FINAL TECHNICAL REPORT

Project A-1251

SPACE VEHICLE ELECTRICAL POWER SYSTEMS STUDY

S. L. Robinette, G. W. Bechtold,
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CONTRACT NAS8-25192

22 September 1971

Prepared for
National Aeronautics and Space Administration
George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama

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Engineering Experiment Station
GEORGIA INSTITUTE OF TECHNOLOGY
Atlanta, Georgia

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FOREWORD

This report was prepared at the Electronics Division of the Georgia Institute of Technology under Contract NAS8-25192. The work was performed within the Communications Branch under the general supervision of Mr. D. W. Robertson, Head of the Communications Branch. The report covers the activities and results of continuation of effort on a project to aid in the development of electrical power systems for future space vehicles.

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ABSTRACT

A survey of literature on advanced electrical power systems of aircraft, ships, and other vehicles indicated that the developing technology can be transferred to the design of electrical power systems for future spacecraft. The natural and induced environments, which will include internal and external atmospheres, radiation fields, magnetic fields, micrometeoroid flux, and ionospheric plasma, will affect the design of spacecraft power systems. The trend in aircraft power system technology is toward higher voltages in order to reduce distribution cable weight, but corona and flash-over conditions will restrict the maximum voltage permissible in spacecraft environments. To meet criteria of reliability and maintainability, a dedicated control system for power distribution (data bus) with self-check capability is recommended. The development of suitable solid-state power controllers for spacecraft power distribution systems is a critical need.

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TABLE OF CONTENTS

PAGE

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LIST OF FIGURES

LIST OF TABLES

I. INTRODUCTION

This Final Technical Report together with the Interim Technical Report of 8 September 1970 [1] and the Second Interim Technical Report of 22 January 1971 [2] delineates the work performed under Contract NAS8-25192.

Initially, an extensive review was made of the literature on advanced electrical power distribution systems. Studies of system sizing, load usage, switching, and distribution were analyzed with the objective of bringing advanced technological developments from related fields to the attention of spacecraft designers. Out of several hundred literature sources that were scanned, a bibliography of ninety-five studies was generated. Findings from this bibliography formed the basis of the Interim Technical Report of 8 September 1970.

For the Second Interim Technical Report some fifteen further major studies were reviewed and model power distribution systems and techniques were developed. Power sources were examined for compatibility with the conceptual models of distribution systems. During the course of investigation it became clear that many investigators were unaware of the dangers of corona and arcing in spacecraft power systems. System voltage specifications were found to be based on minimum corona onset voltage (COV) calculated from Paschen's "Law", although Paschen's results hold only for dry air, low background radiation, and uniform voltage fields. Failure to view worst-case COV as a system constraint has led to mission failures in the past [3,4]. A second deficiency in the planning of advanced power distribution systems for spacecraft resulted from variances in specifications for solid-state power controllers. An advanced data bus system which utilizes solid-state power controllers is currently being developed for a military aircraft, and transfer of this advanced technology to spacecraft is highly desirable. However, data on solid-state power controllers must be made available by manufacturers, and realistic specifications for solid-state controllers must be written before planning of spacecraft power systems can proceed.

Because of the deficiency of information in the two areas, corona onset voltage and solid-state controllers, five lines of investigation were recommended and subsequently became the subjects of this Final Technical Report. The five tasks were:

1. Determine the minimum COV that could be expected in future spacecraft.
2. Compare COV characteristics of flat and round conductors.
3. Determine operational characteristics of solid state switches.
4. Compare dedicated and non-dedicated data bus multiplex systems.
5. Estimate weight and efficiency tradeoff effects of system constraints imposed by COV and solid-state switch characteristics.

The nature of the investigation for this final report was very different from the earlier work. The first two reports were based entirely on what could be found in the literature. The third phase investigation has depended on the pursuit of specific information, much of which could not be found in the literature, and on a limited amount of experimental work. The degree of success has thus been directly related to the ability to find specific published data. To some extent the search for data was pursued at the expense of the experimental program because of the desire not to duplicate work done by other investigators.

II. CORONA ONSET VOLTAGE STUDIES

The atmosphere surrounding Space Shuttle and Space Station will vary greatly over the life of a mission. Before and during the early moments of launch, normal sea level atmosphere (760 mm of mercury) will act on the spacecraft. As the craft ascends pressure will decrease until at about 400 km it will be only 10^{-8} mm of mercury. Upon reentry Space Shuttle will experience the same conditions in reverse order. When Space Station is in orbit the pressure will be vastly different in various parts of the vehicle. There will be compartments where the pressure will approximate that at sea level, and there will be air-locks that will undergo rapid pressure changes while they are being pressurized or depressurized. The pressure constraints and ranges must be considered in the design of the spacecraft electrical systems.

In addition to wide ranges of environmental pressure, the spacecraft will experience great changes in temperature and radiation. It was indicated in an earlier report [1] that external temperature will range downward to about 175°K at 80 km, and upward to a maximum of 1800°K at altitudes above 300 km. It was also indicated that the radiation environment will include electromagnetic fields -- radio frequency, ultraviolet, and infrared -- as well as energetic electrons and protons. At orbital altitudes in the ionosphere, the electron densities seem to be sufficient to support glow discharge with only moderate voltage gradients.

The corona onset voltage (COV) is the voltage at which corona begins to appear; steady-state COV for a particular electrode composition and configuration is a function of a parameter equivalent to atmospheric pressure multiplied by electrode separation distance (PD). In spacecraft power systems COV would also be influenced by the composition of the atmosphere, the background radiation (for insulated conductors), and the frequency or waveform of applied voltage. A curve relating COV to PD for simple electrode configurations (parallel plates) was found experimentally by Paschen in 1889. Paschen's curve is often misused to predict a minimum COV of about 300 volts for any and all electrical systems, but his results are restricted to cases where the field is uniform, space charge effects are absent, and electrode spacing is large with respect to mean free path. It can only be said for every electrode system that the COV is high at very large and at very small ranges of PD, and is a minimum at some intermediate PD value. Equipment located on Space Shuttle or in a Space Station airlock which must undergo pressure changes over many orders of magnitude is particularly vulnerable to corona. If the equipment itself is not continuously pressurized, it may have to be designed so that all voltages lie below the minimum COV, or provision must be made for turning off the equipment at times when corona could form. Failure to adhere to these strictures has caused launch failures in the past [3,4].

A. A Theoretical Discussion [5,6,7]

Breakdown phenomena in gases occur at all current from nanoamps to hundreds of amps. The low current discharges encompass two regions generally termed Townsend discharge and glow discharge. See Figure 1 for a representation of these phenomena. Current in the Townsend discharge range is about 10^{-10} to 10^{-6} amp, and depends only on the rate of production of ions. Townsend discharge has little or no luminosity. Current in the glow discharge range is 10^{-6} to 10^{-3} amp. In both regions changes in current are relatively independent of changes in applied voltage. In the glow discharge region the gas is luminous. In the transition from Townsend to glow discharge, the negative slope of the voltage/current characteristic causes the discharge to be unstable. At currents higher than about 10^{-2} amp the discharge is termed an arc.

Strictly, corona implies a partial breakdown at one electrode caused by a high electric field with no current flowing between electrodes. However, the term is not often used in this sense but is used for all types of low-current discharges, both Townsend and glow discharge regions. In this discussion corona will be used to refer to all low-current discharges, and arc will refer to discharges above about 10^{-2} amps.

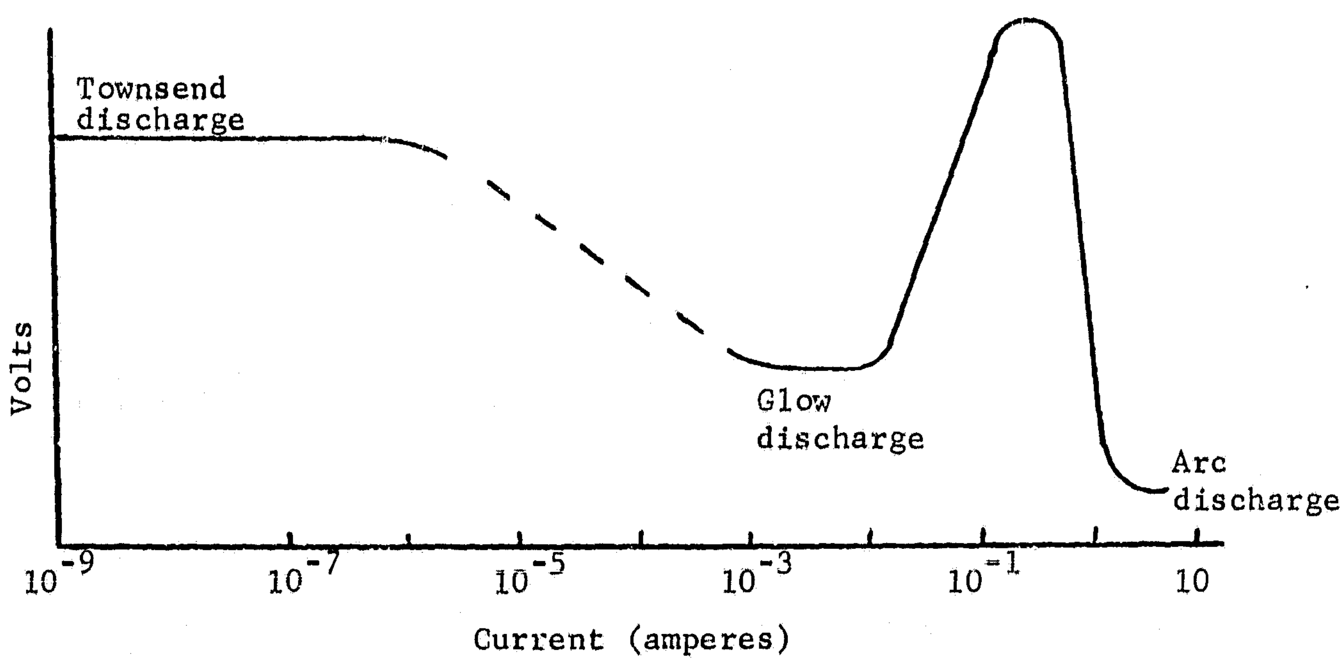


Figure 1. Self-sustaining Gas Discharge [6].

As mentioned earlier Paschen's curve of COV applies only to uniform fields. In non-uniform fields (which implies a spatial variation of field strength) the shape of Paschen's curve is qualitatively the same as for uniform fields, but the voltage at which breakdowns occur differ and are usually lower. These breakdown voltages are dependent on the exact geometrical configuration being tested since each geometry implies a different field strength for equally applied voltages. High field strengths are associated with small radii of curvature, i.e., sharp points. Non-uniform fields can also exist with uniform field geometry if there exist unequal concentrations of positive and negative charges within the gap. In the presence of such space charges non-uniform fields can exist quite easily. Non-uniform fields are often encountered in practice.

As has already been stated the term corona is usually applied to all low-current discharges. It is instructive to investigate the mechanisms which lead to corona and ultimately to arcing in low pressure areas. When a potential difference exists between two electrodes in a gas, the gas acts as an insulator until the potential is increased above a threshold potential, V_o. Current will then flow between the electrodes if ions or electrons are introduced into the gap. In practice there are always some ions present in the gap because of the natural background of ionizing radiation. There is, however, a time lag between the application of a voltage greater than V_o and the onset of corona. During the time interval a complex series of events occurs.

The steady-state value of current is determined by back-scattering of electrons from the gas molecules which lie just outside the electrode surface, and by the mean electron energy, which is related to the mean free path distance traveled by electrons. The basic equation which describes the steady-state current is:

$$I = I_o e^{\alpha d} \tag{1}$$

where I_o = the initiation current,

d = the electrode separation, and

α = Townsend's first coefficient, a characteristic of the gas.

Equation (1) is rigorously correct only for a uniform field, but it serves to describe qualitatively the steady-state current in a non-uniform field.

Equation (1) is only applicable to the primary ionization process; an equation which also takes into consideration the secondary ionization processes is:

$$I = \frac{I_o\ e^{\alpha d}}{1 - \gamma(e^{\alpha d} - 1)} \tag{2}$$

where γ = Townsend's second coefficient, representing secondary ionization effects.

Equation (2) also is strictly applicable only for uniform fields. There are several possible secondary processes. There can be secondary emission from the cathode due to the incidence of positive ions. Electrons can be ejected from the cathode by the photo-electric effect of photons which are emitted by molecules excited but not ionized by the stream of electrons in the gap. A third mechanism for producing secondary electrons is through gas ionization by the positive ions in the gas. A fourth possible mechanism is through the incidence of excited atoms on the cathode. This action is similar to the photo-electric effect except that the process results in a slower transfer of energy to the cathode. Other processes of secondary emissions are also possible. The above examples serve to indicate how an avalanche breakdown can proceed once the required initiation ion is in the electrode gap.

Equations (1) and (2) apply only to breakdown in a static dc field. Townsend's coefficients are constants, not time functions. At low ac frequencies breakdown occurs when the ac voltage reaches the value at which there would be a breakdown in a static dc field (see Figure 2); but when the frequency is increased beyond a certain critical frequency, a decrease in sparking potential occurs. This behavior may be explained by considering ion mobility. When the period of the applied voltage is

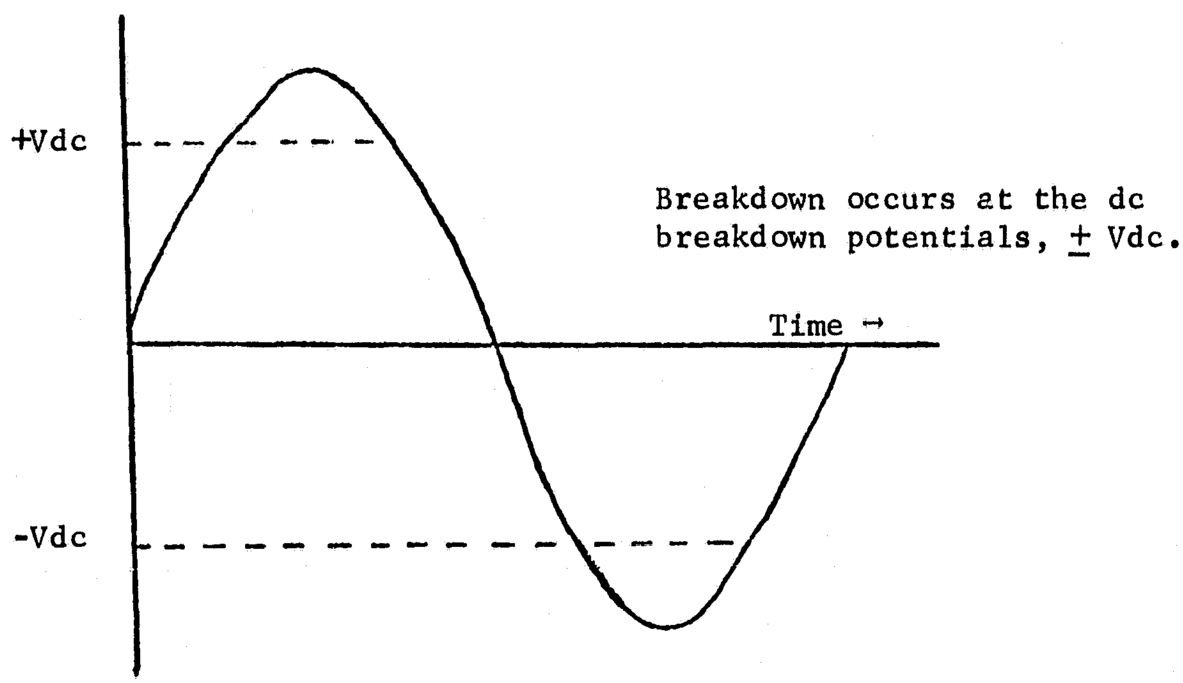


Figure 2. Ac Breakdown at Low Frequencies.

long compared to the ion transit times, the situation is essentially the same as for dc conditions; but when the period is on the order of the transit times of the ions, field reversal can occur before the ions can be collected at the electrodes. This causes the density of ions in the gap to increase, resulting in a distortion of the field. The frequency at which this occurs depends on the pressure and the molecular weight of the gases present. The static Equations (1) and (2) would have to be replaced by time functions to describe dynamic behavior when the applied voltage is high frequency ac or when the applied voltage is a pulse.

It was mentioned earlier that a time lag will occur between the sudden application of voltage and the onset of corona. The time lag can be broken down into two components -- a statistical time lag and a formative time lag. The statistical time lag, t_s, is the time required for an initiatory ion (necessary for the conductive avalanche to form) to appear in the gap after a dc voltage is applied. The appearance of the initiatory ion is a statistical event. When background radiation is the only source of ions, and if the radiation level is low, the statistical time lag will be large; conversely, if the radiation level is high, the statistical time lag will be short. The formative time lag, t_f, relates to the time required for an avalanche to develop after appearance of the initiatory ion. It is a function of ion mobilities. In laboratory experiments when the gap is purposefully irradiated, the time lag will be primarily formative; when natural background radiation is the source of initiatory ions, the statistical time lag may be significant. The total time lag is the sum of t_s and t_f.

The initiatory ion, when the radiation level is low, may originate as a field-induced electron from a cathode surface. In that case, t_s will not be a measure of a statistical radiation event, but will depend on the conditions of the cathode surface.

When there is no solid insulating material present between electrodes, the only effect of increasing radiation is to reduce the time lag to the duration of t_f, the formative lag; but when solid insulation <u>is</u> present, radiation can sometimes cause changes in the dielectric strength of the insulation, and can thus depress the COV values. Radiation can also cause dissociation of heavy organic molecules into lighter molecules, which may escape into the spacecraft atmosphere and form pockets of above-normal pressures. This, too, could cause a corona breakdown at a voltage lower than that at which breakdown would normally occur. For this reason it is useful to study the effects of radiation on equipments subject to corona.

This brief discussion of corona formation has attempted to present certain pertinent factors. There is much more to the theory of ionized gases than the material presented here; only those concepts which are essential to an understanding of corona formation have been selected and discussed.

B. Literature Survey

In order to supplement knowledge reported earlier [1,2] additional literature has been surveyed for more information on specific aspects of the corona problem. This phase of the corona studies deals particularly with the following three areas:

1. Environmental data.
2. Flat conductor cable versus round wire COV curves.
3. Corona onset voltage measurement techniques.

A summary of some of the information pertinent to COV measurements in spacecraft power systems is given in this section.

1. Environmental Data

The most obvious environmental parameters which have effects on corona formation are temperature, pressure, and radiation level. Temperature and pressure are interrelated in that high temperatures cause higher pressures for the same volume and density of gas. Radiation and temperature both contribute to the degradation of insulation materials and can thus decrease dielectric strength and increase outgassing.

Temperatures range from 175°K at 80 km to 1800°K at altitudes above 300 km. Temperature variations of 1000°K can be expected at altitudes above 130 km [1].

Spacecraft components which are exposed to the external environment can be expected to experience approximately the ambient pressure. For reference purposes a portion of the U. S. extension to the ICAU Standard Atmosphere [8] is given in Table I. The critical pressure region for experimental studies of corona formation is approximately from 10 mm Hg to 0.1 mm Hg which corresponds to altitudes from 30 km to 65 km. In a mission profile study [9] it has been predicted that the Space Shuttle will remain in this pressure region for approximately 3 minutes, with vehicle separation taking place at 60 km. As the booster returns to base it will again go through the critical pressure region. Similarly, the orbiter will pass through the critical region when returning from its mission.

Corona can form, of course, under conditions other than the critical pressure region used for laboratory experiments. There are many conceivable circumstances both normal operational and accidental where corona might occur. One normal operational circumstance is the pressurization of airlocks. An example of an accidental circumstance is outgassing due either to high temperatures or radiation damage to insulating materials. Outgassing does not occur instantaneously when the pressure is reduced; it takes place over a period of time which depends on the type and thickness of insulating material. Outgassing even into the near vacuum of an open lock could cause pockets of relatively high pressures, and thus contribute to the formation of corona.

It has been indicated [1] that the ionospheric regions from 70 km to about 400 km contain ionized gases and free electrons. The densities of free electrons range from about 10 electrons per cubic centimeter at 50 km to 90 km (D level) to about 3×10^6 electrons per cubic centimeter at 250 km to 400 km (F_2 level). The plasma conditions in the upper ionosphere can support glow discharge currents (milliamps per square centimeter) at voltages lower than the COV minima. In addition to voltage level constraints set by minimum COV, there will be still lower voltage constraints related to glow discharge and arcing conditions in those parts of spacecraft

TABLE I

1962 U. S. STANDARD ATMOSPHERE [8]

Geometric Altitude (meters)	Pressure (mm Hg)	Geometric Altitude (meters)	Pressure (mm Hg)
0	760.0	44000	1.27
2000	596.3	46000	0.99
4000	462.5	48000	0.67
6000	354.2	50000	0.60
8000	267.4	52000	0.47
10000	198.8	54000	0.36
12000	145.5	56000	0.28
14000	100.3	58000	0.22
16000	77.7	60000	0.17
18000	56.7	62000	0.13
20000	41.5	64000	0.099
22000	30.4	66000	0.075
24000	22.3	68000	0.056
26000	16.4	70000	0.041
28000	12.1	72000	0.030
30000	8.98	74000	0.022
32000	6.67	76000	0.016
34000	4.98	78000	0.011
36000	3.74	80000	0.008
38000	2.83		
40000	2.15		
42000	1.65		

power systems exposed to ionospheric plasma. The design and mode of utilization of mating power connectors between the orbiter and Space Station for example must be compatible with the expected environment. Other points of vulnerability will be external cables, bulkhead connectors, and solar arrays. Radiation degradation of insulation materials over extended mission intervals can cause system failure if external component voltages are higher than the glow discharge constraint.

Prediction of corona in a system as complex as a spacecraft is almost impossible. However, the causes and dangers of corona can be recognized. Design of the spacecraft can be planned to minimize the probability of corona formation, by holding all voltages well below experimentally determined minimum COV. It is doubtful, however, that classical experiments performed with simple geometry electrodes are meaningful for establishing such constraints.

2. Flat Conductor versus Round Wire COV

Corona onset voltage has been determined for teflon-insulated twisted wire pairs, spaced wires, and flat conductor cable [10]. In the experiments the gas was air, the temperatures ranged from 23°C to 287°C, and the pressure range was chosen to include the critical region. In addition to a comparison of flat conductors versus round wire, the COV of several electrical connectors was examined. The minimum COV for a twisted wire pair was found to be higher than for a spaced wire pair with equal wire sizes and insulation thicknesses. Minimum COV was higher for a flat conductor cable than for round conductors, but when flat conductor cable was placed close to a ground plane it exhibited a lower COV than round-wire conductors (at pressures above 20 mm Hg).

3. Experimental Corona Measurement Techniques

There are several methods which can be used to detect the onset of corona the simplest being visual observation. In a dark or almost dark room corona will become visible at a voltage slightly higher than onset voltage. The difference in voltage between actual onset voltage and the voltage at which corona becomes visible depends on several factors including the pressure, the gas present in the system, and the visual acuity of the observer. This method is in general unsatisfactory for it is dependent upon the observer.

Another method of detecting corona is to detect corona-generated radio frequency (RF) noise with a loop placed in the vacuum chamber. This also is not a very satisfactory method of detecting corona because it lacks resolution. It has been reported that some types of dc corona do not produce RF noise.

A corona detection network described in MIL-T-27B is shown in Figure 3. The device under test is in parallel with the high voltage power supply, and at high voltages this detection network suffers the serious disadvantage that it does not distinguish between corona in the capacitor and corona in the device being tested.

In 1929 Quinn devised a method of corona detection in which the device to be tested for corona was placed in the ground return of the power supply. A modification has been made of the Quinn detection network for corona measurements on spacecraft components [11]. Results indicate that this circuit, shown in Figure 4, is capable of detecting all types of corona. Other methods that have been used to detect corona are adaptations of the methods described above. In one a photo electric cell is substituted for the eye for visual observations of corona, but with some gases the initial light emission may be outside the spectral range of the photo electric cell.

C. Experimental Results

The early stages of this program of corona investigation centered on analysis of published experimental results. The literature was searched to find those COV curves which were pertinent to spacecraft power system corona. Curves were found of COV obtained with dc fields and with various combinations of air, H_2O, N_2, and O_2 [5,13]. Curves were also found with 400 Hz ac fields, and with air, N_2, O_2, and CO_2 in a range of mixtures [12]. Some of the environmental conditions which

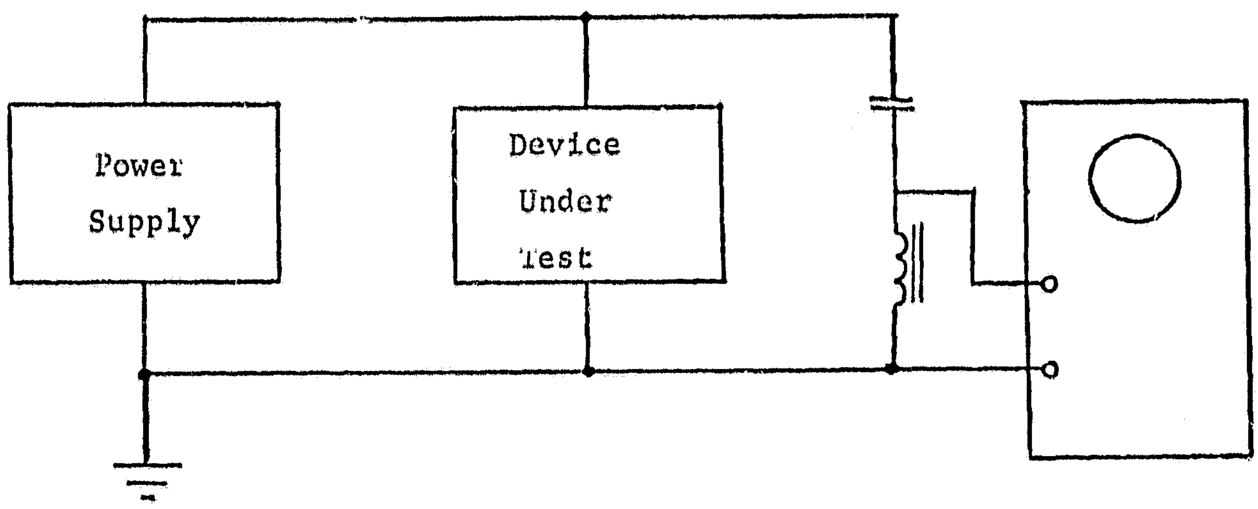


Figure 3. MIL-T-27B Corona Detection Network [11].

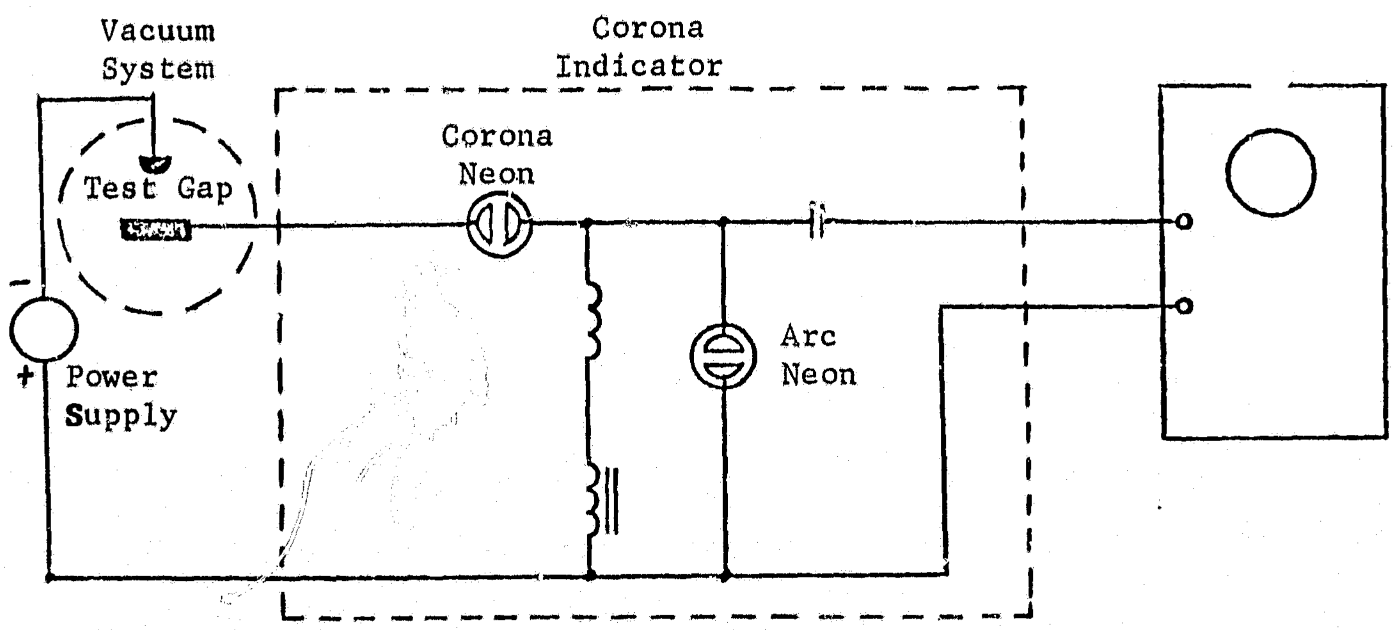


Figure 4. Modified Series Corona Detection Network.

might be encountered in future space missions were not found in reports of investigations. In particular, the effects of radiation were largely ignored. Certain anomalous results were also unexplained in many of the reports, perhaps because the theory of corona formation is not definitive enough to predict minimum COV except under very narrow constraints on geometry and environment.

An experimental program was initiated under this contract to study corona formation over ranges of laboratory conditions which would approximate conditions that might be encountered by future spacecraft power systems. An answer should be sought to the question: What is the minimum corona onset voltage in a spacecraft power system, under forseeable worst-case conditions, consistent with a fail/operate, fail/operate, fail/safe criterion [2]. Table II is a matrix of the experimental program that was initiated. The set of experiments specified in Table II would allow determination of the gas composition and the frequency at which the lowest value of COV occurs. If the radiation level and the test geometry closely model the expected spacecraft power systems, such laboratory experiments will yield data useful for specifying spacecraft voltage limits. Information about expected radiation levels for future spacecraft could not be found in the literature, and time did not permit development of a worst-case model of a spacecraft power system which would be suitable for laboratory experiments. Only a small part of the experimental program of Table II was conducted.

For the experiments that were conducted the equipment setup utilized a typical vacuum system: a bell jar and two roughing pumps capable of maintaining pressures from atmospheric downward to 10^{-2} mm Hg. Figure 5 is a photograph of equipment used. For detecting corona, a modified Quinn system was placed in series with the corona gap ground return. The schematic of the detector circuit, which is similar to the one found in the literature [11], is shown in Figure 4. A hemisphere-to-plane geometry was first chosen so that results could be compared with the results reported in the literature. The hemisphere was fashioned on the end of a partially threaded stainless steel rod, and the flat plate was machined from aluminum. To set up for an experiment the aluminum plate

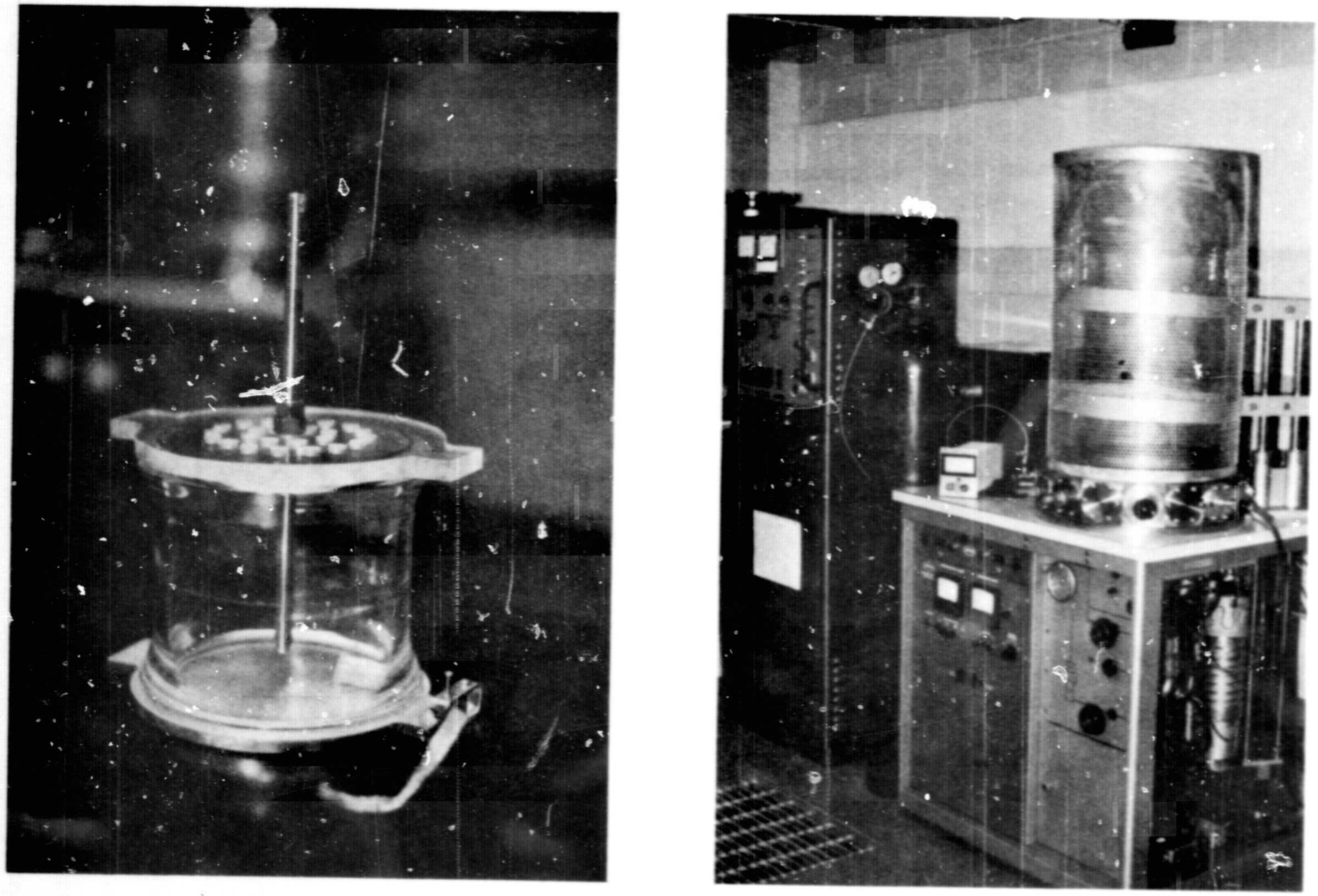


Figure 5. Laboratory Equipment Used for Corona Measurements.

TABLE II

PLANNED EXPERIMENTAL PROGRAM*

Frequency (hertz)	N_2		Air		50% N_2 50% O_2		O_2	
Dc	X	0	X	0	X	...	X	0
60	X	...	X	...	X	...	X	...
400	X	...	X	...	X	...	X	...
1200	X	...	X	...	X	...	X	...

*The experiments listed should be run on actual components or on worst-case models of power system components.

X = planned experiment.

0 = performed experiment.

was placed in the base of the bell jar, on three-inch ceramic insulators, and a glass jar six inches high and open at both ends was placed on the aluminum plate. On the glass jar was placed a plexiglass plate, perforated to equalize pressure inside and outside the jar. A threaded hole centered in the plexiglass cover supported the steel rod which was adjustable for the desired spacing between hemisphere and plane. The radiation source (when used) was housed in a small glass container with its opening oriented toward the gap between the hemisphere and plane. The radiation source was a gas lamp mantle. The radiation was low-level; a reading of approximately 5000 ions per minute was obtained with radiation counters. The isotope in such mantles is thorium 232; the emission is alpha and gamma with some beta from the radium into which thorium decays.

Experimental data was taken with three different gases: air, oxygen, and nitrogen. Air was used first to test the measuring apparatus and to demonstrate that radiation had no effect on COV when no insulating material was placed between the electrodes. (As noted in the theoretical discussion radiation only hastens the start of corona when no solid insulating material is involved.) After taking data with air as the gas, the system was pumped down to 10^{-2} mm Hg and flushed with oxygen by repeated filling and pumping down. COV curves were then run with oxygen in the system. The same procedure was then followed with nitrogen. COV curves obtained with these gases are shown in Figures 6, 7, and 8. The curves are in agreement with results reported in the literature [12].

Corona experiments with flat conductor cable were also attempted, but corona was seen to form only at the cut ends of the conductor, and the results were therefore of no significance.

The experiments emphasized the need for systems oriented tests. Wire, bus, and power transmission line tests should be made on typical wiring systems which include connectors, terminations, and other components [11]. The length of wire can be truncated in such tests, but the system should otherwise be representative of actual wiring harnesses and interconnections. The hemisphere and plane geometry is likewise not an acceptable model of a spacecraft power system. If the specification of maximum system voltage is to be related to measured minima of COV, the criterion should be based on worst-case conditions or on the limits of specified acceptability for critical system components. Measurements should be made on actual system components or on worst-case models of the system.

In addition to systems oriented testing, extensive studies should be made of the electrical breakdown characteristics of insulating materials in radiation environments [14]. Experimental determinations should be made of COV of flat and round conductor power cables over ranges of types of insulation, radiation levels, and connectors. Measurements should also be made under conditions simulating ionospheric plasma.

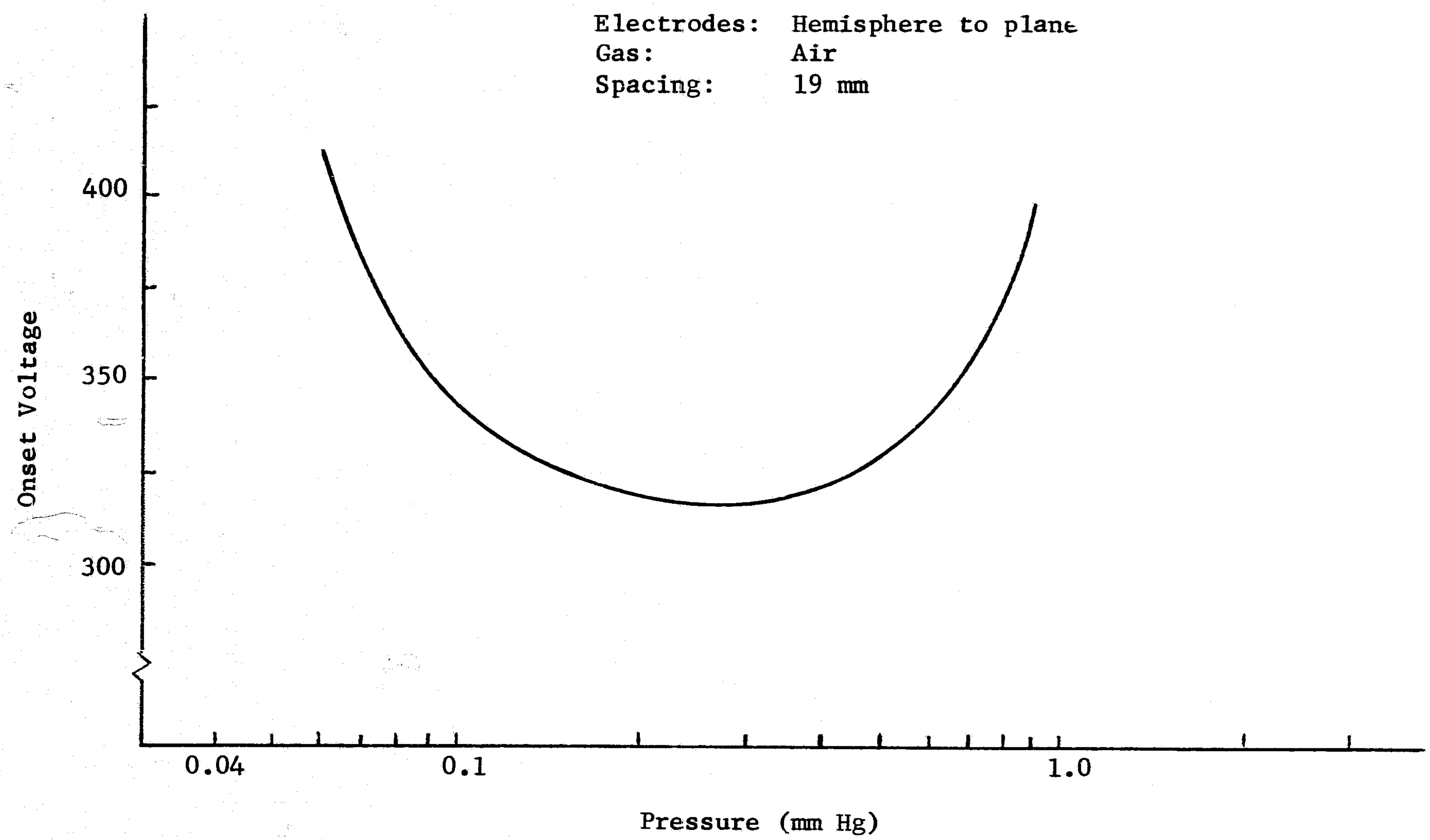


Figure 6. Corona Onset Voltage.

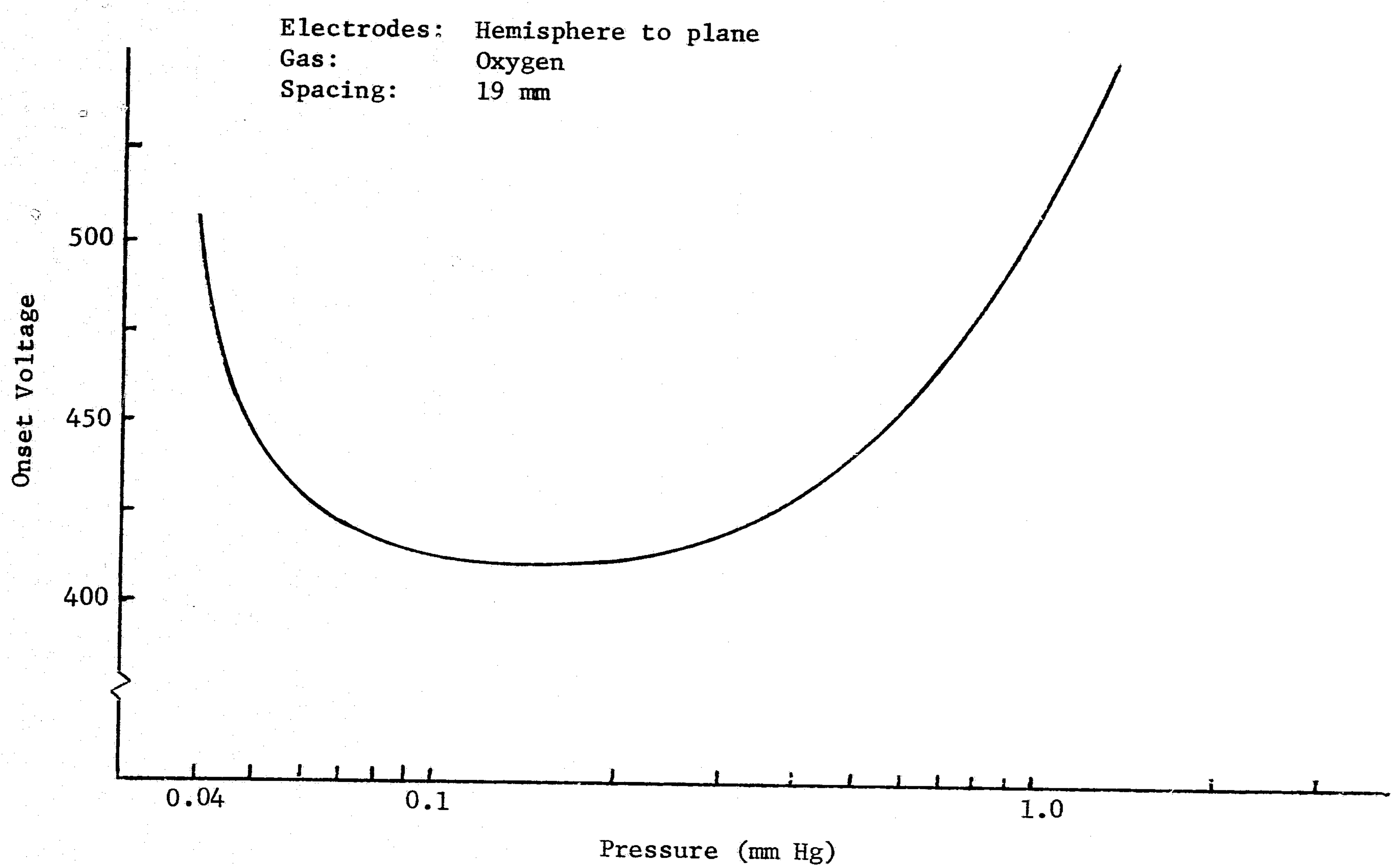


Figure 7. Corona Onset Voltage.

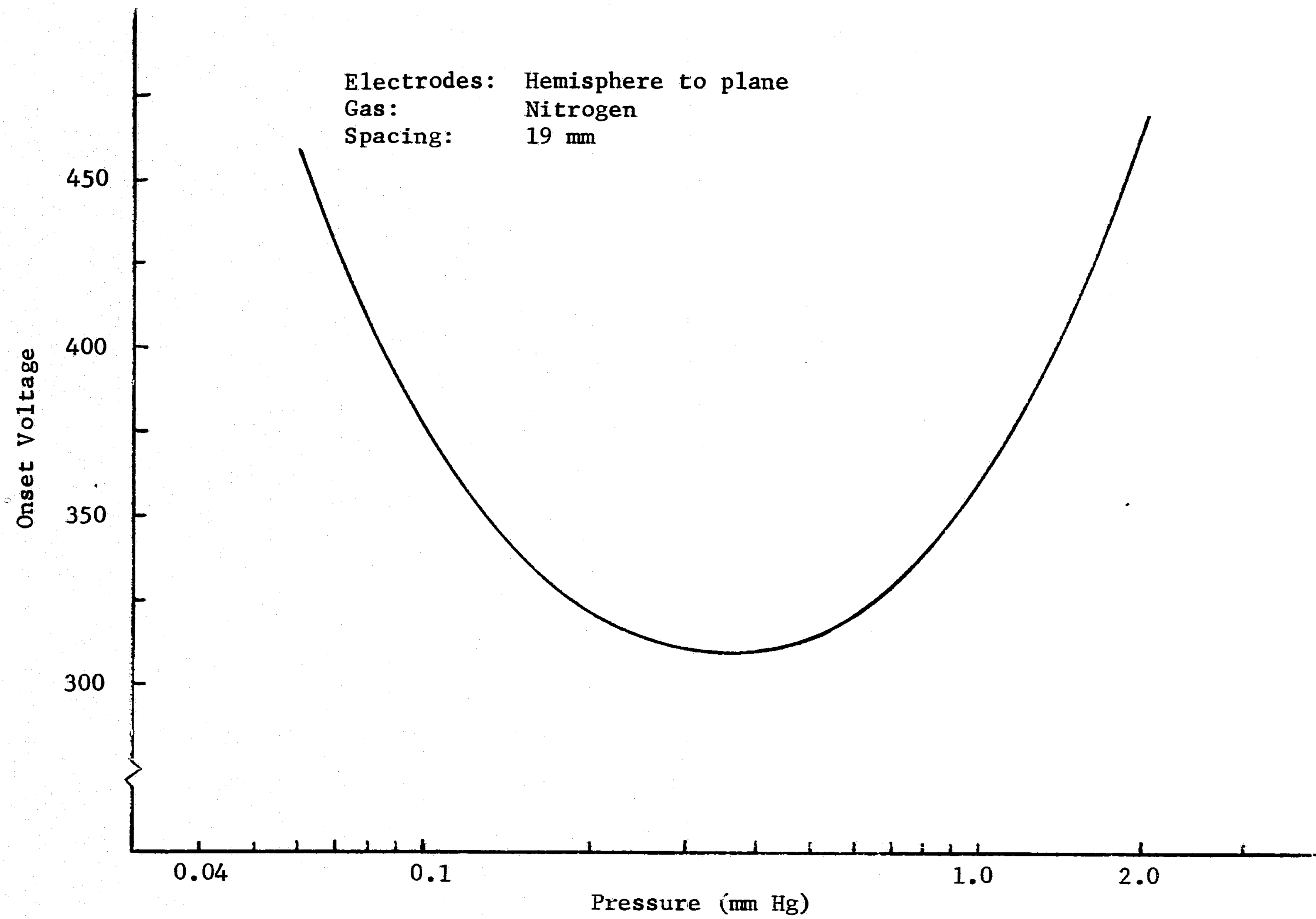


Figure 8. Corona Onset Voltage.

III. SOLID STATE DEVICES AND MULTIPLEX SYSTEMS

Proposed spacecraft power distribution systems for the time period from 1978 to 2000 will utilize a multiplex data bus system to switch solid-state devices which control power delivered to the various loads. Many problems must be overcome before such a system is made operational with the high degree of reliability required for such an essential system. This chapter discusses many of the aspects of a multiplexed power distribution system for future spacecraft. Since many of the problem areas remain unsolved and very real controversy exists in the conceptual system design, a clear approach is not evident at this time. In general many advanced concepts that have evolved in the planning stages need to be verified in experimental models before the designs of future spacecraft power distribution systems are hardened. Some of the unverified design concepts are:

1. Multiplex data bus for power distribution control.
2. Solid-state power controllers for power switching and overload control. By control of turn-on and turn-off time a substantial reduction in electromagnetic interference (EMI) is achieved.
3. Increase in system voltages to reduce wire weight.
4. Computer control of the data bus system to aid in complex load shedding, etc.

Many of these concepts have been well thought out and some have been implemented for tests. Ling-Temco Vought (LTV) has already built a data bus system, designated SOSTEL II, for their A-7 aircraft [15-21]. It is presently undergoing evaluation by the Air Force. Solid-state controllers are being produced on a limited quantity basis but many problems have yet to be answered such as compatibility of solid-state controllers with present MIL specifications, and whether solid-state controllers can be used in systems where voltages greater than 28 Vdc are anticipated. The next section discusses solid-state power controllers and their application in a multiplex power distribution system.

A. Power Controllers for Multiplex Power Distribution Systems

The emergence of solid-state devices to replace the electromechanical relay (EMR) and electromechanical switch (EMS) has been forecast for some time. As discussed in our previous reports [1,2] EMR and EMS devices have limited reliability, generate electromagnetic interference, and have limited life. In addition, the increased number of switching devices and the corresponding complexity of the switching systems for future aircraft and spacecraft make EMR's less desirable. As an example there are 506 circuit breakers in the Boeing 707 aircraft, but there are 952 circuit breakers in the (newer) Boeing 747 aircraft. Electromechanical devices cannot be replaced on a one-to-one basis by solid-state devices. The situation is in many ways similar to the slow replacement of the vacuum tube by the transistor. There remain certain features of the electromechanical devices that are superior to solid-state devices. The EMR's and EMS's can withstand greater overload, are less susceptible to radiation, and have lower voltage drop and higher isolation resistance. Because of these features it may be expected that EMR's will continue to find applications in future spacecraft power distribution systems, except where advantages of the solid-state devices outweigh those of their mechanical progenitors. Many different types of solid-state devices have been developed in the last few years: the hybrid relay, the solid-state switch and the power controller. There are two types of hybrid relays. In the first type solid-state control circuitry drives the coil of an electromechanical relay and the load current flows through the EMR contacts. In the second type an electromechanical relay in the control circuit turns on the power switch which can be either a bidirectional thyristor (triac) for ac or a transistor for dc operation. The solid-state hybrid relay possesses the advantage of switching large currents but has a long response time like the EMR. Solid-state switches have been developed that have a fast response time, but they are usually of limited power rating. However, the most important single device that is being developed is the solid-state power controller since this device performs a multitude of essential functions that include:

1. Switching bus and load power.
2. Overload protection.
3. Reduced electromagnetic interference.
4. Trip indication and other automatic indication required by a multiplex system.

When one adds the desire to extend the voltage and power levels of these devices, it becomes evident that considerable development effort is needed to produce such a device. For these reasons only a limited quantity of power controllers have been produced and the voltage and power rating of these units are somewhat limited.

One manufacturer is offering both ac and dc power controllers; the dc units operate at 28 Vdc with current ratings between 1 and 35 amps, the ac units operate at 115 Vac, 400 Hz, with current ratings between 1 and 70 amps [22].

1. Dc Power Controllers

Dc power controllers use transistors for switching because of their unidirectional properties. The difficulty of switching dc power is well known and is associated with interrupting a constant value voltage. For this reason the rating of dc power controllers has generally been limited to 28 Vdc with current ratings from 0.5 amp to 35 amp, but a 200 Vdc, 5 amp relay has been recently developed [23]. The block diagram of one typical dc power controller, as shown in Figure 9, consists of five interconnected circuits that control the flow of power from the dc bus to the load on command from a control line [24]. The five components are:

1. Power switch.
2. Power control module.
3. Level shifting module.
4. Logic control module.
5. Power regulator.

The combination of basic circuits provides the switching, overload protection and trip indication required of the power controller. The power switch is an NPN transistor operated in the saturated mode to supply the bus power to an individual load. This transistor is mounted on a thermal capacitor

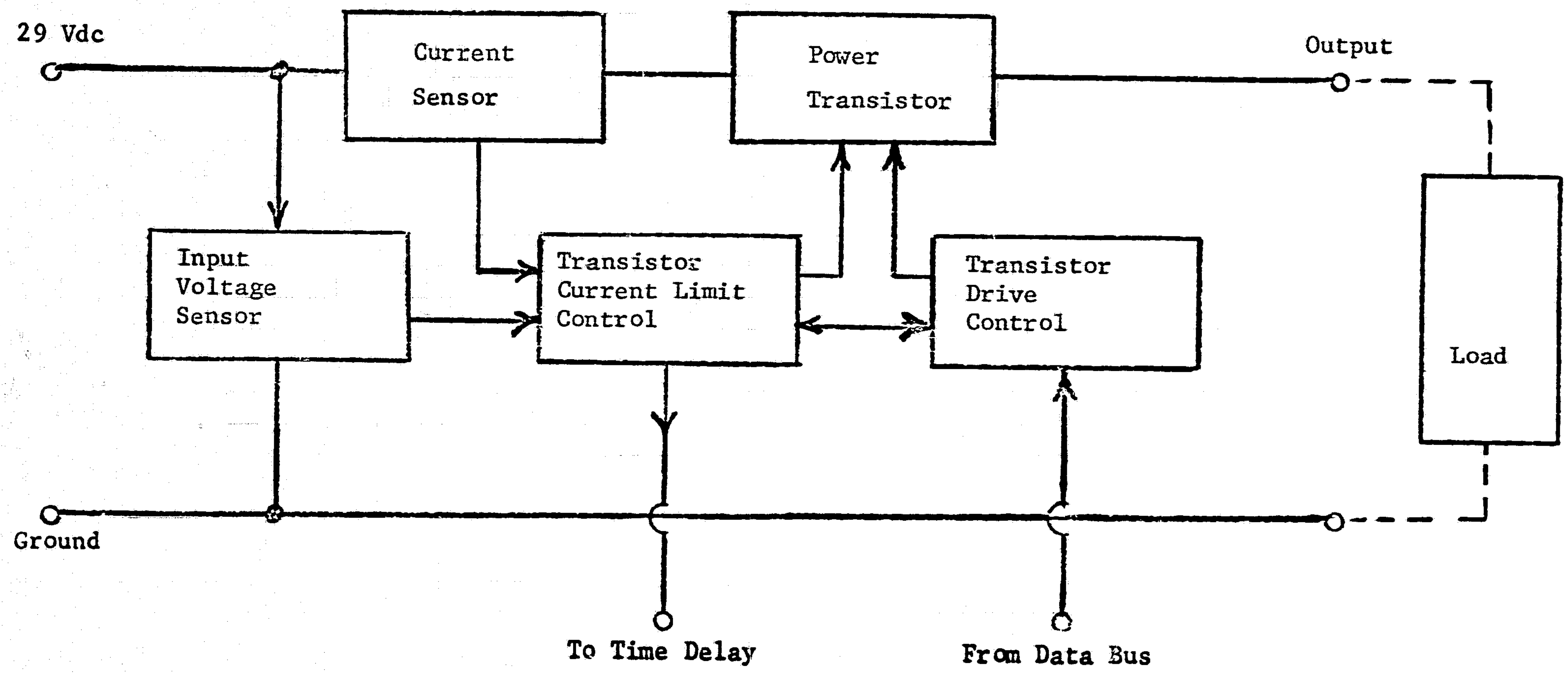


Figure 9 Dc Power Controller [24].

block to prevent the junction temperature from exceeding 150^{o}C. There is much controversy regarding the desirability of the power switch withstand ... the 80 Vdc transient and 600 Vdc spikes required by MIL-STD-704A, and this will be discussed in a later section. The problem areas associated with the power switches include limited voltage ratings, vulnerability to damage from transient voltages and overload currents, and relatively large voltage drops across the switches. Drops of 0.5 Vdc to 1.5 Vdc are typical, and constitute serious power losses and voltage droop.

The voltage transient spike can be reduced by a Zener diode built into the load circuitry; and the mechanism which limits overload current, in the switch of Figure 9 also affords protection. A typical overload characteristic for a power controller is shown in Figure 10. MIL-P-81653 requires that the controller tripout at 130 to 150 percent of rated current. The objective of the characteristic is that circuit trip will occur before the maximum junction temperature is reached. The power control module senses load current and turns the power switch off if overload occurs. The level shifting module is a dc to dc converter. The logic control module processes the control signal to initiate switching, and the power regulator provides tightly regulated power from the unregulated load bus.

In another (28 Vdc) power controller design it is claimed that load current can be limited to 1.3 to 1.5 times rated current, and that the trip-out time constant can be set to 2 to 3 seconds [17]. Superior performance is also claimed for the condition when motors must be started at low bus voltage.

It can be seen that a standard design for the power controller has not been achieved. Also the applicability of MIL-STD-704 is questionable as are some of the specifications of the new MIL-P-81653 that apply to solid-state power controllers.

2. Ac Power Controllers

Where transistors are the switching elements in dc controllers, silicon controlled rectifiers and other bidirectional devices such as triacs are the most promising devices for ac power controllers. Since

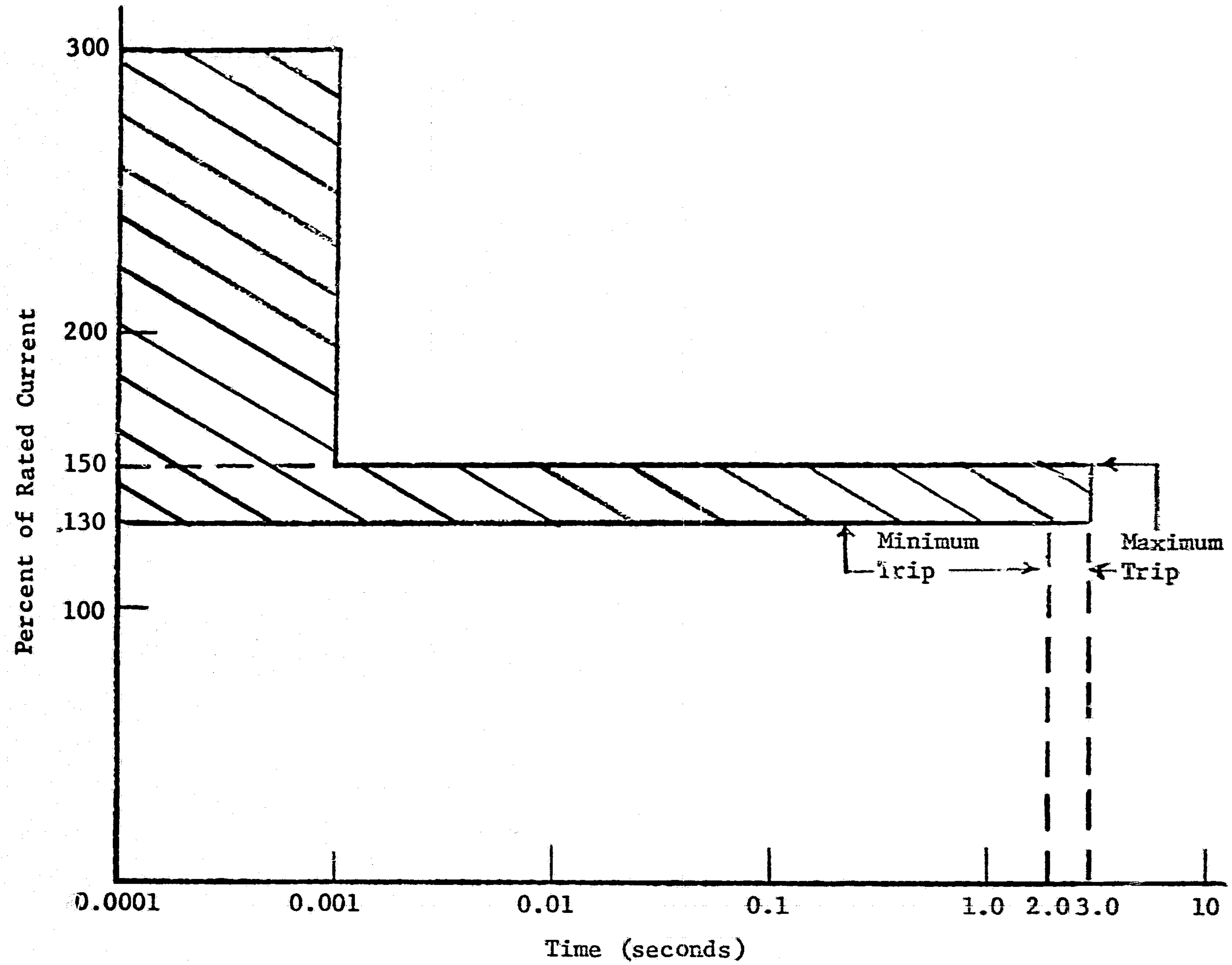


Figure 10. Trip Characteristic for Dc Power Controller [24].

the nominal voltage for ac loads is 115 Vac or higher, an inverse parallel connection of two devices can be used. By switching on at zero voltage and off at zero current, the switching element requirements are minimized so that much higher power loads can be switched with presently developed ac controllers than can be switched with dc controllers.

The circuit configuration of an ac power controller is similar to the dc power controller shown in Figure 9, but there are several exceptions. The ac power regulator must rectify and regulate the ac bus voltage, and the overload protection circuit must switch the power on at zero voltage and off at zero current. A typical overload characteristic for an ac power controller is shown in Figure 11. The protection circuit does not provide current limiting (current limiting is a feature of dc controllers, as shown in Figures 9 and 10). In order to provide overload protection for ac power, the controller must withstand the full short-circuit load current for at least one-half cycle so that the control signal can switch off at zero current. In highly inductive circuits the ac controller must withstand the short-circuit current for a complete cycle (2.5 milliseconds with 400 Hz power) with only the bus impedance and the internal impedance of the controller to limit the current. Normally this means that the power controller must have a rupture capacity of about ten times its rated capacity. This specification, and the need to handle large in-rush current, are state-of-the-art limitations of ac power controllers. As seen in Table III, many ac motors have very large in-rush currents [18].

It can be seen by examining Table III that although the compressor in-rush current is only 5 times the operating current, it has a duration of 4 seconds which is 3200 times as long as a half-cycle time with 400 Hz power. For motors, the power controller must be equivalent to a slow-blow fuse. It is expected that many of the aircraft loads of the space shuttle will have characteristics similar to the compressor; it may be necessary to parallel several power controllers for such power demands. The choice between solid-state power controllers and electromechanical relays for switching large loads is not yet clear.

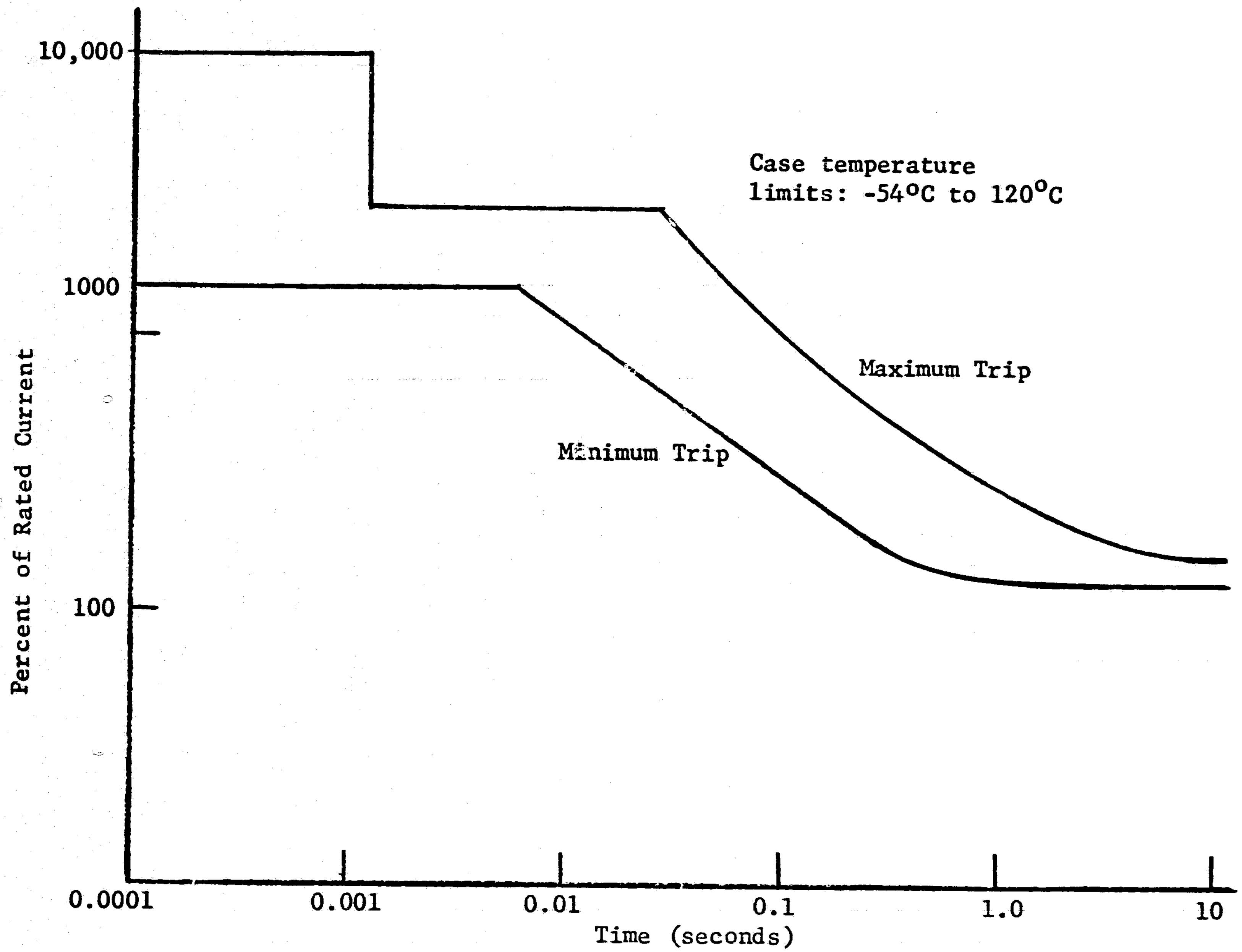


Figure 11. Trip Characteristics for Ac Power Controllers [15].

TABLE III

START AND RUN CURRENTS OF AC LOADS [18]

Load	Running Current (amperes)	In-Rush Current (amperes)	In-Rush Duration (seconds)
Compressor No torque-limit	69	308	4
Condenser Fan	40	121	2.7
Recirculation Fan	37	179	1.2
Stabilizer Trim No load	10	27	0.2
Water Injection	23	200	0.23

3. Bus Controllers

Ac power controllers must switch power into buses as well as into loads. Solid-state bus controllers for 115 Vac, 70 amps, have been developed and it is expected that devices with increased rating will be produced in the near future. Dc bus controllers with ratings of 28 Vdc, 35 amp, and 115 Vdc, 10 amp are also available [22]. This is probably inadequate for the dc bus system for Space Shuttle. Future work in dc solid-state power controllers should aim at ratings of 115 Vdc,70 amps. An alternative would be a hybrid controller which uses solid-state control with an electromechanical relay. The use of electromechanical relays to switch 115 Vdc buses is feasible.

4. MIL Specifications

MIL-STD 704 is still being applied to state-of-the-art and to future designs for aircraft and spacecraft. This specification requires that equipment survive such overloads as 600 Vdc spikes and 80 Vdc transients on 28 Vdc systems. Although most of the solid-state power controllers meet these tests, there is much controversy as to their applicability.

The power form for future spacecraft will be closely regulated. Spikes and transients that in older systems were primarily due to electromechanical relays will be greatly reduced in solid-state systems by overload control, switching at zero voltage and zero current, and the elimination of the noise producing coil of the electromechanical relay. It appears that either MIL-STD-704 must be altered or a new equipment specification must be generated. A new component specification, MIL-P-81653, details power controller performance; some of its features are:

1. Turn on time: 0.1 to 0.5 msec.
2. Turn off time: 0.5 to 5 msec.
3. Control signal: 5 Vdc, nominally 100 mw.
4. Power controller provides a trip status signal to the Master Control Unit (computer).
5. Reset controlled by the Master Control Unit, when a signal from the power controller is present.
6. Overload characteristics of dc controllers adhere to MIL-STD-704 (between 1.3 and 1.5 P.U.).
7. Overload tripout delay times of ac controllers are inversely related to the overload current to protect the wiring. (Power limit rather than voltage limit.)
8. Ac power controllers have a one-cycle rupture capacity of 4000 amps.
9. All power to operate the control electronics in the power controller (except the control signal) must be supplied by the load bus.
10. Voltage drop is 1.0 Vdc max for dc controllers, and 1.5 Vac for ac controllers.

Many of the features of MIL-P-81653 are readily met by controller suppliers but conflicts between this specification and MIL-STD-704 exist in the area of overload control and transient suppression [17]. Because of these conflicts and the very early state of hardware development, many vendors have taken exception to many sections of the specification. One manufacturer [17] designed a controller to use a 28 Vdc control signal and modified the trip indication to be a status indication, reportedly with a saving in a power supply. The prototype power controllers furnished to NASA/MSFC by a second manufacturer do not meet the trip and overload characteristics of MIL-P-81653.

A third manufacturer has designed 28 Vdc solid-state power control hardware that takes exception to many features of the specifications [19]. The biggest innovation in this switch is that power switch junction temperature is sensed and the signal generated by overload current switches the controller off. Since the junction of a silicon device can operate to about 150°C this technique appears to be worthwhile if the design can compensate for the ambient temperature range and the system wiring can be simultaneously protected. (It is quite possible that the junction temperature can remain below 150°C while the bus wire and load wire evaporate.) Perhaps junction sensing should be used as secondary protection. The exceptions to MIL-P-81653 in this third example are:

1. The requirement of 0.5 Vdc drop at rated current is increased to about 0.8 Vdc. The claim is made that this permits simplifying the control electronics without a significant change in dissipated energy.

2. Control and reset impedance are changed to allow control operation between 3 Vdc and 32 Vdc.

3. The trip indicator operates on the same voltage as the control signal.

4. The tripout circuitry is based on junction temperature sensing.

5. The switch turns off at 39 Vdc to protect the power switch. Transient voltage spike protection appears to be deficient however.

As is seen from this discussion of prototype hardware, there are many problem areas to be resolved before new specifications will gain acceptance. A recent article [20] discusses some of the problems connected with resolving the problem of specifications for solid-state power controllers. The conflicts between MIL-STD-704, MIL-SPEC-P-81653, and state-of-the-art should be resolved so that workable, low cost, solid-state power controllers can be designed into future spacecraft and aircraft, including Space Shuttle.

5. Experimental Effort

During this reporting period the experimental effort concentrated on evaluation of solid-state switching devices. Several prototypes of solid-state devices were furnished to Georgia Tech by NASA/MSFC. Switching characteristics at ambient conditions were measured in this evaluation. Of the two types of solid-state relays tested, one had a load rating of 2 amp at 200 Vdc [23]; the other had ratings of 0.25 amp and 1.0 amp at 6 Vdc [25]. The tests consisted of the application of rated voltage at rated current for the 6 Vdc devices. The 200 Vdc devices were found to have excessive contact drop at reduced voltage and were not tested at rated voltage. The manufacturer of the 200 Vdc switch has redesigned the power output transistor circuitry and claims that the redesigned relay has an acceptably low contact drop, 0.5 Vdc. Leakage tests were also conducted on the 6 Vdc switches; the data in Table IV is a mixture of data taken at Georgia Tech and data taken by the vendor when the switches were returned to him for test. The data for the 200 Vdc switch is given in Table V. From the results of these tests it can be readily seen that the solid-state relay field is still in the development stage and that much work remains to be done before production hardware is available. It is encouraging to note that units are available which operate at voltages as high as 200 Vdc, and that isolation resistance and leakage appears to be quite low so that solid-state devices can supply all but the most sensitive loads. The development of solid-state power controllers, and the state-of-the-art in power transistors and silicon controlled rectifiers should be followed closely so that this new technology can be incorporated into the designs for future spacecraft.

B. Multiplexed Power Distribution Systems

Multiplexed systems to control the distribution of power have been discussed in our previous reports [1,2], where it was pointed out that the

TABLE IV

NANOTRON [25] RELAY TESTS

Relay Catalog Number	Tests					
			Drive Circuit			
			Operate			
	Leakage Current (micro-amps) [1]	Contact Drop (volts) [2]	Voltage (volts)	Current (milli-amps)	Release (volts)	Withstanding Voltage (volts) [3]
NSW 2001	0.001	0.8	3.1	0.15	1.1	500
NSW 3001	0.05	0.85	3.95	0.15	1.4	500
NSW 4001	0.05	0.85	3.5	0.15	1.5	500

1. Leakage Test: All terminals shorted except one, positive voltage applied to the unshorted terminal and negative to the shorted terminals.

2. Load Test: Contact voltage drop measured between both normally open and normally closed contacts and the supply terminal. All relays tested at 18 Vdc supply voltage at 250 milliamps.

3. Withstanding Voltage: All terminals to case.

TABLE V

ELECTRONIC SPECIALTY [23] RELAY TEST

Unit	Load Current (amperes)	Contact Drop (volts)	Load Voltage (volts)
No. 1 (14 Vdc operate)	0.5	1.0	6.7
	1.0	7.2	19.0
	1.3	18.0	33.0
No. 2 (19 Vdc operate)	0.5	1.2	7.0
	1.0	3.3	15.5
	1.5	11.5	28.0
No. 3 (17 Vdc operate)	0.5	1.37	7.1
	1.0	5.3	17.5
	1.4	16.0	32.0

advantages of a multiplex system over a conventional system are:

1. Increased reliability.
2. Lower weight.
3. Lower mean time to repair.

A multiplexed power distribution system, the SOSTEL II, has already been built for the A-7 aircraft [21]; and the concepts appear to be directly applicable to the space shuttle as well as to future spacecraft. A mockup of SOSTEL II is being tested by the Air Force. The system consists of a computer (Master Control Unit) that controls the power distribution system; remote input-output units; and solid-state power controllers that are located between the generators and the buses, and also between the buses and the loads. Considerable savings in weight are realizable by transmitting command and data information on the data bus. Binary coded signals can be routed to remote output units on a single pair of twisted cable. The large amount of cabling required in present spacecraft for control and status indication can thus be reduced. Increased reliability is obtained with the data bus system through a built-in test program which continuously monitors the system. In addition, line replaceable units can be used to replace faulted units when they are detected. Safety features of power controllers, which will prevent fires when overloads occur, include current limiting and internal fusing. Goals for improving performance on the space shuttle through the use of a multiplexed system are:

1. Reliability increased five times.
2. Weight reduced to one-half.
3. Mean time to repair reduced to one-fifth.

Obviously the most rewarding feature of a multiplexed system is the increased reliability.

Some of the design features of the SOSTEL II multiplexed power distribution system are:

1. Lossy twisted pair leads for data bus to make the system immune to short-circuit on the data bus.

2. Separate lines for address and reply.

3. Time division multiplex signals.

4. Fault detection in the transmission word code, to isolate faults and to switch operating units.

5. A transmission word structure designed to insure data security.

The transmission word structure, which is shown in Figure 12, requires 240 kilobits per second on the address line and 375 kilobits per second on the reply line [21]. Although this bit rate could possibly be reduced, it can be seen that scanning the entire power distribution system 100 times each second will require a high bit rate. The information rates in Figure 12 are for SOSTEL II; the bit rate would be expected to be even higher for the space vehicle if it has a larger number of remote input/output units.

Many of the principles that are used in the SOSTEL II data handling system can be carried over to Space Shuttle, but the transmission word structure and the type of modulation must be redesigned for system compatibility with the other computer requirements of Space Shuttle. A central computer is presently planned for the entire spacecraft; some

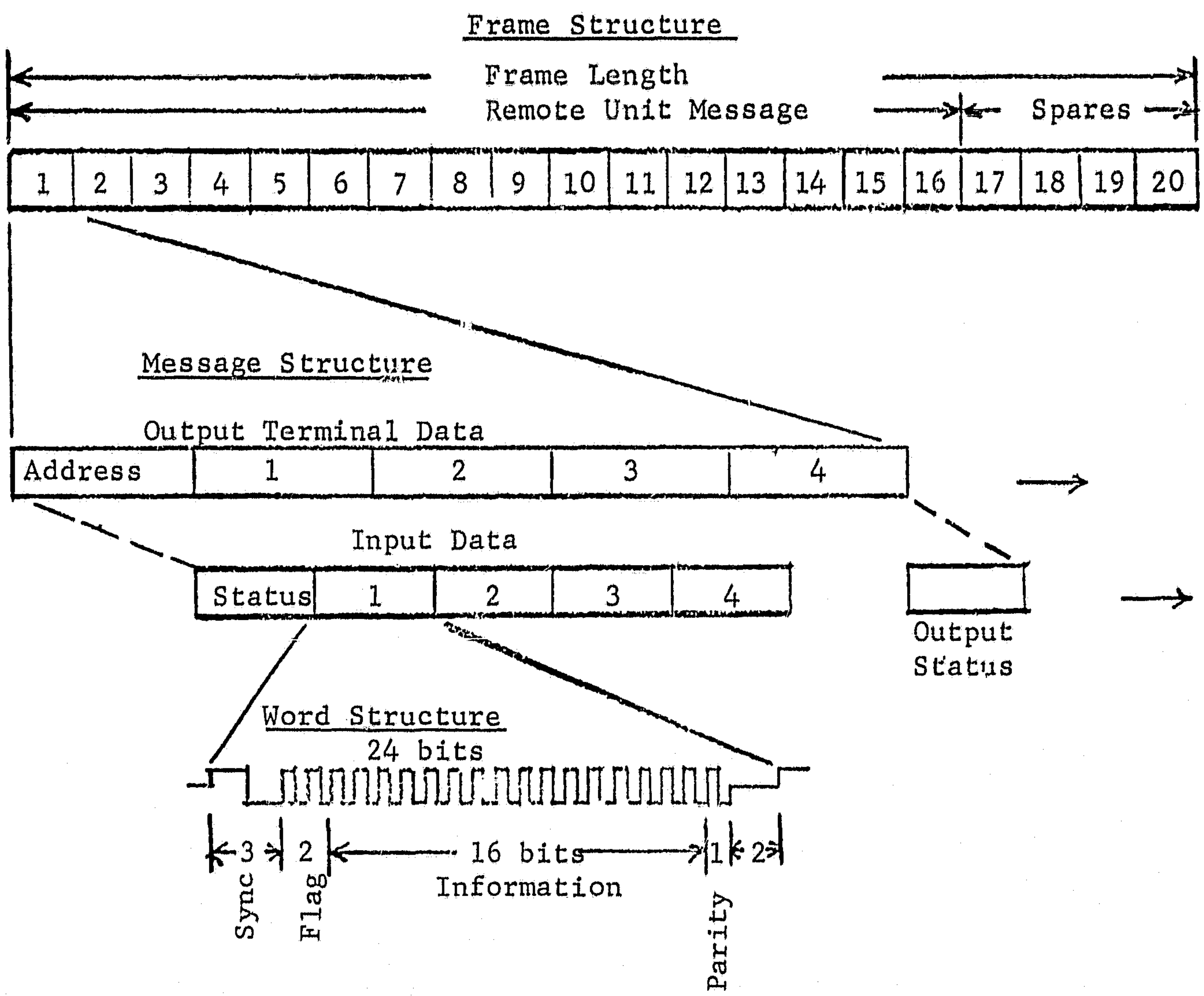


Figure 12. SOSTEL Time Division Multiplex [21].

consequences of integrating the power distribution system into the large computer complex are discussed in the following section.

1. A Dedicated Data Bus versus a Central Computer for Space Shuttle

NASA power system engineers and contractors have generally advocated a dedicated data bus for the following reasons:

1. Lower bit rate: With a single central computer the computer must distinguish between the data from the power systems and the data from other systems.

2. Essential nature of the power system: The power system is vital and its operation should not be affected by other components of the spacecraft.

3. Easier fault detection and isolation.

The only realistic argument for a single central computer is the desire to do all of the computation in one unit. The triple failure criterion [2] for Space Shuttle (fail/operate, fail/operate, fail/safe) will require considerable central computer redundancy. In a dedicated system, three master control units would satisfy the failure criterion and would possibly relax the computer redundancy burden.

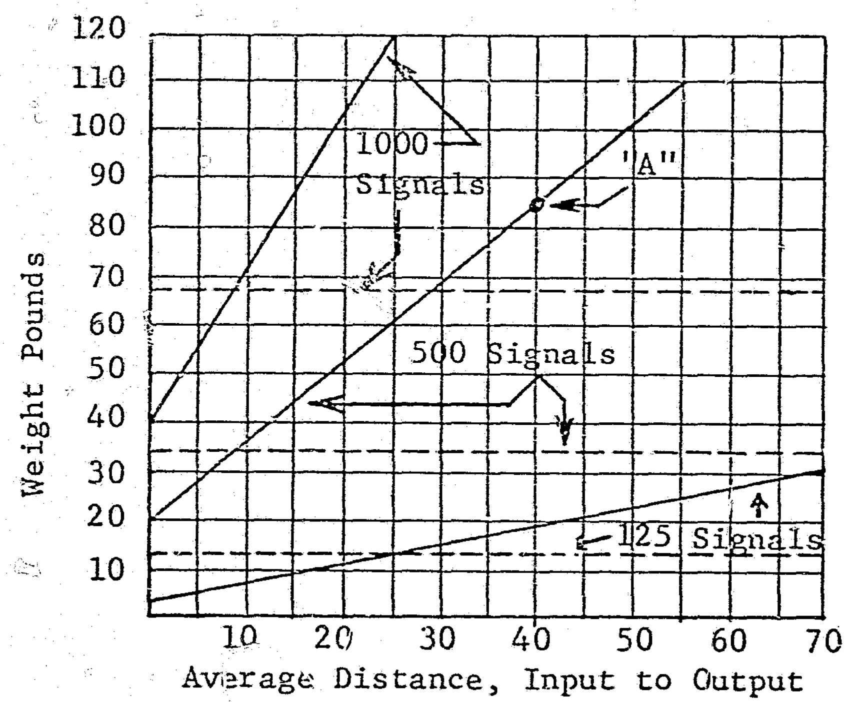
If the master control unit is integrated into the central computer it is expected that, because of the higher-frequency information required by the other spacecraft systems, the power control signal structure will not be optimum for power control. Time division, pulse code modulation was selected for the SOSTEL II system because of the relatively low information rate required by the power distribution system. Signal design changes that will probably be required for a central computer system include modulation, demodulation and detection methods, bit rates, coding techniques, and message format. Frequency multiplex has been suggested.

A full tradeoff study is needed to determine whether the data bus multiplex control of power should be a dedicated, separate system, or share a centralized computer with other systems in future spacecraft. The question of power system dynamic stability should be included in the analysis.

2. Weight and Cost Effectiveness Study of Solid-State Multiplexed Power Distribution Systems

Extensive tradeoff studies of spacecraft power systems have been made by McDonnell Douglas Corporation, and by North American Rockwell Corporation [26,27]. The studies involved the choice of candidate power sources and the choice of voltage or power form for the distribution system. Unfortunately, the two studies often reached different conclusions because the calculations for cost and weight were based on experiences with previous spacecraft and aircraft. Since the basic data is not published, it is difficult to evaluate the conclusions. At best then, published weight and cost effectiveness studies can only serve as guides in planning future spacecraft, for which desired hardware is not yet developed.

For a long time there has been strong motivation to improve aircraft and spacecraft power distribution and control systems. Power form of existing systems have very loose tolerances, serious EMI problems, low reliability and high weight. To be acceptable on a cost effective basis, a new system must show improvements in all the areas where existing systems are weak [28]. The ever increasing complexity of aircraft and spacecraft systems also motivates the redesign of their power distribution systems. One of the first systems that was developed to improve the power distribution system was the SOSTEL II system for use in the A-7 aircraft [16]. This system uses a central computer, a data bus multiplex system, and remote input and output terminals to control power distribution in the aircraft. Figure 13 shows the relative weights of this multiplexed system and a corresponding relay control and distribution system [16]. From the weight curve it is seen that for a 40-foot distance between the input and output the hardwired system weighs 84 pounds and the multiplexed system weighs 36 pounds. It is expected that Space Shuttle will require between 1000 and 1100 signals and will be somewhat longer than the A-7 aircraft, so that this weight advantage of a multiplexed system will be accentuated. The weight penalty is $25,000/lb for the orbiter and $2,500/lb for the booster [27], so that there is a greater cost advantage for the orbiter in using a multiplex system. For Space Station and Space Base



-----Multiplex System

——Hardwired System

Weight Breakdown

Hardwired System:

Signal wiring No. 26 AWG @ 3.2 lb/1000 feet, one run/signal.

Relay logic - 1/16 lb/relay x .64 relay/signal = .04 lb/signal (adds to wire weight).

Multiplexed System:

Signals	125	500	1000
Remote terminals (1.2 lb/terminal 32 signals/terminal)	4.8	19.2	38.4
Central unit	5	6	8
Connecting signal wiring (5 feet per signal)	2	8	16
Additional transmission line, ties to multiplex units (16 lb/1000 feet)	1.6	3.2	6.4
Total weight (pounds)	13.4	36.4	68.8

Sample calculation, point "A":

40 ft./signal x 500 signals x 3.2 lb/1000 feet =
(40)(500)(3.2)/1000 = 64 lb.
0.04 lb/signal x 500 = 20 lb.
Total = 84 lb.

Figure 13. Weight of Multiplexed Systems [15].

the weight advantage of a multiplexed system is even greater since these systems are even more complex and longer.

The improvement of the SOSTEL II over a conventional system is given in Table VI [16]. The listed improvements compare favorably with the goals of the SOSTEL development program. Undoubtedly the most significant improvement is in the reliability. It is somewhat difficult to make a cost estimate of a multiplexed system at this time because of incomplete data, but the increase in reliability and maintainability indicate that the total ownership cost of the multiplexed system will be less than the cost of a conventional system. (Total ownership cost includes the initial cost plus the cost of maintenance over the life of the equipment.)

Figure 14 is a block diagram and Table VII is a description of the SOSTEL II system [21]. It consists of 16 bi-level remote input terminals and 16 bi-level remote output terminals. Each terminal has 64 input or output lines. There are two Master Control Units in the control center. The system is fail/operational since failure of any "black box" can be detected by BITE (built-in test) and redundant units can be switched into the system. A sufficient number of "black boxes" would provide the fail/operational, fail/operational, fail/safe requirement of the space shuttle. Figure 12 shows the structure of the multiplex control signals. Time division multiplexing is employed because of the relatively slow system

TABLE VI

ADVANTAGES OF SOLID-STATE DISTRIBUTION SYSTEMS FOR AIRCRAFT [16]

Parameter	System: Conventional	System: Solid-State	Improvement (percent)
Weight	576 lbs	315 lbs	45
Volume	2756 cu in	1500 cu in	45
Reliability	498 fail/10^6hr	116 fail/10^6hr	77
Maintainability	0.014 mmh/flt hr	0.0024 mmh/flt hr	80

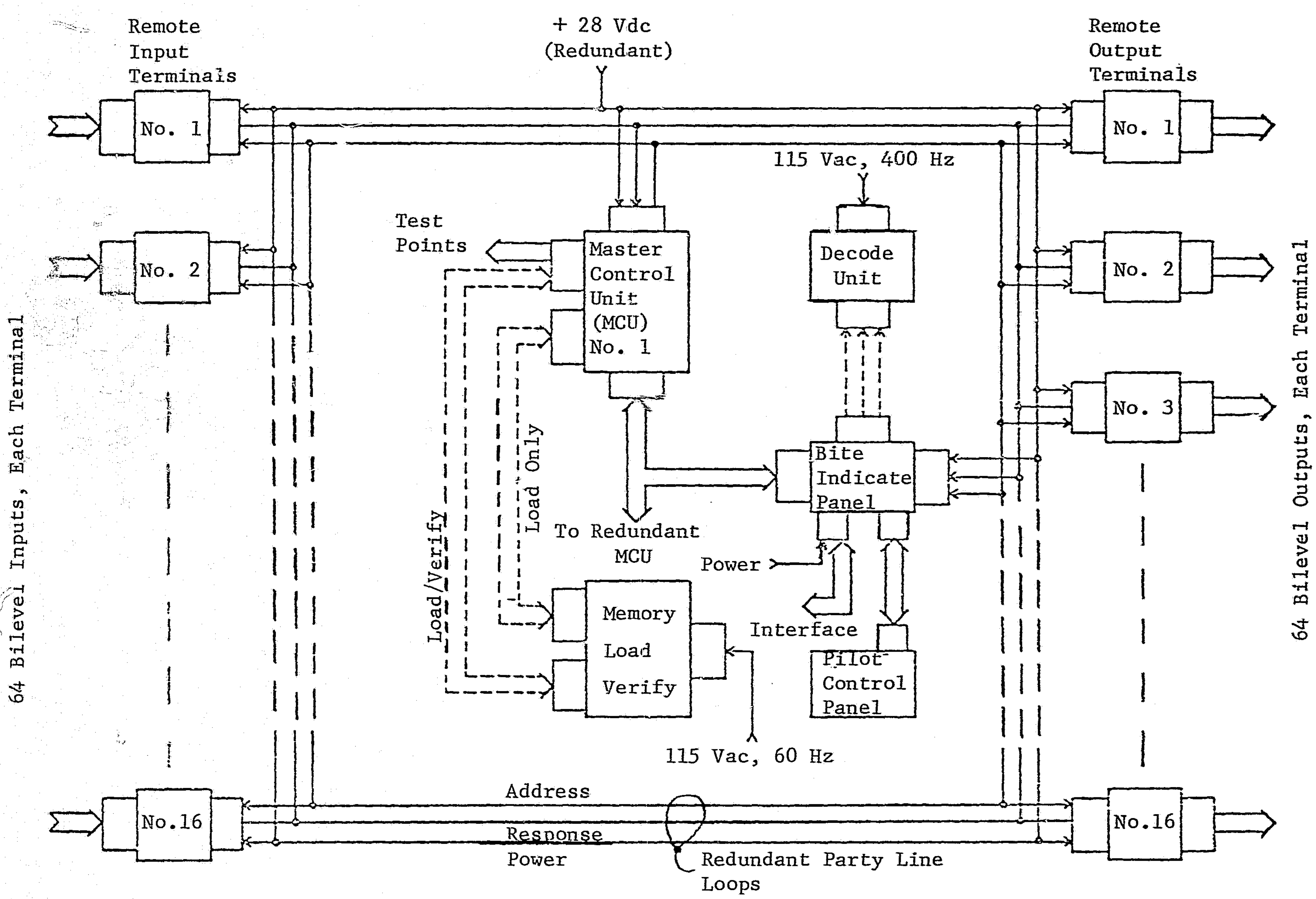


Figure 14. SOSTEL Data Bus [21].

TABLE VII [21]

SOSTEL II - DATA HANDLING SYSTEM CHARACTERISTICS

Remote input terminals (redundant)	16 max (bi-level)
Remote output terminals (redundant)	16 max (bi-level)
Inputs per input terminal	64 bi-level
Outputs per output terminal	64 bi-level
Input channel capacity	1024 max
Output channel capacity	1024 max
Data channels for test	16
Master units	2
Data access rate (frame rate)	100 samples per second (sps)
Effective output rate	50 sps (2 sample integration)
Messages per frame	20 (16 remotes + 4 [illegible]re)
Words per message	11 (5 address + 6 response)
Address bit rate (master to remote)	240 kilobits per second
Response bit rate (remote to master)	375 kilobits per second
Programmable memory capacity	2304 words @ 16 bits per word (expandable to 4096 X 16)
Logic equation capability	500 (expandable to 1000)
Input/output memory capacity	1024 bits
Scratch pad memory size	64 bits
Transmission lines	2 dual redundant party lines (1 set address + 1 set response) full duplex, looped, coupled, lossy, twisted pair shielded
Programmable	Electronically using memory loader/verifier
Bite	Redundant, continuous monitor automatic switch-over with manual override
Maintenance test	Pre/post-flight self-contained test

response time. Two multiplexed signals, one an address message and the other a reply message, are transmitted on separate data buses. The address messages are transmitted on an address bus between the master control unit and remote terminals, while status and sensor data are transmitted on the reply line. A lossy twisted pair cable has been selected for the data bus since it can operate with an open circuit on the line and also is operational to within a short distance from a short-circuit. It is anticipated that many of these concepts could be retained for future spacecraft. The control signal transmission structure might be redesigned to make more efficient use of data; about 20 percent of the time slots in the SOSTEL signal structure are unused. Probably the long synchronization gate (for secure transmission) and two-cycle integration, where two successive transmissions to a remote terminal must match before an output state is changed, should be retained. In BITE all terminals are continually monitored by the master control unit for correct operation. This self test scheme is a vital feature of the multiplexed system and should be retained and expanded in Space Shuttle in spite of its added complexity.

In another study, Westinghouse Electric Company did a tradeoff analysis of a multiplexed power distribution system [29] for Space Shuttle; Table VIII lists data from the study. Comparing the Westinghouse data and the SOSTEL data from Figure 13, it is interesting to note that the weight of the Westinghouse remote terminals (121 pounds) is about three times the weight of the SOSTEL remote terminals (38.4 pounds, from Figure 13) for equal signal capacity. Also, each Westinghouse computer (25 pounds) weighs about three times as much as a SOSTEL unit (8 pounds). Perhaps these weight differences can be accounted for in the applicable semiconductor technology.

The disadvantages of electromagnetic relays (EMR) when compared to the solid-state switches for logic implementation can be summarized:

1. As the amount of logic to be implemented increases, a point is reached at which solid-state combinational logic has a weight advantage over relay logic.

2. Relays do not achieve the reliability figures of solid-state logic systems.

TABLE VIII

WEIGHT ESTIMATES FOR SHUTTLE MULTIPLEX POWER DISTRIBUTION SYSTEM [29]

Item	Rating (amps)	Quantity	Total Weight (pounds)	Total Volume (cubic feet)	Total Power Loss (watts)
Ac power controller	1-10	364	102	0.7	-
Dc power controller	1-15	691	166	0.9	-
Remote input/output units	-	31	121	5.3	744
Switch indicator modules	-	625	-	-	-
Limit switches	-	100	-	-	-
Computers	-	3*	75	1.5	480

*3 computers for triple redundancy.

3. Relays generate EMI from contact bounce and from coil (inductive) transients.

4. Relays cannot offer the same degree of fault isolation capability as a solid-state system.

5. Relays are not as effective as solid-state devices for overload control.

A comparison of relay versus solid-state control circuits for the landing gear of the F-15 aircraft [30] is shown in Table IX. Note that although the two systems are of nearly equal weight the reliability of the solid-state system is about ten times that of the relay system, and fault isolation is much superior.

In conclusion it may be stated that when the number of signals is large and reliability and maintainability requirements are stringent, solid-state logic systems far surpass relay equivalents.

TABLE IX

RELAY VERSUS SOLID-STATE LOGIC COMPARISON
F-15 LANDING GEAR DISPLAY/CONTROL SYSTEM [30]

Device Type	Weight (lbs)	Volume (cu in)	Reliability (1000 hours)	Power Consumption (watts)	Fault Isolation
Relay (21 DPDT)	2.55	84	4.75	61	Poor
Solid-State (6 assemblies)	2.8	122	41.7	30	Good

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IV. CONCLUSIONS AND RECOMMENDATIONS

National priorities have changed and emphasis has shifted from long range planning for Space Station and Space Base to shorter range planning for Space Shuttle. The parameters chosen for the power distribution systems of Space Shuttle will affect parameters of the power systems of Space Station and Space Base when they are constructed, because of the need to interface at docking. It has been urged in the reports of this contract [1,2] that spacecraft power system parameters be selected on a basis of overall system requirements that include Shuttle, Station, and Base. Tentative recommendations were made that future spacecraft power systems standardize on 115 Vdc and 115/200 Vac, 400 Hz, 3Φ. To design the power system for Shuttle on the sole criterion that off-the-shelf 28 Vdc components are lowest in cost will most certainly penalize future spacecraft designs, and will lead to high costs for Space Station and Space Base power distribution systems. The designs of power systems for Space Shuttle should recognize the need for compatibility between successive vehicles, and should utilize current, advanced technology.

Recommendations in earlier reports [1,2] included:

1. The initiation of studies to determine system constraints on maximum frequency and voltage in power supplies for near-earth space vehicles.

2. The analysis of current developments in data bus, multiplexed switching and control systems for aircraft, to determine the feasibility of transfer of the technology to spacecraft power systems.

A literature search was made for an answer to the question, "What is the constraint on spacecraft power system voltage, as set by dangers of corona?" No definitive answer was located in published data. Furthermore, scant description was found of the environment to be expected for power system components in near-earth missions. Basic theory suggests that radiation at moderate energy levels does not influence the magnitude of COV with bare electrodes, but radiation does affect the characteristics of insulation materials. Further analysis of effects of radiation and other environmental parameters is recommended, to determine the COV constraint for future spacecraft power systems.

Problems in transferring current technology of data bus multiplexed power system control of aircraft to spacecraft requirements have been defined. Further effort is recommended, including:

1. A review of current Standards and Specifications which are related to spacecraft power systems, with the aim of initiating changes to remove the conflicts and constraints which hamper development of advanced techniques.

2. An analysis of the dynamic stability of some of the power systems that have been proposed for future spacecraft.

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