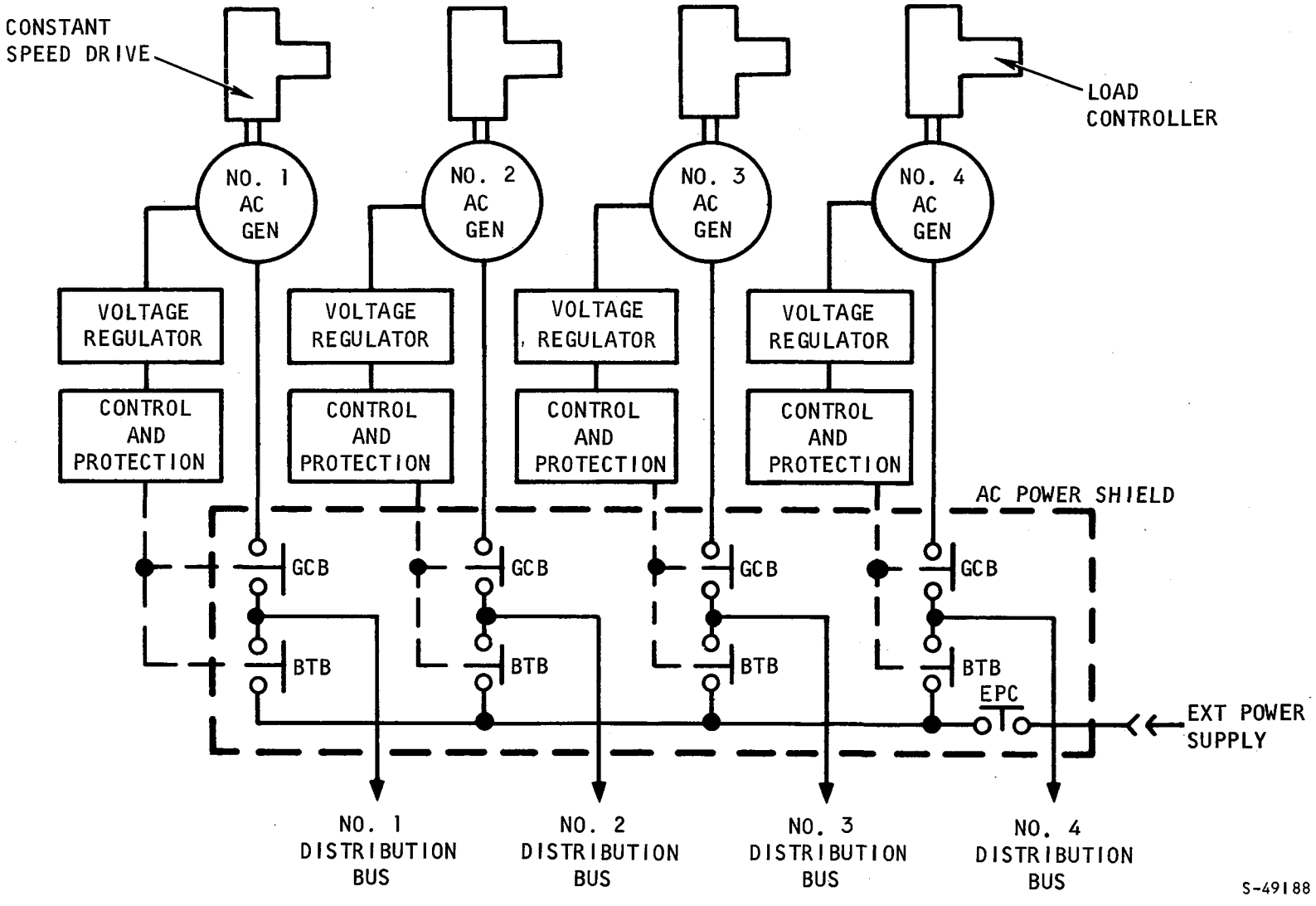


Figure 3-1. Typical Four Generator Bus System

(c) Maintain system capacity (During normal operation, load sharing circuits shall provide maximum system capacity and shall remove only that faulted generating channel which might endanger the overall system or load equipment.)

(d) Maximum protection with minimum components.

In order to provide the necessary degree of system control and to achieve system reliability, each power source, mechanical and electrical, requires controlling, regulating, and protective equipment. Figure 3-2 shows a block diagram of a single generating system with control and protection tie points.

3.1.2 Constant Speed Drive

The alternator drive is a governed hydraulic transmission which transforms the varying input speed from the aircraft engine to constant speed for driving the alternator. Hydromechanical constant speed drives are comprised of basic hydraulic components which can be arranged in various combinations to meet specific performance requirements. The components are generally a variable hydraulic displacement pump and a fixed hydraulic displacement motor. The pumping unit volumetric displacement is varied in relation to input speed to provide an oil flow to the fixed displacement motor unit which will assure a constant output speed. An output speed signal, converted to a hydraulic signal by the governor actuates a servo piston which is connected to a pump wobbler plate, thereby varying the displacement of the pump.

The drive provides a constant output frequency of 400 ±1 Hz for steady state conditions of load and input speed variations (typically 4000 to 8000 rpm). With rapid acceleration or deceleration of input speed or normal changes in load on the aircraft electrical system, transient output frequency is held to 400 ±4 Hz. Under steady state conditions of load and speed when two or more alternators driven by constant speed drives are operated in parallel, the drive output frequency is held to 400 ±1 Hz, and real load division is held within 2 kw (typical for alternators rated 40 kva each). With rapid acceleration and deceleration of input speed, or sudden changes in load on the aircraft electrical system, real load division is held within 12 kw total.

3.1.3 Alternator

The alternator with the typical ratings of 30 kva, 40 kva, and, for more recent application, 60 kva, is generally of the brushless rotating rectifier-type with salient pole construction, and is either air or oil cooled. On the newer type systems the ac main generator incorporates a small pilot permanent magnet generator (PMG) for system self-sufficiency. The PMG output (typically 500 va) serves for main generator excitation and for system control. Alternator speeds are 6000 rpm, 8000 rpm, and, for recent applications, 12,000 rpm.

3.2 SYSTEM CONTROL

When operated singly, the two main controlling functions for each system are frequency regulation (by trimming the output speed of the constant speed

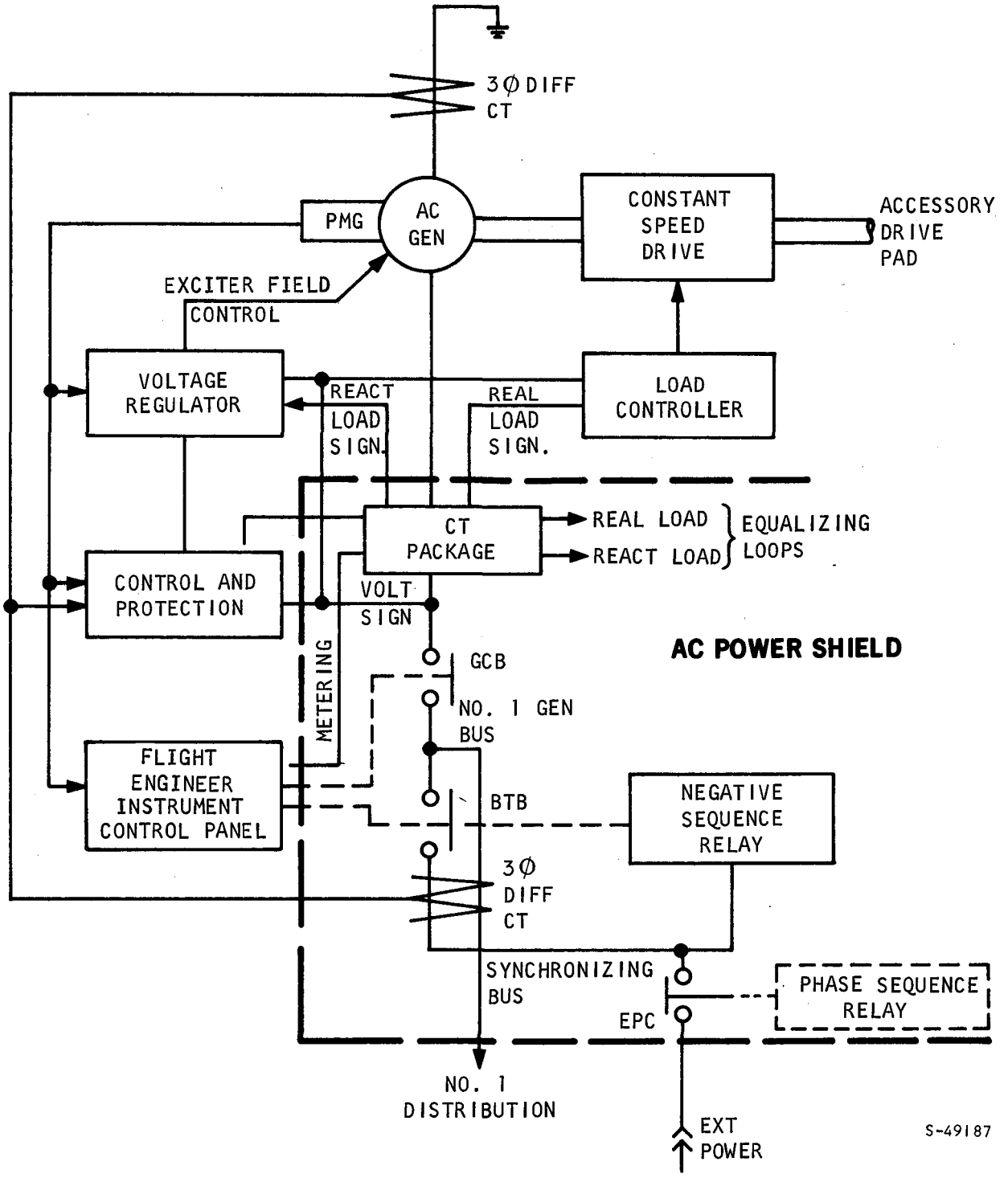


Figure 3-2. Generator Control and Protection Block Diagram

drive) and voltage regulation (by varying generator excitation). Thest two control functions change to real load division control and reactive load division control respectively when the systems operate in parallel.

3.2.1 Frequency Control

A block diagram of the control system is shown in Figure 3-3.

Regardless of the particular trim control method that is used, the basic building blocks of the electrical trim will always include the following elements:

(a) Frequency reference

(b) Converter or comparator to obtain a voltage or current analog of frequency error

(c) Amplifier

(d) Transducer to modify the speed governor setting

The frequency reference is the standard to which the generator frequency is continuously compared. A voltage or current analog of the frequency error is obtained from the comparator. This frequency error signal is then amplified and applied to the magnetic trim governor of the transmission. Current in the trim coil functions as vernier control of the governor. In order to distribute the real load equally between generators, the magnitude of the deviation of each generator load from the average load and the direction of the deviation is sensed by the load division loop. The loop is comprised of a series string of parallel combinations of current transformer secondaries and burden resistors (see Figure 3-4). Each current transformer (CT) is on the same phase of each generator.

An example will best illustrate the operation of the load division loop. Suppose that a voltage E1 is produced in CT1 as a result of the induced current and that E2 is induced in CT2, E3 in CT3, and E4 in CT4. Consider E1, E2, E3, and E4 instantaneous voltages, the induced voltage at one particular instant of time. It is apparent that E1 will appear across R1. E1 is shown as a voltage drop across R1 with an arbitrarily assigned polarity. Figure 3-4 (b) shows the burden resistors with the current transformers removed to show more clearly how the voltage E1 is distributed among the resistors. Since E1 is dropped across R1, it can be seen that E1 will also be dropped across the series combination of R2, R3, and R4. Since all of these resistors have the same value, one-third of E1 will be dropped across each of them.

Note that the polarity of the voltage dropped by each resistor is opposite to the polarity of E2 across R2, E3 across R3, and E4 across R4. In a similar manner, E2 will be dropped across R2, while one-third of E2 will appear across each of the other resistors, but of the opposite polarity with respect to E2. Likewise, E3 and E4 will be distributed among the resistors. As a result of voltages E1, E2, E3, and E4 will be distributed among the resistors. As a

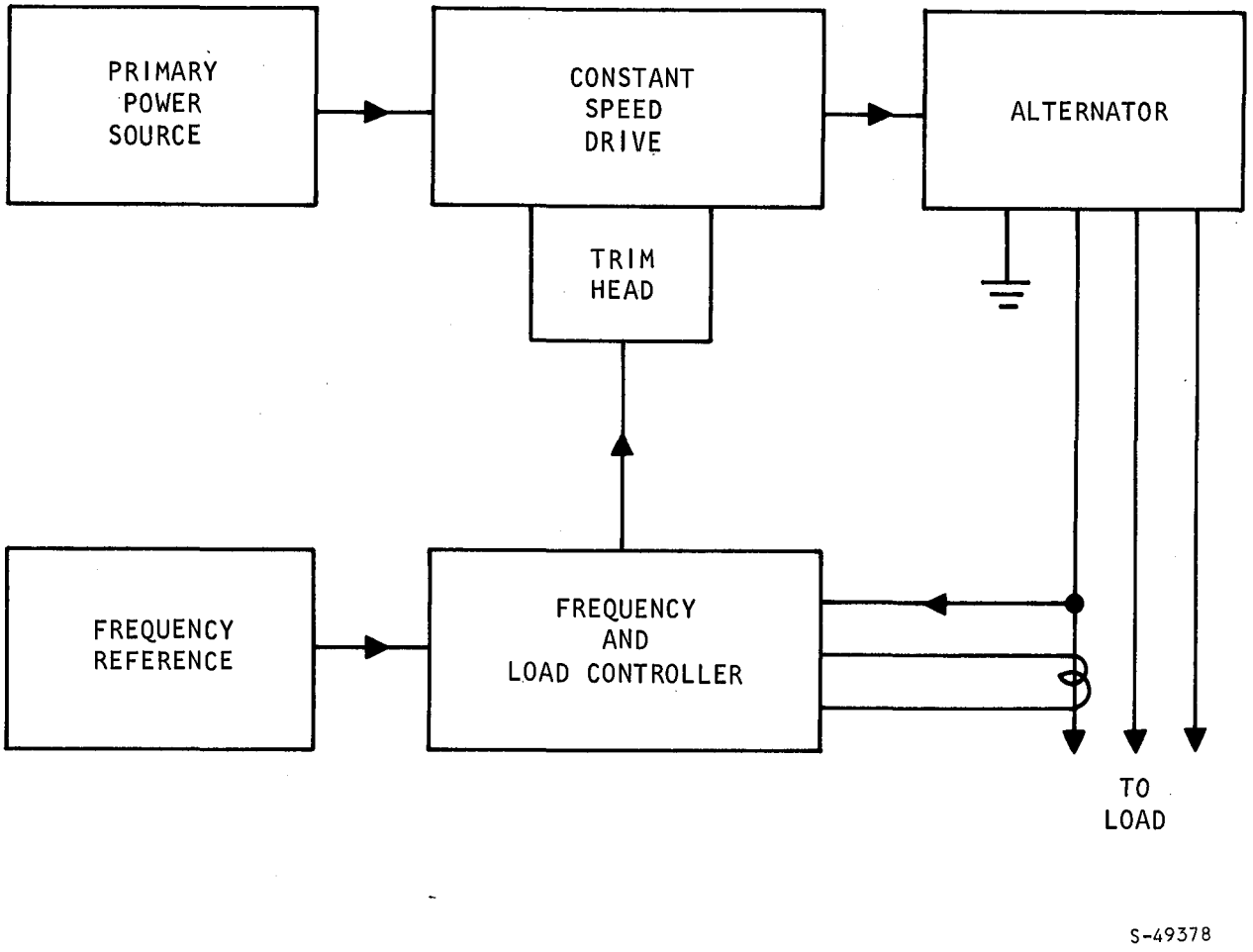
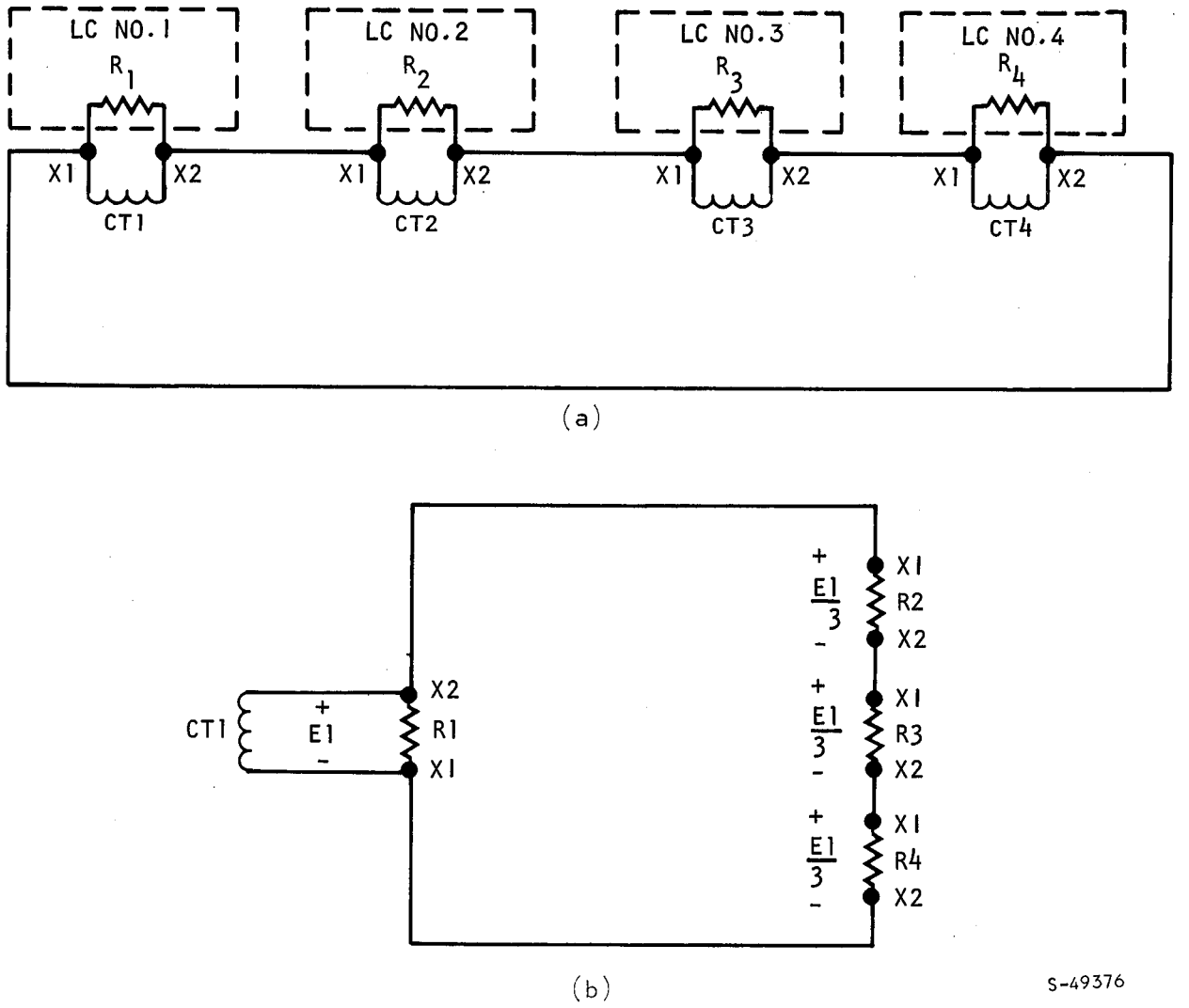


Figure 3-3. Frequency and Real Load Sharing Control

LOAD DIVISION LOOP AND VOLTAGE DISTRIBUTION



LC = LOAD CONTROLLER

Figure 3-4. Load Division Loop and Voltage Distribution

result of voltages E1, E2, E3, and E4 being induced in the load division loop, each resistor drops four separate voltages. On R1 the four voltages are E1, one-third E2, one-third E3, and one-third E4 with E1 opposing the other three voltages. The net voltage dropped across R1 will be the algebraic sum of these four voltages. The net voltage dropped across R2 will be E2, one-third E1, one-third E3, and one-third E4, and so on for R3 and R4. The net voltage across each burden resistor is the input to the load control circuit from the load division loop. Consider first the case where the load is balanced. The output of all four generators will be the same and E1, E2, E3, and E4 will be equal. The net voltage dropped across each resistor will be zero, and there will be no output from the load controllers.

Now consider the case where the load is unbalanced. An unbalanced load condition exists when two or more generators are carrying more or less than their share of the total load. The share that each generator should normally carry is equal to the total load divided by the number of generators. In the load division loop, this share is represented by the sum of E1, E2, E3, and E4 divided by four. In the unbalanced load condition the net voltage drop across R1 is still the algebraic sum of E1, one-third E2, one-third E3 and one-third E4.

Since the voltage induced in the CT are not necessarily equal, the resulting net voltage drop across R1 may have a certain magnitude and polarity. The magnitude indicates how much the output of Generator No. 1 has deviated from its required share of the load, and the polarity indicates whether the generator is carrying more or less than its share of the load. Since the current sensed by the current transformers in the same generator phases represents total or apparent current, reference to the in-phase component with the voltage has to be made within the load controller to obtain the correct real load division. Reference of the in-phase component of the current may be obtained by a phase sensitive demodulator as shown in Figure 3-5. The dc output of the modulator is proportional to the magnitude of the product E1 cos θ, where θ is the phase angle between current and voltage.

It is important that the primary of the modulation transformer is connected to the same phase from which the current is sampled to obtain real load division. If a generating channel does not operate in parallel with the other channels, its corresponding BTB or GCB is open. This closes a normally open contact on the breaker, thereby shortening the output of the current transformer in the noncontributing system.

3.2.2 Voltage Regulation

Except for weight and size, the functional requirements of the voltage regulator have not changed over the years. These requirements are

(a) Compensation for the high no-load to full-load regulation that is characteristic of unregulated ac generators with fixed excitation

(b) Compensation for generator voltage change with change in machine speed

PHASE SENSITIVE DEMODULATOR

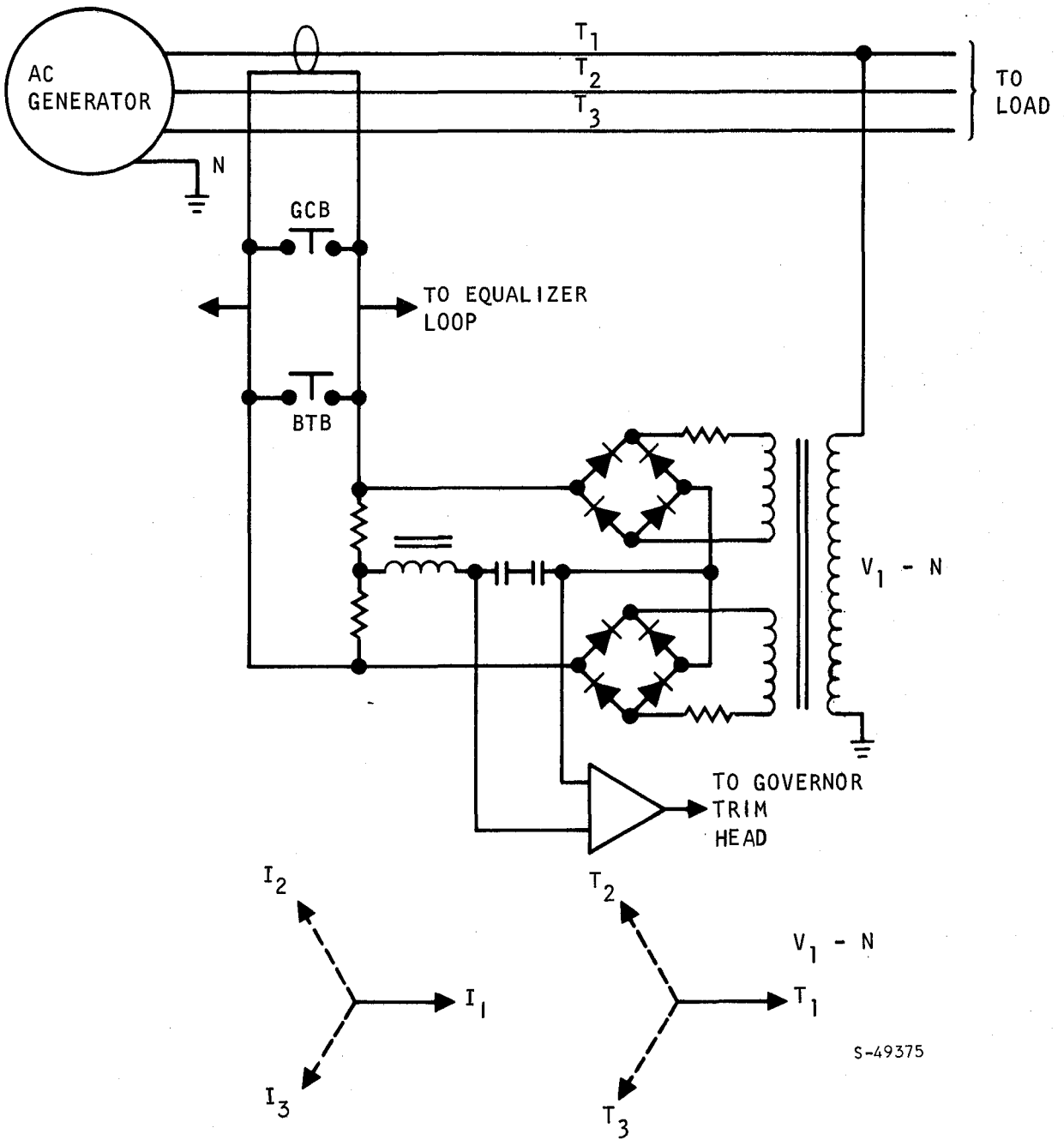


Figure 3-5. Phase Sensitive Demodulator

(c) Compensation for the tendency of generator voltage to drift as the field winding of the machine heats up (resistance change)

(d) Rapid return of generator voltage to normal after transient changes in load occur, i.e. sudden application and removal of one to two per unit load, clearing of faults, etc.

(e) In parallel systems, control of the division of reactive loads supplied by each generator

Although the functions remained the same, the means for accomplishing them changed, bringing along some significant improvements. At one time, the standard regulator designed for use with rotating exciter machines was the carbon pile regulator. Gradually, this type was replaced by the magnetic amplifier regulator. In later applications, a voltage regulator comprised of a transistor amplifier controlling a magnetic amplifier output stage was adopted. Present-day regulators, however, are completely transistorized, including the output stage.

Magnetic amplifier and transistorized regulators each have their own advantages and disadvantages. The advantages of magamp regulators are the ability to stand high temperature environment and unsusceptibility to electrical noises. The advantages of transistorized regulators are fast response and small size and weight. A more detailed comparison of these regulators will be performed later in the program.

Transistor regulators usually are switching type regulators and employ the pulse width modulation scheme. Figure 3-6 shows the block diagram of a typical regulator. The three-phase line to line voltage is sensed at the point of regulation and is stepped down by a three-phase transformer in the regulator. After half wave rectification the ripple voltage is subjected to a wave shaping network to produce a saw tooth waveform. This waveform is compared to a fixed dc reference voltage. The rising or falling ramp of the saw tooth causes a varying pulse with which is amplified and switches the output stage to control the current to the generator field. Power for generator excitation and also for all control functions is obtained from the pilot PMG.

Equal reactive load division for each generator is provided by the reactive load equalizing loop. The principle is the same as that for real load sharing, except that the sine of the phase displacement angle between voltage and current (instead of the cosine) is used as factor to obtain the power product. This is done by connecting the primary of the modulation transformer (see Figure 3-5). Between lines T2 and T3, those lines where the current sample is not taken as input to the modulator.

The resulting modulator output dc signal is added or subtracted to the reference in the pulse width modulator, depending whether the respective generator contributes less or more than its share of reactive load to the system.

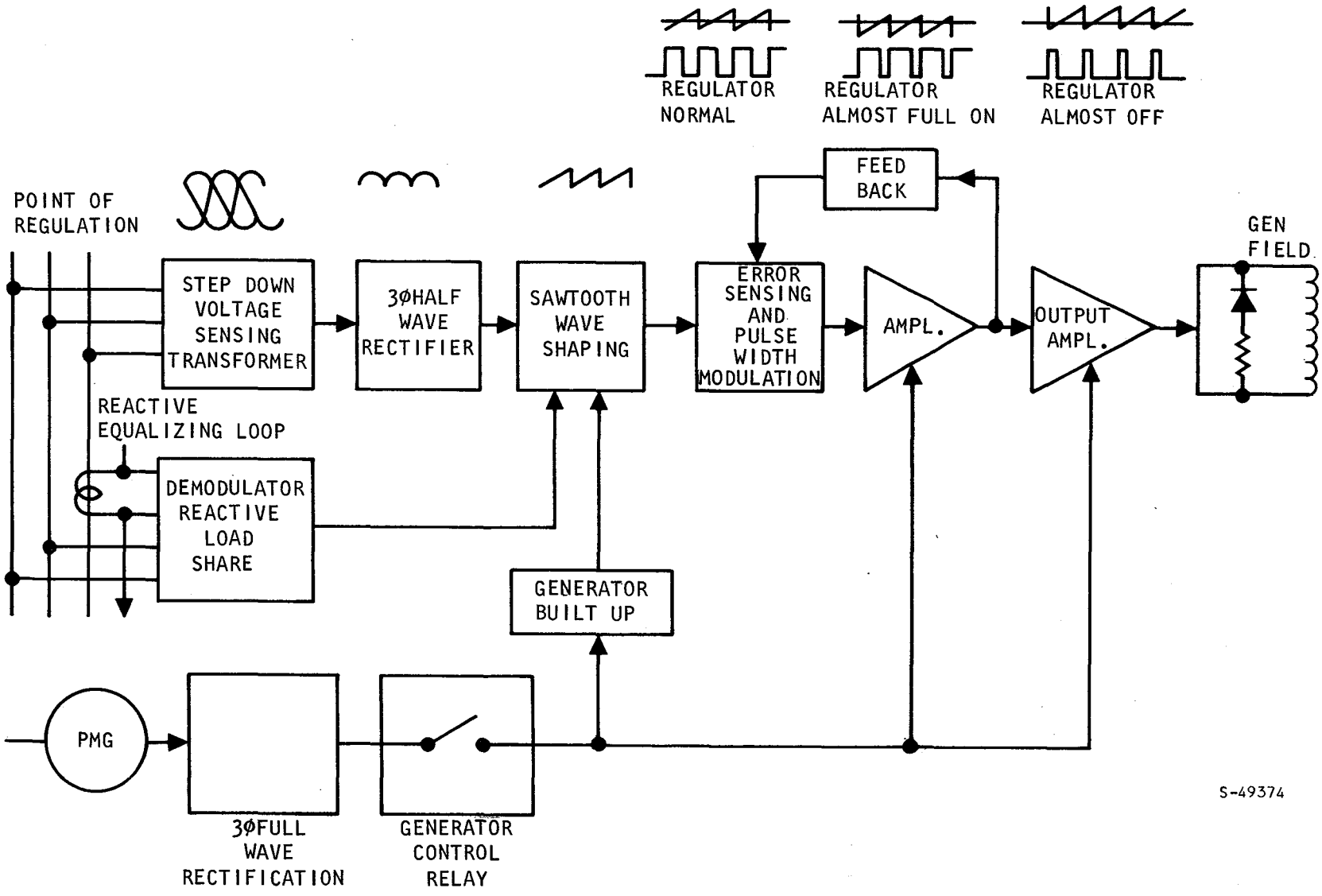


Figure 3-6. Basic Principle Operation of Transistorized Voltage Regulator

3.2.3 Control Panel

The control panel provides system control and protection from abnormal operating conditions for each generating channel, the total generating plant and the equipment load. These control, protective, and warning functions are listed below:

Control Functions:

(a) Generator field control

(b) Generator circuit breaker control

(c) Bus the breaker control

(d) Autoparallel

(e) Time delay functions

Protective Functions

(a) Overvoltage

(b) Undervoltage

(c) Overspeed, over frequency

(d) Underspeed, under frequency

(e) Overexcitation
(f) Underexcitation
} - Excessive reactive load division unbalance

(g) Difference current protection

(h) Differential protection

(i) Open phase protection

(j) Synchronizing bus protection

(k) External power out of tolerance protection

(l) CSD protection

Warning and Emergency Functions

(a) Generator overtemperature

(b) Emergency fire switch

This is believed to be a representative list of control and protective functions which could be provided by a supervisory panel for a conventional electrical system in contemporary aircraft. Existing control panels, however, do not provide all the protective functions listed above. Each individual application will call for an optimum number of protective functions which is influenced by the prevailing protection philosophy of the system designer. Some functions, such as inadequate external power quality and sync bus protection, which are usually not included in the generator control unit, are furnished by a separate bus protection panel.

3.2.4 Control Warning and Emergency Functions

3.2.4.1 Generator Control Relay

The main function of the generator control relay is to open and close the exciter field current. It is generally a latch type relay with both a trip and a close coil, and with auxiliary contacts connected to the generator circuit breaker trip coil circuit. This causes the circuit breaker to open whenever the exciter field is opened. Relay position is usually indicated by a light to the flight engineer. Operation of the generator field relay may be semiautomatic (flight engineer may have a switch for manual control) or fully automatic (flight engineer has no control). In very recent control panels, this mechanical relay has been replaced by solid state switches.

3.2.4.2 Generator Circuit Breaker (GCB) and Bus Tie Breaker (BTB) Control

These contactors are generally of the mechanically latching type. They require two separate input signals, one for closing and one for opening the breaker. These signals are furnished by small slave relays (mechanical or magnetic latch type) which control the main power contactors. Whenever a trip signal is present from a protective command, an inhibit signal prevents the respective circuit breaker from closing, thus providing anticycling as long as the abnormal condition persists. In very recent control panels, all mechanical relays have been replaced by solid state switches; the main power contactors are the only mechanical switches in the system. The status of the GCB and BTB is indicated to the flight engineer by a light.

3.2.4.3 Autoparallel

The automatic paralleling circuit operates to lockout the GCB closing circuit whenever the voltage frequency and phase angle are out of tolerance between the incoming generator and the synchronizing bus. Usually the circuit does not perform any corrective action to bring the two systems into synchronism. It merely prevents the incoming generator from being paralleled until all the conditions for paralleling are met. The circuit does not operate if the frequency difference exceeds 4 Hz or the phase angle difference is greater than 30 deg. If the frequency and phase angle requirements are satisfied, the circuit will operate with a voltage difference of 10 v or less line to neutral between the two voltages.

Some systems are provided with synchronizing lights to permit and aid the operator in tying the system manually via the BTB if the autoparallel circuit malfunctions or during specific emergency procedures.

3.2.4.4 Time Delay Functions

To discriminate the action of certain fault command signals, two different time delay circuits are simultaneously energized as shown in Figure 3-7. One time delay circuit actuates the BTB after 2 to 4 sec, thereby isolating that channel from the rest of the system. The other time delay circuit trips the GCR after 5 to 7 sec, which disconnects the generator from its load bus if the fault still persists. In this case, reclosure of the BTB is desired. On older systems this done manually; on more recent systems it is done automatically.

3.2.4.5 Generator Overtemperature

Some systems incorporate various warning functions; typical of these is generator overheat, which is indicated to the flight engineer by a warning light. The overheat warning could be caused by an overloaded generator, inadequate generator cooling or generator mechanical failure. Whatever the cause may be, the overheat warning is announcing incipient generator failure, and that the generator should be deenergized and removed from its load bus. The practical value of this warning indication has been questioned by some airplane users in the past. If some warning requires a definite response by operating personnel, the corrective action could be done automatically as well.

3.2.4.6 Emergency Fire

In a case of extreme emergency, where it would be desirable to completely deenergize the entire bus system, the flight engineer pushes the fire switch. This sends a signal to the 5- to 7-second time delay and results in tripping of the GCR and GCB. Pushing the fire switch affects all generating channels simulataneously.

3.3 SYSTEM PROTECTION

3.3.1 Protective Circuits Defined

(a) Over- and Undervoltage Protection--Prevents damage to load equipment in the event of excition faults during single generator operation

(b) Over- and Underspeed (Frequency) Protection--Prevents improper operation of load equipment; it is usually included in the constant speed drive; it selects and isolates a faulted drive or an abnormal speed condition during engine idle or shutdown

(c) Over- and Underexcitation Protection--Prevents damage to generating equipment in the event of excitation faults during parallel generator operation; selective isolation of the faulted system prevents power loss to the aircraft loads

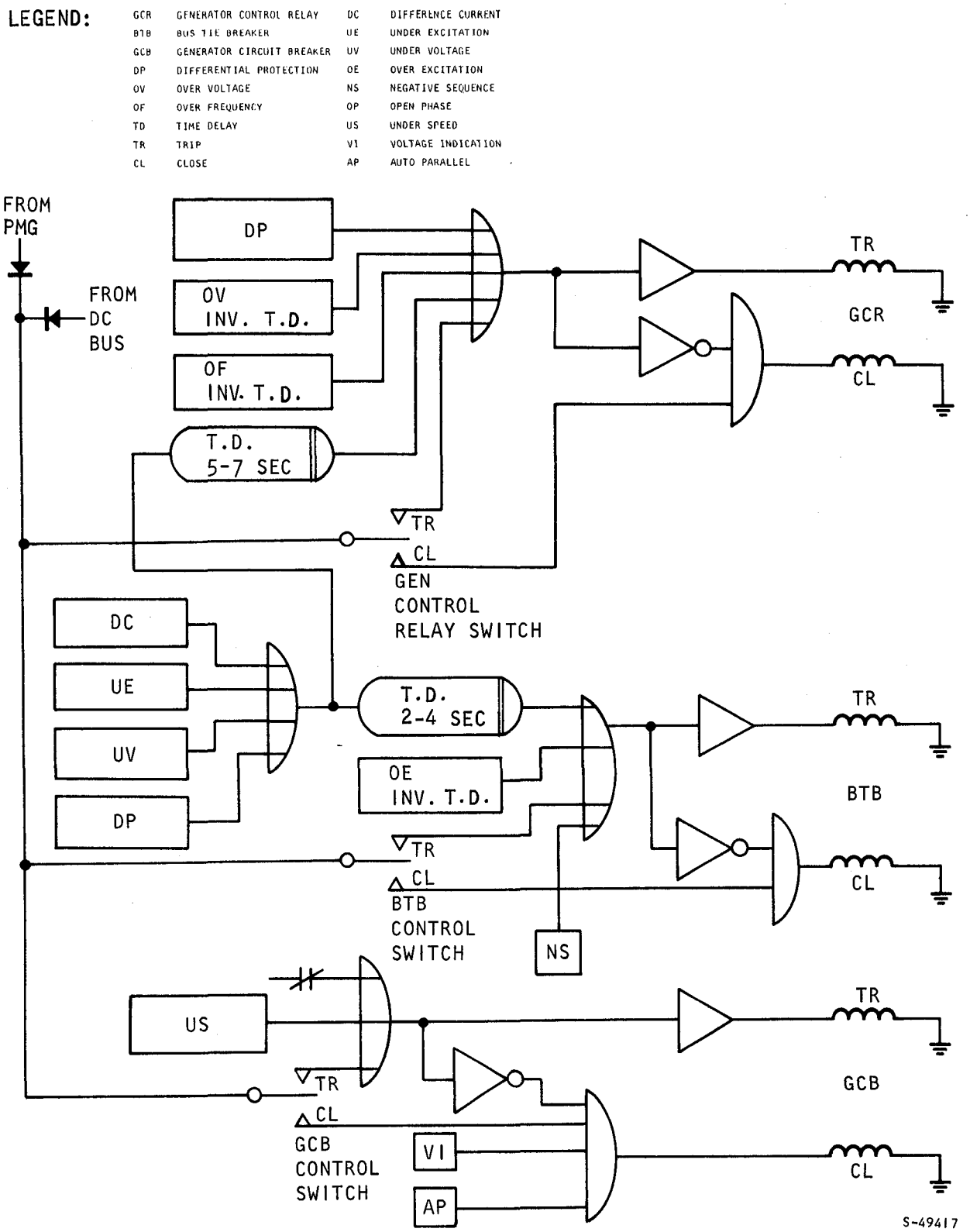


Figure 3-7. Control and Protection Logic Diagram

(d) Difference Current Protection--As for over- and underexcitation, this protection selects and removes that generating channel which contributes significantly more or less than its share of total current to the loads

(e) Differential Fault Protection--Guards against fault in the generating and transmission system only; it selects and disconnects a faulted generator or feeder (line to ground short or line to line short) from the remaining part of the system

(f) Open Phase Protection--Prevents damage to load and source equipment; if an open phase occurs anywhere in the system, and if there is danger of reducing system capacity, its respective system will be deenergized by operation of a zero sequence relay

(g) Synchronizing Bus Protection--A distribution system protection may be obtained by a negative sequency relay on the synchronizing bus, thus providing unbalance voltage protection for that bus and tripping all BTB simultaneously in case a line to ground or line to line fault on the bus occurs

(h) External Power Protection--To prevent inproper operation and damage to load equipment, several protective functions already provided for the aircraft system may also be included for protection of the external power channels. This usually is protection against out of tolerance voltage and frequency. In case of excess deviation, the external power contactor (EPC) is tripped off. Phase sequence protection will prevent closure of the EPC should the ground power phase sequence be opposite to that in the aircraft.

(i) CSD Protection--There are usually three protective devices incorporated in the constant speed drive. A shear section is provided on the input drive shaft for the protection of the main engine accessory gearbox. A solenoid operated emergency drive disconnect permits manual disconnect of the drive shaft in flight in case of generator or drive failure or overheat. The disconnect is controlled by a switch on the flight engineers panel and cannot be reset in flight. Also, an overrunning clutch on the drive input prevents power flow in the reverse direction from the generator back into the drive in case of severe load unbalance or drive failure in parallel operation.

Before discussing the function of each individual control circuit and the sequence of protective actions, a brief description of normal system operation using the logic diagram in Figure 3-7 is in order.

3.3.2 Normal System Operation

3.3.2.1 System Starting

When the aircraft engines are started, and the generators are up to speed, sufficient output power for control and generator excitation is available from the PMG. The CSD, which uses the PMG output as referenced frequency regulates

the system to nominal output speed. If the previous shut down of the system was normal and did not occur because of a fault, the generator control relay is in the close position and excitation current is automatically applied to the generator field. The voltage regulator will control the generator output voltage to 115 ± 1 v at the point of regulation. Before the generator circuit breaker (GCB) can be closed, four conditions must be met for single generator operation.

(a) The generator must be up to speed, and no underspeed signal (US) present

(b) The generator control relay must be closed

(c) Proper generator output voltage must be indicated (VI)

(d) Sync bus voltage must be zero

If all the aforementioned conditions are met, closing of the GCB switch on the flight engineers panel will close the breaker, connecting the generator to its load bus. Normally all BTB are closed, and each oncoming channel after the first is paralleled via the GCB.

3.3.2.2 System Shut Down

If the flight engineer wishes to isolate the load bus or synchronizing bus, the GCB switch and BTB switch can be moved to the trip position manually. If the aforementioned breakers are tripped, the associated generator may be deenergized by moving the GCR relay to the trip position, if the GCR and GCB are both closed, tripping the GCR will both deenergize the generator and automatically trip the GCB through the GCR contacts.

Normally, when no fault occurs, system shutdown is completely automatic. The generator circuit breakers are tripped by the underspeed circuit when the drive underspeed falls below 360 Hz. Thus on shutdown the BTB and GCR both remain closed. Since the bus tie breakers are closed, all load buses can be energized by external power when it is connected to the switch. Also since the generator control relay remains closed, the generators begin to build up automatically on start up as soon as control power is available from the PMG. Consequently the only action that is required for the system described is that on start up the flight engineer would have to close the GCB switches manually.

3.3.3 System Protection for Fault Condition

3.3.3.1 Single Generator System Faults

Overvoltage protection is usually considered to be the most important single protective device in an aircraft. An overvoltage condition can not only cause damage to load equipment (in particular, semiconductors) and decrease life expectancy, but it can also have adverse effect on the generating system

(heating) and may lead to severe reduction in total system capacity. Overvoltage protection is designed to have an inverse time-voltage characteristic because much of the damage caused by overvoltage is due to overheating. To obtain overvoltage protection and at the same time prevent nuisance operations on load switching transients, the circuit must operate very rapidly on extreme overvoltage and relatively slow on slight overvoltages. On completion of the time delay, the generator control relay trips, deenergizing the generator and, consequently, tripping the GCB. Figure 3-8 shows the operating limits of MIL-E-7894 for the overvoltage circuit. The overvoltage circuit senses the average of the three-phase generator output voltages and converts it to a proportional dc signal. This signal charges an appropriate RC timing network which in turn initiates the trip signal to the GCR. Due to the similarity of the overvoltage and overexcitation functions, these circuits are usually combined.

Undervoltage protection is needed to prevent damage to load equipment. Sufficient low voltage could result in high current and excessive I^2R losses in motor loads, causing overheating and eventual burn out. A condition of undervoltage exists whenever the system voltage falls below 100-v L-N. Load switching can result in momentary low voltage; however, time delays prevent nuisance operation by overriding these transients. The undervoltage is usually combined with the underexcitation circuit; its output starts different time delay circuits. In single generator operation, if an undervoltage condition exists for 5 to 7 sec the GCR will be tripped deenergizing that generator and associated GCB. Overfrequency or underfrequency protection is provided in case of excessive drive speed deviation. It is generally initiated when the system frequency increases or decreases 10 percent from its nominal value. This will cause tripping of the GCR and GCB. The circuit operation may be based on the frequency to dc conversion, comparing the analog of the PMG frequency to a fixed reference, or it may utilize the digital principle comparing the period of the frequency to be sensed to a fixed time interval.

Differential protection is provided for protection of the power source and the primary transmission system only. This protection is required to respond instantly whenever the difference current within the differential protection zone exceeds approximately 30 percent of the systems per unit line current. Faults causing this difference current may be due to internal grounds in the generator or phase to phase or phase to ground faults within the protective zone. The differential protective loop extends from three current transformers connected on the neutral side of the generator to three current transformers connected on the synchronizing but just beyond the bus tie breaker. A fault signal on the differential protection circuit will trip the GCR, deenergize the generator and isolate the generator from its load bus by tripping the GCB. Some newer systems employ two or more differential protection loops. One for each generating channel and others to include the synchronizing bus with all its feeder lines from auxiliary and external power.

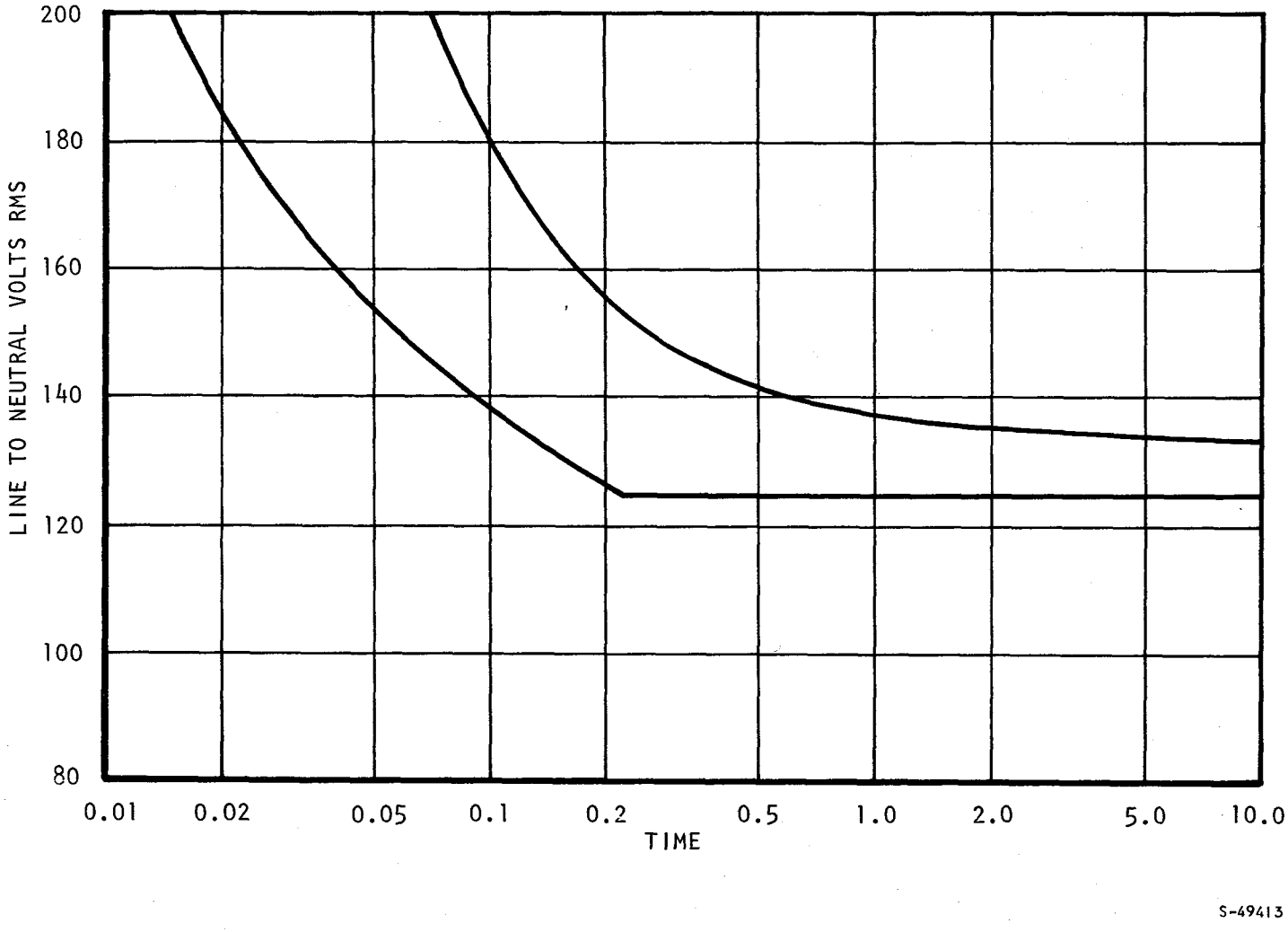


Figure 3-8. Operating Limits for Overvoltage Relay (Per MIL-E-7894)

Open phase protection guards against excessive phase unbalance. It can be shown by symmetrical components that whenever an open phase occurs in a grounded neutral system of paralleled generators, zero sequence current will appear in the neutral connections. Thus, a properly designed sensing network in the neutral connection of the system will sense and detect the zero sequence component, and, if excessive, will initiate a time delay network. In single generator operation the system will be deenergized and removed from its load bus after the condition has persisted for 5 to 7 seconds. Open phase sensing is usually incorporated into the differential protection circuit as a matter of convenience.

3.3.3.2 Parallel System Faults

Parallel operation of a number of small generating units is a factor which cnntributes to the reliability of the service furnished by a power system. Maximum reliability is not attained when each unit supplies an isolated section of the total load as is necessary on a variable frequency generating system. Isolated operation results in a portion of the load being lost at least momentarily, after failure of a generating unit and before the load may be divided among other machines.

In a parallel system, a generator can become overexcited or underexcited. But as it is in parallel with several other generators, the resultant voltage is the average of all the generator voltages and may not reach a value which would cause the overvoltage protection to operate. However, when a generator operating in parallel becomes overexcited it tends to supply considerably more than its share of reactive power to the system. This condition causes the generator to become overloaded, and since this could cause the system capacity to be reduced, the faulty generator should be removed from the system. Overexcitation is any condition that produces excitation over and above normal exciter field current requirements. Any overvoltage condition will also cause an overexcitation when the generators are operating in parallel. The following faults cause overexcitation: an open current transformer in the equalizer circuit, and a shorted equalizer loop could cause the machine with the higher voltage to become overexcited. Overexcitation is overvoltage biased with a current signal and is sensed in exactly the same manner as the overvoltage circuit. Current is sensed by means of the equalizer loop. This provides selectivity and enables the system to select and shut down the system causing the overexcitation.

Underexcitation is any condition which produces excitation below normal exciter field current requirements. When an underexcitation fault occurs on a generator operating in parallel with other generators this generator becomes a load for the rest of the system and heavy reactive circulating currents flow between generators resulting in overheating. The undervoltage on the system due to the underexcitation fault on one generator is not as severe as it would be if only the one generator were operating by itself. When an underexcitation condition of a certain generator is detected (by means of the equalizer loop, same as the condition of overexcitation) two time delays are initiated. One will trip the BTB after about 2 to 4 sec. If the voltage of that isolated generator indeed shows an undervoltage, the second time delay will deenergize the GCR after another 1 to 3 sec.

Difference current protection is another precaution to ensure that equal load contribution by each generating channel is within tolerance. This protection senses the total current difference between paralleled generators and, after an inverse time delay opens the BTB of a channel whose total current differs by a predetermined amount from the average.

Because the tie bus is not included in any of the protective schemes for each generating channel, there is usually a separate bus protection panel provided which protects not only the synchronizing bus but also the external power feeder lines and contactors.

On older type systems, bus thermal protection for the synchronizing bus was provided. Some systems employ negative sequence voltage protection. This type responds to line to ground faults and line to line faults, but it will not respond to a three-phase symmetrical fault.

Recent protective schemes incorporate the synchronizing bus into a separate differential protection zone which includes also auxiliary feeder lines and breakers. A differential fault within this zone will simultaneously trip all BTB and contactors connected to the zone.

4. EMERGENCY ELECTRICAL POWER SUPPLIES IN EXISTING AIRCRAFT

The source of emergency electrical power in existing aircraft is either a battery or an auxiliary electric generator driven by various type prime movers. A brief review of these sources is given in this section.

4.1 AIRCRAFT BATTERIES

Aircraft batteries are used to provide stand-by sources of power and, in small airplanes, also to start the engine. In case of generator failure, the batteries will supply power to radio communication equipment, instruments, essential lighting, and controls for sufficient time to allow the aircraft to land. The batteries are kept fully charged by the main generators during flight. Batteries selected for aircraft application should be lightweight, low cost, rugged, nonspill, tolerant of wide changes in temperature and pressure, and capable of long charge retention and rapid recharge from the normal aircraft charging system.

If engine starting is a requirement for the battery, the battery should be designed with very thin plates, so that very high, instantaneous performance can be obtained at low weight.

Three types of batteries are used in aircraft today:

(a) Lead acid battery

(b) Nickel cadmium battery

(c) Silver-oxide zinc battery

Performance data of these batteries are given in Table 4-I.

4.1.1 Lead Acid Batteries

The lead-acid system has the advantages of high voltages, good cycle life, and low cost. Its disadvantages are weight, bulk (low energy density), poor capacity retention, and low temperature service. The aircraft lead acid batteries are similar to those batteries for automotive starting. Its electrodes are cast lead-antimony grids filled with a paste of lead oxide, water, and sulfuric acid. The paste of the negative plates contains about one percent of expander, which is comprised of barium sulfate and an organic material, such as lignin.

TABLE 4-I

PERFORMANCE DATA OF BATTERIES IN EXISTING AIRCRAFT

Characteristic	Cell Type		
	Lead Acid (Open Rectangular)	Nickle-Cadmium (4 AH Sealed Cylindrical)	Silver-oxide Zinc (Open Rectangular)
Energy Density			
Theoretical W-hr/lb	55	95	196
Reported W-hr/lb	10 to 20	12 to 14	40 to 75
Reported W-hr/cu in.	1.0 to 2.0	1.15 to 1.30	2.0 to 4.8
Cell Voltage			
Theoretical	2.05	1.299	1.672####
			1.560#####
Open circuit voltage#	2.0 to 2.05	1.30 to 1.35	1.86
C/8 to C/5 rates##	1.95 to 2.05	1.25	1.58
C/2 rate		1.20	1.54
C/0.2 rate		0.95	1.37
Cycle Life			
50 % depth of discharge		2000 to 3000	100 to 300

Notes:

#The differences in the OCV and theoretical voltages may be due to the time the cell stands between charge and discharge

##C represents the capacity of the cell. C/N rate of discharge indicates a rate such that the nominal capacity of the battery is removed during a N-hour period.

####Value associated with the higher (+2) oxidation state of silver.

#####Value associated with the lower (+1) oxidation state of silver.

The barium sulfate provides a surface for the deposition of lead sulfate, and the lignin prevents it from crystallizing on the sponge lead surface, thereby improving the high discharge and low temperature characteristics of the batteries. After the plates have been formed, dried, and assembled, they are immersed in a weak solution of sulfuric acid and charged. The lead oxide of the negative plate is reduced to sponge lead, and the lead dioxide is formed at the positive plates.

Recent progress in lead acid batteries is achieved by using thinner plates, thereby increasing surface area for the same weight. The increase in surface area enables a reduction in working surface current density, resulting in lower internal resistance. This, in turn, reduces the internal I^2R loss and the internal voltage drop.

Other recent improvements of lead acid batteries include the use of a lightweight polystyrene container and the arrangement of passing the intercell connectors through the cell walls. Both of these help to reduce the weight of batteries.

4.1.2 Nickel-Cadmium Batteries

The nickel-cadmium battery is mechanically rugged and capable of withstanding electrochemical abuse such as overdischarging, overcharging, and standing idle in the discharged state. Although Ni-Cd batteries operate at slightly lower voltages, they exhibit very little self-discharge and can be used over a temperature range of -55^0 to 80^0C. These batteries evolve practically no gas on discharge or idle stand so that for some purposes they can be hermetically sealed. In addition, Ni-Cd batteries are characterized by good cycle life and can be kept fully charged by a very small current (floating charge), resulting in little water consumption. By changing the construction of the plates, it is possible to build batteries for low or high rate applications. The positive and negative plates are of the same mechanical design and external appearance.

In one type of construction (pocket type), the active materials are tightly packed in perforated flat tubes or pockets and mounted horizontally in nickel-plated steel frames. The plate separators are rods of insulating material that rest in vertical grooves pressed into the faces of the plates. The active material in the positive plate is initially nickelic hydroxide with about 25 weight percent of a specially treated graphite, the latter acting as a current conductor. The negative material, cadmium, in some instances contains about 25 weight percent iron oxide, which acts as a disperser to prevent the packing together of metallic cadmium particles formed on reduction of cadmium oxide (CdO) during charge. The graphite and iron take no part in the cell reaction, and the electrochemical reactions that occur in the cell during charge and discharge are very complex and not fully understood.

The second type of construction (sintered-plate) uses porous plates made from nickel powder, which is molded into shape and sintered (at 900°C) in a reducing atmosphere onto nickel or nickel-plated supports, such as screens, perforated metals, or expanded metals. This produces porous plates with pore volumes or porosities in the region of 70 to 90 percent. These plates are then impregnated with active material by soaking them in a solution of nickel salts to form the positive plates or a solution of cadmium salts to form the negative plates.

The pocket-plate design is more rugged, but the sintered-plate design, because of the thinner plate construction and larger surface area, has better high-rate discharge and low-temperature performance characteristics. At low drain rates, the pocket- and sintered-plate batteries have similar characteristics. The Ni-Cd battery is particularly suitable for services where the charge and discharge rates fluctuate considerably. Aircraft Ni-Cd batteries are mainly sintered-plate design. The limitations of Ni-Cd batteries are low energy output per unit weight and volume, moderately high cost, the need for an overcharge to obtain full capacity output, and the time required (12 to 16 hr) to recharge the sealed cells. Rapid charging was possible only through the use of complex chargers or supplementary devices such as auxiliary electrodes that added to weight, volume, and cost. However, recent developments, such as the fast-charge cell and new charging innovations, have made it possible to charge Ni-Cd batteries rapidly and safely without the need for external control devices, reducing the charge time for some applications to approximately one-half hour.

4.1.3 Silver-oxide/Zinc Batteries

The silver-oxide/zinc (AgO-Zn) battery is representative of a new class of secondary batteries that was not commercially available until after World War II. Although the potentialities of this electrochemical system were known for many years, it was not until the early thirties that the silver-oxide/zinc battery became practical. Up until that time, the major difficulty with this system was the intermixing of the active materials during charge and discharge. This problem was later resolved to a considerable extent by interposing a semi-permeable cellulosic membrane separator between the plates.

In these cells in the charged state, the cathode consists of silver oxide electrolytically formed on silver or a silver-plated copper screen. The anode consists of sponge zinc on a suitable collector, usually silver or silver-plated copper foil. The electrolyte is an aqueous solution of potassium hydroxide (KOH) in concentrations over the range of 30 to 45 weight percent. The separator materials used most frequently are cellophanes or various modified cellophanes. Most of the electrolyte in this type of cell is absorbed by the porous electrodes and separator materials. In the fully charged state, the electrolyte level reaches to approximately one-third the height of the plates and reaches all the active surface of the electrodes by capillary action.

On discharge, the silver oxide is eventually reduced to metallic silver and the zinc is oxidized to form the hydroxide. The charge and discharge characteristics are complicated at times by the presence of two distinct voltage plateaus, corresponding to the monovalent and divalent states of silver. This phenomenon is not noticeable at high rates due to polarization. An initial high voltage level is observed at low drain rates and can be eliminated by preloading the cell until the voltage falls to the lower value.

For the combination of high energy density and high power density, the AgO-Zn battery is undoubtedly the leader among present-day practical rechargeable systems. These batteries deliver high currents and give three to five times higher watt-hour capacities per unit weight and volume than lead-acid or nickel-cadmium batteries. Other advantages of the AgO-Zn battery are good charge retention, high charging efficiency, good voltage regulation, and good high-temperature operating range.

Although the AgO-Zn system may exhibit all the desirable features listed above, it also has the property of being one of the most thermodynamically unstable. Finely divided zinc is highly susceptible to oxidation in the presence of moisture. Fortunately, in the absence of moisture, the rates of the degradation (and passivation) reactions involved are very slow. For this reason, AgO-Zn batteries are usually shipped in dry form and sealed to keep out moisture. Another undesirable characteristic of this system is the high solubility of the electrode reaction products in the electrolyte. This zinc discharge product in particular is highly soluble and does not redeposit in its original form or position on the electrode during charge, resulting in rapid deterioration of performance. Another serious weakness of the zinc electrode is its tendency to form dendrites or "trees," which, if they remain attached to the anode, cause shorting of the cell. Other limitations are short storage and cycle lives, sensitivity to overcharge, poor low-temperature performance, and high initial cost.

The silver-oxide/zinc batteries have not found wide aircraft application because of their short cycle life and the requirement of special charging facilities because they are not suitable for constant-potential charging. High discharge currents can generate high internal temperatures which may damage the separators and possibly even destroy the cells.

4.2 AUXILIARY POWER UNITS

Auxiliary power units generate emergency power independently of the aircraft main engines. These units can supply power to the airplane while on the ground, when all engines are out during flight, or for engine starting. The prime movers for large units are usually gas turbines, and for smaller units are gasoline engines or ram air turbines. The ram air turbine is classified as one kind of auxiliary power unit in this study. Electrical loads while the airplane is on the ground include galley services, cooling fans, recirculating fans, lighting, fuel pumps, instrumentation and ignition.

4.2.1 Gas Turbine-Generator APU

Usually the APU gas turbine uses the same fuel as the main engines. Unlike the main engines, these APU turbines run at constant speed. If the APU serves as a standby source of power in the event of failure of the main engines in flight, the turbine must be capable of being re-lit and operation at all altitudes. The gas turbine APU can provide a continuous supply of bleed air in combination with shaft power. Gas turbine APU can be placed advantageously anywhere in the airplane. The prime considerations are fire safety and noise.

The weight of a gas turbine varies appreciably with design requirements. For reference purpose, the weight of a gas turbine and gearbox versus shaft horsepower (sea level and static) is given in Figure 4-1. The specific fuel consumption of a gas turbine also depends on operation conditions. Approximately at sea level and static condition, the fuel rate is about 0.5 lb/hp-hr for gas turbines in the 200 to 600 hp range. Fuel rate increases rapidly as the turbine rating decreases to below 100 hp. Fuel rate also increases with altitude, but decreases as the airplane speed increases.

The gas turbine APU has much larger energy density than storage batteries. For example, consider a 70-hp, 60-kva (0.8 pf) gas turbine APU.

Weight of 70-hp turbine and gearbox (Figure 4-1) = 75 lb

Weight of 60-kva generator = 60 lb

Assume the total installed weight = 155 lb

(The weight of the starting air bottle, battery charger, and control panels are all ignored in this comparison.) The fuel rate of the turbine is about 0.7 lb/hp-hr. For a one-half hr operation, the fuel weight is 35 lb.

APU and fuel weight 190 lb

Emergency capability 48 Kw for 1/2 hr

The same weight of Nickel-Cadmium battery has energy storage of approximately:

$$\frac{190\ (\text{lb}) \times 13\ (\text{w-hr/lb})}{1000} = 2.47\ \text{Kw-hr}$$

Thus the gas turbine APU in this given condition is a much better source of emergency power. However, if the emergency power requirement is low and the time duration for emergency power is short, the battery will become competitive with the gas turbine APU as an emergency power source.

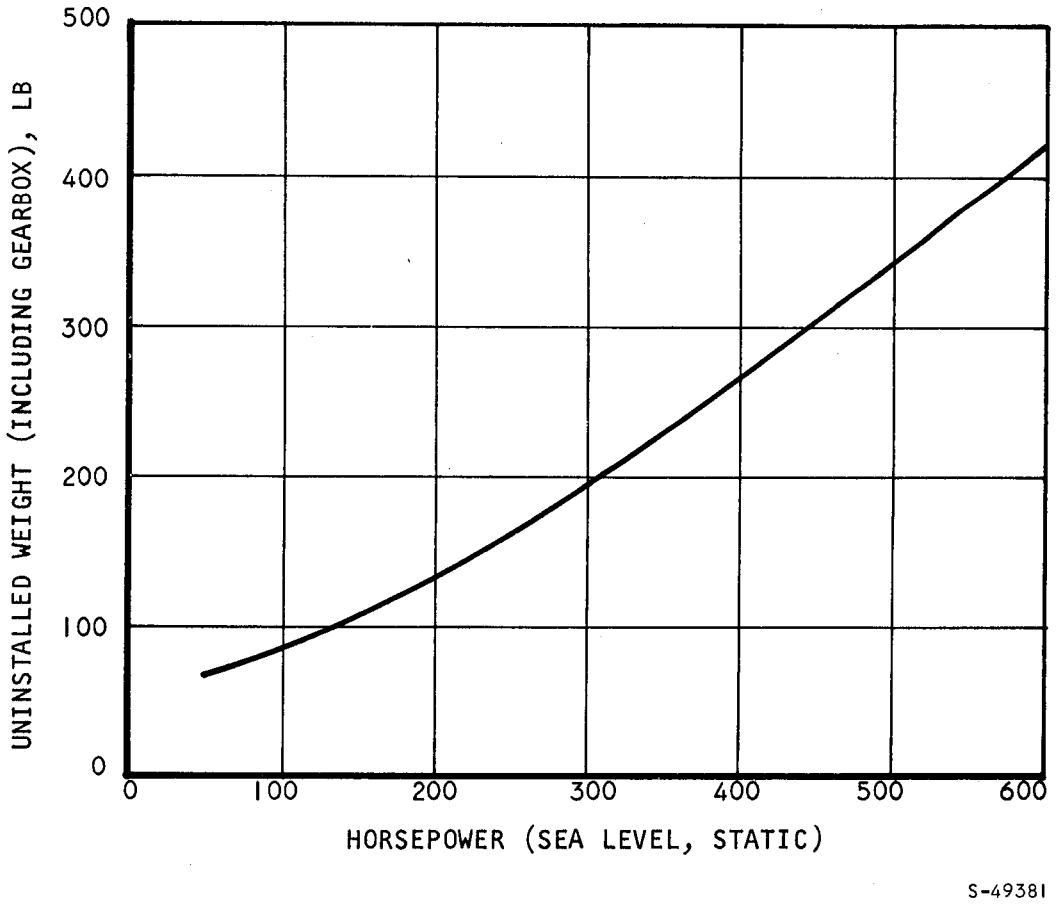


Figure 4-1. Approximate Weight of APU Gas Turbine with Gearbox

4.2.2 Ram Air Turbine

The ram air turbine is another independent source of power for electrical, hydraulic, and shaft power for aircraft. These units generally mount in the fuselage of an aircraft and are flipped into the airstream when emergency power is required. If desired, these units can be retracted after use. Present ram air turbines can be used for flight envelopes ranging from as low as 80 knots to Mach 1.4 at sea level and up to Mach 2.4 at 60,000 ft altitude. The speed of the turbine is controlled by a governor, presently at ±2 percent.

The ram air turbine may have various numbers of blades. In general, eight to ten blade units are best for hydraulic power systems where high starting torque and quick acceleration under load are required. These turbines develop high power at low rotational speeds. Two and four blade units are particularly suited to electrical power systems where a close speed control is required and high speed (12,000 rpm) generators can be used. These units are outstanding where light weight, low drag, and low costs are important requirements. Five-bladed units can fulfill the same requirements as two- and ten-bladed units. They have good starting torques and are designed for rotational speeds up to 9000 rpm.

The weight of ram air turbine varies with design condition. If the turbine supplies only electric power, the specific weight of turbine and generator combination is approximately 2 lb/kva for a 50-kva unit to approximately 6 lb/kva for a 5-kva unit, at about 150 knots indicated air speed and 12,000 rpm. The approximate diameter across the tips of the turbine blade is given in Figure 4-2.

Consider another example of providing 10 kva (0.8 pf) of emergency power for 10-min duration. A comparison of the weights of gas turbine APU, battery, and ram air turbine is given below:

(a) Gas turbine APU--12.5 hp turbine 10-kva (8-kw) generator

Turbine and gearbox	60 lb
Generator	20 lb
Approximate combined installed weight	≈ 95 lb
Fuel for 10-min operation	≈ 5 lb
Total	100 lb

(b) Nickel-Cadmium battery

$$\text{Energy required} = 8000 \times \frac{1}{6} = 1334 \text{ w-hr}$$

$$\text{Wt of battery} = \frac{1334}{13} = 103 \text{ lb}$$

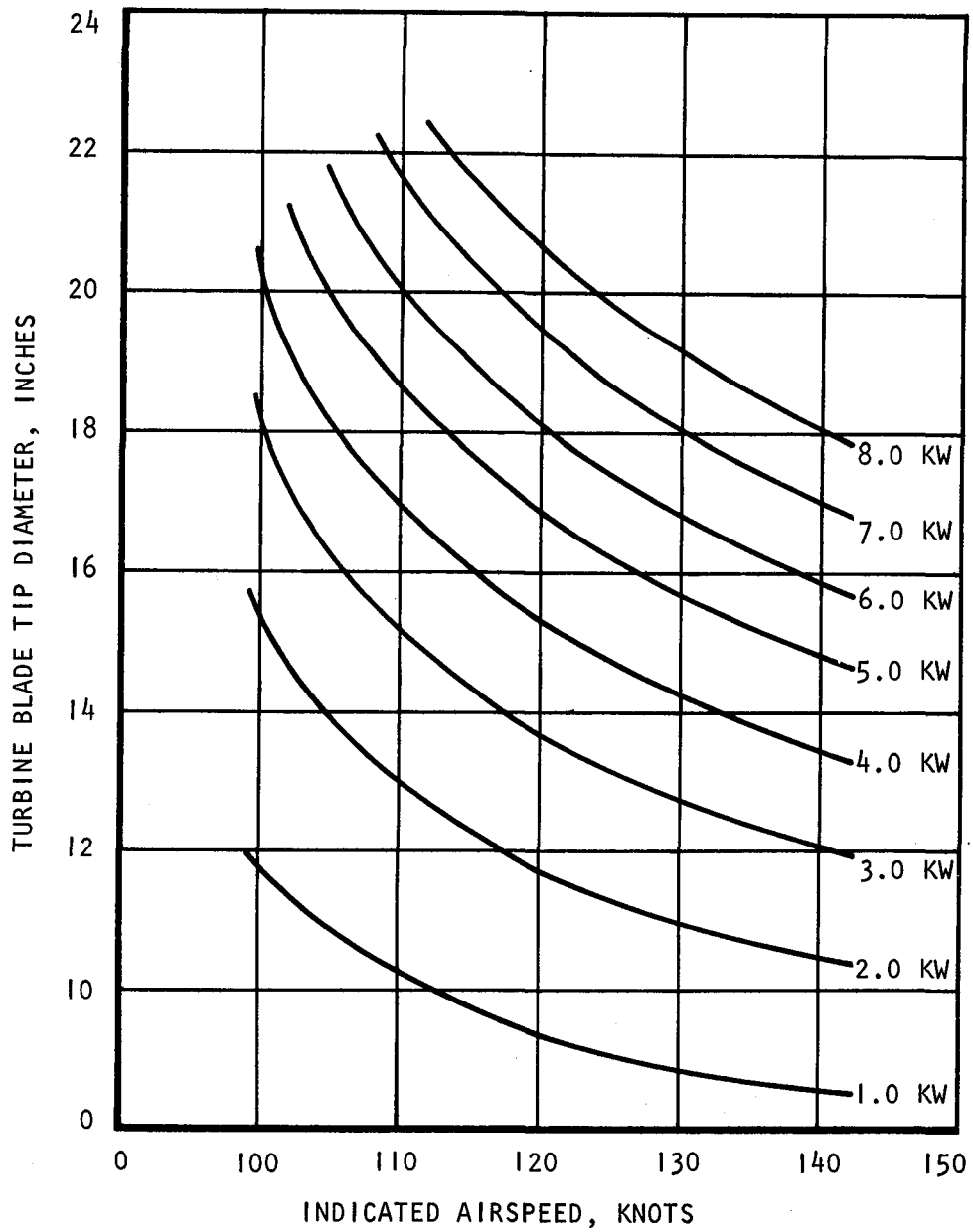


Figure 4-2. Turbine Blade Tip Diameters S-49416

(c) Ram air turbine

Estimated weight of turbine generator = 45 lb
(10 kva, 12,000 rmp, 150 kias)

It can be seen that at low power level, the ram air turbine APU has higher energy density than the nickel-cadmium battery and the gas turbine APU.

5. INFORMATION DISPLAY

Higher performance aircraft and more complex avionics have given the pilot and flight engineer more work to do and less time to do it. More subsystem components and flight aids have caused cockpit readout congestion. As a result, the display of electrical power system information in current aircraft is rather simple. In a large commercial transport, the following information are usually displayed:

Ac Power

Phase voltage - A voltmeter with phase selection switch frequency

Power and reactive power - One single phase wattmeter per channel. By push button action, this wattmeter can be changed to read reactive volt-amperes

Dc Power

Voltage

Current

Operational Status

Generator circuit breaker, "on" or "off"

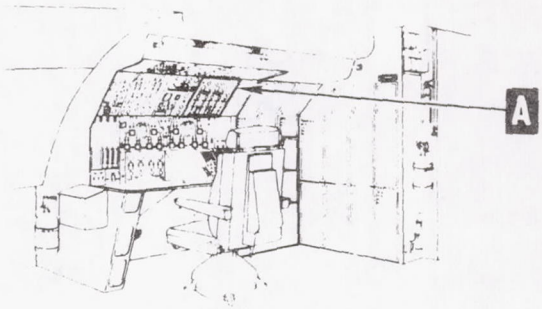
Bus tie circuit breaker, "on" or "off"

APU generator, "on" or "off"

Current philosophy is that a certain piece of information will not be displayed unless that information will help the flight crew to perform some corrective action during flight.

A photograph showing the ac system control panel in C-5A airplane is shown in Figure 5-1. C5-A represents the latest aircraft design.

The use of system and component failure prediction, detection and compensation on future advanced aircraft will require a very different display system.



FLIGHT ENGINEER'S STATION

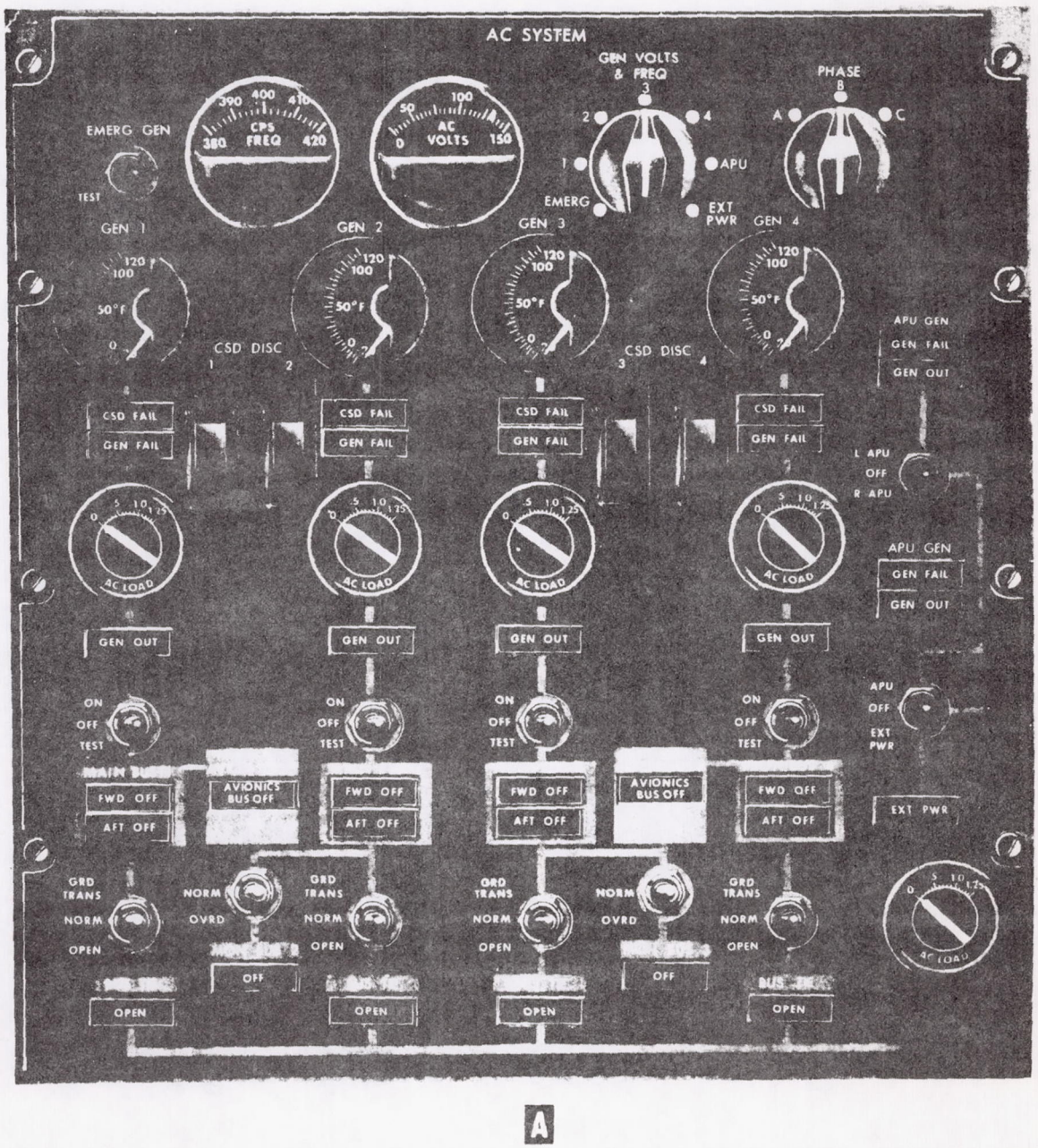


Figure 5-1. Flight Engineer's AC System Control Panel

6. HEAT TRANSFER TECHNIQUES

6.1 INTRODUCTION

In order to meet the specified performance and reliability criteria of an electric or electronic component, cooling is normally necessary to maintain the component ambient temperature at or below the manufacturer's maximum rated temperature for at least 90 percent of the life of the unit.

Reliability requirements are usually expressed as a mean time between failure (MTBF), the reciprocal of which is defined as the failure rate. From failure-rate curves, such as shown in Figures 6-1 and 6-2, the allowable component ambient temperature can be determined for a required MTBF for a specific component.

The recommended component ambient temperatures as specified in MIL-E-5400 is reproduced in Table 6-1. For continuous operations, temperatures of 55, 71, 95, and 125°C are not to be exceeded for Classes 1, 2, 3, and 4 environments, respectively, under the worst heat dissipation conditions.

The problems of cooling electric and electronic equipment in aircraft, in general, have become increasingly difficult because of

(a) Requirements for operation at increasingly high altitudes and flight speeds

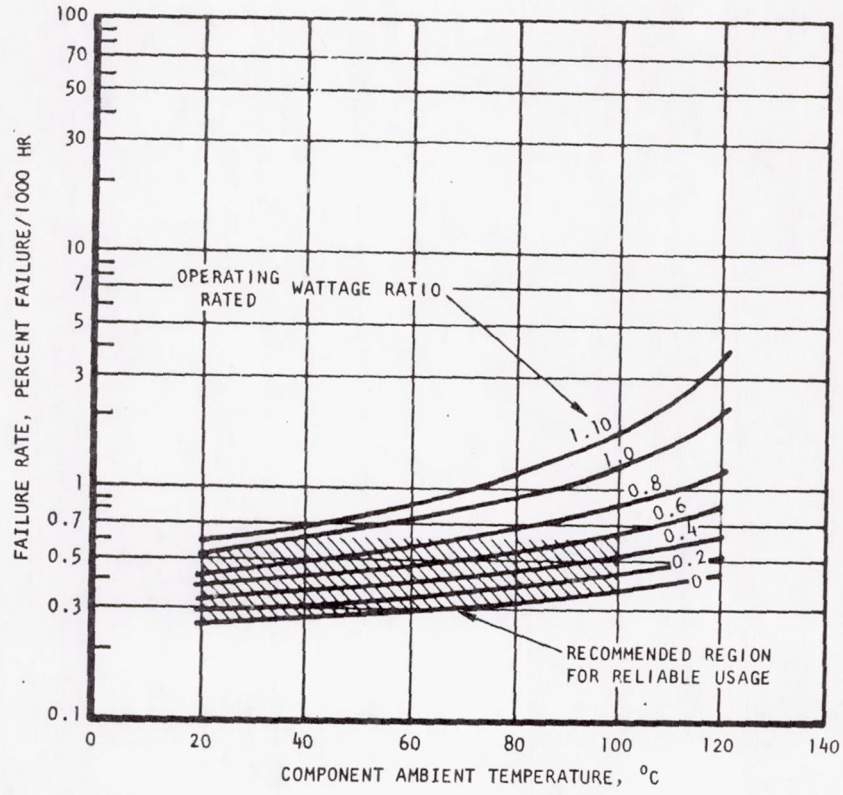
(b) The size-reduction of equipment to fit allowable space envelopes

(c) An increase in the power ratings of components

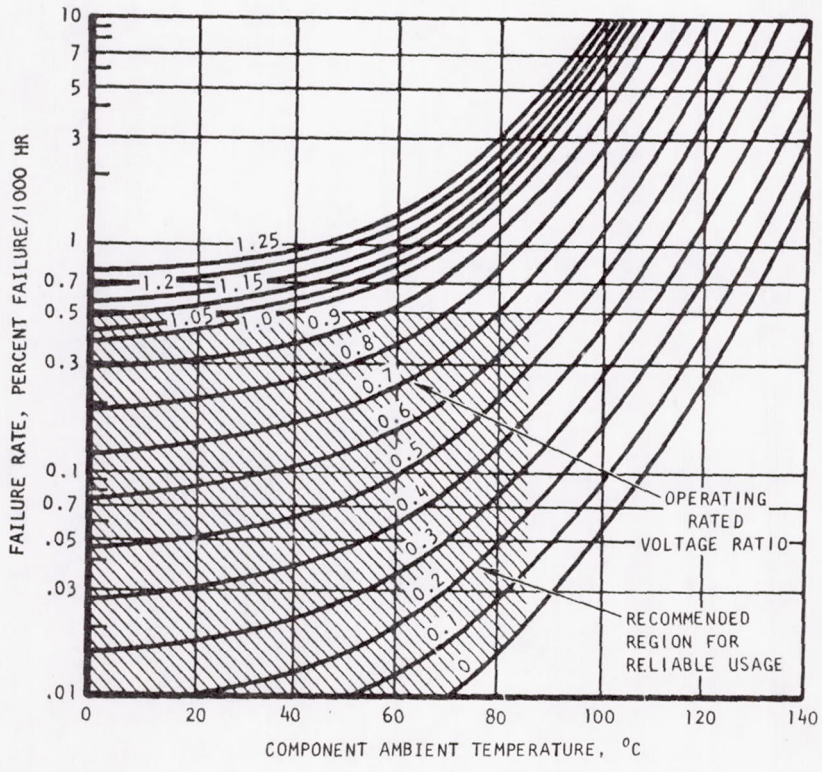
(d) The desire to increase the present service life components by operating at lower temperatures

Some methods of cooling electrical system components now in use on aircraft are briefly described in Section 6.2 and illustrated by sketches and photographs. The approximate cooling capabilities of each individual method or their combinations, as summarized from past hardware experience, are shown in Figure 6-3 and Table 6-2.

Various cooling systems are described and illustrated in Section 6.3. Variations and modifications of these basic systems are possible but are not described here. The system control that is an integral part of the basic system is not discussed in detail. No performance comparison is considered, but a sample output of an AiResearch computer program showing its capability of providing this piece of information is included.



A. SILICON DIODES



B. CERAMIC CAPACITORS

Figure 6-1. Typical Electronic Component Failure Rate Curves

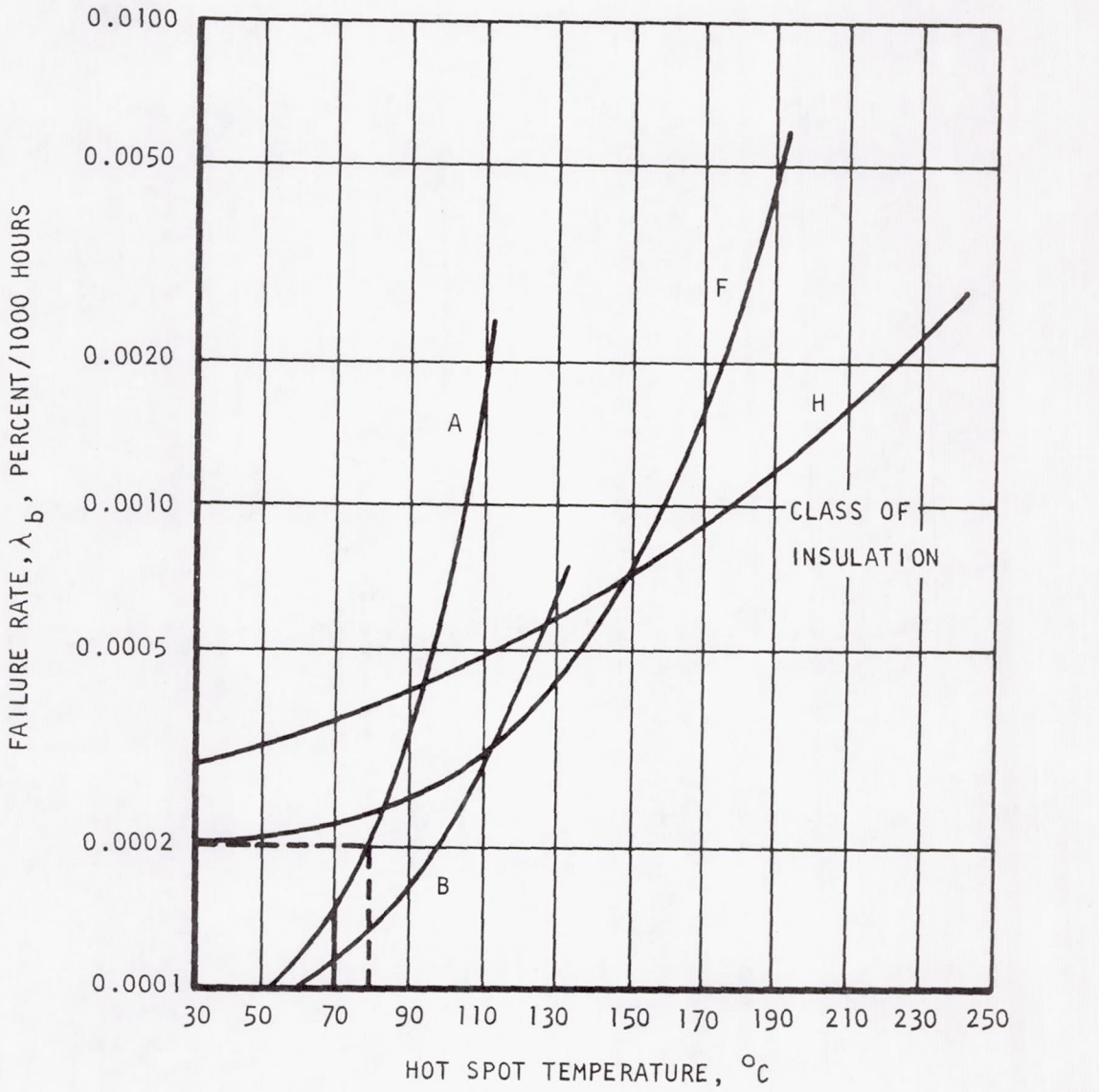


Figure 6-2. Failure Rates from Temperature and Class of Insulation for Motors and High Speed Rotating Devices Technical Report No. RADC-TR-67-108, 1967

TABLE 6-1

RECOMMENDED COMPONENT AMBIENT TEMPERATURES (MIL-E-5400)

Environment	Ambient Temperature Limit, °C			Heat Flux, w/sq in.	Applications
	Preferable Continuous Operation	Maximum Continuous Operation	Maximum Intermittent Operation		
Class 1	25	55	71	0.1 to 0.5	Integrated circuits, capacitors, resistors, and power transistors
Class 2	40	71	95	0.3	Integrated circuits, transistors, resistors, and high-temperature capacitors
Class 3	70	95	125	~	Motors, generators, transformer, inductors, diodes
Class 4	90	125	150	~	High-temperature components

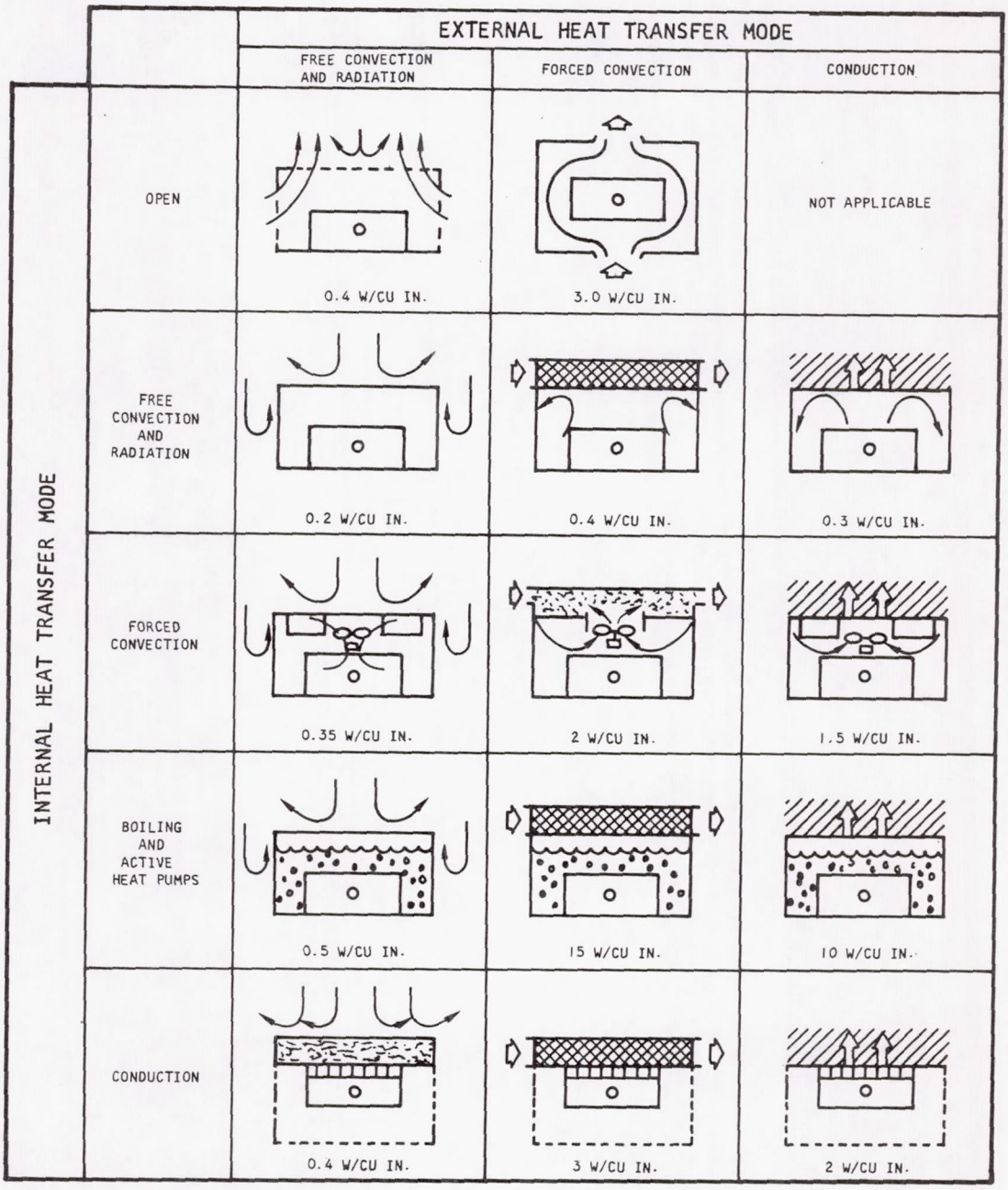


Figure 6-3. Internal/External Heat Transfer Modes and Heat Dissipation Capability

TABLE 6-2

APPROXIMATE COOLING CAPABILITIES OF DIFFERENT METHODS FOR ELECTRIC ROTATING MACHINES

Cooling Method	Approximate Cooling Capability, w/cu in.
Self-cooling without heat sinks	3
Self-cooling with heat sinks	5
Forced gas convection	5 to 9
Forced liquid convection	10 to 20
Expendables	50 or more

In Section 6.4 are presented improvements that can be incorporated for aircraft application. Some of these methods have already been used successfully for ground applications. In the next reporting period the recommended research and development trends will be described.

6.2 AIRCRAFT EQUIPMENT COOLING METHODS

The cooling methods now being used in aircraft electrical equipment are grouped subjectively into the following seven categories. These methods are generally special cases or combinations of the basic heat transfer mechanisms, i.e., conduction, convection, and radiation.

(a) Self-cooling (Section 6.2.1)

(b) Conduction or heat-shunt cooling (Section 6.2.2)

(c) Forced gas cooling (Section 6.2.3)

(d) Forced liquid cooling (Section 6.2.4)

(e) Thermal attenuation or thermal damping (Section 6.2.5)

(f) Thermoelectric coolers and heat pumps (Section 6.2.6)

(g) High-temperature heat exchangers (Section 6.2.7)

6.2.1 Self-Cooling

By this method the equipment is cooled by natural convection between the equipment surface and the surrounding fluid. It is always aided to some degree by radiation. Extended surfaces are very often used, and production-type finned heat sinks are available where more convective area is needed.

Power transistors are often cooled by this method. Figure 6-4 shows some arrangements and a serrated surface by which the convective area is further increased.

This method is good when the required heat dissipation rate is in the order of 0.5 w/cu in. Conventional motors, generators, and transformers quite often use this method of cooling. Concorde's static inverter also uses this method.

Figure 6-5 shows a self-cooled electric motor. Cast fins are an integral part of the housing. Black paint can be used to promote radiation heat dissipation.

6.2.2 Conduction or Heat-Shunt Cooling

This is a method in which heat is dissipated by connecting the heat source to the heat sink with a good conductor like metal or Hot spots are avoided by covering the whole component exterior with a metal sheath.

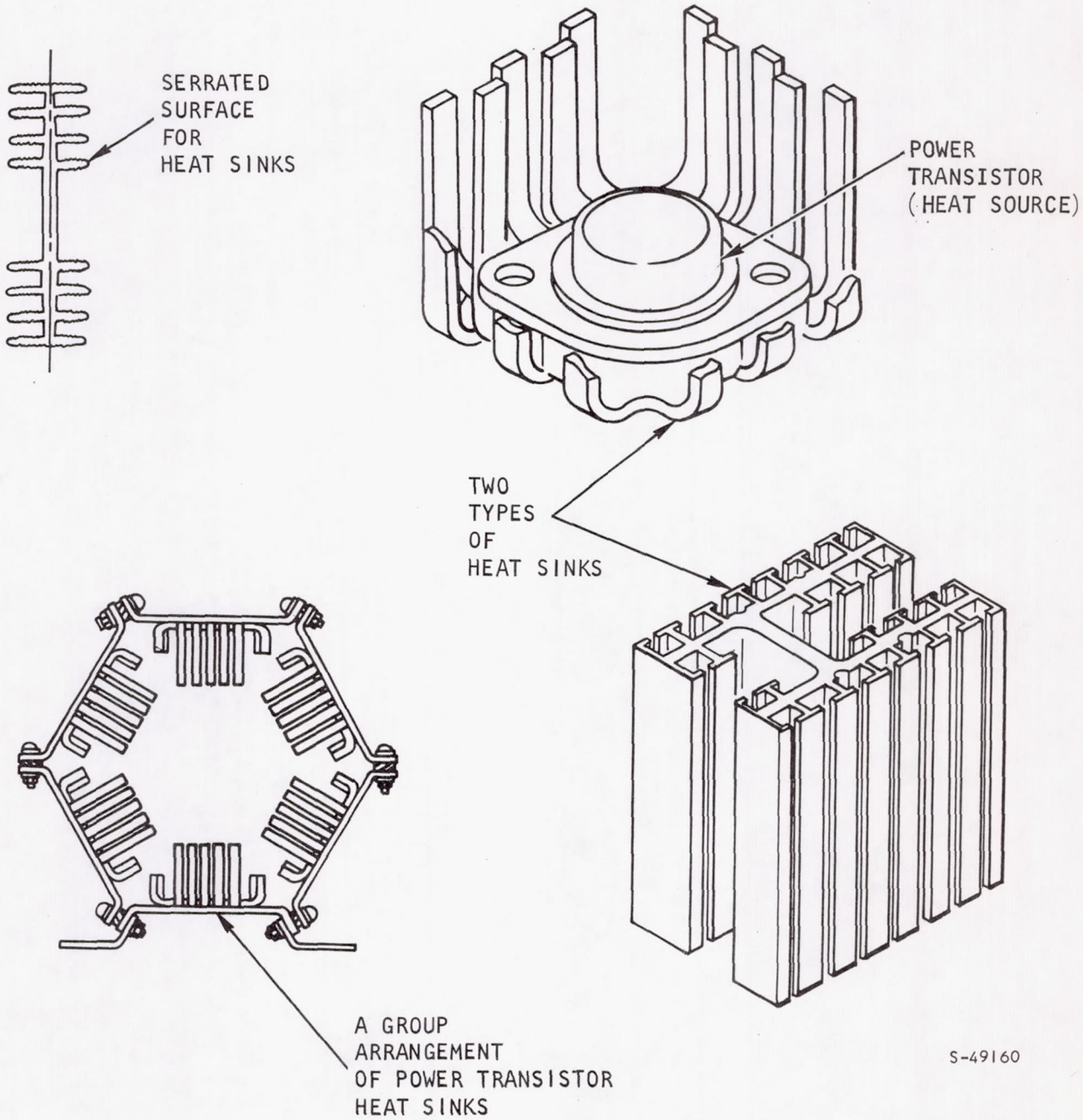


Figure 6-4. Power Transistor Heat Sinks for Self-Cooling

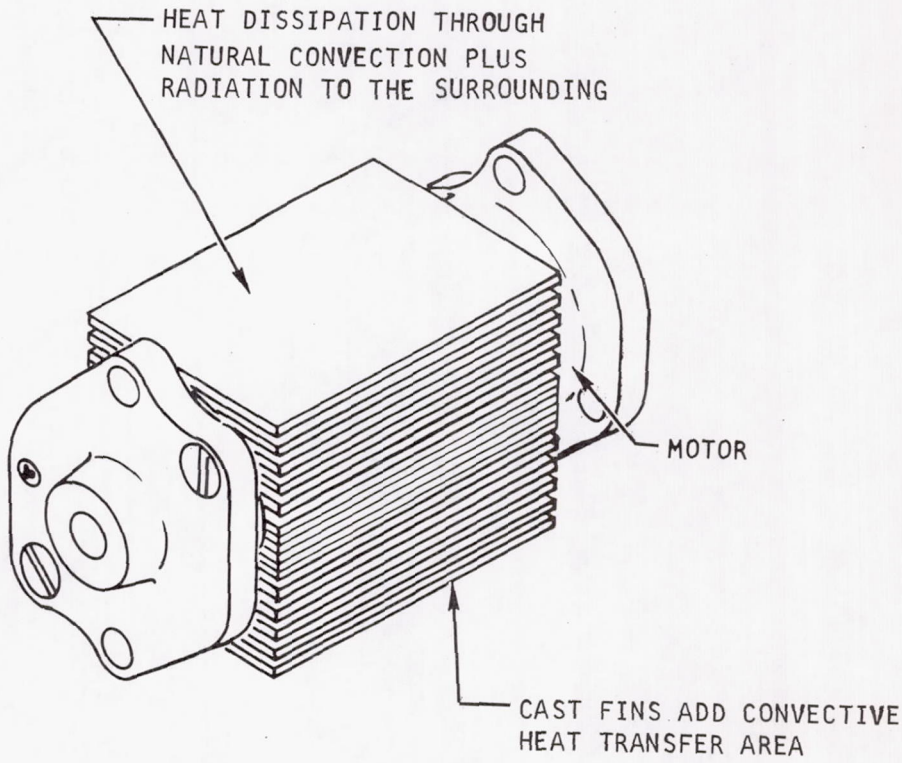


Figure 6-5. Flap Actuator Motor

In the application of this method, good contacts between parts are not always available. Contact resistances, i.e., conduction through a mechanical joint, can significantly reduce the thermal conductance. In these cases thermal-joint compounds like silicone grease are used to improve the heat transfer.

In cases where high dielectric strength is a requirement, materials like beryllium oxide with safe operation up to 1500-v or more and with thermal conductivity in the same order as that of copper (200 Btu/hr-ft-^{o}F) are available.

Figure 6-6 shows several ways that this method has been used in an electronic cooling.

6.2.3 Forced Gas Cooling

In cases where self-cooling or conduction cooling is not sufficient, the motion of the gas coolant can be mechanically agitated. This widely-used method has an advantage of simplicity because all it requires is a fan or a blower with or without ducting. Conventional motors and generators that have built-in fans are cooled by this method. Concorde's transformer-rectifier unit has an integrated fan and is thermally switched.

In order to dissipate large amounts of heat, cold plates as shown in Figure 6-7 are often provided. The gas from the fan is forced to flow over the equipment and the extended surfaces to dissipate the heat generated.

Another good example of cooling by forced air convection is illustrated by Figure 6-8 where the rotation of the rotor is utilized to pump or shear the air for cooling purposes.

High-specific-heat gases, e.g., hydrogen, are preferred to those with lower specific heats because the former require less coolant flow rate and provide a larger convective heat transfer coefficient. Inert gases are preferred to reactive ones for safety and reliability.

6.2.4 Forced Liquid Cooling

Forced liquid cooling differs from forced gas cooling (Section 6.2.3) merely by the replacement of the coolant by a liquid. Liquid cooling normally requires a closed-loop arrangement and thus loses the simplicity of the air system. In most cases, liquid cooling systems require an accumulator to take up liquid expansion in the system, and a heat exchanger to reject the heat from the liquid coolant to the ultimate heat sink such as air or engine fuel.

However, liquid pumps are normally smaller than the fans for air because of incompressibility. Liquids are more readily adaptable when internal coolant passages prove beneficial because the passages can be made very small. In addition, heat transfer rates can be orders of magnitude higher than those in gas systems. When large heat loads are to be dissipated or when high power density (equipment compactness) is a requirement, liquid convective cooling should give a good lightweight system.

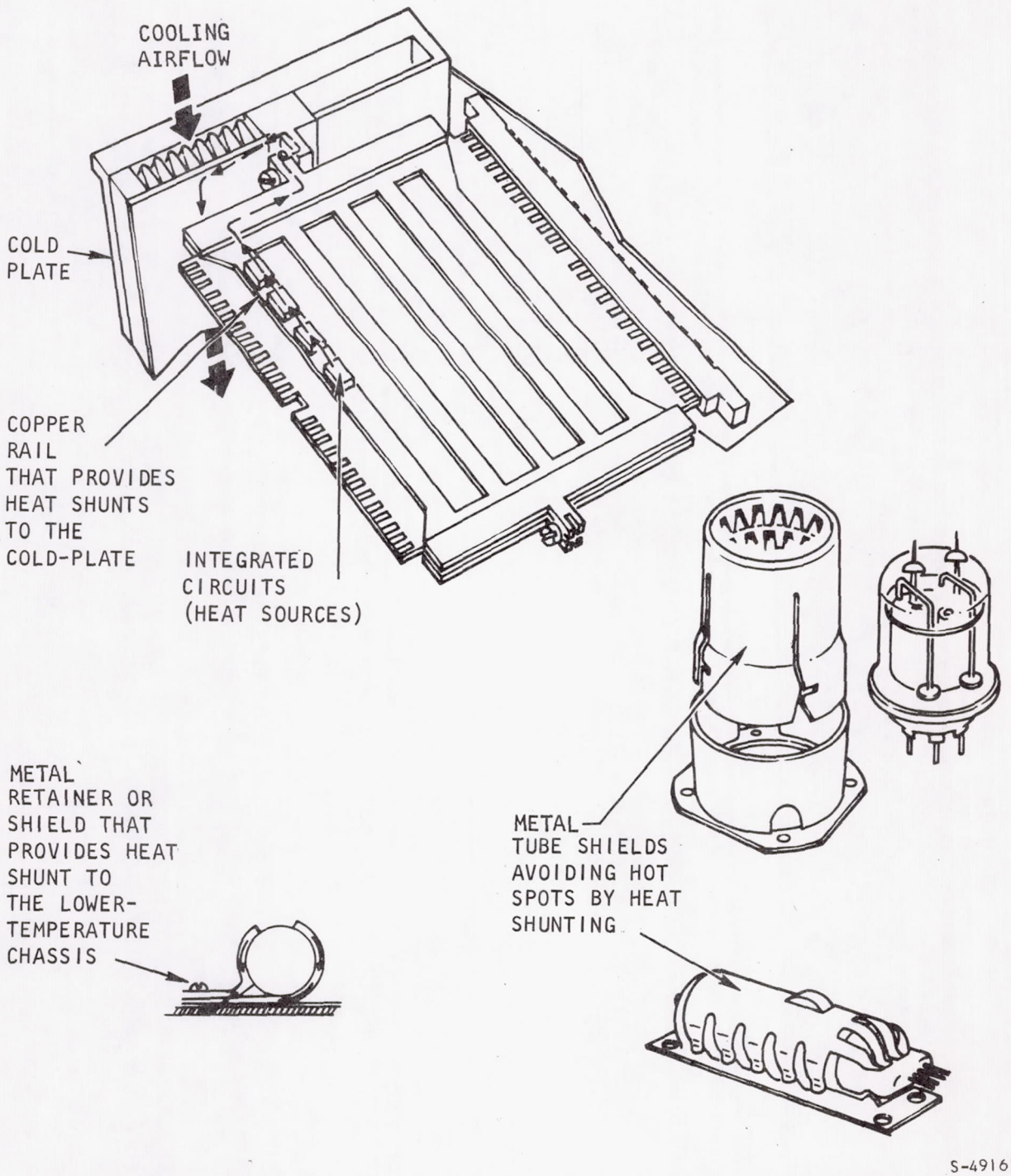


Figure 6-6. Cunduction (Heat Shunt) Concept and Its Applications

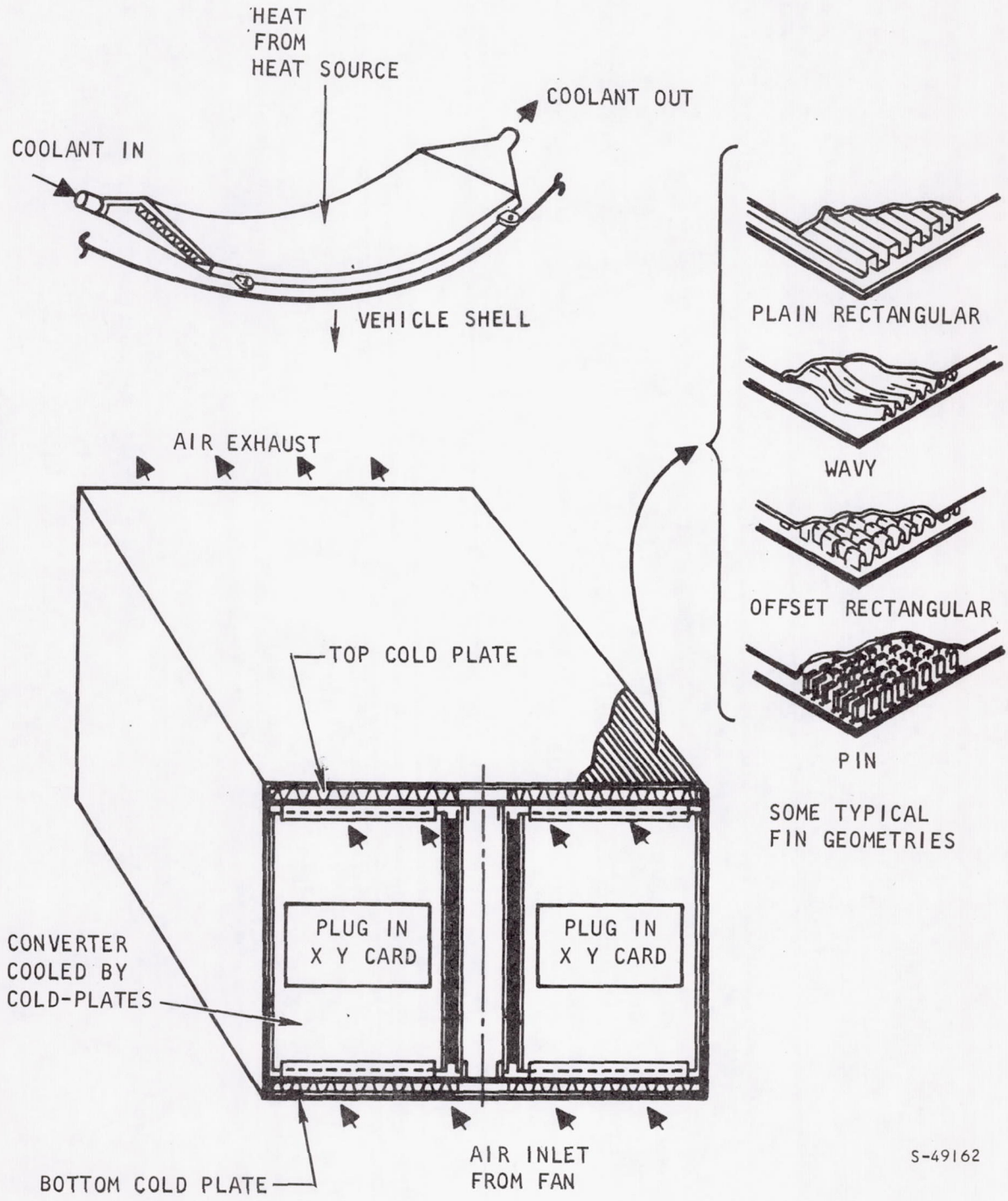


Figure 6-7. Forced Gas Cooling and Cold-Plate Concept

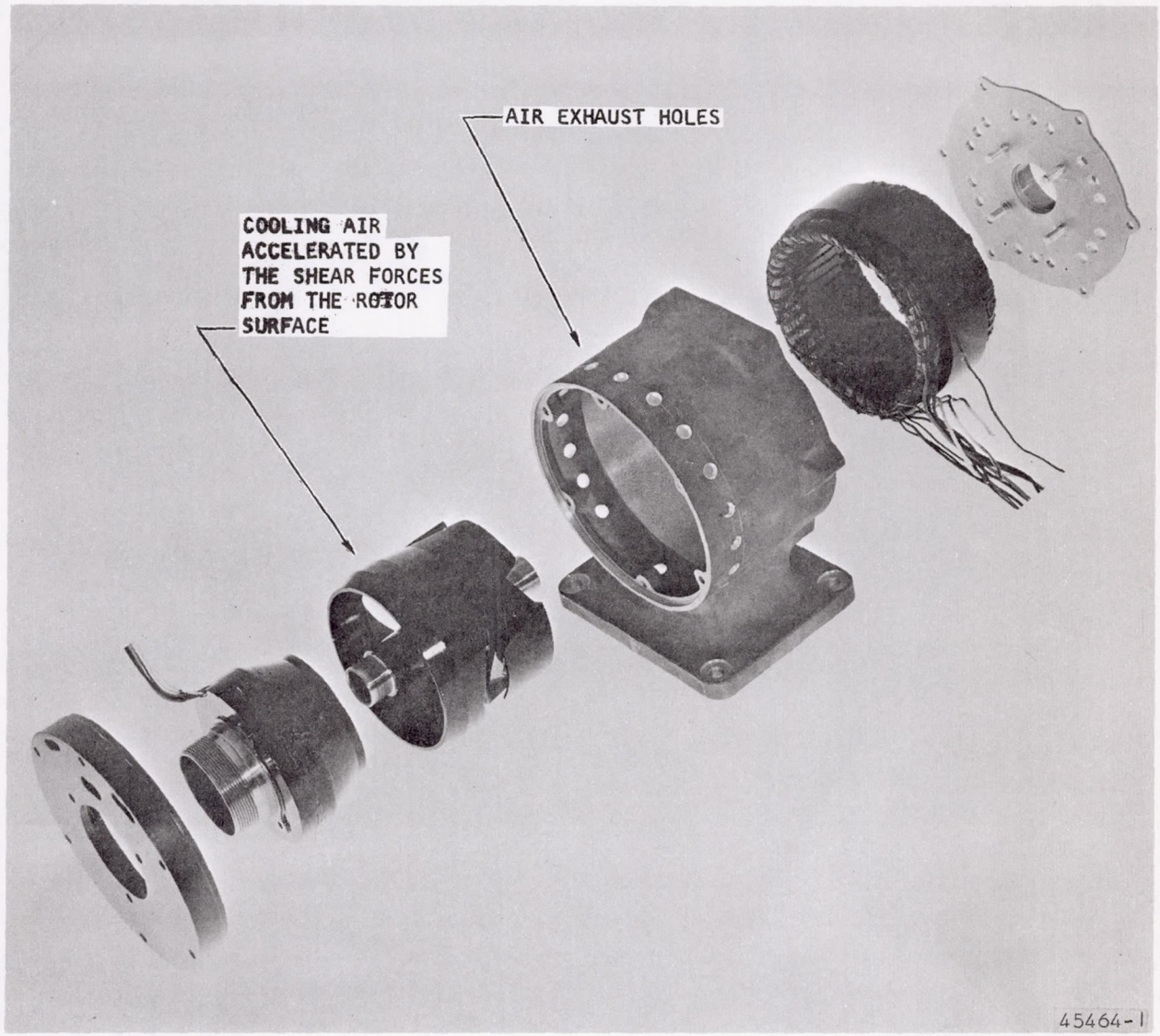


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Figure 6-8. Forced Convective Cooling by Air Accelerated by Shear Forces (Ram Air Turbine)

Figures 6-9 and 6-10 illustrate some techniques of using this method. Concorde's generators are also oil-cooled.

6.2.5 Thermal Attenuation (or Thermal Damping)

This is a technique that is often used to minimize the effects of the varying or oscillating temperatures of the environment, or to shield off the radiation and/or aerodynamic heating. A damper in its simplest form can be just a metal housing. But the optimum weight and the time lag are two items to be considered in design. Melting materials that absorb and reject the latent heat of fusion at a constant temperature give optimum thermal damping for some applications. Another application is to reduce errors in gas temperature measuring.

A damper that provides the constant temperature environment required for temperature-sensitive equipment like gyroscope in inertial guidance systems is shown in Figure 6-11.

6.2.6 Thermoelectric Coolers and Heat Pumps

This method or its equivalent is necessary whenever the temperature of the surrounding is higher than that of the allowable component temperature, and no nearby heat sink at lower temperatures is readily available. Under such circumstances useful energy is a required input. The application of thermoelectric devices is particularly suited to low heat loads like spot-cooling of individual electronic components or infrared detectors. Figure 6-12 shows two units designed for 100-w range.

6.2.7 High-Temperature Heat Exchangers

In high-speed flight or cruise-dash-cruise type of flight, ram air is often too hot to provide adequate cooling. Two approaches can be taken to solve this problem. One is the design and manufacture of high temperature heat exchangers (for air cycle refrigeration systems). Heat exchangers with metal temperatures higher than 1800^{0}F have been developed. Another approach is to incorporate nozzles in the heat exchanger to spray water or other liquids to cool the ram air down before it is used for other purposes.

6.2.8 Conclusions on Electrical Equipment Cooling Methods in Existing Aircraft

Figure 6-3 shows the relationships between the basic cooling methods and their heat-dissipation capabilities as normally encountered in electronic equipment. The method of boiling heat transfer is also included to obtain more combination of internal and external cooling modes for comparison.

Table 6-2 shows the approximate cooling capabilities of several methods that have been used successfully for electrical rotating machines. The numbers indicated are somewhat higher than the corresponding ones in electronic cooling of Figure 6-3. This is because in rotating machines higher temperature differences are normally feasible for cooling purposes.

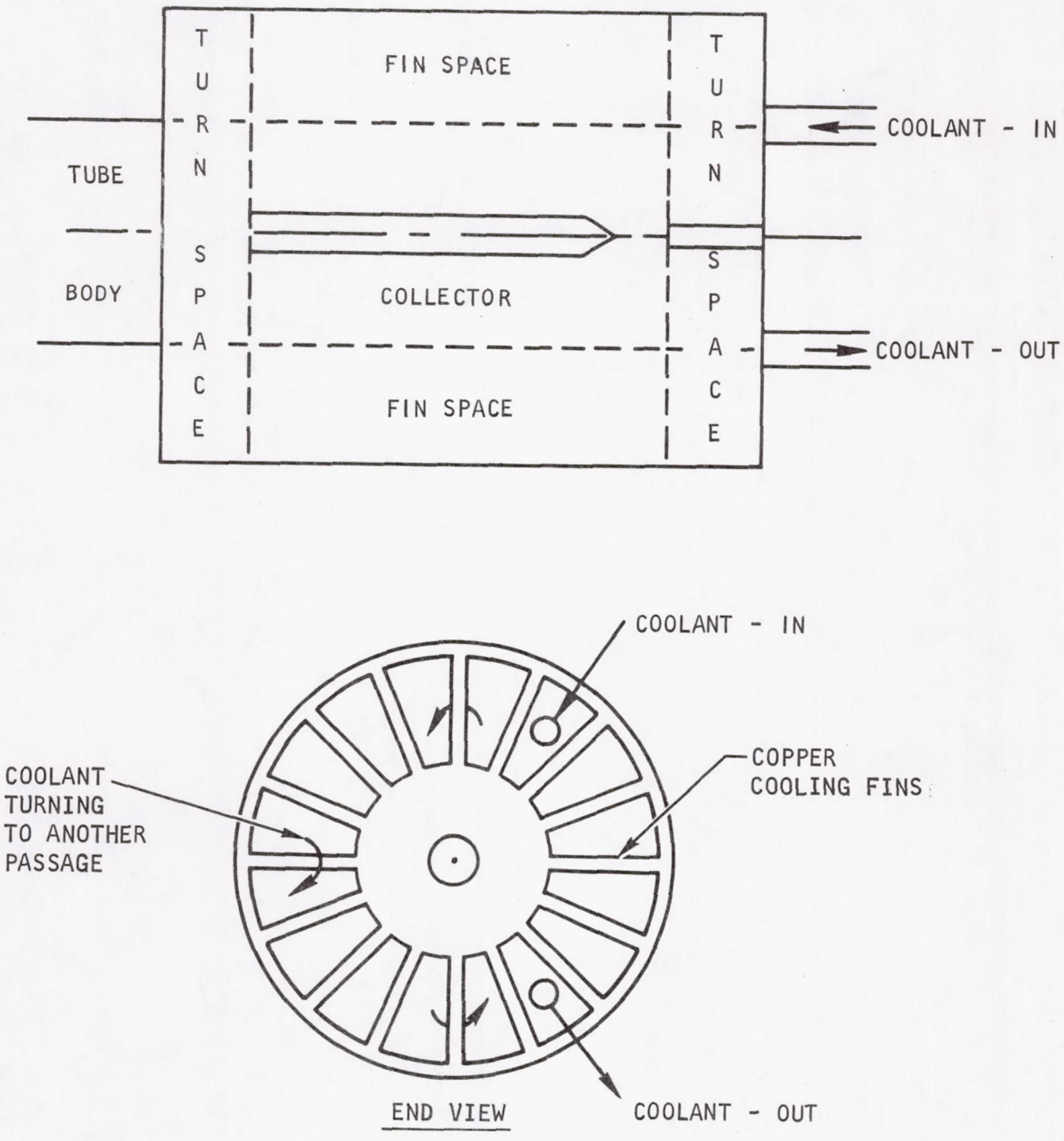

Figure 6-9. Forced Liquid Cooling for Traveling Wave Tube Collector

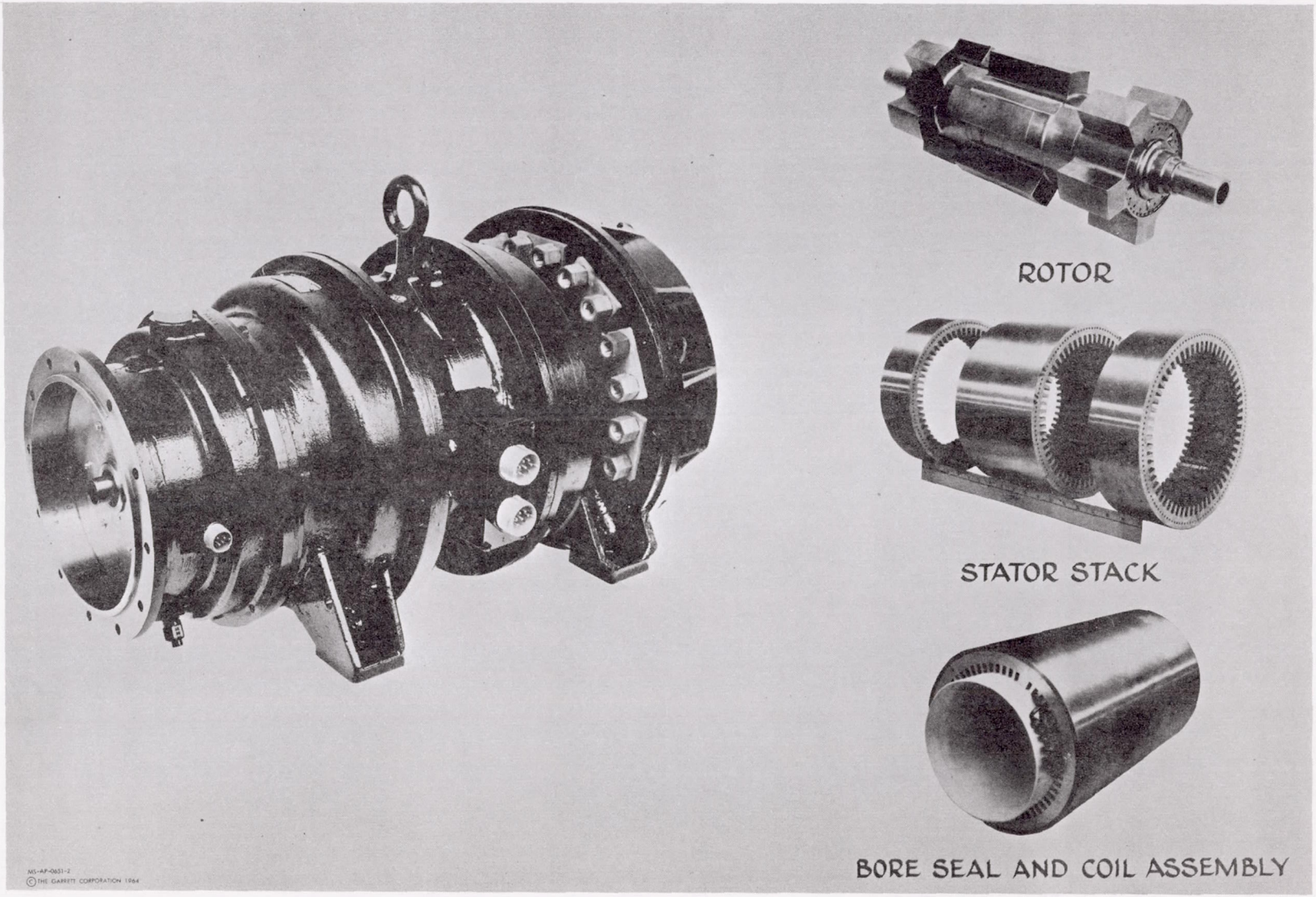

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Figure 6-10. Forced Liquid Cooling for a Homopolar Generator (by flowing oil through axial holes both in rotor and stator)

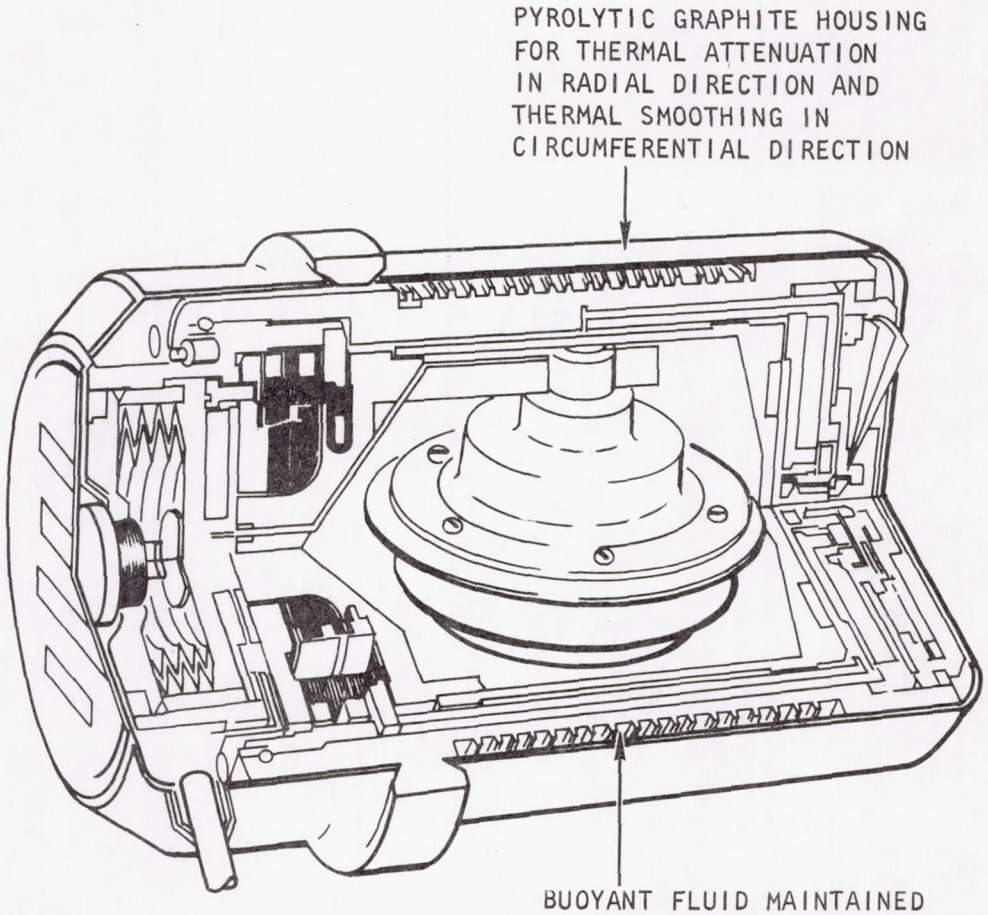
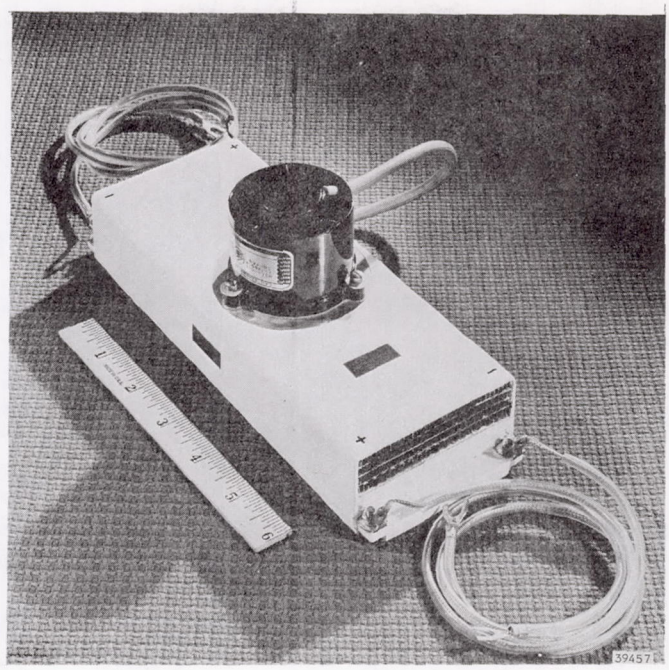


Figure 6-11. Cutaway View of a Gas Bearing Gyroscope Showing the Attenuation of Environmental Thermal Perturbation



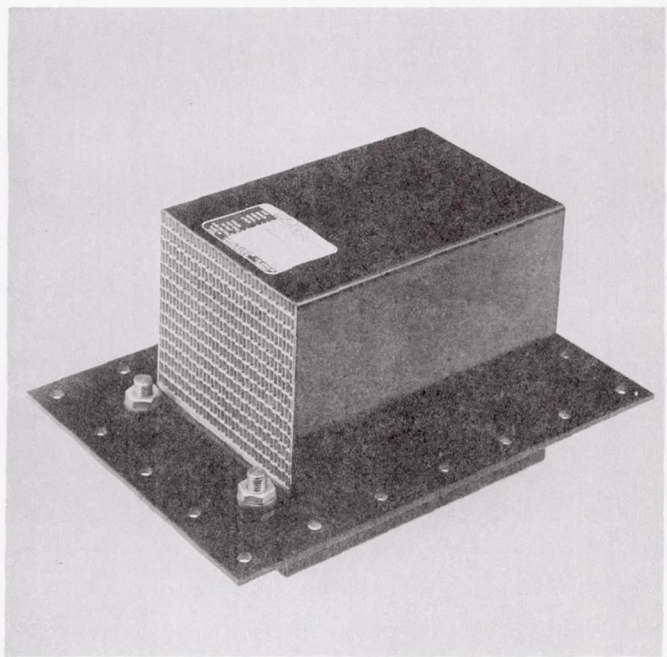


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Figure 6-12. Thermoelectric Coolers for 100-watt Cooling Loads

The factors that may influence the cooling capability of a particular application are so many that the heat dissipation capabilities shown in Figure 6-3 and Table 6-2 may only be regarded as a guide.

6.3 COOLING SYSTEMS IN EXISTING AIRCRAFT

A brief description of the cooling systems that have been used for different aircraft applications is given below.

They are arranged in the following order:

(a) Simple ram air cooling system (Section 6.3.1)

(b) Expanded ram air cooling systems (Section 6.3.2)

(c) Simple bleed air cooling system (Section 6.3.3)

(d) Bleed air regenerative cooling systems (Section 6.3.4)

(e) Bootstrap air cycle cooling systems (Section 6.3.5)

(f) Fuel cooling systems (Section 6.3.6)

(g) Expendable cooling systems (Section 6.3.7)

(h) Vapor-cycle cooling systems (Section 6.3.8)

(i) Cooling systems that are coupled with environmental control systems (Section 6.3.9)

This list consists of two ram air systems, three bleed air systems, and four others. These systems are selected for general considerations and may be impractical in some applications.

6.3.1 Simple Ram Air System

In this system, the gas or liquid in the transport loop transfers the heat from the thermal load to the ram air via the heat exchanger and the heated ram air is discharged overboard. A schematic diagram is shown in Figure 6-13.

This system is simple. It does not require bleed air from the engine and thus does not affect the engine performance.

When the aircraft speed becomes supersonic, a changeover to fuel as ultimate heat sink can be made or, alternatively, liquids like water can be injected into the ram air stream to reduce its temperature before it reaches the heat exchanger. The system may require supplementary cooling systems at low aircraft speeds and ground static conditions when sufficient ram air is not available.

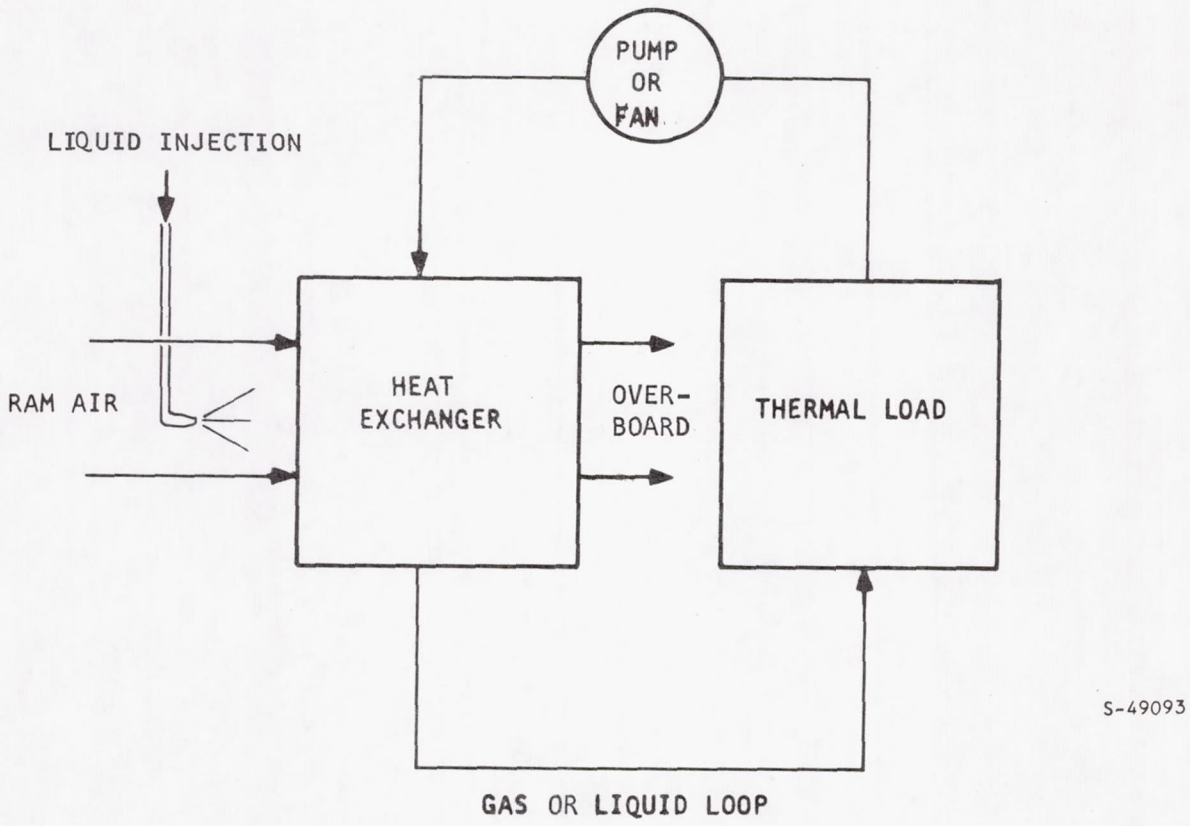

Figure 6-13. Simple Ram Air System

6.3.2 Expanded Ram Air System

Figure 6-14 shows a schematic diagram of this system which expands the ram air through a turbine and thus reduces the ram air temperature for cooling purposes. The ram air heat exchanger that is required in the simple ram system is replaced here by the cooling turbine and thus is suited for supersonic flight conditions.

The same advantages and disadvantages that are associated with the simple ram air system are applicable to this system, namely, it is simple, but at low flight speeds or ground static conditions supplementary cooling is a requirement. Another disadvantage is that the turbine, compressor, and ducting may become prohibitively large because of the low air densities involved.

6.3.3 Simple Bleed Air System

Figure 6-15 shows a schematic of a simple bleed air system. The bleed air from the engine is first cooled by ram air in a heat exchanger before entry into the water boiler. In the boiler, the air is cooled by the latent heat of vaporization of water, thereby further reducing the air temperature. The air then passes through the cooling turbine and water separator and finally reaches the equipment chamber. The turbine drives the compressor which draws ram air through the heat exchanger.

This system is easy to install from the standpoint of controls and ducting. It is simple mechanically and operates with the lowest turbine inlet pressures. A further advantage is that it uses high discharge temperatures and will therefore tend to be free from icing difficulties. This system can provide the cooling on ground.

If the aircraft Mach number is less than 2.0, a simple system is capable of doing the job, and there is no need for the water boiler. If the Mach number is greater than 2.0, a boiler becomes desirable, and if the Mach number is equal to 3.0 or more, the use of fuel as a heat sink is desirable.

If the cooling load variation is in the order of 5:1, the high point being at low altitude and high speed, then a simple system can be used since the bleed pressures for cooling are normally controlled to that ratio to avoid high-temperature bleed air and the cooling can be dealt with by various types of flow control devices. If the cooling requirements are higher at altitude than at sea level, then pressure regulation may be the best approach for control, or a variable area turbine may be adaptable.

6.3.4 Bleed Air Regenerative Systems

Two types of bleed air regenerative systems are shown in Figure 6-16. In this system the bleed air from the engine is first cooled by a regenerator, the cold side of which is the same bleed air, but after it has expanded through the turbine and picks up the electric or electronic heat loads. The cooled bleed air can be used to cool the equipment directly as in part (a). A closed gas or liquid coolant loop can be incorporated as in part (b). With the scheme shown in (a), a water separator may become necessary if moisture presents a problem in the application.

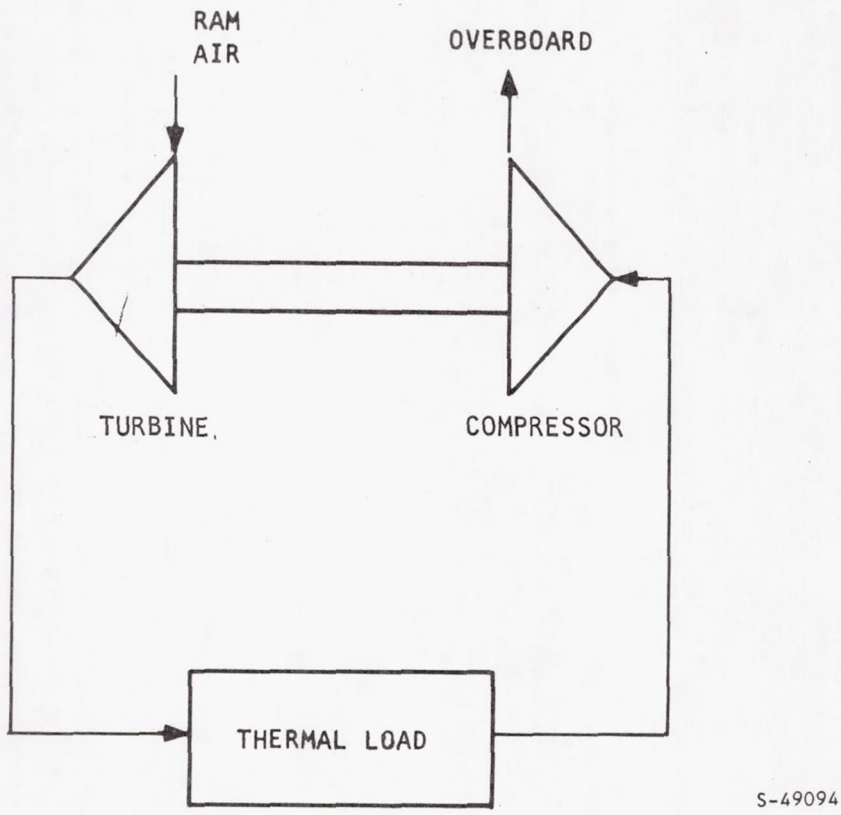

Figure 6-14. Expanded Ram Air System

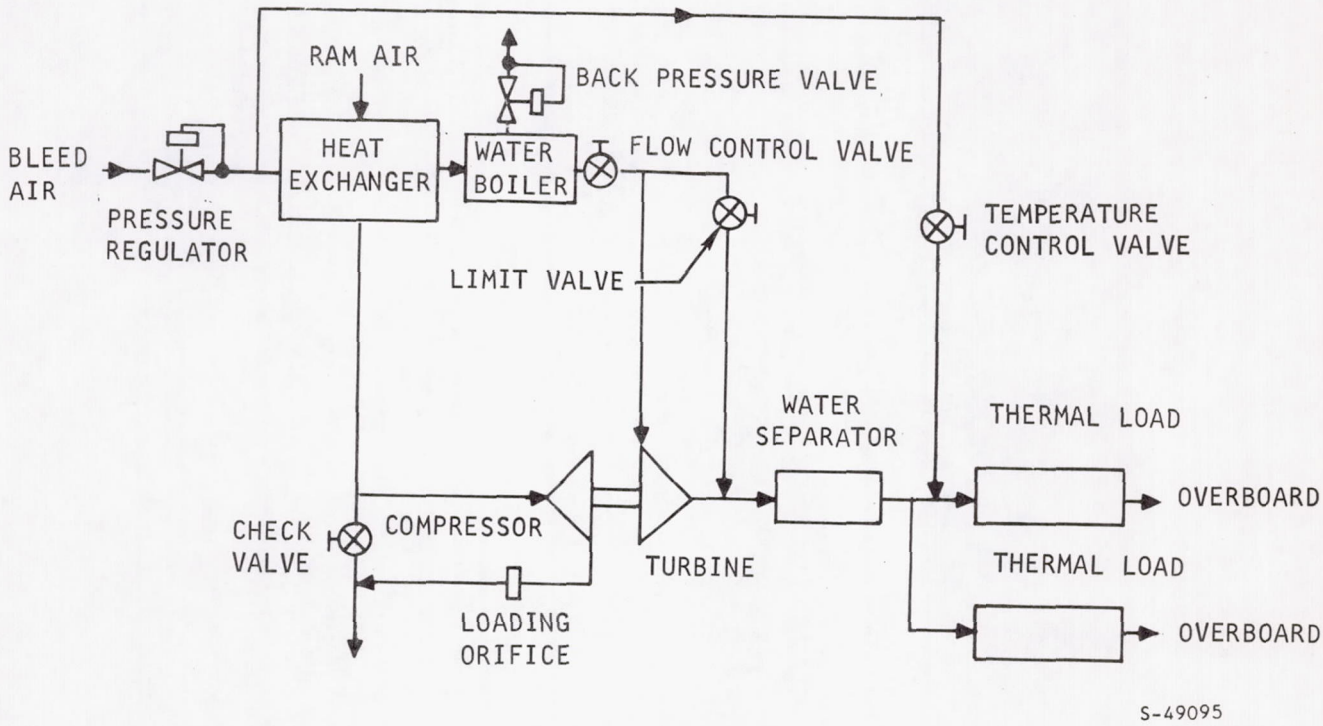

Figure 6-15. Simple Bleed Air System

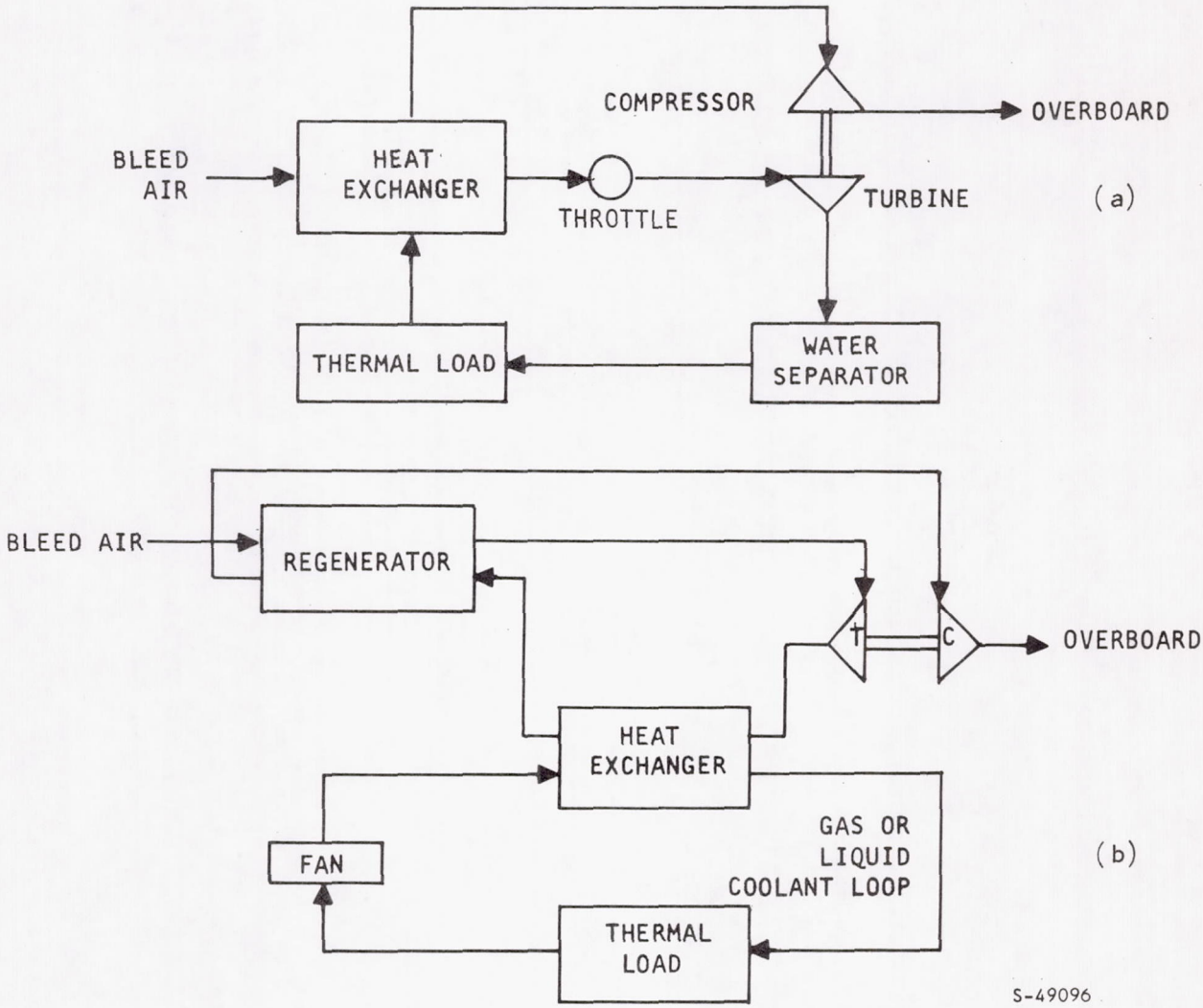

Figure 6-16. Bleed Air Regenerative System

This system eliminates the ram air cooling completely, and therefore is not restricted to subsonic flights. It has performance similar to that of the simple bleed air system and also provides cooling on ground.

6.3.5 Bootstrap Air Cycle System

If the cooling load is relatively constant (±20 percent) over the flight envelope of the aircraft, and if the air being used for cooling is at a constant supply temperature, then the mass flow required will be essentially constant with altitude. This would suggest the use of a bootstrap type of system with a constant gauge inlet pressure regulator, a schematic diagram of which is shown in Figure 6-17. This type of system, when sized properly, will provide a flow variation of approximately 20 percent under all flight conditions.

This system is more efficient than the simple bleed air system for a given amount of high pressure bleed air but is more complicated mechanically.

Fans or jet pumps may be put in the ram air stream for low speed conditions or cooling on ground.

In cases where low moisture content in the cooling air is a requirement, one high-efficiency water separator or two in series can be put in. In addition, a reheater which increases the air temperature beyond moisture-saturation but not over the required component ambient temperature of the electronic equipment can be incorporated.

6.3.6 Fuel Cooling Systems

Fuel cooling systems utilize transport liquids to transfer the heat from the equipment to the fuel via a heat exchanger as shown in Figure 6-18.

Since the same amount of fuel must be carried by the aircraft even with the electrical and electronic equipment generating zero heat, fuel is used as heat sink whenever it is feasible or safe to do so. A liquid-to-liquid heat exchanger is in general smaller than the air cycle heat exchanger, and the system has all the advantages of forced liquid convection and even boiling/condensing processes. The fuel cooling system may very likely be a lightweight system.

Fuel temperature usually rises during letdown at the end of each mission. Special thermal conditioning to the full storage environment or special cooling consideration to specific items of equipment may be required during this period.

Photographs of compact heat exchangers and electronic cooling packages are shown in Figure 6-19. These packages include the heat exchangers, fans or pumps, and necessary ducting.

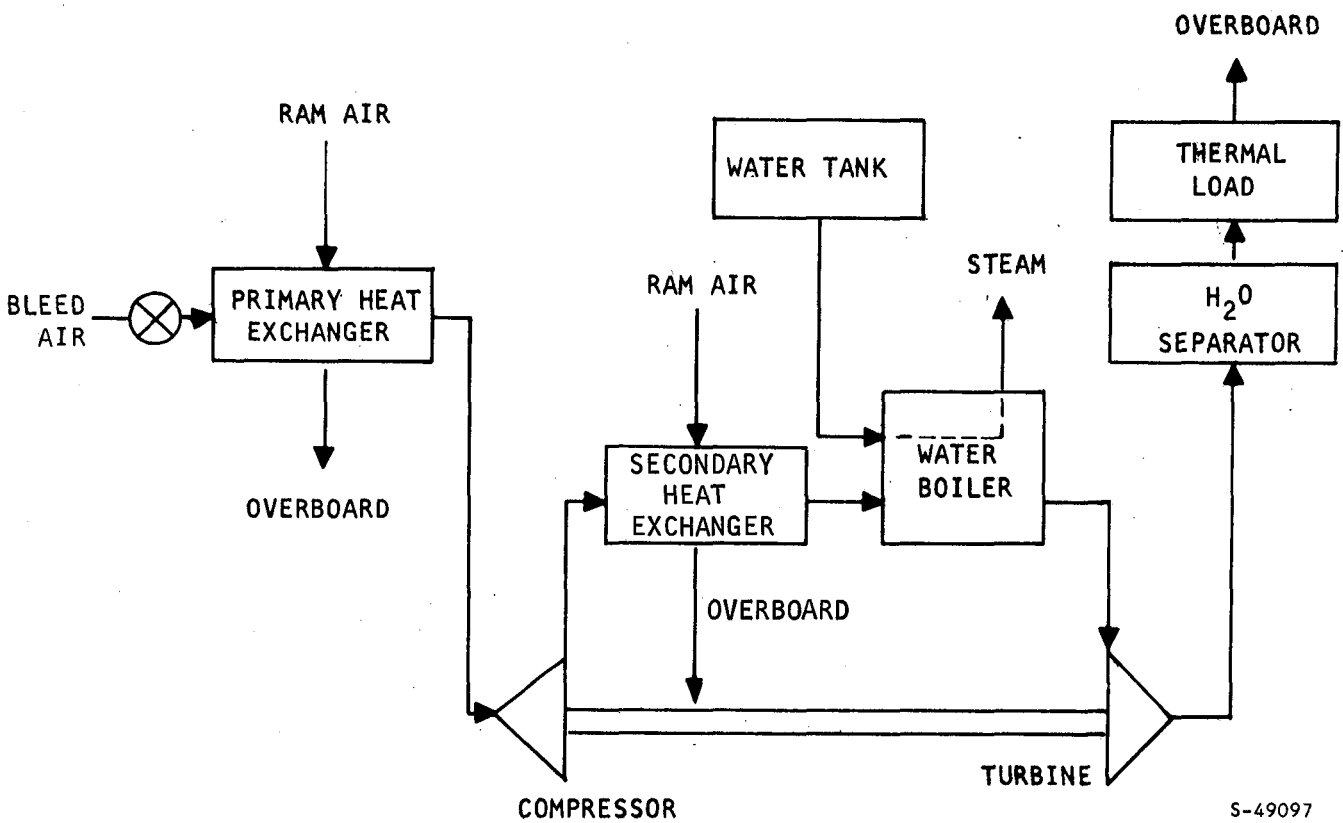

Figure 6-17. Bootstrap Air Cycle System

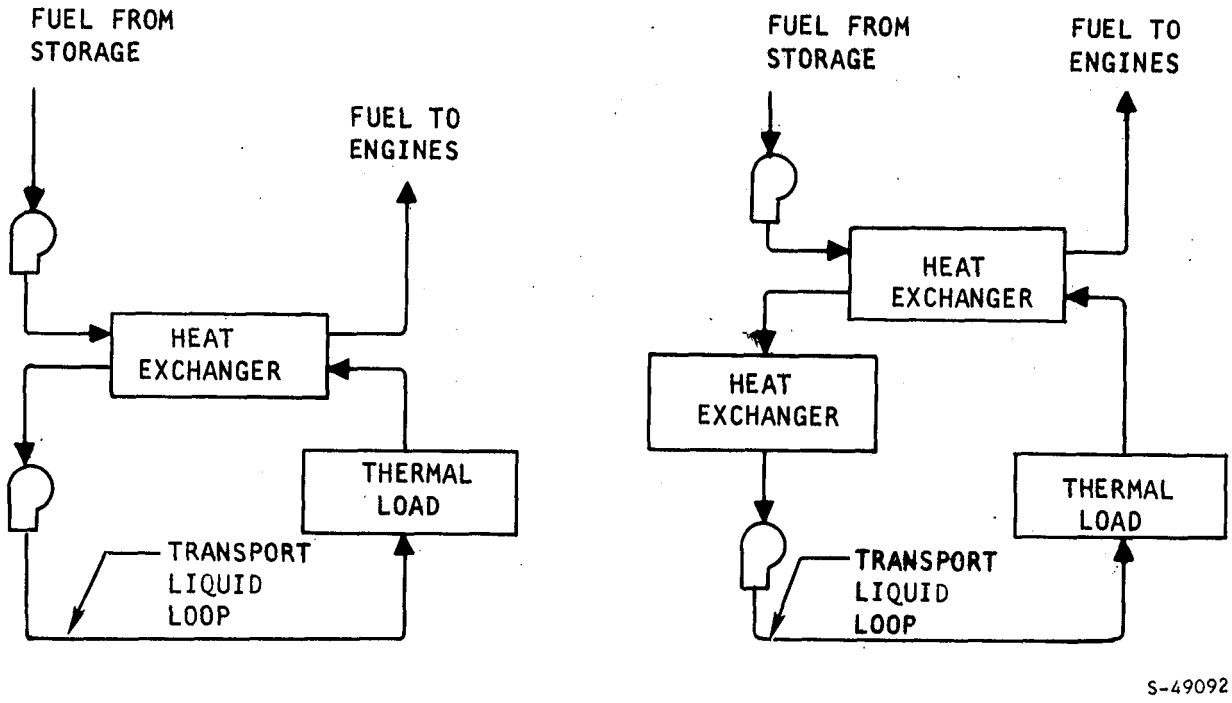

Figure 6-18. Fuel Cooling System

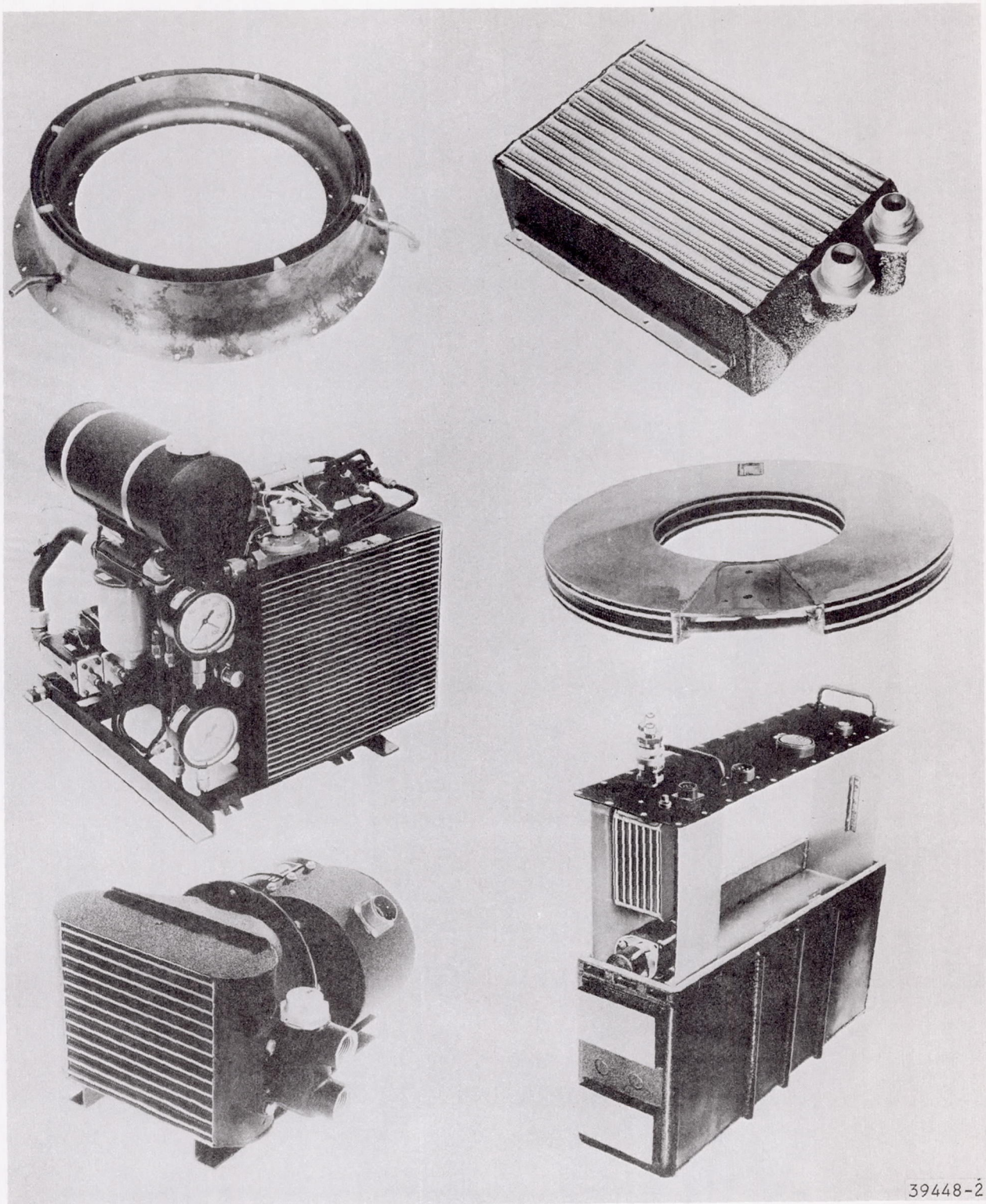

Figure 6-19. Cooling Packages Consisting of Fans, Heat Exchangers for Gas Loops and Pumps and Plumbing for Liquid Loops

6.3.7 Expendable Systems

The expendable systems make use of the large heat transfer rate of boiling process and the large heat of vaporization of liquids to cool the equipment down to a desired constant temperature by presetting the expendable pressure.

Two such systems are shown schematically in Figure 6-20. Pressure regulation of the expendables is normally a requirement. Pressurization of the transport loop (right figure of Figure 6-20) may be necessary when leakage is a possibility or the fan gets too big. The biggest drawback of this system is that the required amount of expendable may be prohibitively high for long supersonic flights. The expendable systems are most suited, therefore, when used in conjunction with other systems. It is particularly attractive when cryogenic temperatures are required, but the refrigeration plant is too much weight penalty to be practical.

Figure 6-21 shows two expendable system arrangements that can be applied to electronic cooling. Reservoirs can be used to extend the service time period as shown in part (a).

6.3.8 Vapor Cycle Systems

In essence the vapor cycle system uses the principle of refrigeration to provide the required cooling. One possible configuration is depicted in Figure 6-22 where a closed loop for the refrigerant (vapor loop), another closed loop for the transport fluid (liquid or air loop), and an open loop for the ultimate heat sink are shown. In this particular arrangement the ultimate sink is shown as a combination of ram air and expendables.

This system is especially good when bleed air or low-temperature ram air is not accessible for cooling purposes, but shaft or electrical power is available. Small cooling loads (in the order of 1 ton) could be handled by shaft-drive air-cycle units and larger loads by hydraulically driven units.

One variation of this method is shown in Figure 6-23. It represents basically the elimination of the transport loop. The refrigerant makes direct contact with the equipment to be cooled for maximum cooling efficiency. In this arrangement the blower replaces the compressor, the spray nozzle replaces the expansion valve, and the electronic compartment replaces the evaporator in a typical vapor cycle system.

Still another variation of this system is one described by Drexel* and shown schematically in Figure 6-24. The whole vapor cycle is carried out within the box itself without any ducting. The refrigerant (FC-75, in this example) is evaporated by the heat generating component at the bottom of the box, the vapor rising to the cooled cover by diffusion process (mass transfer due to density difference). The vapor condenses at the cover and transforms into liquid droplet form and is returned to the bottom by gravity forces.

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*Drexel, W. H., "Fluorochemical Cooling for High Heat Dissipation," Electronic Design, May 24, 1961

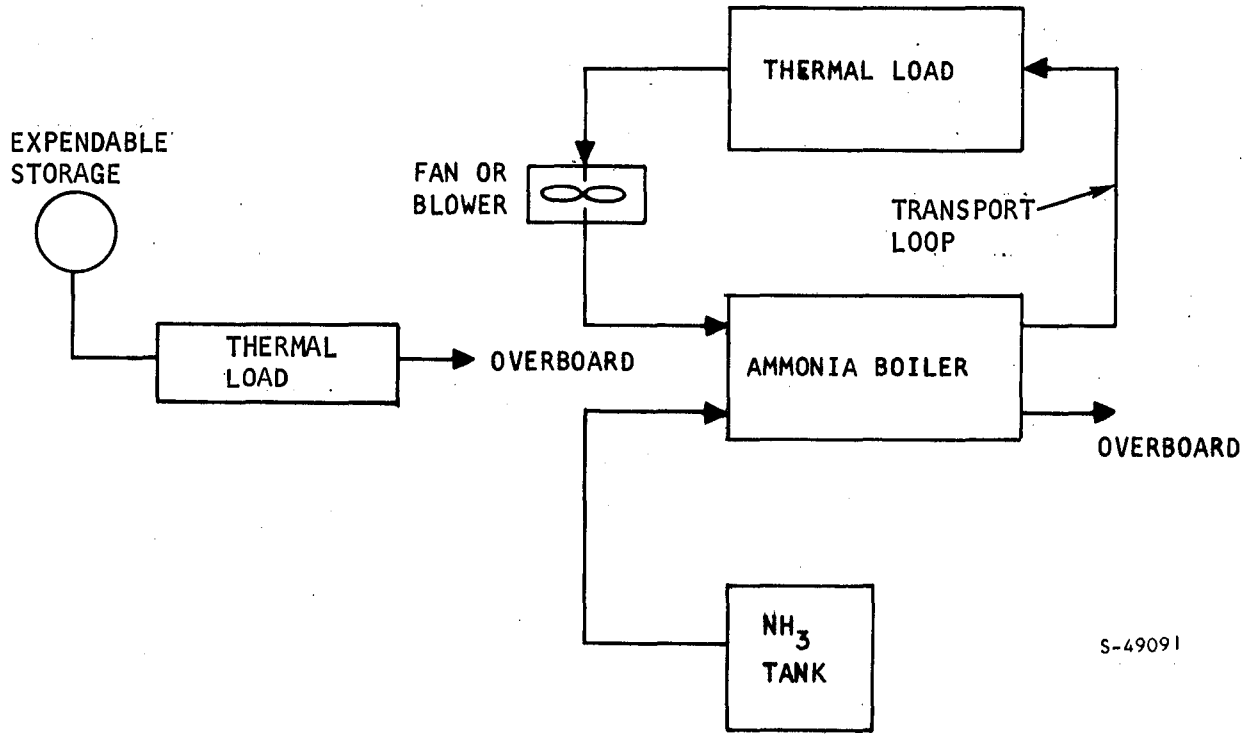
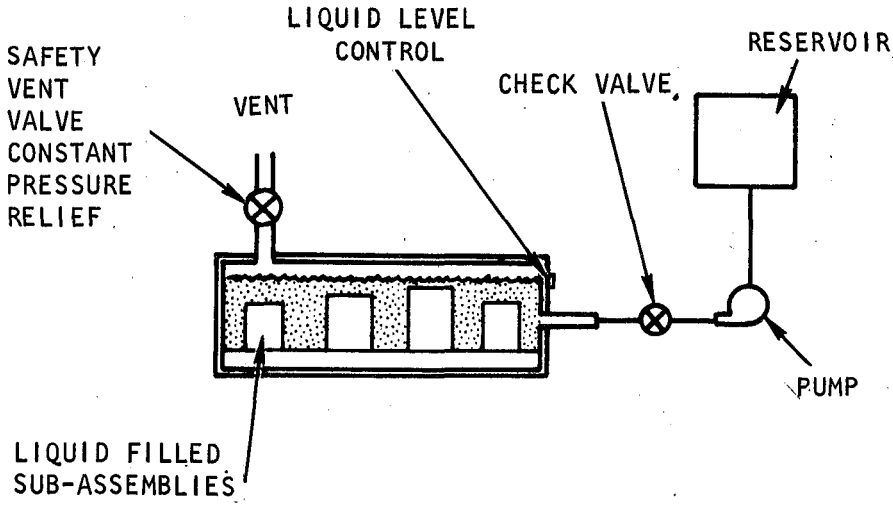
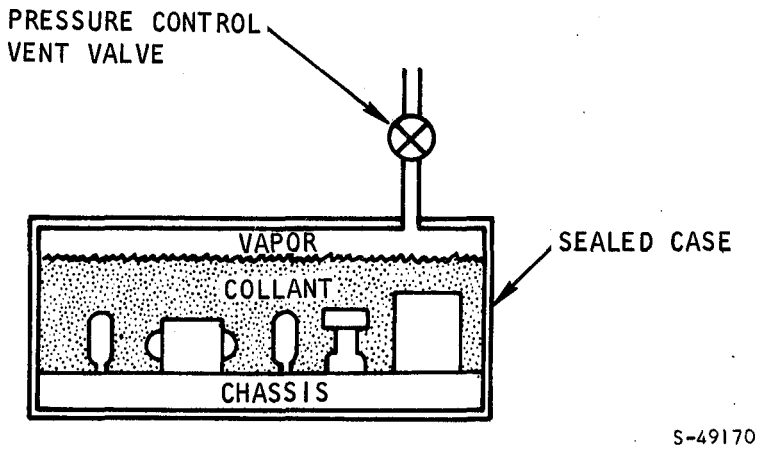


Figure 6-20. Expendable Systems



(a)



(b)

Figure 6-21. Direct Expendable Vaporization Cooling System

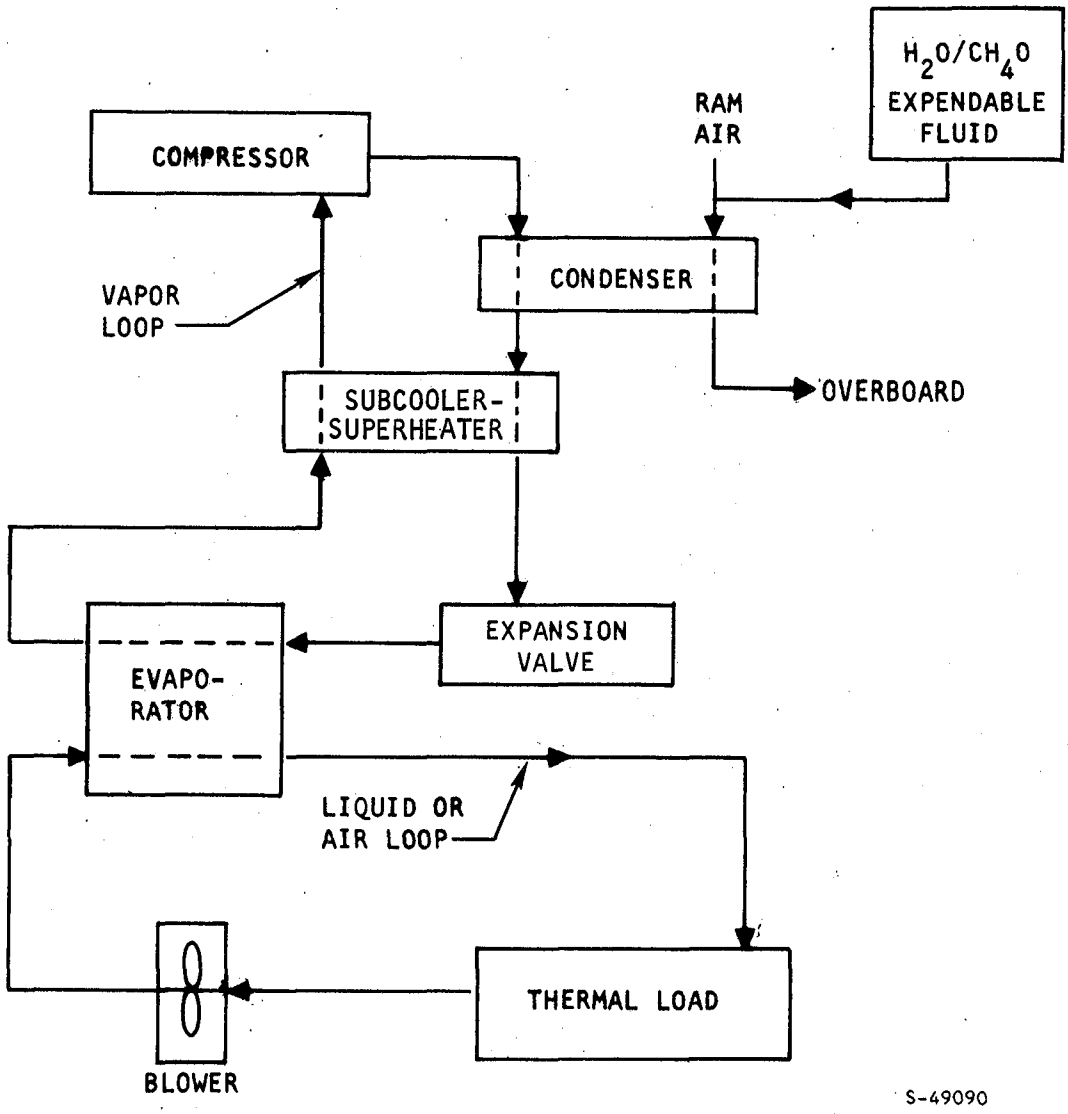


Figure 6-22. Vapor Cycle System

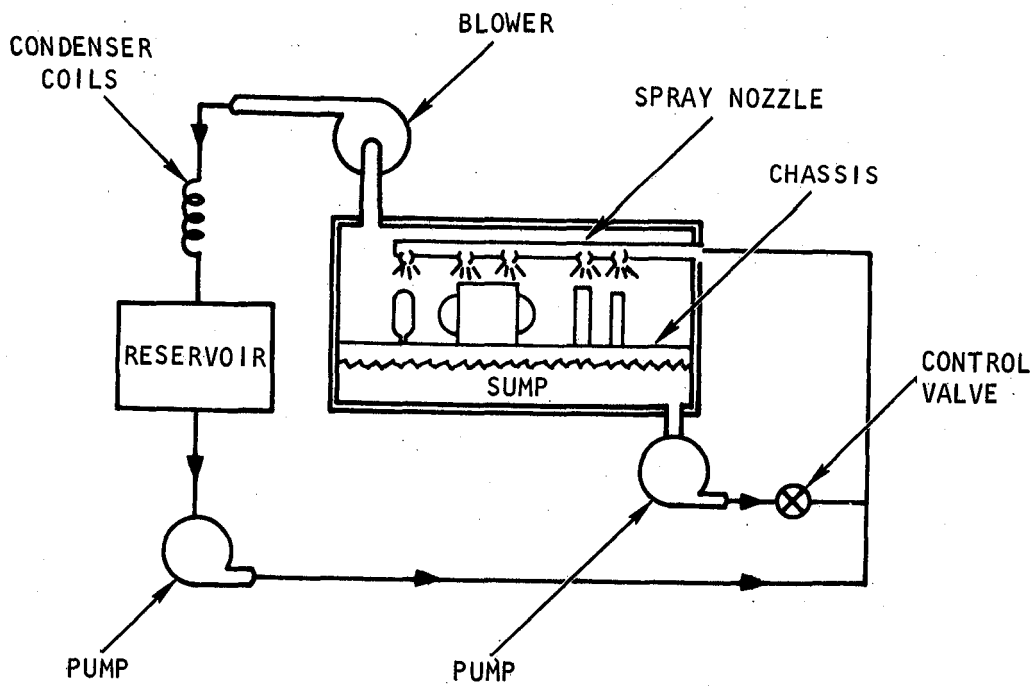


Figure 6-23. Electronic Boxes Cooled by Evaporation and Condensation (A Direct Evaporative Spray System)

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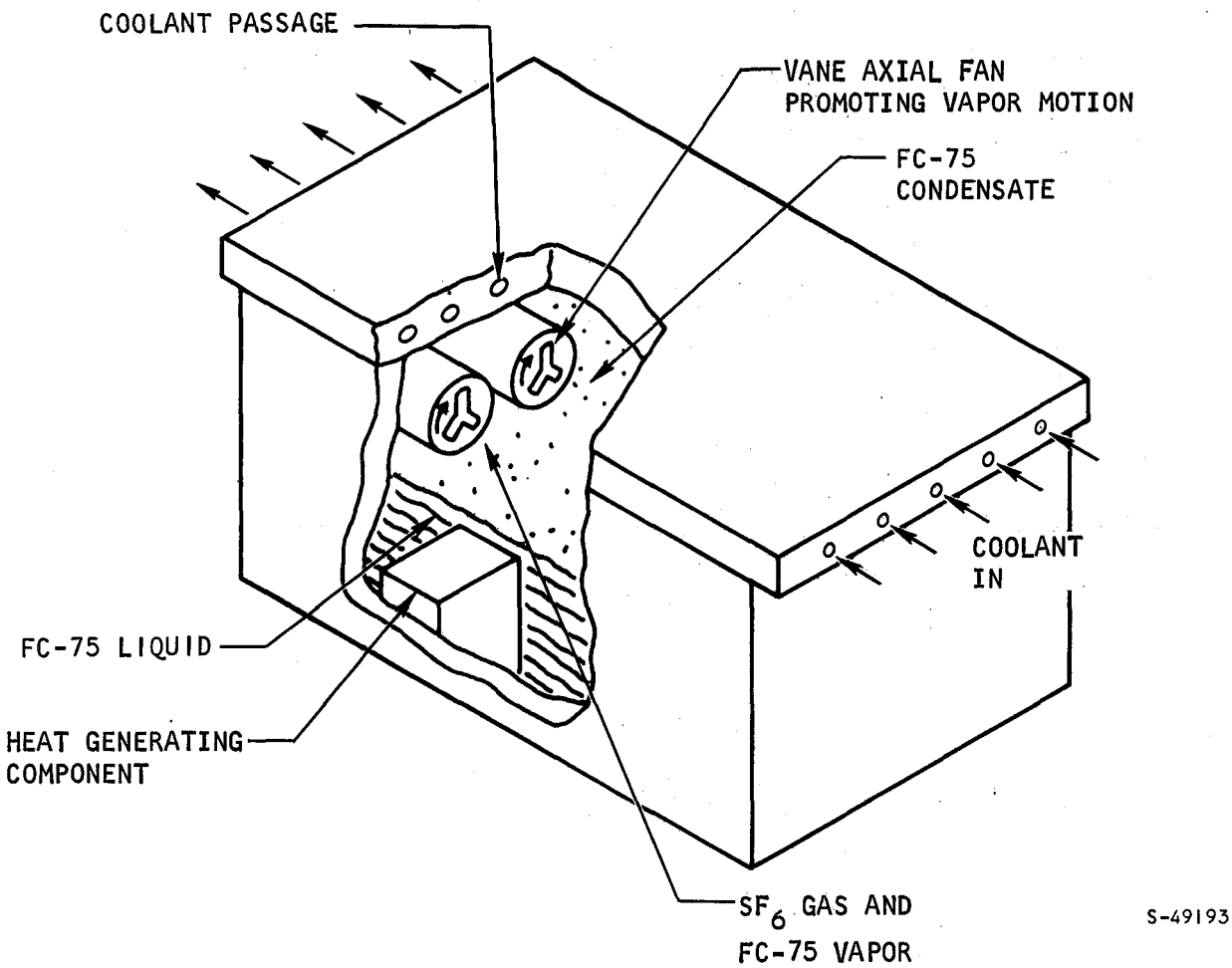


Figure 6-24. Evaporation-Condensation Cooling System for High Voltage Power Supply

In this particular application, SF_6 (sulphur hexafluoride) was used for its high dielectric strength at low temperature (down to -65°F) and vane axial fans were used to promote heat transfer at the cover.

6.3.9 Electric Cooling System Coupled with Environmental Control System

The biggest advantage of coupling electrical and electronic cooling with the cabin environmental control system (ECS) is that the main ducting of the air distributing system is already in existence and all that is necessary in order to incorporate electrical cooling is to resize the ECS to include the additional cooling loads and to provide the extra ducting needed to branch off the cool air to the particular locations.

As shown in Figure 6-25 the cooling air can be taken either before or after it enters the cabin. The weight penalty in such a modification can be very small as compared to the whole environmental control system.

6.3.10 Conclusions on Cooling Systems in Existing Aircraft

It is highly improbable that one single cooling system will be optimum for all flight conditions. A combination of several systems and cooling techniques, e.g., bootstrap air cycle system for environmental control, liquid loop to fuel for electrical cooling plus cryogenic expendables for a few components and thermoelectric spot-cooling for some microcircuits, may turn out to be the lightest weight system for a particular aircraft for a specific mission.

Again the computer program (H-0610) mentioned previously may prove to be very useful in the optimization procedure. A sample summary output for a 160-kw thermal load and 5.5 engine compression ratio from this program is shown on page 6-37. It is a map showing the minimum weight system for each combination of Mach number and altitude. For example, at Mach number of 1.5 and altitude of 60,000 ft, the minimum weight system indicated by a circle would be expendable evaporants (expendable system abbreviated as VAP). The approximate mission profile for an F-111 aircraft is plotted overlapping the computer outputs, showing the best solution to every flight condition for the complete mission. As it is indicated in the output, the minimum weight cooling system during takeoff would be a simple bleed air system (SIM). For climb, cruise, and descent, either an expendable system (VAP) alone or an integration of simple ram air system with a liquid transport loop (LIQ), expendable system (VAP), and bleed air regenerative system (REG) could be used.

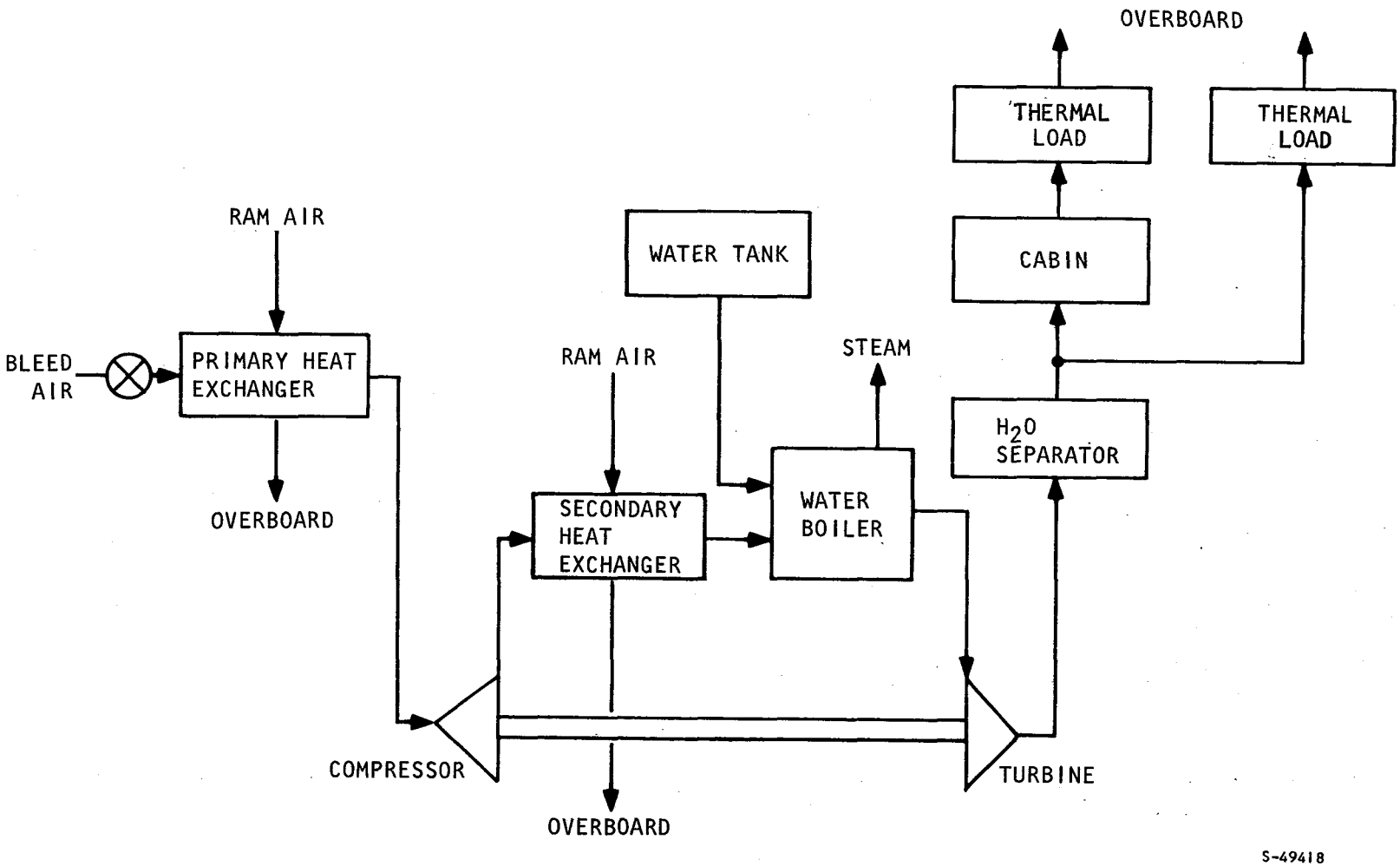
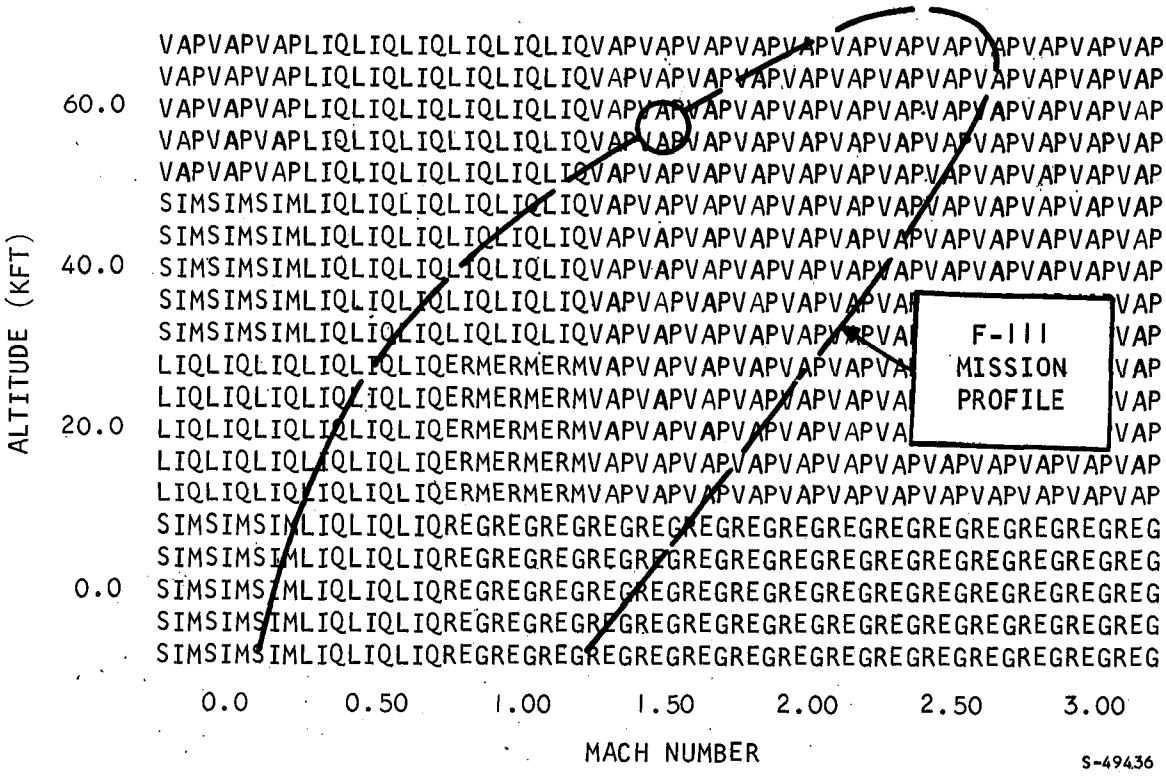


Figure 6-25. The Coupling of Electrical Cooling System with the Environmental Control System (Bootstrap Air Cycle System in this Case)



OUTPUT DATA PLOT OF MINIMUM WEIGHT SYSTEMS

6.4 POSSIBLE IMPROVEMENTS ON COOLING METHODS USING CURRENT TECHNOLOGY

The eleven items listed below are selected from a viewpoint of possible application to aircraft cooling designs. The selection is subjective as in Section 6.3.

(a) Transient state cooling by thermal capacity (Section 6.4.1)

(b) Impingement cooling and special techniques of convective cooling (Section 6.4.2)

(c) Vortex (Hilsch) tubes and other miniature refrigerators (Section 6.4.3)

(d) Thermal shielding (Section 6.4.4)

(e) Cooling by phase-change (Section 6.4.5)

(f) Heat pipes (Section 6.4.6)

(g) Cooling by expendables (Section 6.4.7)

(h) Cryogenic cooling (Section 6.4.8)

(i) Improved heat rejection (Section 6.4.9)

(j) Internal heat ducting from hot spot (Section 6.4.10)

(k) Thermal network synthesis (Section 6.4.11)

6.4.1 Transient Cooling by Thermal Capacity

When the equipment is used for only a short duration, the equipment can be designed to take care of its own heat generation by providing sufficient thermal capacitance (product of mass and specific heat). Its energy-storing capability plus the natural convection and radiation to its surroundings can maintain the maximum temperature for the whole duration below a required level.

One concern about this approach is the existence of internal hot spots. An overall lump-parameter analysis does not always provide this information. The absence of hot spots is ensured by proper conduction-shunting from point of heat generation to area of greatest thermal capacitance as discussed in Section 6.2.2.

Figure 6-26 shows a turboalternator set designed on this principle.

6.4.2 Impingement Cooling and Special Techniques of Convective Cooling

Some additional design features of convective heat transfer that can be applied to aircraft electric cooling are shown in Figures 6-27, 6-28, and 6-29. They represent the considerations of high heat transfer coefficients of an impinging jet (Figure 6-27), direct contact of coolant and heat source for best efficiency (Figure 6-28), or utilizing the centrifugal field of rotating machinery (Figure 6-29).

Figure 6-30 shows some typical convectively-cooled rotating electric machines design features.

Power cables can be cooled by an arrangement depicted in Figure 6-31 where a concentric outer conduit contains the coolant that flows along the cables. The spacers keep the cables from contacting each other, thus providing the maximum surface area for convection cooling. Additional surfaces can be provided by connecting the cables with the conduit, and then connecting the conduit with the structure by good conduction paths.

6.4.3 Vortex (or Hilsch) Tubes and Other Miniature Refrigerators

A vortex tube is an expansion device that utilizes the energy and momentum transfer between concentric gas-layers in a cylinder to produce a lower-temperature core as the gas stream spirals down the tube. The high pressure gas is introduced into the tube in a tangential direction as shown in Figure 6-32. The energy and momentum transfer takes place as the gas layers tend toward constant angular velocity.

The temperature difference that is obtainable in a vortex tube is in the range of 100^{0}F if properly designed. This device, although not efficient compared with some other devices, has the big advantage of its small size. In most cases the tube can be located adjacent to the point of use. The unit has no moving parts and can be located in inaccessible places.

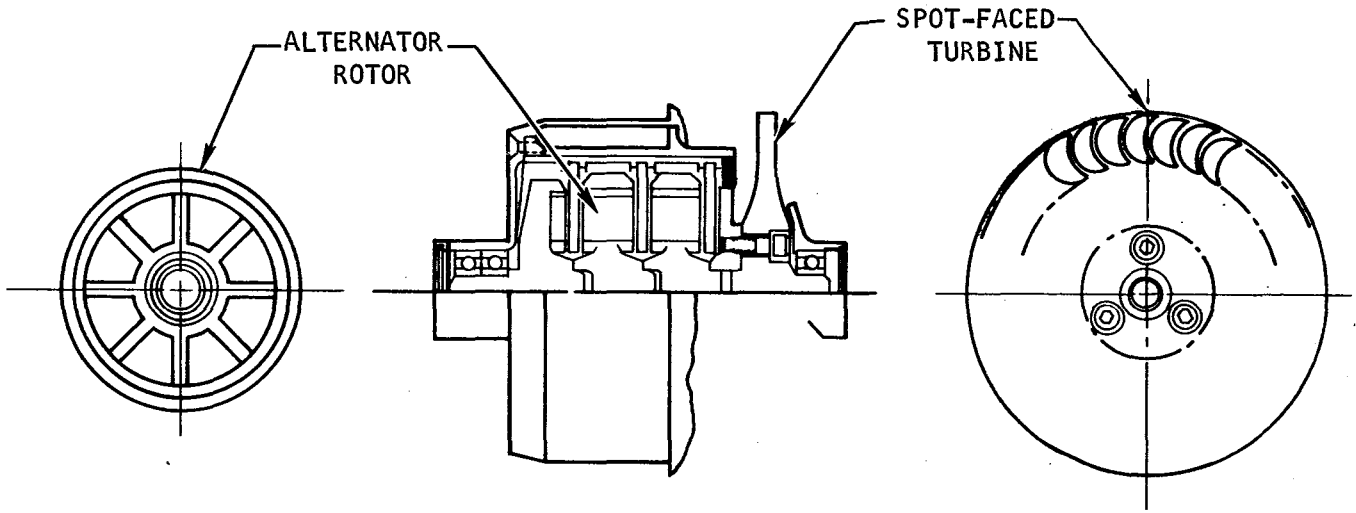


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Figure 6-26. Good Transient Thermal Behavior Provided by the Thermal Capacity of a Turboalternator Set

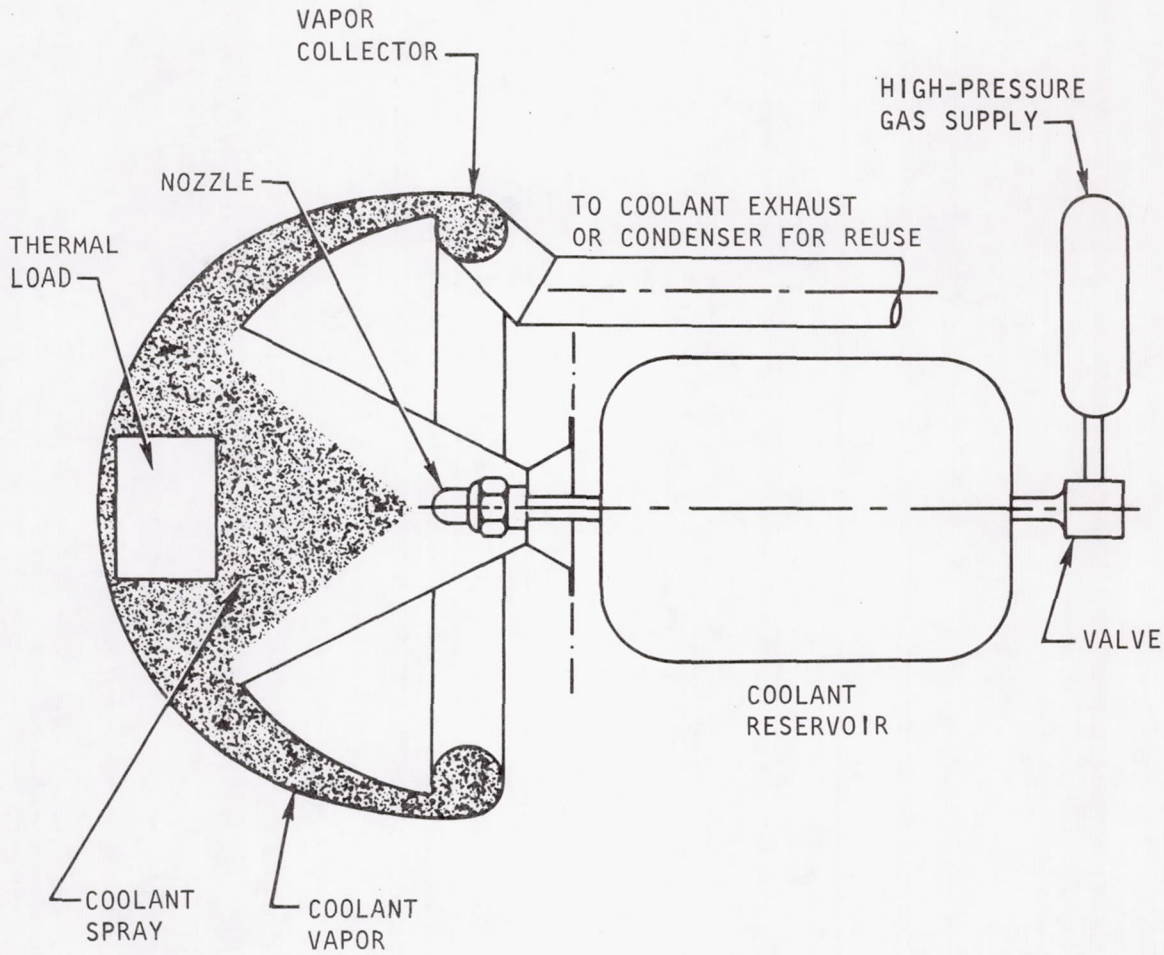


Figure 6-27. Impingement Method Shown as Liquid Spray Cooling System (The same method can be applied to gas systems)

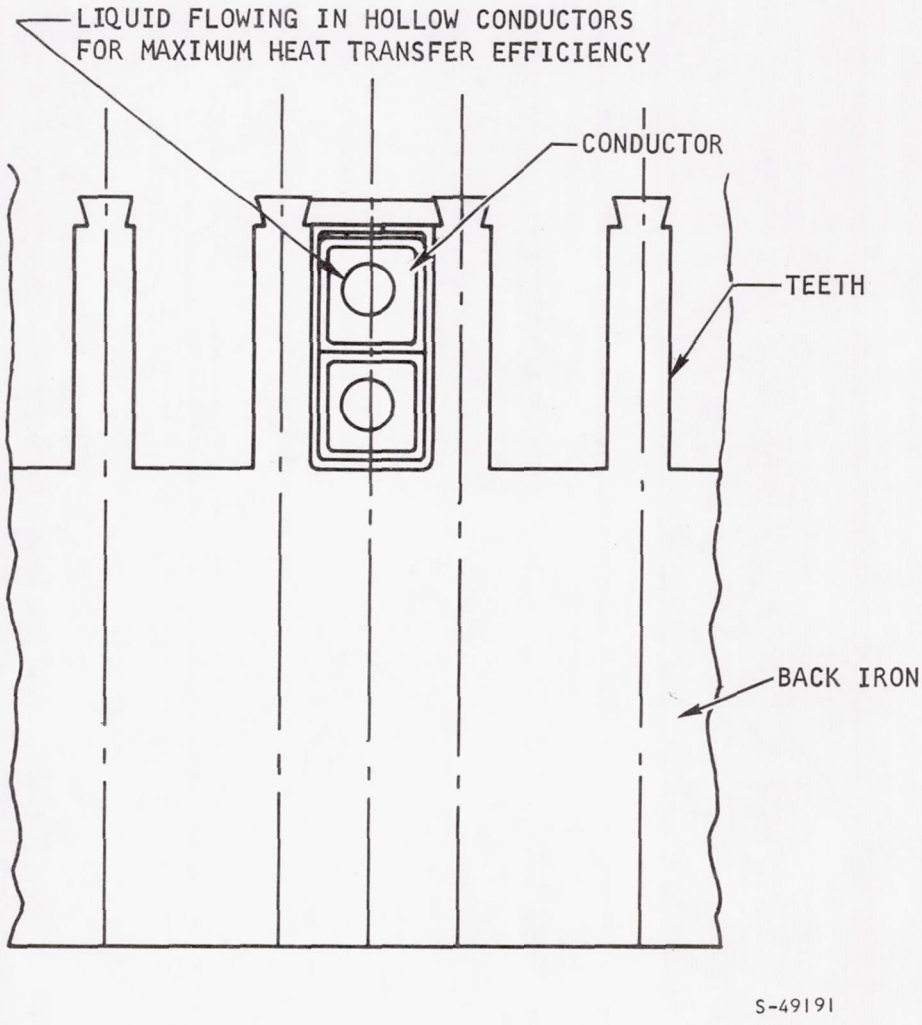


Figure 6-28. Forced Liquid Cooling for Electric Machines

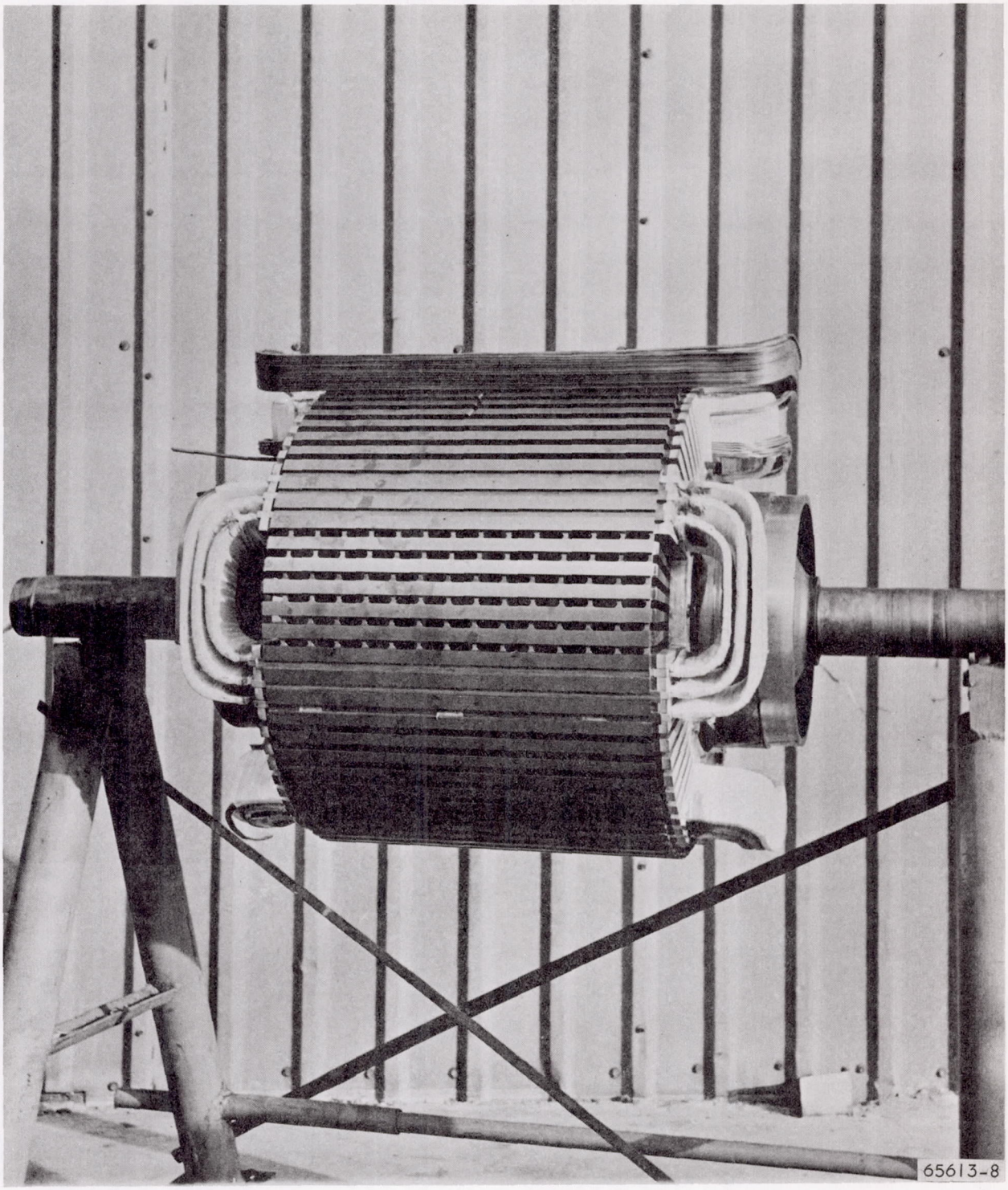
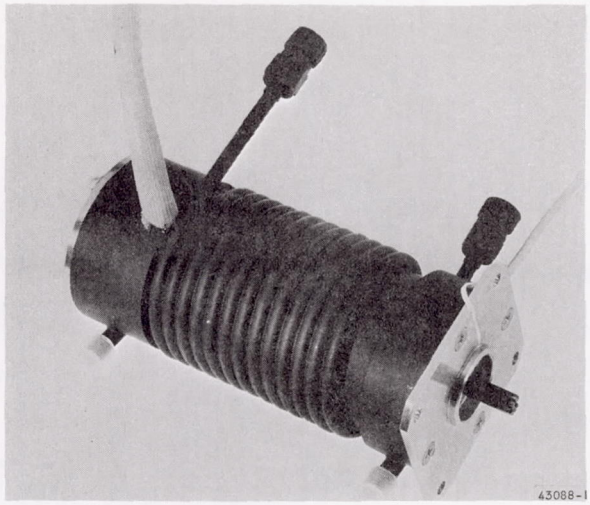
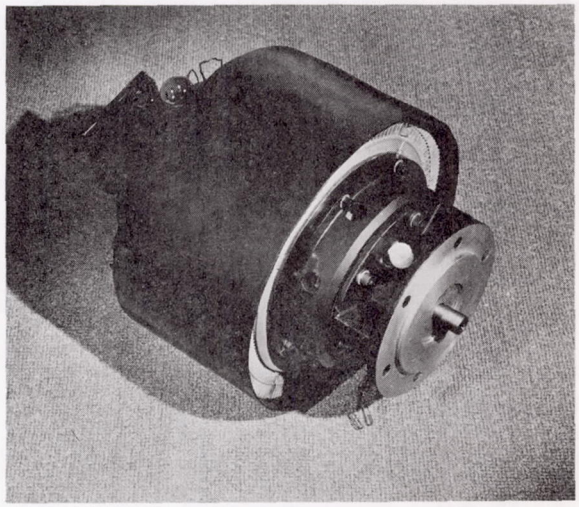


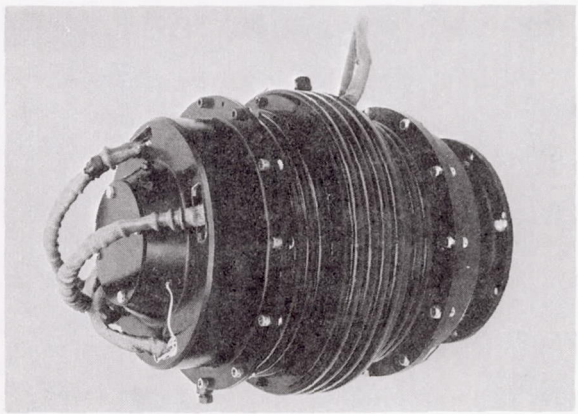
Figure 6-29. Void Slots on the Rotor of an Electric Machine Generating Centrifugal Field for Air Cooling



LIQUID-COOLED CAGE-ROTOR MOTOR, 5 HP, 62,000 RPM, 3200-CPS



AIR-COOLED HOMOPOLAR ALTERNATOR, 7.5 KW, 48,000 RPM



LIQUID-COOLED HOMOPOLAR INDUCTOR ALTERNATOR,
95 KVA, 3400 CPS, 40,800 RPM

F-10909

Figure 6-30. Typical Air-Cooled and Liquid-Cooled Rotating Machines

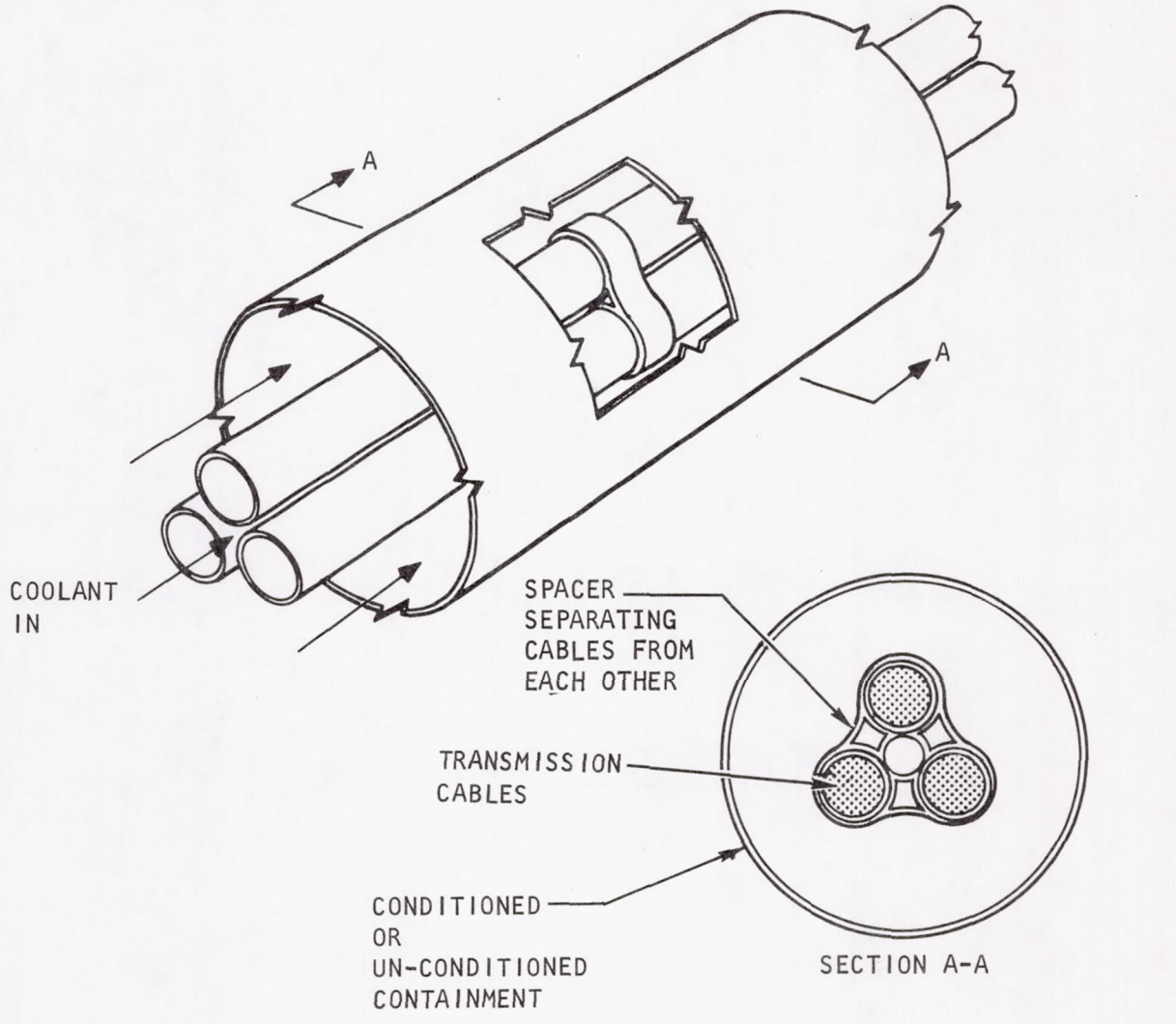


Figure 6-31. Transmission Cables Cooled by Forced Convection

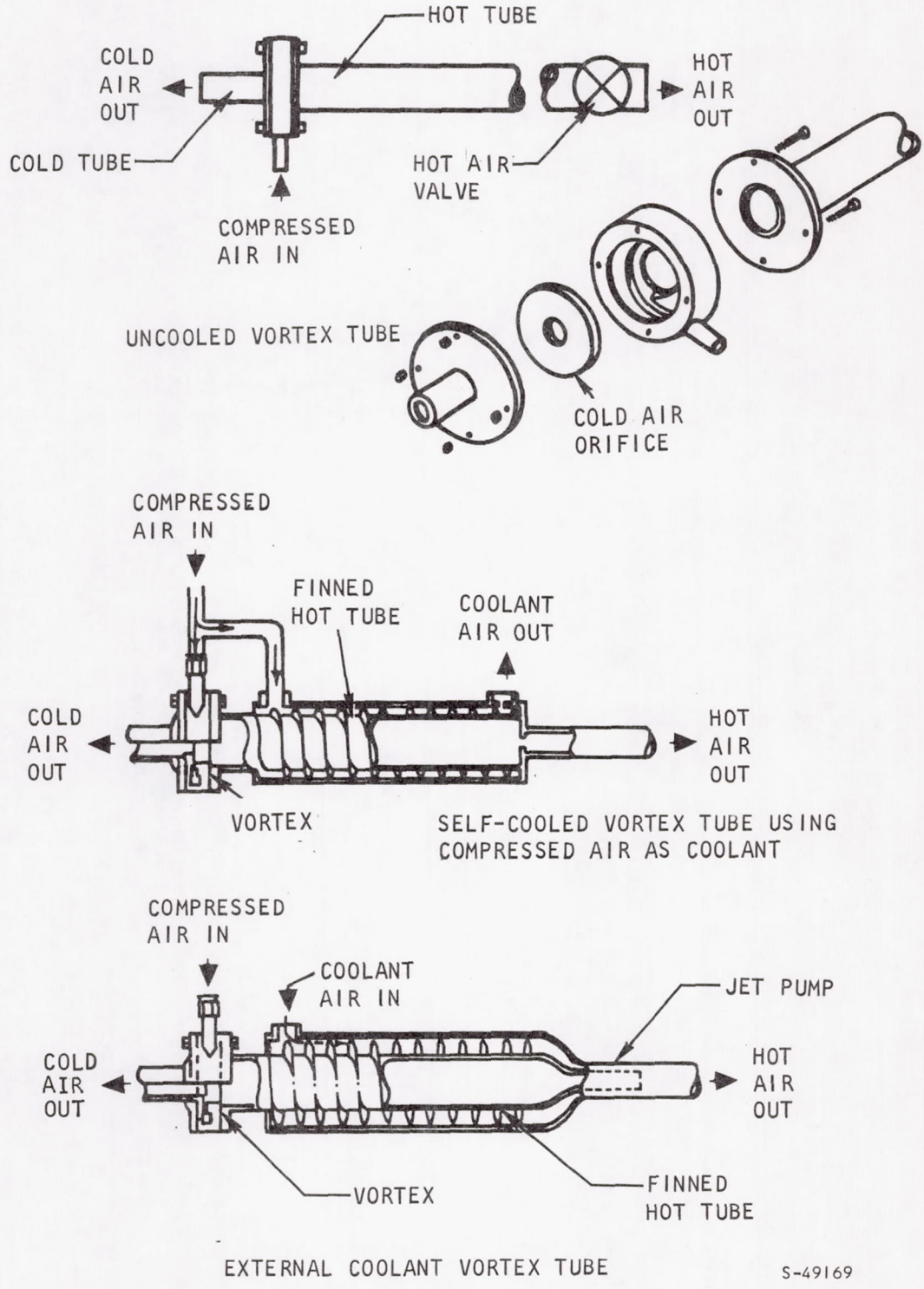


Figure 6-32. Three Types of Vortex (Hilsch) Tubes

Other devices that utilize the compression and expansion processes of a gas to produce lower-temperature gas or liquid for cooling purposes such as pulse tubes and miniature Joule-Thomson refrigeration systems can also be used for appropriate applications.

6.4.4 Thermal Shielding

To protect electric or electronic equipment at a special location, for instance, close to the leading edge of a pod or wing section, from extreme aerodynamic heating, solar heating or cold of space, the conditioning of its containing structure may be important. Convective, film, transpiration, insulation, ablation, or a combination of these can be used for this purpose. Figure 6-33 shows two schemes that this method can be applied.

6.4.5 Cooling by Phase-Change

Evaporation, fusion, and sublimation have been used for various cooling applications. The basic transport mechanisms can be quite complicated, but the transport rate can be phenomenal if properly designed. Another feature of cooling by phase-change is that the temperature at which phase change occurs is a function of pressure alone. If pressure is more easily controlled in one application, constant temperature is a direct consequence under equilibrium conditions.

Example of evaporation/condensation applications are shown in Figures 6-21, 6-23, and 6-24 of Section 6.3, all utilizing the high heat transfer rate of phase-change processes. Figure 6-34 shows the photograph of an aqua-ammonia evaporative oil cooling system for a typical supersonic aircraft to supplement the cooling by fuel.

In forced convective boiling, swirlers are often used to prevent film-boiling for more efficiency. The swirlers generate a centrifugal field and move the entrained liquid droplets toward the hot metal wall, keeping it always wet. Swirlers may not be practical in some applications, but film-boiling should always be prevented.

6.4.6 Heat Pipes

A specialized application of evaporation/condensation processes for cooling purpose is the heat pipe, in which wicks are used to provide capillary forces to return the condensate to the evaporative end. The basic principle and one application are shown in Figure 6-35. Heat flux densities as high as 400 w/sq cm have been obtained using liquid silver as the working fluid and operating at 2000°C*.

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*"High Thermal Conductance Devices Utilizing Boiling of Lithium or Silver," Los Alamos Scientific Laboratory Rep. LA-3211, Oct. 1964.

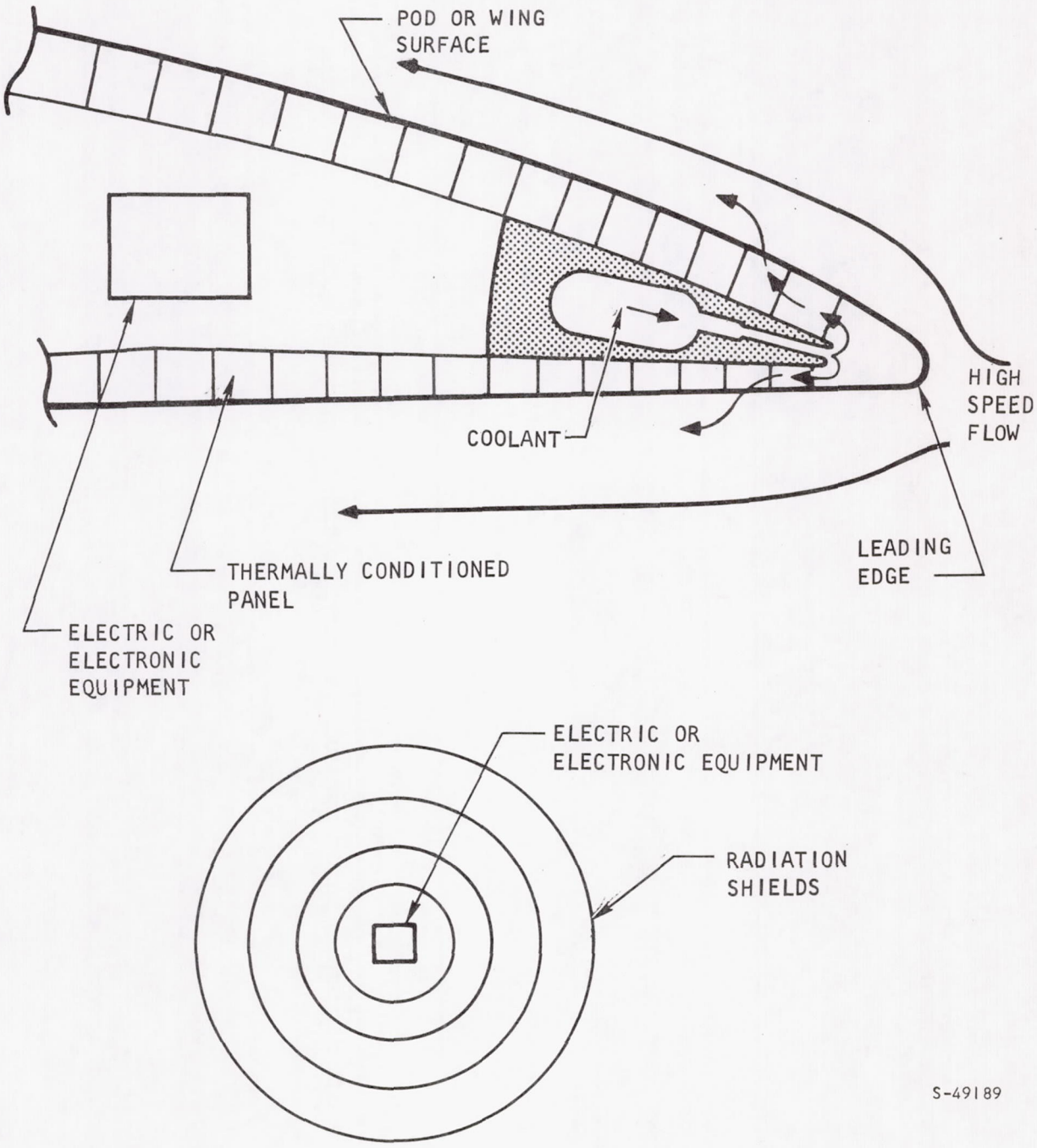


Figure 6-33. Thermal Shielding

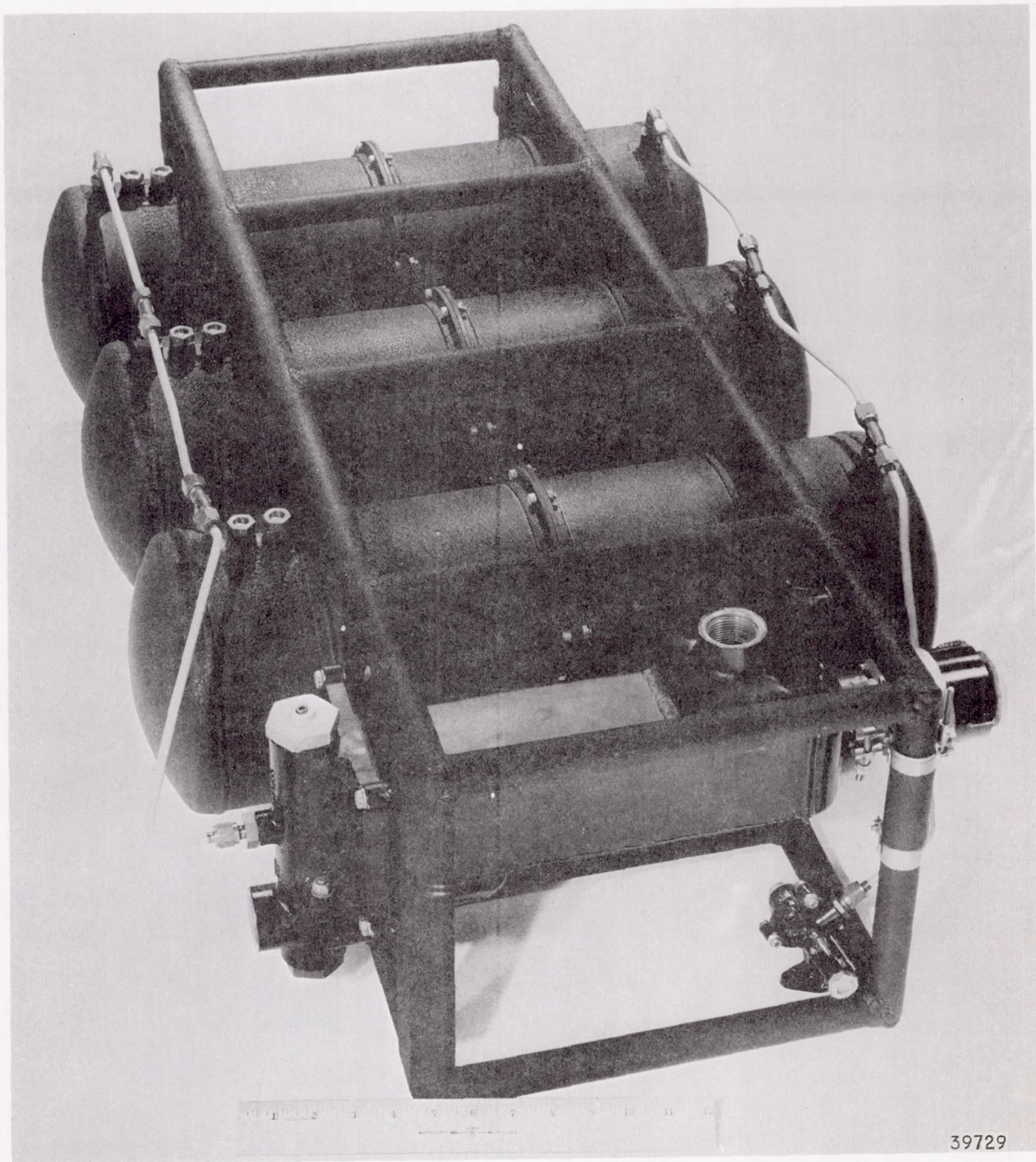
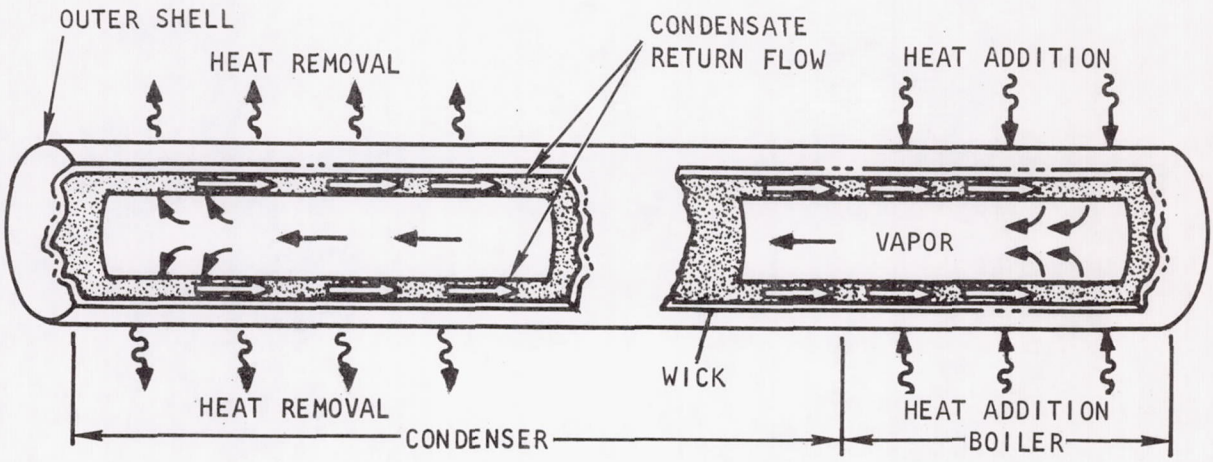


Figure 6-34. An Aqua-Ammonia Evaporative Oil Cooling System (50-KW Capacity)

BASIC HEAT PIPE: HEAT IS BEING CARRIED BY THE VAPOR FROM THE HEAT ADDITION (BOILER) END TO THE HEAT REMOVAL (CONDENSER) END AT ENORMOUS RATE. THE RETURN OF THE CONDENSATE IS ACCOMPLISHED BY THE CAPILLARY ACTION INSIDE THE WICK



HEAT PIPES AS AN INTEGRAL PART OF RADIATION FINS, WITH THE HEAT PIPES KEEPING THE FINS NEARLY ISOTHERMAL

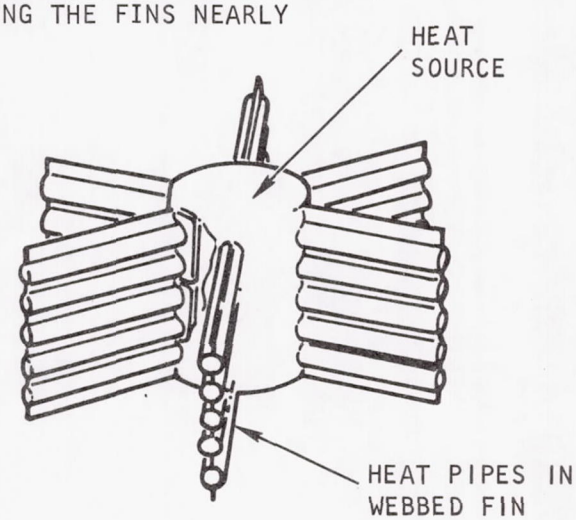


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Figure 6-35. Cooling by Heat Pipes

6.4.7 Cooling by Expendables

As in the system consideration in Section 6.3.7 the method of expendables utilizes the phenomenal heat transfer rate of the boiling process without the weight and cost penalties of compressors, condensers, and boilers of a refrigeration plant. The vapor produced, as shown in Figure 6-21 of Section 6.3.7, is dumped overboard.

The problems involved with this method are that (a) the system may need servicing after each flight, and (b) sealing and corrosion may present difficulties if the operating condition has high pressure and high temperature.

Figure 6-36 shows a stator-rotor assembly that is cooled by the method of expendables. Water flows along the passages in the stator housing and the vaporized water is dumped overboard.

6.4.8 Cryogenic Cooling

One reason cryogenic fluids are attractive in electrical and electronic cooling is the fact that the electric resistances of a lot of conductor materials decrease almost linearly with temperature. This fact has been successfully utilized to transport enormous amounts of electricity over long distances in ground applications. Figure 6-37 shows such an arrangement where 3000 mva is transmitted using liquid hydrogen. Cooling is not a problem in this type of application but the prohibitive reason for aircraft usage is that the refrigeration plant amounts to too much of the system weight. On the other hand, the expendable type of arrangement should eliminate the complete refrigeration plant and retain all the advantages of cryogenic cooling.

6.4.9 Improved Heat Rejection

For improved heat rejection from high power dissipation electrical components, a liquid or air cooled heat exchanger commonly known as a heat sink is attached to the component. For smaller components such as TO-5 and TO-8 case size transistors, a circular finned heat sink is pressed on the housing. Power transistors, small diodes and SCR's may be bolted to an extruded aluminum heat sink. Large SCR's and diodes may be stud or clamp mounted to large extruded aluminum heat sinks. Improvements in heat sink weight and volume requirements may be achieved by using pin-fin or plate-fin forced convection air-cooled heat sinks or liquid-cooled cast, machined, or plate-fin heat sinks. Table 6-3 gives the thermal conductance per unit weight and volume for both natural convection and forced convection cooled heat sinks. This table indicates that higher thermal conductance per unit weight and volume is obtained with the small component heat sinks. Forced convection improves the thermal conductance per unit weight and volume by a factor of 2 to 4. Forced convection pin-fin and plate-fin air cooled heat sinks are 4 to 9 times better than extruded heat sinks per unit weight and volume. Liquid cooled heat sinks are 3 to 12 times better than extruded aluminum air cooled heat sinks.

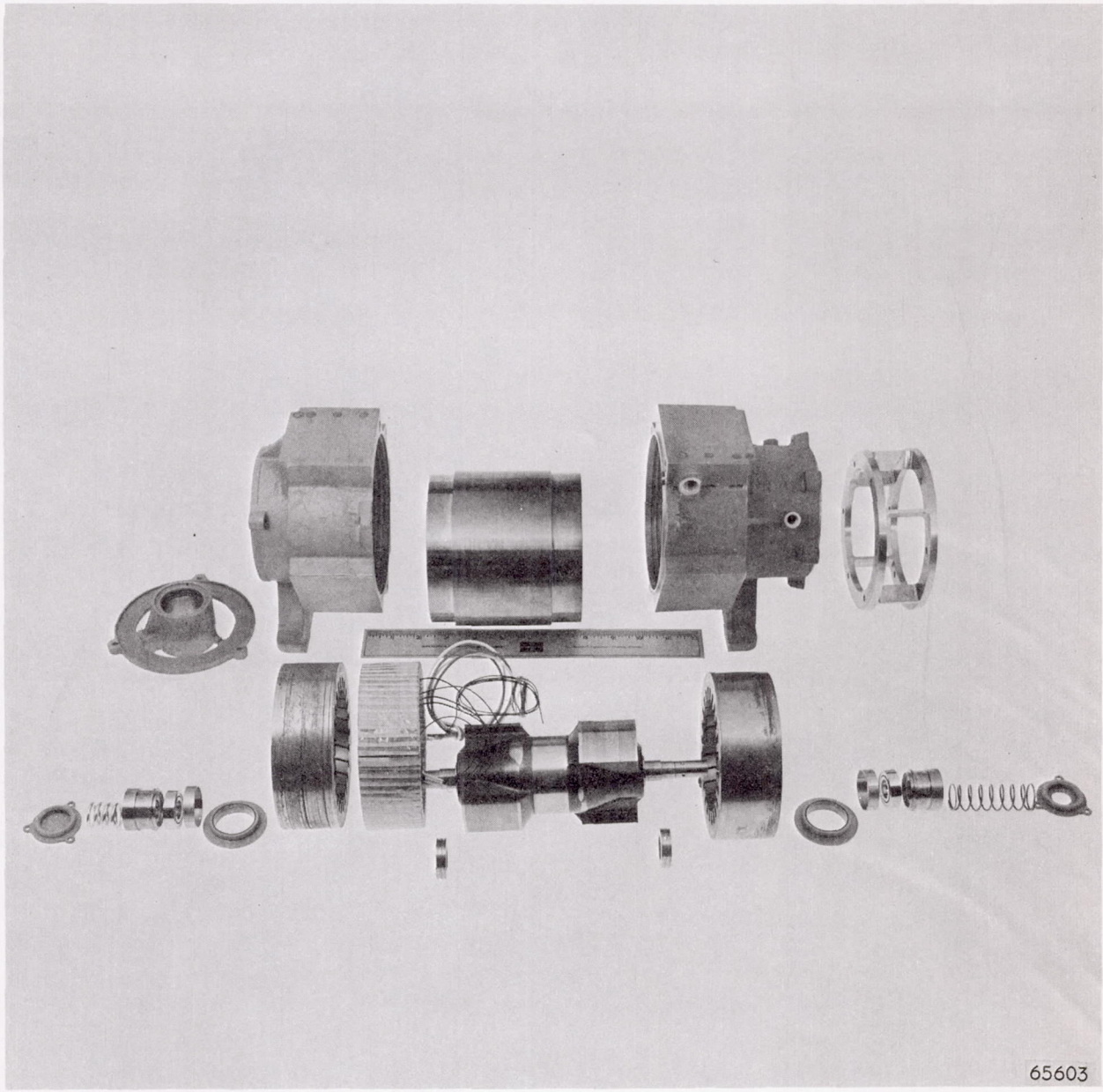

Figure 6-36. Stator-Rotor Assembly Cooled by Evaporating Water Flowing Through Internal Passages in Rotor and Stator

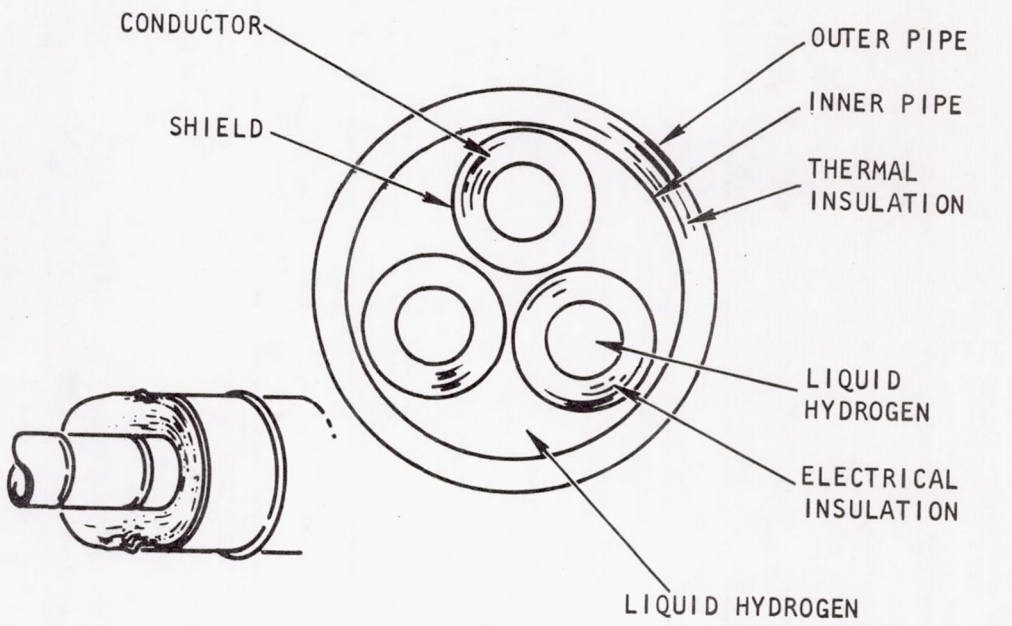

Figure 6-37. AC Transmission Cables Cooled by Liquid Hydrogen

TABLE 6-3

SUMMARY OF HEAT SINK THERMAL CONDUCTANCE
PER UNIT OF WEIGHT AND VOLUME

Type of Heat Sink		Coolant	Thermal Conductance per Unit Weight Watts/C° lb	Velocity Air ft/min	Thermal Conductance per Unit Volume Watts/C° $in.^3$
Natural Convection	Small transistor (TO-5, TO-8) heat sink	Air (1 atm pressure)	6.23	—	0.34
Natural Convection	Power semiconductor (TO-3, TO-36, 2N1016) extruded aluminum heat sinks	Air (1 atm pressure)	1.786	—	0.03938
Natural Convection	Large SCR, diode extruded aluminum heat sinks	Air (1 atm pressure)	0.619	—	0.0109
Natural Convection	Press-pak semiconductor extruded aluminum heat sinks	Air (1 atm pressure)	0.47	—	0.0159
Forced Convection	Large SCR, diode extruded aluminum heat sinks	Air	1.03 1.95	200 1000	0.0181 0.0343
Forced Convection	Press-pak semiconductor extruded aluminum heat sinks	Air	1.09 2.33	200 1000	0.0192 0.041
Forced Convection	Pin fin heat sinks for power semiconductors	Air	7.42 16.63	200 1000	0.2351 0.5269
Forced Convection	Plate-fin heat sinks for power semiconductors	Air	9.6 19.0	200 1000	0.47 0.93
Forced Convection	Cast or machined passage (aluminum) for large SCR's and diodes	Oil	5.8	—	0.42
Forced Convection	Cast or machined passages (aluminum) for large SCR's and diodes	water	28.3	—	1.265
Forced Convection	Plate-fin (aluminum) for large SCR's and diodes	Oil	6.3	—	0.526

6.4.10 Internal Heat Ducting from Hot Spot

Internal ducting into the heat source has two advantages. One is that additional heat transfer area is obtained as a gain and the other is that the transport medium can make direct contact with the heat source for maximum cooling efficiency. This technique has been successfully used to cool windings of generators of large ratings for the past sixteen years, using pressurized hydrogen gas. Recently water has been used in place of hydrogen gas to obtain even higher performance. The application of this technique to machinery of smaller ratings is certainly feasible but needs further trade-off and design studies.

In case internal ducting can result in the shifting of a hot spot to another location, thermal de-coupling (or isolation) of the heat source from the other portion of the system may be necessary.

Heat pipes described in Section 6.4.6 can certainly be investigated for future applications in conjunction with this technique.

6.4.11 Thermal Network Synthesis

It should be a big imporvement in analyzing methods if optimization techniques can be incorporated into existing performance-predicting thermal analysis computer programs to provide information on optimum thermal design and trade-off comparisons. AiResearch has computer programs that give detailed steady-state and transient temperature distributions for systems of any geometrical complexity with a maximum of thirty fluid streams flowing through them. These programs use analogous electrical network for thermal calculations and thus are readily adapted to network synthesis techniques.

7. RELIABILITY, SAFETY, AND MAINTAINABILITY

7.1 OBJECTIVES

As applied to aircraft electrical power systems, the term "reliability" is employed to indicate the probability of not experiencing a component, channel, subsystem, or system failure, while the term "safety" is employed to indicate the implications of failure. While it is recognized that individual components must be made inherently as reliable as feasible to minimize required maintenance actions, it is mandatory that any system, to be suitable, be designed to tolerate both component failure and channel failure, since they are certain to occur, even if infrequently. Thus, a primary system objective is established, which shall be identified as a "failsafe criterion." (Failsafe means in this report either (1) the system has a backup which continues to function if the electric supply fails, or (2) the failure does not produce secondary damage.)

A corollary of the primary failsafe objective, and one with strong economic implications, is termed "dispatchability." This term is defined in current airline practice as follows: no single component failure (or channel failure) can be permitted to preclude a legal revenue dispatch of a commercial transport aircraft. The latest thinking on aircraft dispatchability is shown in Table 7-I on p 7-7 . This second objective should not automatically apply to small commercial or military aircraft, but its applicability should be determined by the FAA for the categories of smaller subsonic commercial aircraft.

These two objectives imply some degree of redundancy. Since increases in redundance are accompanied by increases in initial cost, corrective maintenance frequency (the incidence of component, not system, failure determines the incidence of maintenance frequency), installed weight, complexity of control, etc., it is clear that any future system need not, and should not, possess redundancy appreciably in excess of that needed to attain full conformance to the two objectives of failsafe and dispatchability. In-flight reliability, defined as the ability to continue to the scheduled destination after an in-flight failure is also important, but usually is achieved if the failsafe and dispatchability objectives are realized.

The objective of a maintainability study is to minimize the cost and out-of-service time of electrical system maintenance.

7.2 CRITERIA

Certain concepts which are understood in general by electrical engineers in the aerospace field are defined more precisely to indicate their importance to reliability, safety, maintenance, and the cost of ownership. Technical statistical terms have been intentionally avoided throughout this report, and an engineering approach adopted.

One example is given of the danger in adopting a technically correct but very misleading procedure of reliability engineering, almost universally employed for the evaluation and comparison of redundant systems. Redundant systems are, by definition, systems not incapacitated by one or more failures. It is commonplace for reliability people to estimate component failure rates and compute an equivalent single thread system meantime between failures. This latter value is then used for comparative evaluation of competing system configurations.

The trouble arises from the basis for calculation of relevant probabilities; these probabilities are almost always computed for one flight. Thus, the wingsweep angle-changing mechanism for the earlier configuration of the Boeing SST was estimated at approximately 14 million hr MTBF, based upon redundant channels each with values of 2,500 hr MTBF. Such high values no longer remain high when, instead of one flight, the life of the airplane is used for computation.

There are simpler and more valid techniques which are more realistic. One of the best methods is comparison of competing redundant system concepts and designs with a standard set of criteria such as those presented below:

(a) Total installed weight and associated aircraft penalty, a direct function of degree and sophistication of system redundancy and designed overcapacity

(b) Initial cost of installed system, a direct function of degree of system redundancy

(c) Mean aircraft flight time between component unscheduled removals (MTBUR) and resultant maintenance actions, generally but not always a direct function of degree of system redundancy

(d) Dispatch reliability, a direct function of degree of redundancy, is a probability value commonly expressed by its complementary probability of delay in number of delays or cancellations per year or per arbitrary number of scheduled dispatches or departures; a recent study conducted by Garrett/AiResearch for Airbus disclosed that cancellation of one DC-10 or L-1011 flight involved a loss of revenue to an airline of 56 thousand dollars

(e) In-flight reliability or the probability of not aborting to a field other than the scheduled destination; this parameter is closely associated with dispatch reliability, as it implies system tolerance of two or more component failures, one before flight and one (or more) during flight

(f) Safety of flight, which defines the requirements for the emergency system (dc battery, deployable ram air turbine, onboard auxiliary power unit, etc.)

(g) Accessibility, or required access time to prepare to replace failed components, always additive (in time) to line maintainability

(h) Maintainability (line) or the time required to replace a failed component with a spare, and thus restore a system to its original fully operational state; time for access is never included here

(i) Maintainability (shop and depot) or the required time, technical manuals, trained personnel, test equipment, and spare parts required to repair failed or worn components, whether they be LRU (line replaceable units) or an entire subsystem; this parameter, expressed in units of maintenance manhours per flight hour, is commonly translated into direct labor cost (in dollars per flight hour). It has attained the importance of being specified for new aircraft systems, and guaranteed maximum values are presently being required by the airlines of their airframe suppliers, who, in turn, require them of their component and subsystem suppliers

(j) Failure prediction, detection and compensation (failure detection, isolation and identification), a necessary function which involves all the previously enumerated functions

Each of the above enumerated parameters is important. Some, such as accessibility, are known to present acute problems in SST, but none can be represented as bona fide scientific problems; they are, prima facie, engineering problems of the most practical type, with profound economic impact. Therefore, the current study cannot ignore any of them, and some comment is warranted at this time. Three terms to be employed are defined as follows:

(a) Single Thread Channel, or simply Channel (Singular)--the power generator and all associated hardware driven from any one engine

(b) Subsystem--two (or more) associated channels

(c) System--the total electrical power generation (and associated hardware) of all engines, including the emergency power supply

It is assumed (or laid down as Tentative Axiom Number One) that loss of two engines on a four-engined aircraft shall be allowed to pose more of a thrust-required-for-flight problem than the loss of associated electrical generating power, and that this rule thus imposes a constraint on the minimum designed overcapacity of any four-engined aircraft (SST or Subsonic Jet Transport). This rule is apparently already recognized in current jet transport design practice.

The above distinctions are quite important in distinguishing between the advantages and disadvantages of competing channel designs versus competing system configurations. An example will suffice: two radically different approaches in power generation or distribution may be amenable to identical system designs, and any differences in the system parameters would then be a reflection only of differences in the approach used for channel design.

Concurrent with the current study it seems wise to commence to essay establishment of feasible reliability goals for each item of hardware of each candidate channel, subsystem, or system design. Only in this way will quantitative values be elucidated for dependent parameters already listed, such as dispatch and in-flight reliability, line and shop maintainability, and mean time between unscheduled removals.

The mathematical relationships required for this work are all too well known to bear repetition. However, some discussion of the method of establishing values for the reliability performance of new concepts and conceptual hardware certainly is in order, since there are numerous pitfalls for the unwary and very little, if any, emphasis in the literature on the necessary mathematical connection between mean time between failures (MTBF), mean time between unscheduled removals (MTBUR), time between scheduled overhaul (TBO), and design life.

Mean time between failures (MTBF) is the quotient of cumulative flight or equipment operating time divided by cumulative confirmed failures. Mean time between unscheduled removals (MTBUR) is simply the quotient of cumulative flight or equipment operating time divided by cumulative removals. This frequently significant difference between MTBF and MTBUR is a reflection of deficiencies in effectiveness of failure detection, isolation, and identification.

Time between scheduled overhaul (TBO) is an arbitrary but essential definition of scheduled remanufacture to (nearly) a "zero time condition."

Design life is a term rarely defined and frequently confused with service life. The latter term implies, "... with scheduled and unscheduled overhaul (remanufacture) whereas design life is usually employed synonomously with time before overhaul.

All this leads to the concept that, for any population of hardware, some fail prematurely or before this design life (TBO). The probability of this happening is simply $\left(1 - e^{-\frac{TBO}{MTBF}}\right)$. For very rugged components (a heavy duty gearbox, for example) the MTBF will tend to exceed the TBO by several times. For highly stressed or very complicated hardware (some, but not all, constant speed drives, for example) the TBO tends to exceed the MTBF; the implications are abvious.

Applying the listed set of ten criteria to the electrical systems leads to certain tentative inferences that indicate the status of thinking at the time of writing:

(a) The overall penalty ascribable to total installed weight (and associated aircraft penalties) is an inverse function of aircraft optimum cruise lift over drag ratio. This ratio, which approximates sixteen for current high performance subsonic transports, will be reduced to approximately eight for SST; installed weight, thus, is of sharply increased significance for SST

(b) The total initial cost of newly conceived systems for power generation and distribution will naturally be appreciably higher than for existing state-of-the-art components. Therefore, some compensation in cost of ownership is highly desirable or even required. The most attractive area is that of increased reliability, with concomitant reduced incidence of failure and reduced line and shop maintenance costs; this approach appears feasible with VSCF cyclconverter and with DC

(c) The VSCF approach has a potential reliability advantage over the CSD approach, and the weight disadvantage of the former at present is not pronounced, leading to the belief that it can be eliminated

(d) Dispatch reliability can be made identical for any desired value, but with possibly different levels of system redundancy (weight and costs) required

(e) The comment for (a) applies equally to in-flight reliability of the system

(f) Safety of flight may, in the final analysis, depend upon a good dc emergency source, and improved generation of dc power is employed in past and present manned spacecraft; thus, it appears that the dc approach makes applicable not only much that already has been accomplished but may also possess the greatest growth potential for the primary power generation and not just the emergency source

(g) Accessibility is such a difficult problem that any improvement in SST is a plus. Avoidance of a CSD falls within this concept, not only for accessibility but also for most of the ten previously listed criteria

(h and i) No comments are required on these two criteria, except that both are improved with improved reliability

(j) Since any fault (failure) detection, isolation, and identification concept itself is part of the loop, its complexity can pose just as serious a reliability problem as anything else, and the simpler it can be made the better. In aircraft air conditioning systems, no attempt is made to fault isolate certain components, because the failure frequency of the fault detection and isolation equipment would actually be appreciably greater than that of the monitored component (for example, a heat exchanger). This task may lend itself to sophisticated mathematical analysis and optimization and may conceivably become an important desideratum in final choice of the prime candidate mode of generation and distribution. Current work on fault detection and isolation on major subsystems of the McDonnell Douglas DC-10 large commercial jet transport clearly indicate a void exists on a scientific approach to optimization of such fault detection as isolation. Currently, the methodology and techniques employed can only be described as primitive and cut and try. Some of the BITE (Built-In Test Equipment) has more than doubled the failure rate of certain electronic components, and extensive re-design to reduce the complexity has been required.

7.3 POWER SYSTEM CHANNEL CAPACITY

Availability of channels or complete systems for dispatch upon their intended mission is a probability combining reliability with accessibility and line, but not shop or depot, maintainability. In spite of some weapons systems engineers' having evolved some very sophisticated approaches to certain facets of their specialized problems, availability or reliability with maintenance, simply is "up time" divided by real or calendar time. Up time is usually approximated by mean time between failures, and calendar time is the sum of up time (MTBF) plus scheduled and unscheduled down time. Usually, down time equals the mean time to repair a system (MTTR) or to restore the system to its normal, fully operational status. Thus, from the point of view of a using airline (or armed service), availability is the salient characteristic which combines many of the previously listed characteristics.

It is important to note that availability is dependent upon the ratio of MTBF/MTTR where the denominator is not the mean time to repair individual components (line replaceable units) but is the mean time to restore the system to its fully operational state. Thus, MTTR as employed in this context must reflect access time. It should also be noted that availability as defined above is equal to the maximum utilization factor (U_{max}), and that scheduled, as distinct from unscheduled, maintenance can reduce this factor. However, there exists a strong trend against scheduled maintenance and towards "on condition" maintenance. Also, electrical generating systems, like other systems, tend to receive their scheduled maintenance only when the aircraft as a whole is scheduled for such maintenance.

The importance of availability cannot be exaggerated, and the airlines and armed services have become hypersensitive to impaired availability. This is true for the dual reason that not only is it costly to maintain equipment, but the delays or cancellation of flights means that these users require more aircraft to maintain their intended level of operational capacity. If a four engined aircraft has four channels of power generation and distribution each with an availability of A_1 and an unavailability of U_1 (the probability complement), then the availability of two (or more) channels out of four would be the summation of the terms of second power and higher in the expansion $A_{2/4} = (A_1 + U_1)^4 = A_1^4 + 4A_1^3U_1 + 6A_1^2U_1^2 + 4A_1^1U_1^3 + U_1^4 = 1$. The symbol $A_{2/4}$ indicates acceptable availability when two out of four channels must be fully operational (without fault). For the case $A_{3/4}$ (for dispatch) only the third and higher powers of A_1 are summed.

Table 7-I indicates various power capacities and their probable number of operational channels needed for a legal revenue dispatch. This binomial makes abundantly clear a method of determining individual channel designed power generation capacities. If in a four-engined aircraft each channel were

TABLE 7-I

RECOMMENDED AND MINIMUM ALLOWABLE CHANNEL AND SYSTEM DESIGN CAPACITIES

Number of Engines	Number of Channels	Design Channel Capacity	Dispatch with One Channel Inoperative	Abort with One Channel Inoperative	Comments
4	4	0.25	No	No	Not sufficient margin for commercial aircraft
		0.333	Yes	No	Minimum contingency factor
		0.50	Yes	No	Adequate margin for most emergencies
3	3	0.333	No	No	Second failure critical (see text)
		0.50	No	No	Second failure critical (see text)
		0.75	Yes	No	Unacceptable contingency factor
		0.67 N/1.0E	Yes	No	Adequate margin for most emergencies (Note assumption that 1/2-hr emergency capacity = 1.5 normal)
2	2	0.50	No	No	Second failure critical
		0.67	No	No	Unacceptable contingency factor
		0.67 N/1.0E	Yes (if APU is running and has same capacity as main generator)	No	Acceptable with 1/2-hr emergency rating of 1.5 (0.67) Adequate margin without flight engineer

Notes: N = Nominal continuous capacity
E = Suggested emergency 1/2-hr rating = 1.5 N

to have only 25 percent of design capacity, the summation of terms in the expansion might quite possibly be limited to the one (A_1^4) term, indicating that all four must be operational for dispatch. If, at the other extreme, each channel possesses a design capacity of 100 percent, all but the last (U_1^4) term might be summed to indicate the potential for continuing an already air-borne flight to the scheduled destination with three failed channels. If, as is almost certainly the only prudent approach, the on-board battery emergency supply is considered to be no more than an ultimate back-up supply then (N-1) channels should be required for safe FAA approved dispatch for.

It is clear that best practice lies between the two extremes cited and, in fact, lies between channel design capacities of 33 and 50 percent for four-engined aircraft. The logic employed for such problems can be somewhat different in emphasis between sub- and supersonic aircraft. The logic employed in studies the past six years for cabin pressurization, refrigeration, and ventilation (usually referred to as the environmental control subsystem) for both the British-French SUD Aviation CONCORDE and the U.S. Boeing SST has been that take-off shall be at 100 percent design capacity with one channel inoperative. The final design decision seems to rest upon the answer to this question: After a dispatch with one channel inoperative, and following an in-flight loss of a second channel, what is the minimum capacity required for a safe continuation of the flight without serious change in flight plan or serious discomfort to the passengers or, more importantly, to the flight crew. The answer has usually lain between 67 and 100 percent of design capacity.

At the time of writing, the investigation of the nighttime crash of a United Airlines Boeing 727 three-engined aircraft into the sea of Los Angeles International Airport on January 18, 1969 was being ascribed by the National Transportation Safety Board Chairman J. J. O'Connell, Jr. to the following set of conditions:

(a) Aircraft took off with its No. 3 generator inoperative, a fact known to the pilot and not by itself a hazard.

(b) But, said Mr. O'Connell (on Thrusday, February 6, 1969, in testimony before the House Interstate and Foregin Commerce Committee), the fact combined with loss of a second generator when the pilot reported he was "shutting down an engine" ... could have resulted in a gross electrical system failure. This would have occurred when the plane's entire electrical load was transferred to the one remaining generator.--(UNI dispatch datelined Washington DC and published on Page 1, Part II, of the Los Angeles Times of Saturday, February 8, 1969).

It is evident that certain design problems discussed herein and oriented to four-engined aircraft require different solutions for three- and two-engined aircraft, and it appears that these different solutions relate to channel generating capacity. Table 7-I summarizes the order of magnitude first estimates of required capacity of various aircraft engine number. It is noteworthy that certain aerodynamic problems also relate to the number of engines, and it has been recognized for sometime that twin-engined commercial aircraft pose unique problems of design simply because an engine loss (on takeoff or climbout) constitute a 50 percent loss of power.

Existing aircraft have no provision for automatic removal of non-essential loads when the generating equipment is overloaded. The foregoing example shows how important automatic load switching would be to the airplane safety. This subject is included in the present study effort, it will be investigated later in the program.

7.4 RELIABILITY OF ELECTRICAL SYSTEM COMPONENTS IN EXISTING AIRCRAFT

Many airlines have kept good maintenance and service records on aircraft components. Typical statistical data on component failure and repair are shown in Table 7-2. This table, taken from Eastern Airlines Fleet Reliability Report, January 1969, shows the premature removal and failure rates for the top 40 components, ranked by failure rate, during the month of January 1969. The MTBF of each component is given by (1000/FR) hours. The typical MTBF of generator and CSD shown in Table 7-2 are 3700 and 2700 hr, respectively. Many of the electronic components have only several hundred hours of MTBF.

A similar tabulation of unscheduled removal and failure rates of electric power generation components is shown in Table 7-3. Delays and cancellations due to mechanical irregularities of various subsystem for various airplane fleets are shown in Table 7-4 through 7-8. It can be seen from these tables that the electric power system (including generation and distribution) is not a major source of contribution to flight delay and cancellation.

7.5 TESTING FOR RELIABILITY

If, as a result of this study, a new concept of or approach to power generation or distribution becomes attractive, certain procedures become mandatory. Previous experience indicates clearly that laboratory simulation and consequent exposure to and solution of problems is far more economic than attempting to shortcut this necessary, indeed indispensable, step.

Today, no one contemplates building an aircraft without extensive wind tunnel testing. Analogously, in the aircraft ventilation and air conditioning area, half-system testing is performed, whereby half of an entire aircraft ship set is erected in a suitable laboratory facility and extensive testing is conducted at design points and during transients. Once such a test set-up exists, it quickly becomes apparent that new components not only can be tested, to their own problem statements by means of acceptance test procedures, but also can benefit from the half-system setup to test interactions between the component and the rest of the half-system.

TABLE 7-2

TOP 40 COMPONENTS, RANKED BY FAILURE RATE, DURING JANUARY 1969*

Component	Type Aircraft and No. of Units in Service	No.of Sched. Remov.	No.of Prem. Remov.	PRR** 1000 Unit Hrs.	FR*** 1000 Unit Hrs.
ELECTRONIC COMPONENTS:					
Panel, Auto-pilot Control	DC9(84)	0	74	4.39	3.26
Vertical Gyro	B720(30),L188(74)	0	51	3.09	2.12
Receiver, VHF Comm.	All DC8(99),B720(45),L188(74)	0	94	2.98	2.03
Transceiver, VHF Comm.	B727(225),DC9(252),DC8-61(48)	0	252	2.04	2.00
Computer, A/P Air Data	DC9(84),DC8-61(16)A/C #781	0	83	3.92	1.89
Receiver, VHF Nav.	DC8-21/51(34),B720(30),L188(74)	0	79	3.38	1.80
Indicator, Turn & Slip	DC9(168)	0	54	1.60	1.30
Recorder	All except CV440(242)	0	76	1.44	1.21
Transceiver, DME	DC9(168),B727(150,B720(30), All DC8(66),L188(37)	0	186	1.76	1.14
Gyro, Directional	DC9(168),B727(150),B720(30), All DC8(66)	0	144	1.50	1.00
Indicator, Course Deviation	B727(150),DC9(168),DC8-61(32)	0	103	1.26	.96
Transmitter, Radar	B727(75),DC9)84),DC8-61(16)	0	122	2.99	.88
Reproducer, Tape	All A/C except CV440(242)	0	60	1.23	.88
Indicator, Horizon	DC9(168),B727(150),DC8-61 & #781(34)	1	87	1.06	.73

*Eastern Airline Fleet Reliability Report, January 1969
**PRR - Premature Removal Rate
***FR - Failure Rate

TABLE 7-2 (Continued)

Component	Type Aircraft and No. of Units in Service	No.of Sched. Remov.	No.of Prem. Remov.	PRR* 1000 Unit Hrs.	FR** 1000 Unit Hrs.
Computer, Pitch	DC9(84)	0	66	3.91	.59
Receiver, ADF	B727(150),DC9(168),DC8-61(32)	0	61	.75	.50
Transponder, ATC	DC9(168),B727(150),All DC8(66), B720(30),L188(74)	0	119	1.13	.38
Gyro, Vertical	DC9(252),B727(105),DC8-21/51(35) DC8-61(48)	2	64	.91	.38
Transceiver	DC9(84),B727(75),DC8-61(16), A/C #779 & 781	0	44	1.06	.36
Receiver	DC9(252),B727(225),DC8-61 & #781(51)	0	115	.93	.29
COMPONENTS OTHER THAN ELECTRONIC:					
Control, Heat & Cab. Press.	L188(34)	0	24	5.86	3.42
Synchronizer, Prop. Phase	L188(34)	0	28	6.84	2.44
Controller, Cabin Temp.	L188(34)	0	43	10.50	2.20
Indicator, Fuel Qty. Center M.	L188(136)	0	34	2.08	1.04
Clock	B727(150)	0	55	1.39	.88
Pump, Hydraulic	B727(150)	7	37	.93	.81
Indicator, EPR	DC8-21(56)	0	29	2.27	.78
Clock	DC9(176),DC8-61(48),B727QC(75)	0	52	.61	.53
Indicator, T/M	L188(136)	0	27	1.65	.49

*PRR - Premature Removal Rate
**FR - Failure Rate

TABLE 7-2 (Continued)

Component	Type Aircraft and No. of Units in Service	No.of Sched. Remov.	No.of Prem. Remov.	PRR* 1000 Unit Hrs.	FR** 1000 Unit Hrs.
***Control Panel, AC Gen.	DC9(252)	0	40	.79	.47
***Transmission, CSD	B727(225)	0	28	.47	.37
***Control Panel, AC Gen.	DC9(252)	0	32	.63	.32
Indicator, Fuel Flow	DC8-21/61(120)	0	35	1.20	.31
***Generator	B727(300),B720(60)	33	58	.60	.27
***Generator Control Panel	B727(300)	0	31	.39	.27
Amplifier, Vibration	DC8(33),B727(75),B720(15)	0	51	1.60	.25
Indicator, EPR	DC9(176)	0	33	1.96	.24
Indicator, Aux. Fuel Qty.	DC9(176)	0	31	.92	.18
Power Supply, Fuel Flow	DC9(164)	0	31	.92	.18
Indicator, Turbo Comp. RPM	DC8-21/61(60)	0	31	2.12	.14

*PRR - Premature Removal Rate
**FR - Failure Rate
*** Electrical Power Components

TABLE 7-3

UNSCHEDULED REMOVALS*

Part Number Description	Aircraft Type	October 1968			November 1968			December 1968			January 1969			FR P/S
		UR	URR	FR	UR	URR	FR	UR	URR	FR	UR	URR	FR	
ELECTRIC POWER (ATA 24)														
21-GE-0002 A.C. Generator	DC8-51 & #781	2	.15	.08	5	.37	.07	3	.18	.06	0	0	0	
21-GE-0003 A.C. Generator	DC8-51/-55	3	1.98	-	4	2.22	.56	1	.38	-	4	1.58	.79	
21-GE-0013 A.C. Generator	CV-440	1	.24	.24	1	.27	-	0			0			
21-GE-0016 A.C. Generator	DC-9	22	.53	.36	11	.25	.16	6	.12	.08	10	.20	.20	.25
21-GE-0017 A.C. Generator	L-188	11	.61	.28	11	.62	.34	18	1.03	.57	6	.37	.18	.35
21-GE-0018 A.C. Generator	B-720/B-727	21	.24	.10	26	.29	.17	47	.50	.32	58	.60	.27	.15
21-GE-0020 A.C. Generator	DC8-21	1	.08	-	5	.38	.08	1	.07	-	4	.33	.33	.15
21-TR-0101 CSD Transmission	DC8-21/ B-720	8	.30	.15	12	.43	.18	17	.55	.23	17	.59	.24	.40
21-TR-0103 CSD Transmission	DC8-51/55 (-51,2 AC)	1	.66	-	3	1.66	.55	2	1.15	-	4	2.36	.59	
21-TR-0105 CSD Transmission	B-727	27	.49	.46	31	.56	.45	39	.67	.60	28	.47	.37	

*Eastern Airline Fleet Reliability Report, January 1969

LEGEND: UR = Unscheduled Removals URR = Unscheduled Removal Rate FR = Failure Rate
P/S = Performance Standard (3 month average) 00 = Above P/S, one month

TABLE 7-4

MECHANICAL IRREGULARITIES - DELAYS AND CANCELLATIONS (B-720) JANUARY 1969*

ATA Code	Number of Events					Percent Departures Delayed or Cancelled		
	Delays	(Min)	Cancelled	Other	Total	Report Month	3-Mo Avg	Perf Std
32 Landing Gear	22	1520	2	4	28	1.12	.64	.70
71-80 Power Plant	8	364	-	10	18	.42	.42	
71-80 Aircraft	5	190	1	20	26	.38	.38	
29 Hydraulic Power	4	476	3	-	7	.33	.31	.30
34 Navigation	5	120	1	9	14	.28	.24	.20
21 Air Conditioning	3	119	-	27	30	.14	.18	.12
26 Fire Protection	1	67	1	-	2	.09	.13	.10
27 Flight Controls	2	249	-	-	2	.09	.11	.20
33 Lights	1	34	-	5	6	.05	.10	.10
24 Electrical Power	1	5	-	22	23	.05	.10	.11
28 Fuel	-	-	1	12	13	.05	.06	.15
30 Ice & Rain Protection	1	67	-	2	3	.05	.05	.03
38 Water & Waste	1	49	-	-	1	.05	.03	.02
22 Auto Pilot	1	99	-	11	12	.05	.03	.07
23 Communications	1	80	-	5	6	.05	.03	.07
57 Wings	1	39	-	-	1	.05	.03	N.A.
52 Doors	-	-	-	1	1	-	.03	.15
05 Miscellaneous	1	62	-	-	1	N.A.	N.A.	N.A.
TOTALS	59	3677	9	127	195			

*Eastern Airline Fleet Reliability Report, January 1969

TABLE 7-5

MECHANICAL IRREGULARITIES - DELAYS AND CANCELLATIONS (DC8-21-51) JANUARY 1969*

ATA Code	Number of Events					Percent Departures Delayed or Cancelled		
	Delays	(Min)	Cancelled	Other	Total	Report Month	3-Mo Avg	Perf Std
32 Landing Gear	10	1116	-	-	10	.61	.70	.70
71-80 Power Plant	11	1155	1	13	25	.70	.65	
71-80 Aircraft	3	75	-	25	28	.55	.55	
27 Flight Controls	5	700	2	8	15	.43	.47	.35
34 Navigation	6	295	-	11	17	.36	.41	.35
52 Doors	-	-	2	5	7	.12	.21	.15
28 Fuel	2	77	-	28	30	.12	.19	.16
24 Electrical Power	-	-	1	8	9	.06	.17	.18
29 Hydraulic Power	2	89	-	1	3	.12	.13	.30
33 Lights	2	70	-	-	2	.12	.13	.10
35 Oxygen	1	12	-	1	2	.06	.11	.08
38 Water & Waste	1	22	-	-	1	.06	.06	.05
56 Windows	1	275	-	-	1	.06	.06	N.A.
36 Pneumatic	1	14	-	2	3	.06	.06	.12
54 Nacelles/Pylons	2	305	-	1	3	.12	.04	N.A.
26 Fire Protection	2	233	-	-	2	.12	.04	.10
21 Air Conditioning	-	-	-	24	24	-	.02	.12
22 Auto Pilot	-	-	-	10	10	-	-	.07
05 Miscellaneous	4	122	-	7	11	N.A.	N.A.	N.A.
TOTALS	53	4560	6	144	203			

*Eastern Airline Fleet Reliability Report, January 1969

TABLE 7-6

MECHANICAL IRREGULARITIES - DELAYS AND CANCELLATIONS (DC8-61) JANUARY 1969*

ATA Code	Number of Events					Percent Departures Delayed or Cancelled		
	Delays	(Min)	Cancelled	Other	Total	Report Month	3-Mo Avg	Perf Std
71-80 Power Plant	3	481	-	10	13	.27	.65	
71-80 Aircraft	5	289	-	12	17	.55	.55	
32 Landing Gear	4	150	1	-	5	.34	.39	.70
34 Navigation	4	231	-	17	21	.27	.34	.35
29 Hydraulic Power	5	445	-	2	7	.34	.22	.30
52 Doors	5	250	-	10	15	.34	.20	.15
24 Electrical Power	3	120	-	6	9	.20	.17	.18
27 Flight Controls	2	207	-	2	4	.13	.15	.35
28 Fuel	2	46	-	32	34	.13	.10	.16
21 Air Conditioning	2	99	-	33	35	.13	.10	.12
25 Equipment & Furnishings	-	-	-	1	1	-	.10	.05
33 Lights	-	-	-	6	6	-	.10	.10
26 Fire Protection	1	90	1	-	2	.13	.07	.10
30 Ice & Rain Protection	1	182	-	3	4	.07	.05	.08
36 Pneumatic	-	-	-	7	7	-	.05	.12
23 Communications	-	-	-	5	5	-	.02	.12
38 Water & Waste	-	-	-	1	1	-	.02	.05
22 Auto Pilot	-	-	-	7	7	-	-	.07
31 Instruments	-	-	-	2	2	-	-	.04
05 Miscellaneous	1	64	-	2	3	N.A.	N.A.	N.A.
TOTALS	38	2654	2	158	198			

*Eastern Airline Fleet Reliability Report, January 1969

TABLE 7-7

MECHANICAL IRREGULARITIES - DELAYS AND CANCELLATIONS (L-188) JANUARY 1969*

ATA Code	Number of Events					Percent Departures Delayed or Cancelled		
	Delays	(Min)	Cancelled	Other	Total	Report Month	3-Mo Avg	Perf Std
61-80 Power Plant	14	1118	1	27	42	.35	.58	
61-80 Aircraft	6	109	1	7	14	.48	.48	
32 Landing Gear	15	709	2	-	17	.35	.28	.25
34 Navigation	8	249	-	34	42	.17	.16	.20
24 Electrical Power	4	179	-	8	12	.08	.12	.13
21 Air Conditioning	7	285	1	31	39	.17	.09	.18
33 Lights	8	145	-	4	12	.17	.08	.07
29 Hydraulic Power	4	194	-	5	9	.08	.07	.08
56 Windows	1	157	2	-	3	.06	.05	N.A.
23 Communications	3	20	-	4	7	.06	.05	.10
30 Ice & Rain Protection	2	218	-	6	8	.04	.05	.05
28 Fuel	2	119	-	19	21	.04	.03	.07
25 Equipment &	1	12	-	3	4	.02	.03	.02
27 Flight Controls	1	77	-	-	1	.02	.03	.10
52 Doors	1	42	-	13	14	.02	.03	.125
36 Pneumatic	2	102	-	1	3	.04	.02	.025
38 Water & Waste	-	-	1	-	1	.02	.01	.025
22 Auto Pilot	-	-	-	9	9	-	-	.015
05 Miscellaneous	2	14	-	2	4	N.A.	N.A.	N.A.
TOTALS	81	3749	8	173	262			

*Eastern Airline Fleet Reliability Report, January 1969

TABLE 7-8

MECHANICAL IRREGULARITIES - DELAYS AND CANCELLATIONS (CV-440) JANUARY 1969*

ATA Code	Number of Events					Percent Departures Delayed or Cancelled		
	Delays	(Min)	Can-celled	Other	Total	Report Month	3-Mo Avg	Perf Std
61-82 Power Plant	14	1364	10	1	25	1.30	1.25	
61-82 Aircraft	2	239	1	6	9	1.10	1.10	
32 Landing Gear	8	382	-	-	8	.35	.26	.30
26 Fire Protection	7	237	-	-	7	.31	.23	.15
24 Electrical Power	4	57	2	-	6	.26	.11	.175
33 Lights	4	91	-	2	6	.18	.11	.10
34 Navigation	2	73	-	1	3	.09	.11	.12
21 Air Conditioning	1	86	-	1	2	.04	.09	.10
23 Communications	2	19	-	-	2	.09	.08	.10
29 Hydraulic Power	1	86	1	-	2	.09	.05	.05
56 Windows	-	-	1	1	2	.04	.05	N.A.
52 Doors	1	139	-	1	2	.04	.04	.125
30 Ice & Rain Protection	-	-	-	1	1	-	.03	.03
27 Flight Controls	1	60	-	-	1	.04	.01	.03
35 Oxygen	1	40	-	-	1	.04	.01	.01
05 Miscellaneous	1	13	-	1	2	N.A.	N.A.	N.A.
TOTALS	49	2886	15	15	79			

*Eastern Airline Fleet Reliability Report, January 1969

The half-system contemplated as a follow-on to this study would consist of two paralleled channels of power generation, distribution, and utilization, each channel being driven by a small gas turbine, these turbines would simulate two aircraft jet engines. Such a test would investigate the effects of a whole schedule of simulated faults in an airborne system, and the ability of paralleled channels automatically to fault detect, identify, and isolate (disconnect) in order that only the faulty channel be lost and the good channel remain connected. In particular, the speed with which such fault isolation can be accomplished, and the effects of delay, can be measured and recorded for different system configurations. Once such a test setup was in existance many problems not even thought of would be discovered and investigated. In conclusion the primary function of the test set-up would be to qualify a new concept or approach as suitable, safe, and reliable for incorporation into FAA-certified aircraft.