FENN COLLEGE OF ENGINEERING
CLEVELAND STATE UNIVERSITY

Electrical Engineering

A HISTORICAL OVERVIEW
OF THE ELECTRICAL POWER SYSTEMS
IN THE U.S. MANNED
AND SOME U.S. UNMANNED SPACECRAFT

Final Technical Report

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Principal Investigator: James E. Maisel

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Cleveland State University
Cleveland, Ohio 44115

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16. Abstract

A historical overview of electrical power systems used in the U.S. manned spacecraft and some of the U.S. unmanned spacecraft is presented in this investigation. A time frame of approximately 25 years, the period for 1959 to 1984, is covered in this report. Results indicate that the nominal bus voltage was 28 volts dc in most spacecraft and all other voltage levels were derived from this voltage through such techniques as voltage inversion or rectification, or a combination. Most spacecraft used solar arrays for the main source of power except for those spacecraft that had a relatively short flight duration, or deep spaceprobes that were designed for very long flight duration. Fuel cells were used on Gemini, Apollo, and Space Shuttle (short duration flights) while radioisotope thermoelectric generators were employed on the Pioneer, Jupiter/Saturn, Viking Lander, and Voyager spacecraft (long duration flights).

The main dc bus voltage was unregulated on the manned spacecraft with voltage regulation provided at the user loads. A combination of regulated, semiregulated, and unregulated buses were used on the unmanned spacecraft depending on the type of load. For example, scientific instruments were usually connected to regulated buses while fans, relays, etc. were energized from an unregulated bus. Different forms of voltage regulation, such as shunt, buck/boost, and pulse-width modulated regulators, were used. This report includes a comprehensive bibliography on spacecraft electrical power systems for the space programs investigated.

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1. Introduction

Background

The United States Space Program under NASA started in 1959 with an unmanned satellite and will continue into the twenty-first century with manned space stations capable of supporting humans for long periods of time. This progress could only occur if the program was a well-integrated set of space operations where each mission solved technical problems for future missions. The NASA space flight program can be divided into manned and unmanned programs with each contributing and ultimately setting the stage for the space shuttle and future space stations. A chronology of the flight program is presented in Figure 1-1.

All spacecraft, whether manned or unmanned, have common features. They must have a guidance system, process data, and communicate with Earth or other spacecraft. These features, in turn, depend on a source of electrical power. The level of power that must be generated in order to sustain spacecraft systems has been increasing over the years. At the end of this century, space stations will require power levels in the megawatt range. Hence, it is very judicious that the design of future space station electrical power system should be well thought out to meet these large power demands.

In order to gain an insight into the design of these megawatt power systems, a historical investigation will be presented in this report on the manned (Chapters 2-7) and some of the unmanned spacecraft electrical power systems (Chapters 8-14). In the Summary, a table listing the salient electrical characteristics of each spacecraft system is presented.

Voltage Levels

The majority of electrical power systems that were studied had a nominal 28-volt dc bus voltage. Skylab had the highest array voltage of 70-115 volts and generated 16 kilowatts of power [1]. All ac electrical systems were derived from this bus voltage via solid-state inverters. For relatively lower power levels, spacecraft electrical power systems based on the 28-volt bus voltage were and still may be adequate. At the megawatt level, using a 28-volt bus for the main distribution system would be prohibitive because of size, weight, and cost of the power bus lines.

Increasing the bus voltage from a low to higher value is dependent on the technology-readiness level and constraints that might be inherent in a particular electrical power system. The technology-readiness level has continuously changed as new electrical components are developed. For example, the main source of power in the Mercury spacecraft was a battery which was adequate for the length of the mission. By the time Project Gemini was completed, fuel cells were the main source of electrical power, permitting the mission to be extended to days.

Electrical Power System Organization

Spacecraft electrical power systems that employ solar panels for the main source of electrical energy have an inherent upper bound on the input or array voltage. If the spacecraft solar array experiences occultation, the array voltage can vary approximately 2 to 1 due to temperature. This, along with other important considerations, must be incorporated in analyzing the spacecraft or space station requirements. Figure 1-2 illustrates a block diagram of the electrical systems options available in the design of an electrical power system. In the case of solar/photovoltaic configuration, a power transfer mechanism such as slip rings are required because the solar

panels are outside the spacecraft. For the other energy sources and conversions, the main source of power can be contained within the spacecraft. The power management block can be programmed via ground support and/or spacecraft crew support.

All three energy sources and two conversions (photovoltaic and thermoelectric), as listed in Figure 1-2, have been employed in either the manned and unmanned missions discussed in this report. Both dynamic conversion (Brayton, Rankine, and Stirling generators) and mechanical (flywheels) are still under development and may be adopted in the future development of spacecraft and/or space stations.

Power conditioning, distribution, and power management have been employed in one form or another in all spacecraft considered and become more complex as the power demand level increases.

The choice of the type of main energy source is dictated by the required power and the mission duration [1] as shown in Figure 1-3. This assumes that there are no other constraints such as the need of fresh water (a by-product of fuel cells) or thermal heating of the spacecraft (a by-product of nuclear sources). For long-life missions that require a high level of power, the likely primary energy source candidates are nuclear reactors, solar, and radioisotopes. Of the three energy sources, the solar array is the most frequently used primary source and will probably hold this position for the near future [1].

Solar Array Constraints

Future missions, whether low earth orbit, geostationary or escape trajectory, dictate the following constraints on the solar array [2]:

. Effect of eclipses

. Effects of particle and light radiation from the Sun

. Effect of micrometeorites and trapped particles in
  the magnetic fields of Earth or other planets

. Effect of pollution in space

The eclipse is the principle mechanical constraint on the solar array. The temperature variation and frequency depends upon the orbit of the spacecraft. For example, the temperature range and frequency could vary from -160°C to +70°C, and from 80 to 5,000 cycles per year depending on the orbit.

Radiation from the Sun in the form of particles such as protons and radiation in the ultra-violet and infra-red portion of the spectrum cause the solar cells to degrade. Ultra-violet and infra-red radiations degrade the transparency of the adhesives and cause a temperature increase of the solar array, respectively.

Micrometeorites can impact with the solar cells causing physical destruction which in turn reduces the power output of the solar array.

Advances in solar array technology will also produce improved power systems. For example, in the area of photovoltaic energy conversion, the thin silicon cells (100 or 50 microns), with essentially the same efficiency (~16%) as a cell of conventional thickness, offer a gain in mass performances of the solar array and a better resistance to space radiation.

Gallium arsenide cells with efficiency approaching 20% are presently under development. Besides the increase in efficiency and better radiation resistance, gallium arsenide offers the possibility of annealing at relatively low temperature (~200°C).

Electrical Storage Considerations

Batteries were and will be used as means of storing electrical energy on the spacecraft and/or space station. The nickel-cadmium battery has been used quite extensively on past missions and will continue to be used along with nickel-hydrogen and silver-hydrogen batteries.

As an example, the life cycle duration for nickel-cadmium batteries depends on the type of orbit (low earth or geostationary) and the depth of discharge. The life cycle duration decreases with increasing depth of discharge for either orbit, but the number of cycles is less in the case of the geostationary orbit when compared to low earth orbit, assuming the same depth of discharge and battery temperature.

Secondary batteries on geostationary spacecraft experience on the order of 60% and an expected 80% depth of discharge for nickel-cadmium and nickel-hydrogen batteries, respectively. Although batteries in geostationary orbit have, in general, a greater depth of discharge when compared to batteries in low earth orbit, the number of discharge cycles are considerably smaller (80 versus 5000 cycles per year). According to [1], the design life for both the nickel-cadmium and nickel-hydrogen batteries will be approximately nine years by 1990 with the former battery operating at a depth of discharge of somewhere between 20 to 40% and the latter somewhere between 30 to 60%.

Battery lifetime can be extended substantially by decreasing the depth of discharge and temperature of the battery. Lowering the depth of discharge requires a heavier battery and more lift-off or launch energy. A wiser choice is to operate the battery at a lower temperature because it does not cost as much in mass. For example, the life span of a nickel-cadmium battery in a geostationary orbit can be increased from four to eight years by reducing the operating temperature from 20°C to 10°C.

Electrical Power Distribution Methods

The distribution of dc power from the main or secondary source, i.e. solar-array/batteries, can be either ac, dc or a combination of ac/dc. For a multi-hundred kilowatt system, the life-cycle cost according to [1] are projected to be $0.32 per kilowatt hour for an ac system and $0.40 per kilowatt hour for a dc system indicating that an ac system has an advantage on a dollar per kilowatt hour basis.

As the power demand becomes larger, the cable weight and power losses increase dramatically if the low solar array voltage (150-200 volts) concept is maintained. The upper bound on the array voltage is set by array-plasma interactions which are most severe in low earth orbit where the plasma density is near its maximum value.

By converting the array voltage to ac via inverters, the distribution voltage and frequency can be increased substantially. This approach offers the following advantages:

- Permits the distribution voltage to increase with power demand
- Offers zero current crossover fault switching
- Provides ability to change voltage with transformers
- Uses rotary transformers instead of slip rings

Various studies have shown that for both ac and dc there is a significant savings in the cost and weight of the power distribution system for a given power demand at higher distribution voltage, subject to constraints on insulation, safety, etc.

Besides the references after each section, a bibliography section has been included in this report. It is comprised of the references as well as other reports. The literature search covers the time period from 1962 to 1984.

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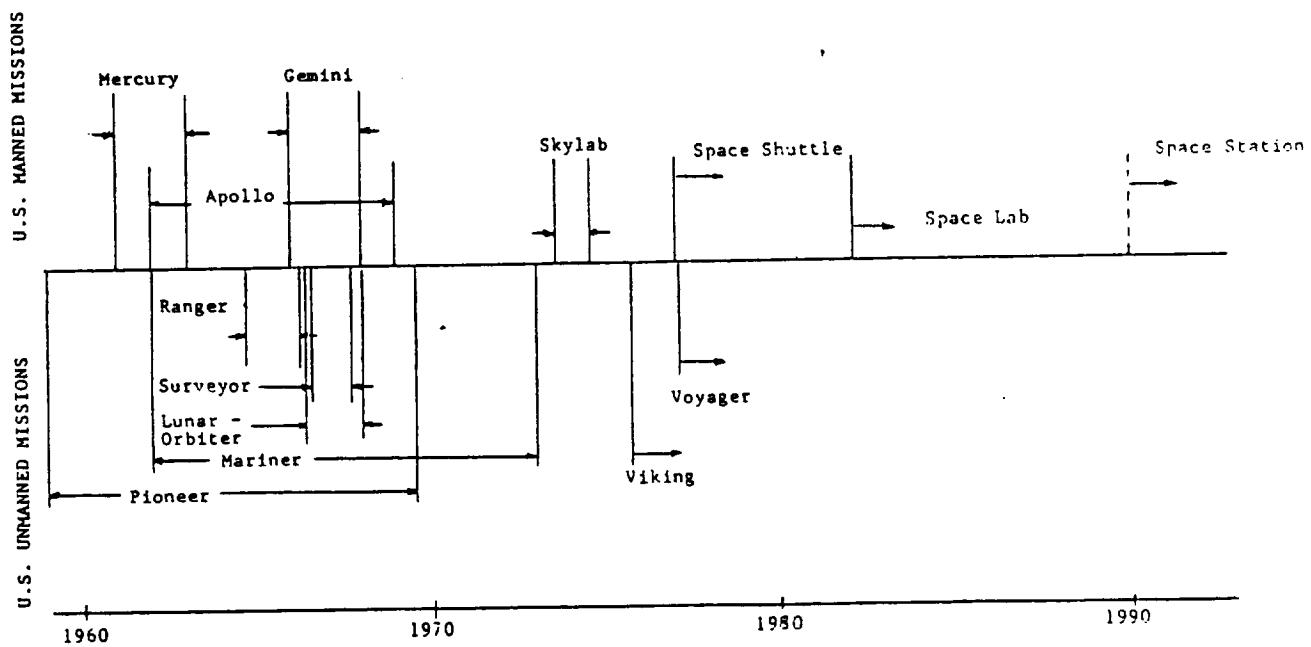
Figure 1-1. Historical Development of the U.S. Space Program

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----------ENERGY SOURCE:  SOLAR, NUCLEAR, CHEMICAL
   |
   |
   |          CONVERSION:  PHOTOVOLTAIC, THERMOELECTRIC, THERMIONIC, DYNAMIC
   |
   |
   |                 POWER MANAGEMENT AND DISTRIBUTION
   |
   |   ENERGY STORAGE:     POWER CONDITIONING:
   |   CHEMICAL,              AC/DC, DC/DC, REGULATION
   |   MECHANICAL
   |
   |
   |                 DISTRIBUTION: AC, DC, SWITCHING, PROTECTION
   |
   |
   |
   |
   |               AC  LOADS        DC  LOADS
   |
   |
   |
   |___
   |   |
   |   |  POWER MANAGEMENT
   |   |
```

_____ Power Bus

---------------- Control Bus
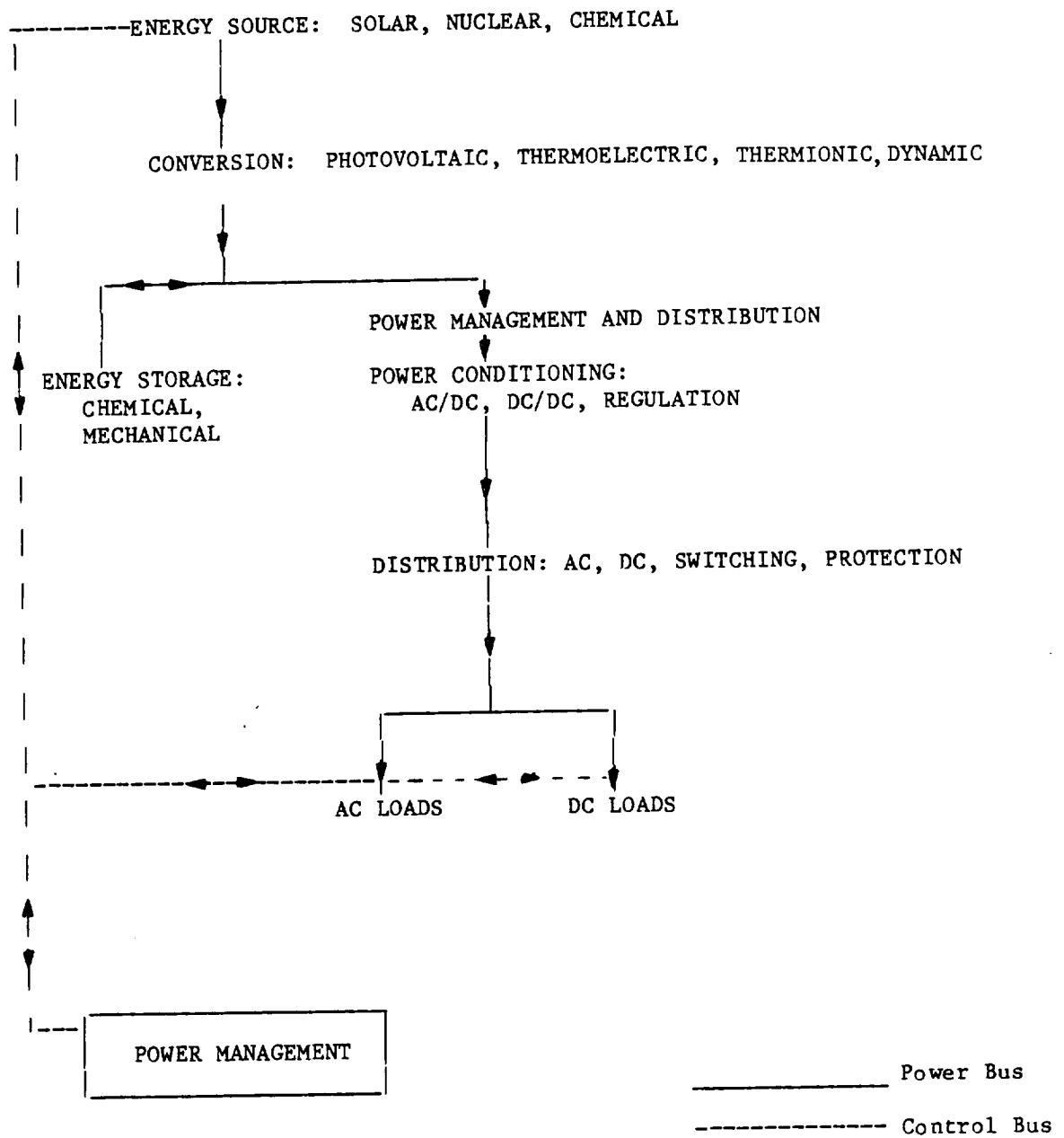
Figure 1-2. Block Diagram of the Electrical Systems Options Available
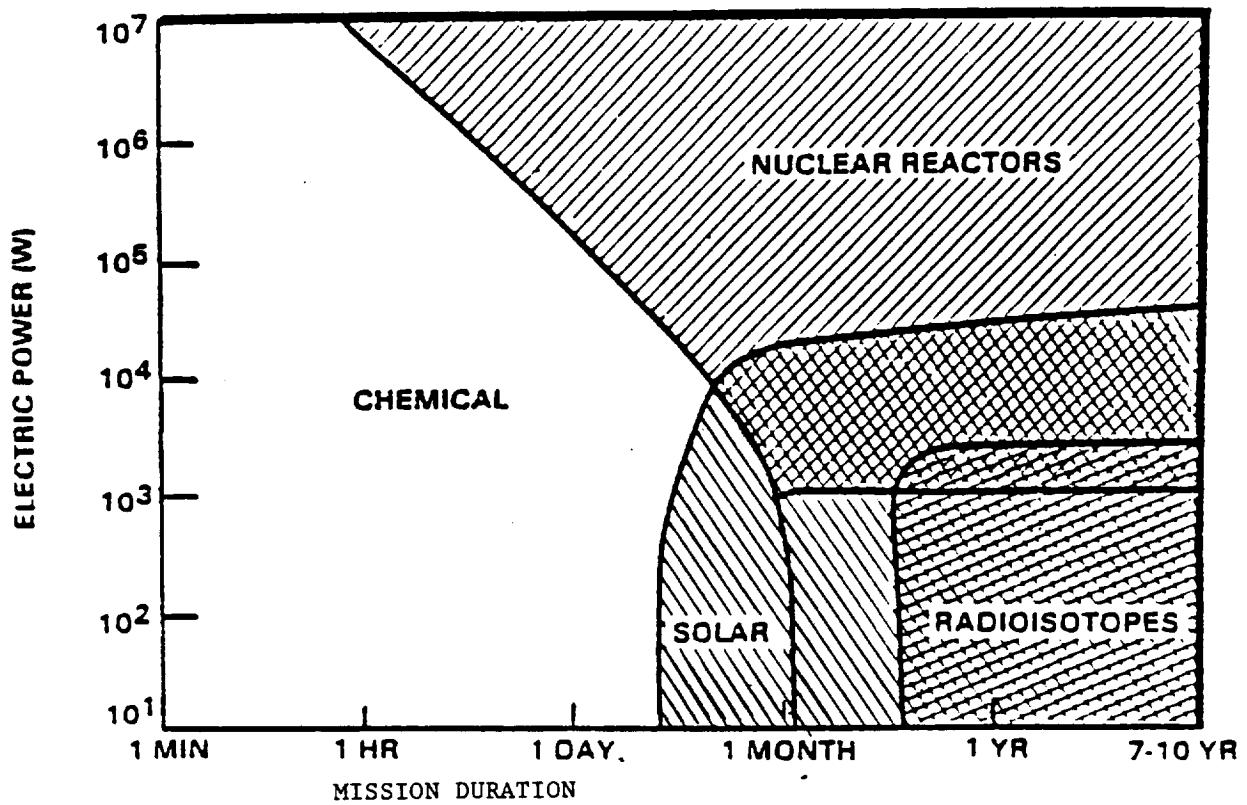
Figure 1-3. Electric Power Versus Mission Duration

2. PROJECT MERCURY

The primary objectives of Project Mercury were to place a manned spacecraft into a controlled earth orbit in order to investigate the performance of man and his capacity to withstand the environment of space and to successfully recover the spacecraft. The first manned flight, Mercury/Atlas-3, MA3, was launched on May 5, 1961, and the first U. S. manned orbital flight, MA6, was launched on February 20, 1962. The project was completed with MA9 that was launched on May 15, 1963, just slightly over two years from the MA3 mission [1].

The basic principles employed in initiating the project were to utilize the simplest and most reliable approach by keeping new innovations to a minimum. The method selected to support this philosophy was as follows [2]:

- Use a ballistic reentry vehicle

- Use an existing ICBM booster

- Use a retro-rocket for deorbit

- Use a parachute as a landing system

The spacecraft was complicated by redundant systems which were added in order to increase safety. The astronaut could take complete control of all the functions of the automatic control system, even to the point of manually flying the capsule if the automatic system failed.

Electrical Power System of Mercury Spacecraft

Figures 2-1 and 2-2 show respectively the dc and ac power schematics for the Mercury Spacecraft [3]. The power supply consisted of three 3000 watt-hour main batteries, two 3000 watt-hour standby batteries, and one 1500 watt-hour battery. All batteries were of the silver-zinc type.

The main batteries developed 24 volts dc and were connected in parallel by a switch on each battery. An isolation diode was connected in series with the positive battery terminal in order to protect against discharge through a discharged or faulty battery. The main batteries provided power to the main bus when the battery switch was in the on position.

Each standby battery had electrical taps at 6, 12, and 18 volts that were connected to the 6-volt, 12-volt and 18-volt standby busses. Isolation diodes were used for reverse current protection on all positive voltage battery terminals.

The 1500 watt-hour isolated battery provided emergency audio and squib power to the rest of the circuits in case the main and standby batteries were depleted. The battery had reverse current protection.

During pre-launch operations, external dc power was supplied through an umbilical cable in order to have all batteries fully changed at launch.

The main 115-volt, 400 hertz ac power was provided by two inverters with ratings of 150 and 250 volt-amperes. The 250 volt-ampere inverter supplied the altitude control system and the other inverter provided power to the fan bus. A standby 250 volt-ampere inverter provided backup for either, or both, the altitude control system and fan busses. The output was 115 units ± 5% volts, single-phase to ground, with a frequency of 400 ± 1% hertz and a sinusoidal waveform.

Because of the inherent overload protection in the inverters, ac loads were not fused. A short circuit fault at the output of an inverter will not damage the inverter or the conductors involved in the short circuit.

Table 2.1 indicates the electrical power consumed in MA-6 mission [3].

The dc and ac electrical systems performed well during the mission. Because the inverter cooling systems did not operate according to specifications, the inverter temperatures exceeded their design temperatures slightly, reaching temperatures over $200°F$. However, the performance of the inverters was excellent even at higher temperatures according to postflight inspections.

The ampere-hour battery ratings for the MA-9 mission was modified by the replacement of two 1500-watt-hour batteries with two 3000-watt-hour batteries. This increased the total power source energy from 13,500 to 16,500 watt-hours [4].

Table 2.1. Watt-hours of Power Consumed for Mercury Mission

Battery System	Prelaunch	Orbital	Postlanding	Total
Main in parallel with standby	606	2480	260	3346
Isolated	30	50	40	120

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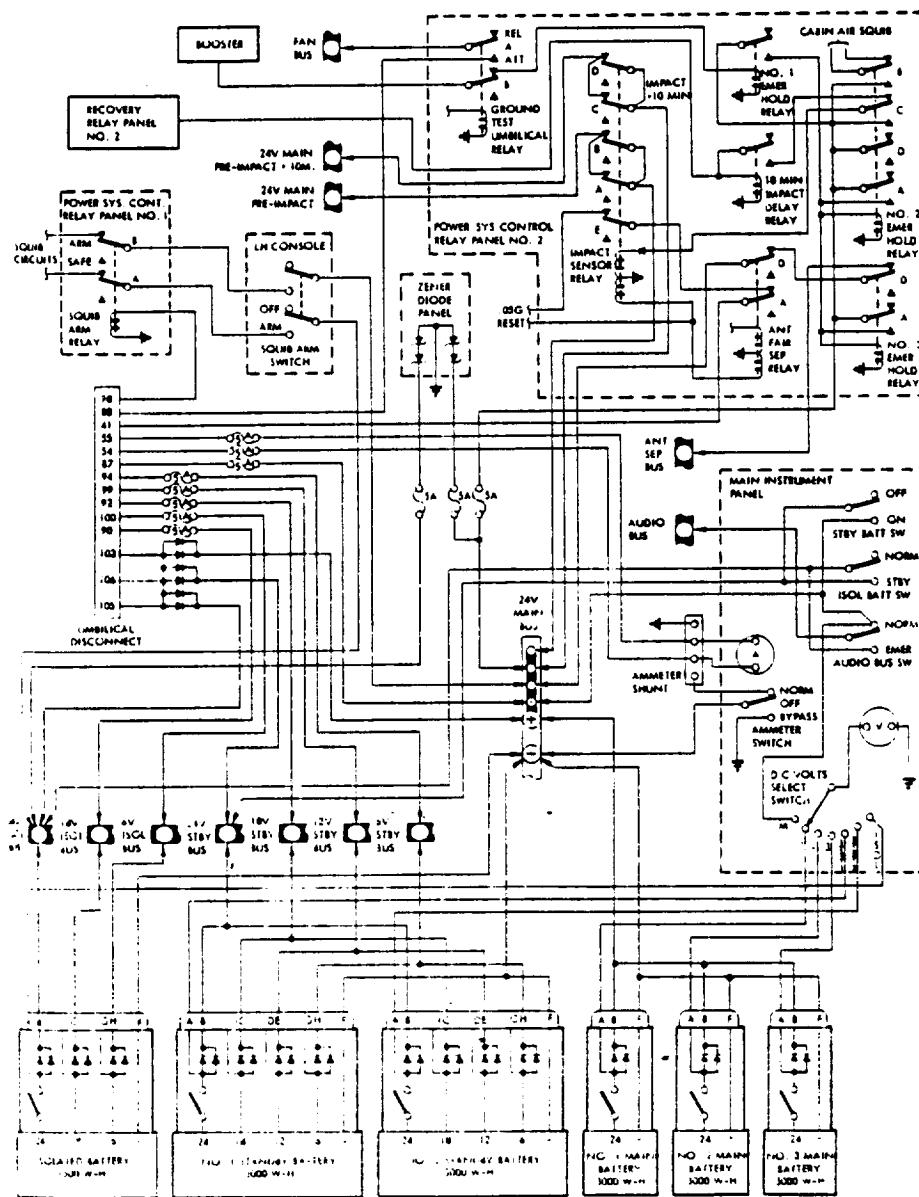
Figure 2-1. D-C Power Control Schematic
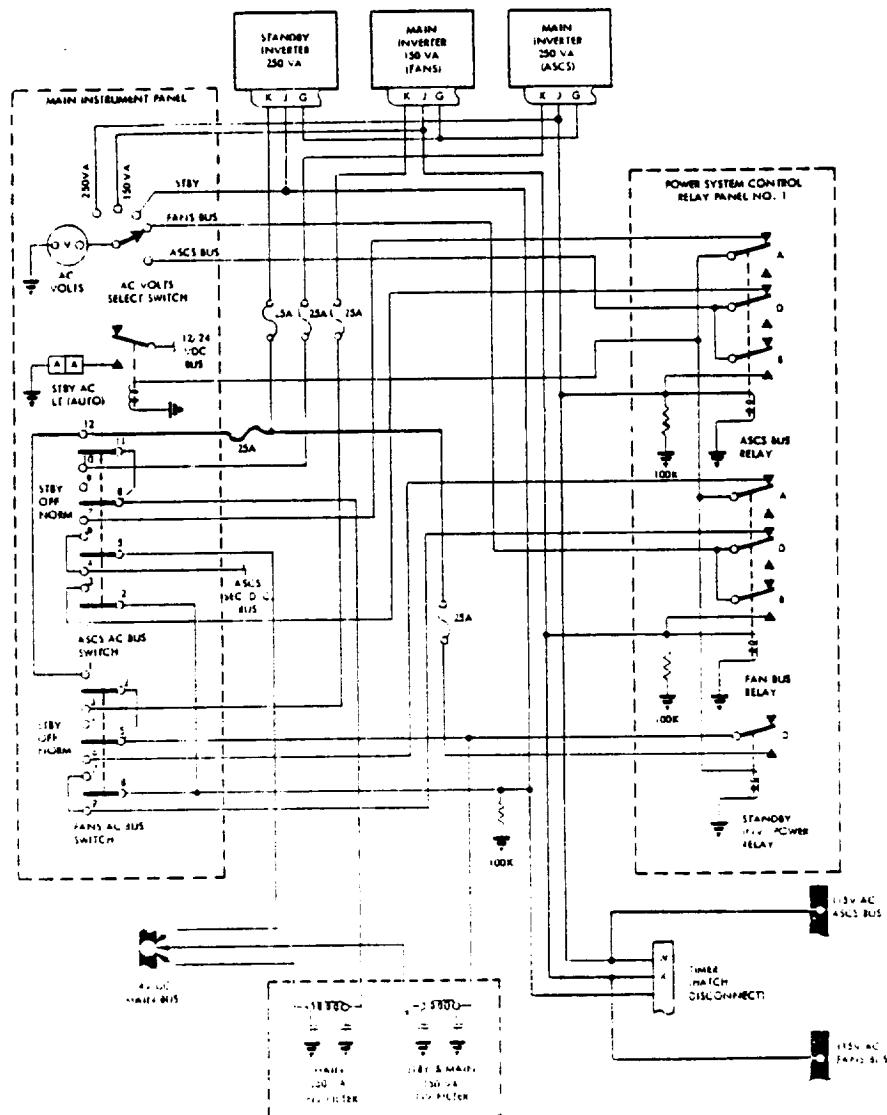
Figure 2-2. A-C Power Control Schematic

3. PROJECT GEMINI

Experience from Project Mercury demonstrated that failure propagation can occur when the systems are designed such that there is an interdependency or when systems are installed in a stacked fashion with common interfaces. In order to remove this constraint, Gemini development program emphasized rigorous:

- Component Testing
- Subsystem Testing
- Integrated Systems Testing

The Gemini systems were almost exclusively installed in a modular structure outside the inner pressure vessel and were accessible through panels in the outer skin.

Table 3.1 lists a summary of the manned Gemini missions [1]. Only two unmanned flights were necessary prior to the first manned mission.

Electrical Power System of the Manned Gemini Spacecraft

The main electrical power sources in the manned Gemini Spacecraft were either batteries (Spacecrafts 3,4, and 6) or fuel cells (Spacecrafts 5,7,8,9, 10,11, and 12) [2]. A multiple bus dc system whose bus voltage varied from 22 to 30 volts supplied power to subsystems that contained their own power processors. Some of these subsystems required tightly regulated dc or ac voltages. The power systems consisted of a main bus, two squib buses, and one control bus. The interconnection of the sources was controlled by the two-man crew, allowing for more redundancy and optimum power utilization from all electrical supplies.

During prelaunch, the spacecraft electrical power was supplied via an umbilical connected to an external ground supply. Switching from external to internal electrical supply took place prior to launch. This arrangement

prevented undue depletion of the spacecraft power supply, especially if the launch were placed on hold. The batteries had sufficient capacity for two hours and 36 hours for pre- and post-launch period, respectively. Also, the batteries could provide emergency power for the suit compressor (12 hours) after the landing of the spacecraft.

The fuel cell system provided the main bus electrical power in the later missions (Spacecrafts 5 and 7-12) with a dc voltage ranging from 22 to 30 volts. In order to insure power during prelaunch and launch phases, the fuel cells operated in parallel with the main silver zinc batteries.

The fuel cell system consisted of six electrically independent stacks with each stack comprising 32 cells that were connected electrically in series. Three stacks were grouped in two cylindrical containers and provided a peak power of 1 kilowatt. The stacks were electrically arranged so that individual stacks could be shut down at will or the selection of any combination of stacks within the system could be chosen by the crew members. Each fuel cell generated 1 kw at 26.5 volts at the beginning of life and 23.3 volts at the end of rated life.

Although the use of fuel cells was planned for all Gemini missions, the fuel cells were not available for early missions (Spacecrafts 1,2,3,4, and 6) thus forcing the project to use batteries as the primary source. Batteries were an inefficient source of power for flight periods exceeding four days [3,4]. Figure 3.1 illustrates the optimum utilization of power as a function of duration [5]. Fuel cells were necessary to support 8 and 14 day flights. From the same studies [3,4], it was demonstrated that batteries, solar cells, and fuel cells would increase their weight ratio by a factor of 5.1, 1, and

1.78, respectively, when a 14 day mission was compared to a 2 day mission. For the two day mission, the weights were 647, 739, and 279 pounds for batteries, solar cells, and fuel cells, respectively. Table 3.2 lists the significant anomalies for the fuel cells.

Three types of silver-oxide zinc batteries were used on Gemini spacecraft. They were:

- Main Batteries (4, 45 ampere-hours)

- Squib Batteries (3, 15 ampere-hours)

- Adapter Batteries (3, 400 ampere-hour, Spacecrafts 3,4, and 6, specified fuel cells were not available for these missions)

Electrical power was supplied from the power sources to the main, squib, and control bus systems for Spacecrafts 3 through 12. Figure 3.2 illustrates a simplified electrical block diagram of the Gemini spacecraft [6]. The main bus power was supplied by either silver-zinc battery or by fuel cells. Relays, powered from a common control bus, connected the power source to the main bus. Main bus power was fed through circuit breakers or fuses to the equipment. The crew controlled spacecraft equipment that was powered from the main bus based on an evaluation of the instrument panel display or at direction of mission control. The squib/control bus system consisted of three diode-isolated silver zinc batteries which supplied power to two isolated-redundant squib buses and the the control bus through the series diodes. Bus-tie switches permitted application of electrical power to the control and squib buses from the main bus in case an emergency developed.

The overall design and performance of Project Gemini were satisfactory and provided an important base line or reference for the development of the next generation of spacecraft, namely Apollo.

Table 3.1. Project Gemini Flight Summary

Spacecraft Number	Launch Date	Description	Results
3	March 1965	3-pass orbital qualification	Demonstrated launch structural integrity
4	June 1965	4-day orbital	1st U.S. extravehicular activity
5	August 1965	8-day orbital	Qualified rendezvous radar operation
6	Dec. 1965	1-day rendezvous	1st U.S. closed-loop rendezvous; with spacecraft VII
7	Dec. 1965	14-day orbital	Qualification for design duration; 1st controllable entry
8	March 1966	3-day rendezvous	1st rendezvous and docking with another vehicle (Agena); short circuited roll thruster terminated flight early
9	June 1966	3-day rendezvous	3 types of rendezvous; extended extra-vehicular activity
10	July 1966	3-day rendezvous	Onboard navigation only for rendezvous; 1st docked propulsion maneuvers (Agena); extravehicular activity
11	Sept. 1966	3-day rendezvous	1st tethered flight with another vehicle (Agena); extravehicular activity
12	Nov. 1966	4-day rendezvous	Gravity-gradient stabilization with tethered Agena

TABLE 3.2. SIGNIFICANT ANOMALIES AND CORRECTIVE ACTION FOR FUEL CELLS

Spacecraft	Remarks
2	• All fuel cell stack hydrogen inlet valves were closed prior to launch because a prelaunch facility malfunction made timely activation impossible. • In-flight information was obtained on launch effects on the pressurized static reactant supply system.
5	• Tests indicated an activation and storage limitation. • Corrosion of the fuel cell shutoff valve and the spacecraft plumbing was eliminated by changing the material of the valve and water connections and by flushing and drying the system after the first activation. • During flight, the fuel cell performance was nominal (system problems led to unusual modes of fuel cell operation). The following modifications were made on later spacecraft: 1. Coolant pump inverters were redesigned to give a high or low flow capability for each loop (conserve power). 2. Coolant loops were reconnected to establish an independent coolant flow to each section.
7	• An apparent restriction developed in the water management system which affected the performance of the three stacks of one fuel cell. However, all phases of the mission were accomplished satisfactorily.
8	• Except for a hydrogen vent modification, the fuel cell system was the same as Spacecraft 7. Fuel cell performance was nominal.
9	• Because fuel cells were activated approximately 15 hours before launch and the failure of the Agena Target Vehicle to achieve orbit, the mission was postponed for two weeks. A new fuel cell system was installed.
10	• In order to provide room in the adapter section for two additional orbit altitude and maneuver system bottles, the fuel cell system was modified. Fuel cell system was nominal during mission.
11	• The launch was delayed twice because of a launch vehicle problem. During the delay, the fuel cells remained activated and operated at 3 amperes/stack. The C stack of section 2 failed at approximately 54.5 hours into the mission. Failure was attributed to burnout in spite of the fact that it was impossible from the mission data to determine this failure. Mission requirements were met even with the failed stack.
12	• At approximately six hours into the mission, there were indications of an anomaly in the water management system. While a definitive answer of the failure was established, indications of a depletion of water storage volume had occurred. This was most probably oxygen leaking into the water system. • Fuel cell flooding occurred. Two stacks had to be shut down and two others experienced a significant loss of power as a consequence of the above problem. • Remaining stacks and batteries provided sufficient electrical power to accomplish all mission objectives.

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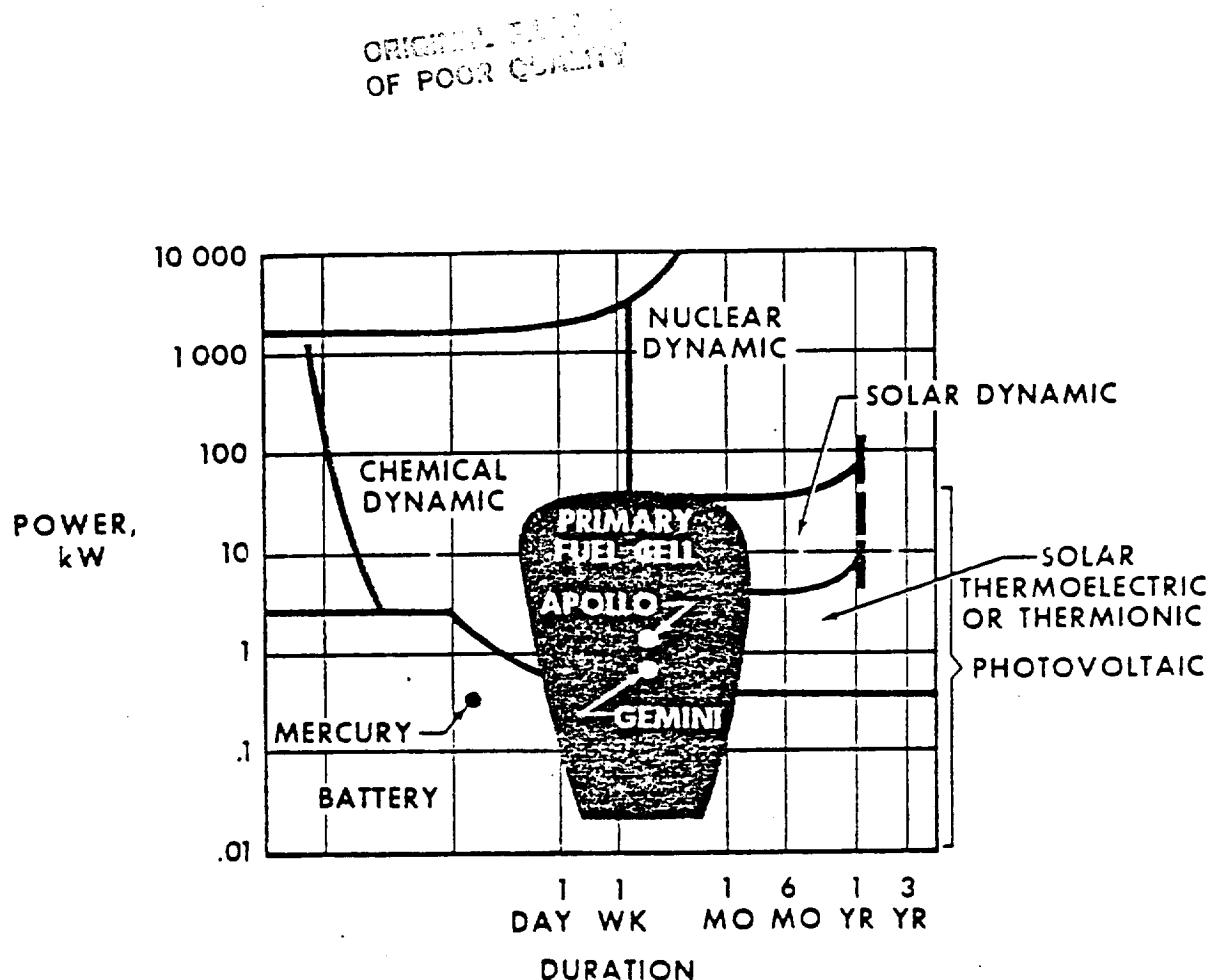


FIGURE 3.1. Optimum Utilization of Power Sources
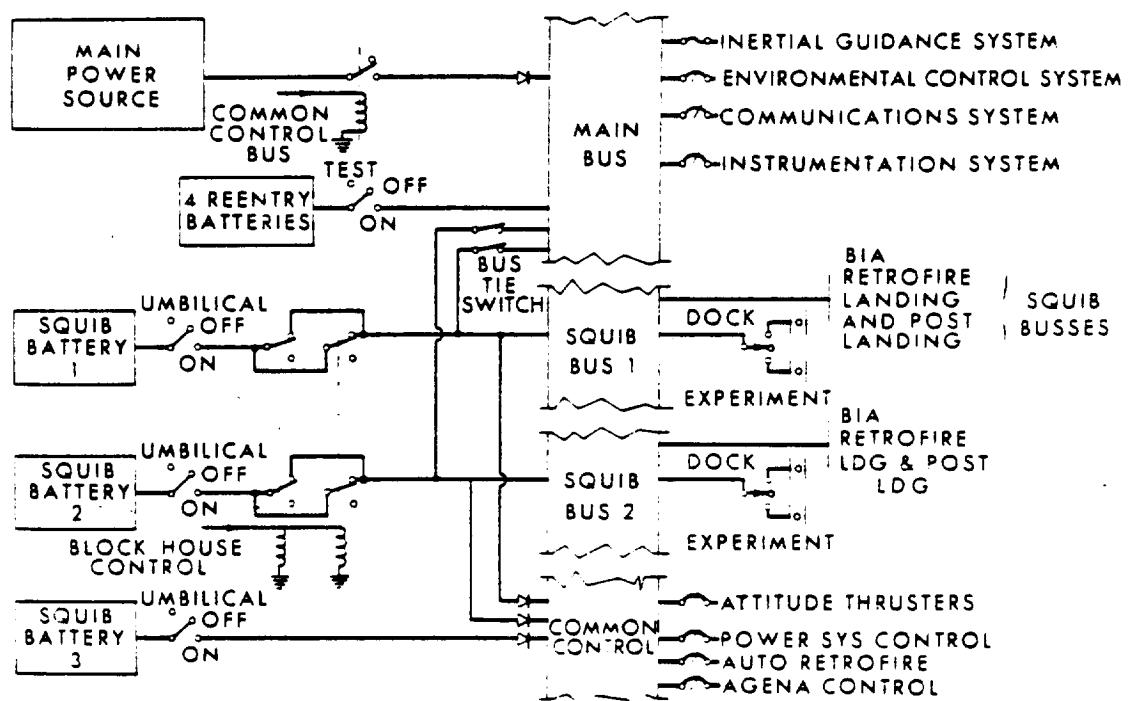
FIGURE 3.2. Gemini Electrical Power System

4. APOLLO PROGRAM

The first Apollo mission launch date was October 27, 1961 and it verified the Saturn I's aerodynamic and structural design. A series of five more launches were conducted before the first manned Apollo launch was made on October 11, 1968. See Table 4.1 for details of Apollo missions [1]. On July 16, 1969, man landed on the moon (Apollo 11 mission) followed by five more moon landings. The last launch was on December 7, 1972. See Table I for details of Apollo missions.

Electrical Power System of Apollo

The Apollo command and service module electrical power system (CSM-EPS) was designed to operate from any combination of seven direct-current sources [2-5]. The electrical sources are:

- Fuel Cells (3, 575 kilowatt-hours each)
- Entry Batteries (3, 40 ampere-hours each, silver oxide-zinc)
- Service Module Battery (1, 400 ampere-hours, added after Apollo 13)

The three fuel cells located in the service module (SM) supplied the primary source of power; two of the three entry batteries, located in the command CM, supplemented the fuel cells during high electrical demand; and the service module battery could be used if a fuel cell failed. Because of the failure of the cryogenic oxygen system in Apollo 13, a 400 ampere-hour service module battery was installed in the remaining Apollo missions. If required, this battery could have provided 12 kilowatt hours of additional or emergency energy via the command module main buses.

After the spacecraft attained orbit, the entry batteries were disconnected, recharged, and used to supplement the fuel cells during service propulsion system (SPS) burns. The load varied from 60 to 80 amperes between SPS burns, which is well within the fuel cell rating; however, during SPS burns when the gimbals were operated, the load current could reach a level of 120 amperes, which required the additional capacity of the two entry batteries.

The EPS of the Apollo CSM was designed to deliver nominal 28 volts dc and three-phase 400 hertz 115 volts ac derived from one of three inverters each having sufficient capacity to supply all alternating-current power required by the system.

The basic dc distribution system as shown in Figure 4-1 has two redundant buses and a single point ground that is connected to the spacecraft structure [6]. The two main dc buses, marked A and B, are energized by the fuel cells and/or the entry and post landing batteries labeled A, B and C. Battery buses A and B are powered by their respective entry and post landing battery. The third battery C can be connected to either or both buses in the event that batteries A or B fail.

The flight and post landing bus was energized from both main dc buses and diodes or directly by the three entry and post landing batteries via diode pairs.

The flight bus received power from both main buses A and B through isolation diodes and the nonessential bus (marked 1 and 2) was energized from either main bus A or B depending on the position of the mechnically coupled single-pole double-throw switch.

The pyrotechnic buses A and B, which were isolated from the main electrical via a normally open switch, are powered by the pyrotechnic batteries. If the pyrotechnic batteries malfunctioned, entry batteries could be connected to pyrotechnic bus A or B.

The battery charger was a constant-voltage current-limited charger with the current limited to 2.8 amperes for a battery voltage less than 36 volts. The charger operated in a continuous mode. At 36 volts the battery charger entered a cycling mode. The internal impedance of the battery increased with increasing battery voltage causing the charging current to decrease. At 39 volts minimum, the current was negligible and the battery reached its fully charged state.

The ac power distribution system illustrated in Figure 4-2 was a three phase four-wire system with the ac neutral connected to the single point ground. Two ac redundant buses, 1 and 2 provided power to the ac spacecraft loads.

Ac power was supplied by one or two solid state inverters rated at 115/200 volts 400 hertz. They produced 1250 volt-amperes each. Inverter 1 and 2 were respectively powered through main bus A and B and inverter 3 through either main A or B. The AC control (6 motor switches) operated contacts to connect or disconnect the inverter from the ac buses such that no two inverters were connected to the same ac bus at the same time. Inverters were automatically disconnected if an overvoltage or overload were present.

The inverter was designed to meet the following specifications [2]:

. Phases: three-phase 120° $\pm$ 2° displacement

. Voltage: 115 $\pm$ 2 volts ac (steady state)

. Frequency: 400 hertz with 6400 hertz external timing or
400 $\pm$ 7 hertz when free running

A major portion of the ac generated was used to power the fuel cell pump motors which presented a highly inductive load to the inverters. A capacitor bank was added to compensate for the lagging power factor of the inductive loads. When the fuel cell pump motors were redesigned with a larger power factor (less inductive), it was demonstrated that the power factor correction bank was overcompensating. Some of the capacitance was removed instead of redesigning the box.

The Lunar Excursion Module (LEM) electrical power system [7] supplied all required power for the LEM during its lunar mission. Prior to separation from the orbiting portion of the Apollo spacecraft, power was provided by the CSM-LEM docking umbilical cable. The electrical power system of the LEM consisted of a dc and ac section with the primary dc power being supplied by six silver oxide-zinc batteries (four in the descent stage and two in the ascent stage).

During the descent phase, all four batteries, rated at 400 ampere-hours each at a nominal output of 28 volts, supplied the electrical power in order for the LEM to complete its mission exclusive of the ascent phase. If only three descent batteries were functional, a protracted mission could be executed. However, if two of the four descent batteries were nonoperational, the LEM mission would be aborted. Either ascent battery, rated at 300

ampere-hours, was capable of supplying all ascent electrical power demands during normal mission operations and an abort.

The ac electrical power was generated by two solid-state inverters with each inverter rated at 115 ± 2 volts rms with an input of 28 ± 4 volts dc. A 6400 hertz master timing pulse supplied by the LEM guidance computer set the output frequency of the inverter at 400 ± 4 hertz. In the absence of the timing pulse, the output frequency tolerance increased to ± 10 hertz. The inverter output waveform was sinusoidal with less than 5% total harmonic distortion.

The dc and ac systems are shown in Figures 4-3 and 4-4.

TABLE 4.1. MANNED APOLLO MISSIONS

Mission	Launch Date	Mission Duration	Remarks
Apollo 7	October, 1968	10 days, 20 hours	First manned flight test of the command service module
Apollo 8	December, 1968	147 hours, 42 seconds	First manned test flight of the Apollo Saturn vehicle in lunar orbit
Apollo 9	November, 1969	241 hours, 53 seconds	To test lunar landing hardware while in earth orbit
Apollo 10	May, 1969	192 hours, 3 minutes, 23 seconds	To test lunar landing hardware while in lunar orbit
Apollo 11	July, 1969	195 hours, 18 minutes, 35 seconds	To undertake the first manned lunar landing
Apollo 12	November, 1969	244 hours, 36 minutes, 24 seconds	Second lunar landing Perform surface experiments · Investgigate the remains of Surveyor III
Apollo 13	April, 1970	142 hours, 54 minutes, 41 seconds	Third lunar landing aborted
Apollo 14	January, 1971	216 hours, 1 minute, 58 seconds	Third lunar landing Perform surface experiments
Apollo 15	July, 1971	295 hours, 11 minutes, 53 seconds	Fourth lunar landing Use lunar rover vehicle
Apollo 16	April, 1972	265 hours, 51 minutes, 5 seconds	Fifth Lunar landing Exploration of the Descartes region
Apollo 17	December, 1972	301 hours, 51 minutes, 59 seconds	Sixth lunar landing Exploration of the Taurus-Littrow region

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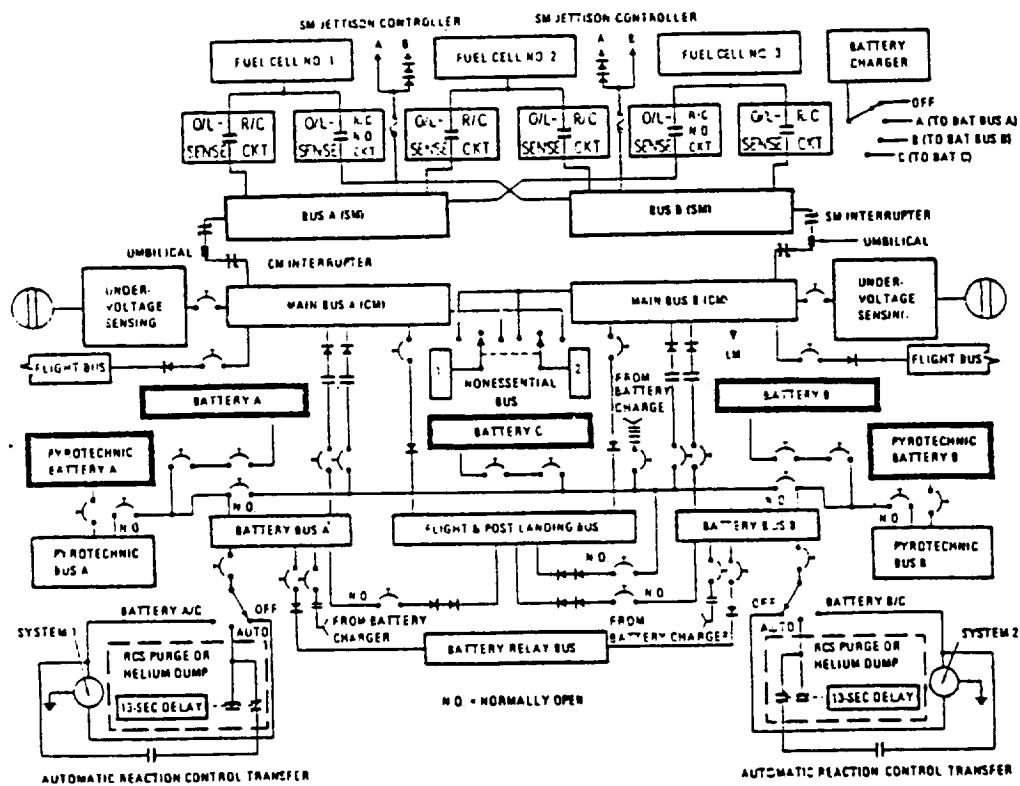
FIGURE 4-1. DC Power Distribution System of the Orbiter
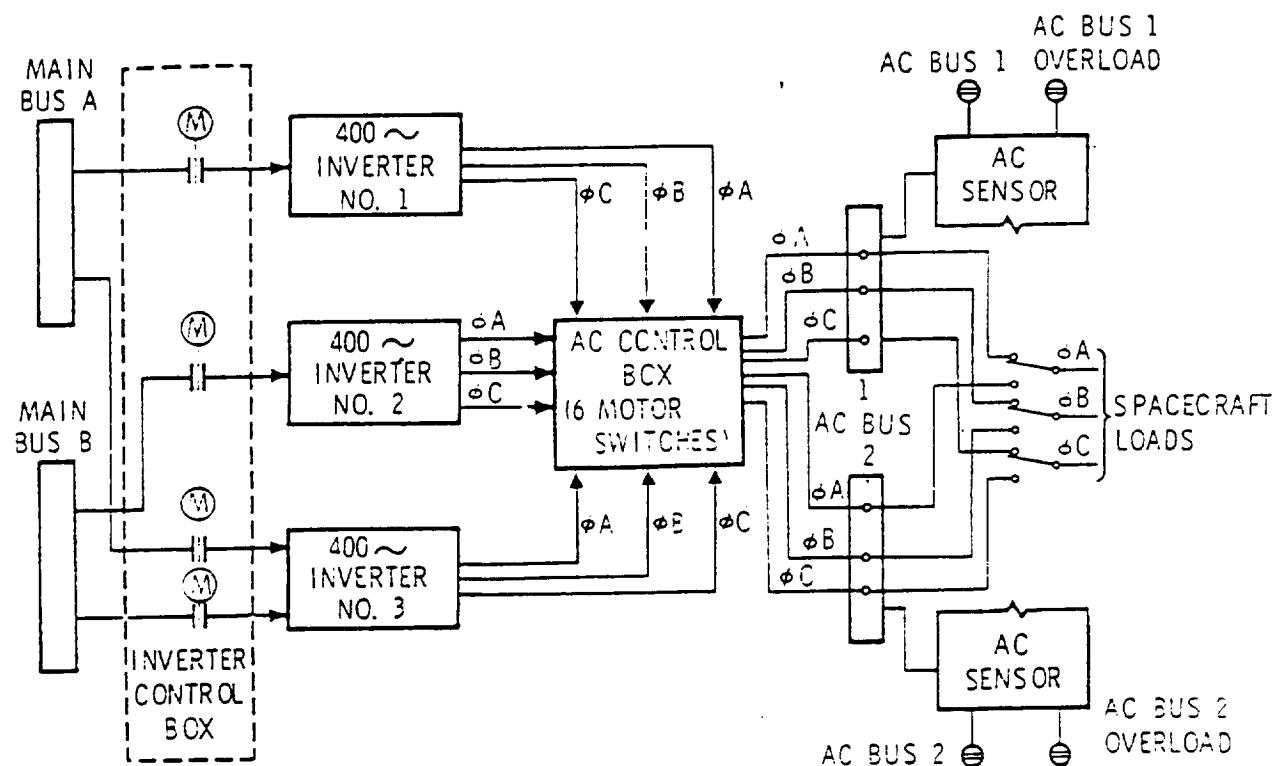
FIGURE 4-2. AC Electrical Power Distribution System of the Orbiter
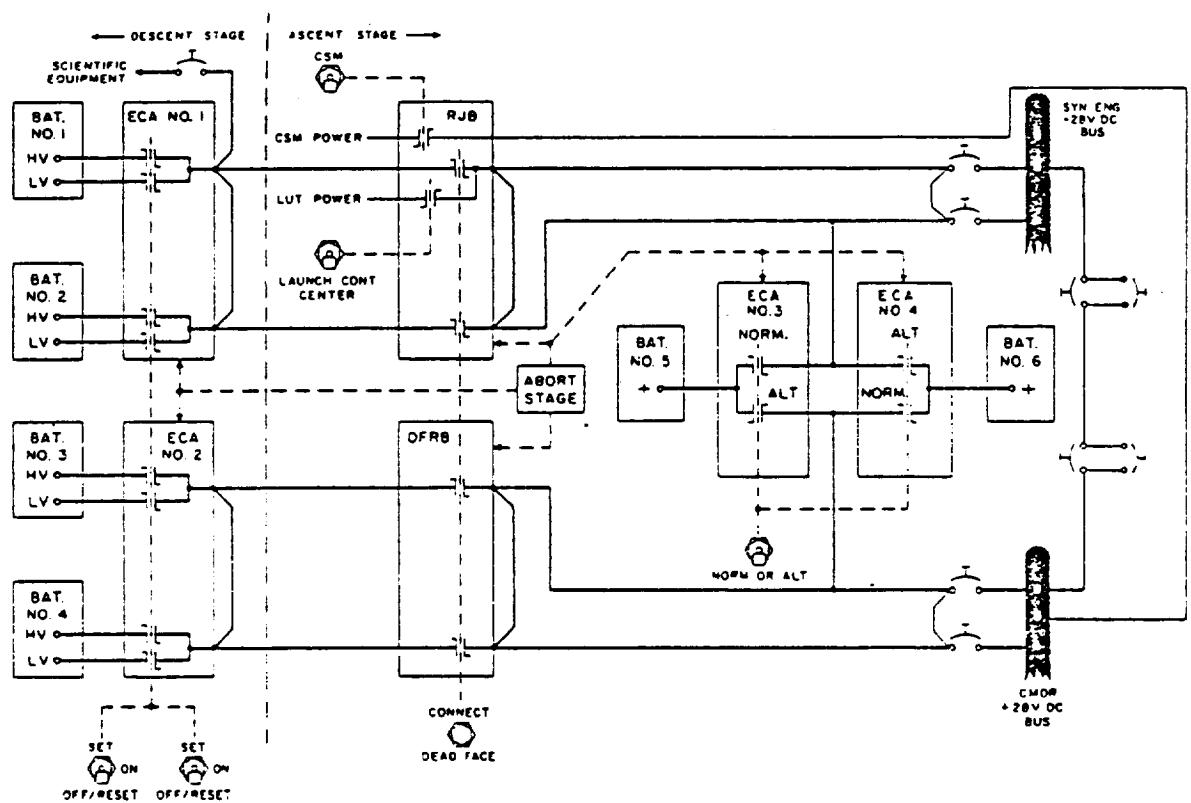
FIGURE 4-3. DC Electrical Power System of the LEM
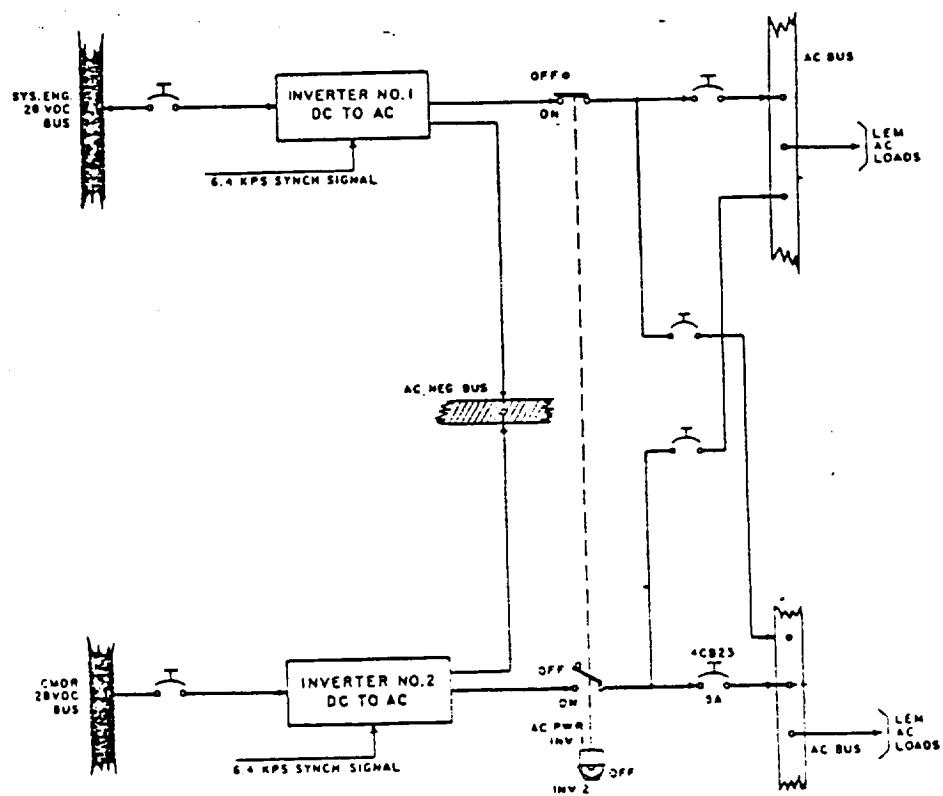
FIGURE 4-4. AC Electrical Power System of the LEM

5. SKYLAB

After approximately 3900 orbits of the earth with 171 days of manned operation from its launch in May 1973, Skylab 1 was a project of unparalleled scientific scope and breadth [1]. There were three visits to Skylab 1 as indicated in Table 5-1 [2]. Upon launching Skylab 1, its meteroid shield was torn away from the exterior of the cylindrical workshop along with one of the retracted solar wings. The second solar panel had not been properly deployed, resulting in an overheated and underpowered Skylab 1. The second solar wing was deployed during Skylab 2 mission.

Even with these major problems, that were partially corrected, the objectives of Skylab 1 were attained. These objectives were [3]:

- Study the Earth's crust, oceans, and mountains

- Study the Sun

- Study the Comets

- Manufacture alloys, grow crystals, and learn to exist in zero gravity for long periods of time

Skylab 1 was a manned modular space station [4] composed of five modules:

- Orbital Workshop (OWS)

- Airlock Module (AM)

- Multiple Docking Adapter (MDA)

- Apollo Telescope Mount (ATM)

- Command/Service Module (CSM)

Electrical Power System of Skylab

The Electrical Power System (EPS) for Skylab consisted of two independent power systems, located in AM-OWS and ATM. These systems were designed to function in parallel, allowing power sharing in either direction. A third EPS, located in the CSM, was available but it was only temporary until the

cryogenics were depleted. This occurred, however, between the 12th and 20th day after the CSM docked with the rest of Skylab 1 [5,6]. Table 5.2 indicates the rating of the orbital assembly power sources and their locations.

Originally AM-EPS design was a simple primary battery system that became a complex solar array/secondary battery system because of the changes in mission goals and design requirements [7]. Initially, the AM-EPS was required to supply a very small amount of power during the predocking mission phase which would last a period of approximately 11.5 hours. At that point the AM-EPS configuration consisted of silver-zinc primary batteries and a power distribution system.

Because the mission duration was lengthened and the complexity of the OWS enlarged to assist the growing experiment program, the AM-EPS design concept shifted to a solar array/secondary battery system with silver-zinc primary batteries to be employed during the preactivation power only. The solar arrays were mounted on the AM in the early designs, but were eventually located on the OWS in order to assist the increasing array size. Through a series of trade-off studies comparing silver-cadmium to nickel-cadmium batteries, it was demonstrated that the nickel-cadmium batteries reduced development risks since they had a better record with more ground test data and flight history. Hence, the nickel-cadmium battery became the principle type of battery that was used on Skylab 1.

Different combinations of solar array/secondary battery system designs were evaluated with the principle goals of increasing the reliability and overall efficiency.

Buck regulation was chosen to maximize efficiency for both the voltage regulator and battery charger. Also, the modular regulator configuration was

selected for both the battery charger and voltage regulator with objectives of high efficiency, reliability and redundant control circuitry. After the establishment of this design approach, the AM-EPS was composed of four Power Conditioning Groups (PCGs), each group including:

- A Battery Charger

- A Voltage Regulator

- A 30 cell, 33 ampere-hour Nickel-Cadmium Battery

Power requirements increased causing the number of AM-PCGs to increase to six and then finally to eight. The power source for the PCGs was from the solar arrays mounted on the OWS. The solar array was quite similar to the existing Agena design and was electrically connected in series in order to achieve the high input voltage required for buck regulation configuration.

The final OWS solar array configuration consisted of [6]:

- 616 solar cells per module

- 4 modules per panel

- 15 modules from wing 1 were connected in parallel per array group (wing 2 was torn off prior to orbital insertion)

- 10 panels per wing section

- 3 wing sections/wing

Nickel-cadmium batteries were preferable because of weight, volume, and proven performance. Eight nickel-cadmium 33 ampere hour batteries were chosen, one for each PCG.

Initially, three types of power sources were considered for the ATM-EPS. They were:

. Fuel Cells

. Radioisotope Thermoelectric Generator (RTG)

. Solar Array/Secondary Battery System

The fuel cell systems did not have an 18-month proven life capability and for systems with a rating of 2 to 4 kw they required an active coolant loop to remove waste heat. At the beginning of the program, the maximum power output of an RTG was approximately 500 watts with a 5% conversion efficiency. Active cooling would be required in order to remove the thermal heating due to the power loss (low efficiency). Solar cells with documented reliability and performance were readily available and were attractive on sun-oriented missions. The array had to be designed to meet the charging capacity of secondary batteries to supply power during earth occultation periods.

The ATM-EPS requirements provided for the ATM to supply electrical power to both the Lunar Excursion Module (LEM) ascent stage and the ATM systems from 24 solar panels via charger/battery/regulator modules (CBRM). From power conditioning study evaluations, it was shown that a 20 solar module, panel/power module configuration was regarded as acceptable.

The required surface area for the ATM-EPS solar cell array was achieved by employing four deployable wing-type assemblies that formed a cruciform pattern. This structure was chosen to minimize reaction forces while the assembly was being deployed and wing assembly was rotated 45 degrees to the Saturn Workshop (SWS) longitudal axis (X-axis) for minimum shadowing of other SWS areas.

Initially, the solar wing assembly panel layout consisted of 6 solar panels (16 modules/panel). Two modular configurations were used:

. 2x6 cm solar cells with two cells in parallel

. 2x2 cm solar cells with six cells in parallel

Both configurations had 114 cells in series. Solar cell module environmental tests indicated a maximum of 114 series connected solar cells determined the upper bound because at extremely low temperature high module output voltages were experienced. These voltage levels were of sufficient magnitude that could damage electrical components within other ATM systems. The maximum panel output voltage was set somewhere between 70 and 80 volts at the expected orbital low temperature.

The AM/MDA electrical power system is shown in Figure 5-1 [8] and consisted of the OWS, Solar Array System (SAS), eight AM Power Conditioning Groups (PCG), power distribution, control, and monitor provisions external to SAS and PCG. A detail block diagram of PCG1, as shown in Figure 5-1, consisted of a battery, battery charger, and voltage regulator. The OWS solar array was portioned into eight groups with one array group and one PCG constituting one of eight independent power subsystems that supplied power to the AM buses. The power distribution system was connected to the OWS, AM, MDA, CSM, and ATM. Electrical loads were supplied power from both the AM and ATM systems when the AM and ATM power systems were operating in parallel.

The battery charger, a subsystem of the PCG, charged the battery and supplied regulated array and battery power to the bus voltage regulator. If the charger failed, it could be bypassed in order to supply power directly to the bus regulator.

The ATM-EPS, as shown in Figure 5-2, supplied power between 26.5 and 30.5 volts dc to the ATM system and experiment-type electrical loads [8]. The 18 solar panels acted as separate sources and supplied unregulated power individually to 18 CBRMs. Each CBRM contained a battery that supplied power during occultation, a charger that processed the solar array power and controlled battery charging, and a regulator that regulated battery and/or array voltage and that regulated power drain or sharing between batteries. Each CBRM was connected to the ATM buses via diodes.

Nickel-cadmium batteries were preferable because of weight, volume, and proven performance. Batteries with a 20 ampere-hour rating were chosen because they had the largest capacity available and proven performance to fulfill the original mission requirements.

Because of the structural failures (solar array wing 2 was torn off, solar array wing 1 could not be deployed, and the meteoroid shield was torn away) during launch, Skylab 1 EPS capability was substantially reduced from premission planning, thereby initiating real-time power management. Most of the electrical burden fell to the ATM solar array. The mission rule (prelaunch) required that an average ATM Depth of Discharge (DOD) of 30% be maintained. After the mishap of losing wing 2 of OWS, the rule was revised to permit the batteries to operate within balance energy for each orbit and that each battery be completely recharged before Skylab 1 entered an orbital night. The new criteria or rule increased the ATM-EPS output capability from 4800 watts with a 0 degree departure from solar inertial to 2400 watts for a 60 degree departure. According to premission load profile predictions, the average load for the first unmanned period was 4500 watts. This power demand

could be met if the departure angle from solar inertial was near zero; otherwise power management techniques were necessary.

During the first 14 days of the first manned period, the ATM-EPS supplied the total Skylab power requirements with the exception of the CSM because it received its power from the CSM fuel cells. The average load requirement for this period was managed at 4000 watts which was within the premission predicted load of 5500 watts.

Two CBRMs failed. One CBRM was lost because a contactor failed to respond upon a closure command while the other CBRM automatically disconnected from the load buses. Both CBRMs were lost for the remainder of the Skylab mission reducing the total CBRMs from 18 to 16 CBRMs. The energy balance capability for the ATM-EPS with only 16 CBRMs functioning and the degradation specified at 4200 watts required the power management of the loads to be reduced to the 4000 watt level.

On the fourteenth day of the first manned mission, the OWS solar wing 1 was deployed by the astronauts. After activating the eight AM-PCGs, the rigorous power management techniques were relaxed and a return to the premission plans for the spacecraft systems operations was readopted. The average load demand for the remainder of the mission, as well as for the next two manned missions (Skylab 3 and 4), was 4700 watts (CSM fuel cells active) and 5800 watts (CSM fuel cells inactive). The capability of the ATM-EPS and AM/OWS-EPS operating in parallel was adequate to provide a positive power margin for each solar orbit.

During specified mission periods, it was feasible to maintain all planned astronaut tasks with the power available without compromising the mission objectives.

The power from the ATM solar array was supplied to the two-load buses via a buck switching charger and a boost-buck switching regulator. The charger sensed:

- Solar Array Voltage

- Solar Array Current

- Battery Temperature

- Charge Current

- Third Electrode Voltage

- Output Voltage

For a 20% DOD the average charger efficiency was 92.5% as compared to the design specification of 92%. The load regulator had the following electrical characteristics:

- Input Voltage: 25.5 to 80 volts

- Output Voltage: 26.5 volts at full load
                  30.5 volts at no load

- Short-Circuit Output Current Limit: 20 amperes

- Regulator Efficiency: 89% (design requirement)
                        92.4% (during sunlight based on a 20% DOD)
                        89.3% (during earth occultation)

The OWS solar array system (wing 1) was required to deliver 5248 watts within a voltage range of 51 to 125 volts dc at the end of a mission. After several orbits following the deployment of wing 1, the average array power was 6700 watts. No measurable degradation was detected during the mission (prelaunch prediction for performance from all causes was 8.3%).

TABLE 5.1

Mission	Date	Remarks
Skylab 2	5-25 to 6-22 1973	Docked with Skylab 1 for approximately 28 days Repaired damage station 3 spacewalks Erected sunshade Collected data on 45 of 55 experiments
Skylab 3	7-28 to 9-25 1973	Docked with Skylab 1 about 59 days Completed repairs Erected larger sunshade 3 spacewalks Exceeded premission plans for scientific experiments Replaced rate gyros
Skylab 4	10-16 to 2-8 1973/74	Docked with Skylab 1 nearly 84 days Replenished coolants Repaired antennas Observed the comet Kohoutek 4 spacewalks

TABLE 5.2. ORBITAL ASSEMBLY POWER SOURCES

Power Source	Rating/Number	Location
Batteries	33 amp-hours/8	AM
Fuel Cells	(2)	CSM
Entry Batteries	40 amp-hours/3	CSM
Descent Batteries	500 amp-hours/3	CSM
Pyro Batteries	40 amp-hours/2	CSM
Batteries	20 amp-hours/18	ATM
Solar Array	20 amp-hours/18 panels	ATM
Solar Array	15 modules/group/wing	OWS

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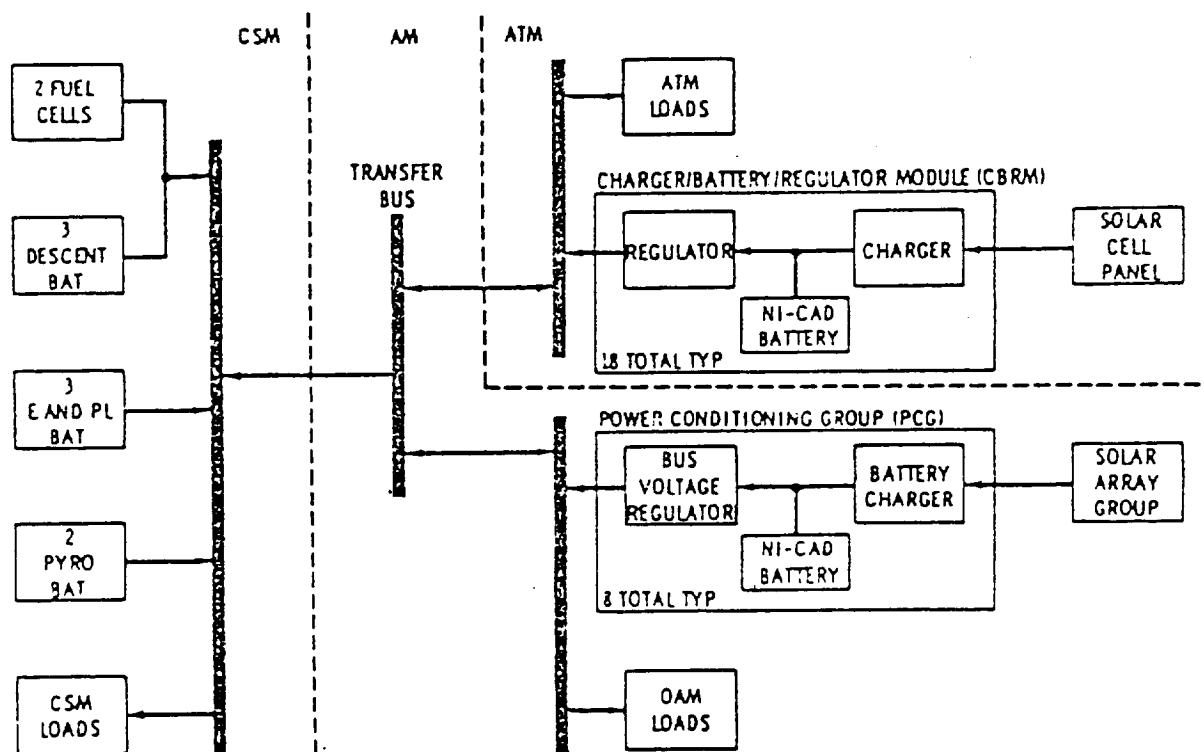
FIGURE 5-1.   AM/MDA Electrical Power System
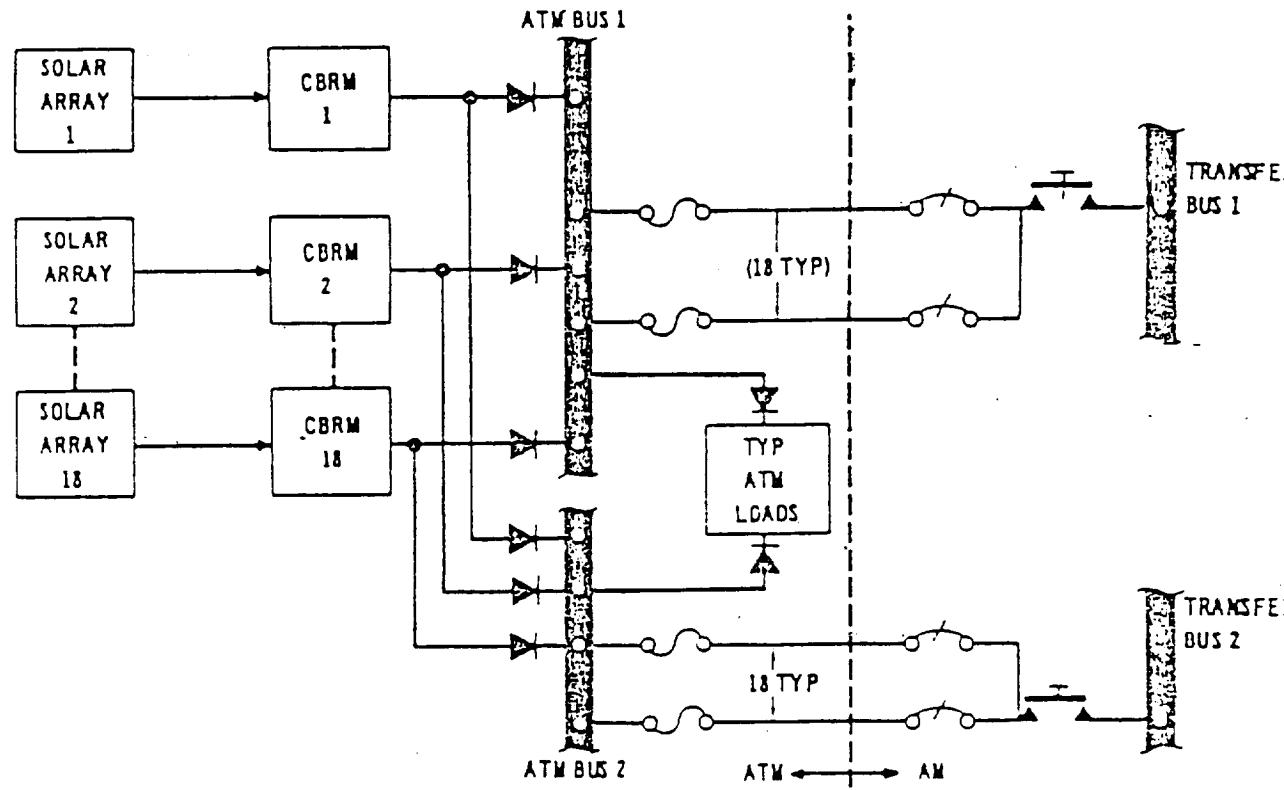
FIGURE 5-2. ATM Electrical Power System

6. SPACE SHUTTLE PROGRAM

The Space Shuttle represents a second generation system when compared to its predecessors such as the Apollo and Skylab Program. Although much of the technology was based on previous programs, there were technological areas that set this program apart from the other space programs. These areas are the thermal protection system of the Orbiter on entering the atmosphere and the Space Shuttle main engines.

The Space Shuttle is comprised of the Orbiter, a pair of solid rocket boosters, and the external tank. Of the three components mentioned, the Space Shuttle Orbiter will be discussed because it contains the electrical power system for the shuttle.

The flight operations of the shuttle consists of four phases [1]:

- Lift-off to orbit

- On-orbit operations

- De-orbit to land

- Ground turnaround operations for the next flight

According to [2], the Space Shuttle Program has so far named five Orbiters. Table 6.1 lists the Orbiters.

Electrical Power System of the Space Shuttle Orbiter

Electrical power for the Orbiter was generated by three fuel cells at a nominal 28 volts dc [3]. The fuel cells were connected to a three-bus system that distributed dc power to the forward, mid, and aft sections of the Orbiter [4]. Figure 6-1 shows the electrical power distribution block diagram. The main dc buses designated MNA, MNB, and MNC are the primary source of power for all dc loads on the Orbiter. Each bus provides power to three single-phase

solid state inverters which are interconnected on the ac side to form a three-phase system. The three three-phase inverters are connected respectively to a three-phase ac bus designated AC1, AC2, and AC3.

The inverters are rated at 750 volt-amperes with an output voltage of 117 volts at 400 hertz. The voltage and frequency regulation is +3/-1 volts and ± 2 hertz. The inverter efficiency is slightly larger than 75%.

All three fuel cells supplied power during average and peak power demands, but only two fuel cells were used when the demand was at its minimum. Two of the three fuel cells were connected to the three buses, while the third fuel cell was placed in a standby mode. If the power demand exceeded the power capacity of the two on-line fuel cells, the third fuel cell could be switched instantly from standby to an active mode in order to support the increase in electrical power [4].

Table 6.2 shows the electrical characteristics of the fuel cells as a function of power [3]. Each cell is rated at 12,000 watts peak and 7,000 watts continuous. The fuel cell system is capable of delivering 21,000 watts continuously with 15 minute peaks of 36,000 watts. The Orbiter power demand is approximately 14,000 watts, leaving a 7000 watt margin for payloads. During preorbit, approximately 10 minutes after launch, and deorbit to landing (30 minutes), the Orbiter could provide 1 kilowatt average to 1.5 kilowatts peak to the payload. Most of the experiment hardware was either on standby or turned off during this period.

Table 6.3 lists the voltage and power (average and peak) at various interfaces for different mission phases [4].

TABLE 6.1. THE NAMES OF THE SPACE SHUTTLE ORBITERS

Name	Comments
Enterprise	Test vehicle for landing tests in 1977.
Columbia	First vehicle to fly in space on 4-81.
Challenger	Originally designed as a test vehicle. Later it was modified and finished as a flight vehicle.
Discovery	Named after Hudson's ship that was used in search for the Northwest Passage and Captain Cook's ship when he discovered Hawaii.
Atlantis	Named after Woods Hole Oceanographic Institute research ship (1930-66).

TABLE 6.2. ELECTRICAL CHARACTERISTICS OF THE ORBITER FUEL CELLS

Power Watts	Voltage Volts	Current Amperes
2,000	32.5	61.5
12,000	27.5	436

TABLE 6.3. PAYLOAD POWER INTERFACE VOLTAGE AND POWER CHARACTERISTICS

Mission Phase	Interface	Voltage Range Min./Max.	Power, kW Average	Peak	Remarks
Prelaunch operation	Dedicated fuel cell connector	24/32 27/32	1 7	1.5 12	Normal checkout Orbiter powered down
	Main bus connector	24/32	1 5	1.5 8	Normal checkout Orbiter powered down
	Aft (bus B) Aft (bus C)	24/32 24/32	1.5 1.5	2 2	May be used simultaneously
Ascent/descent	Dedicated fuel cell connector	27/32	1	1.5	Power limited to a total of 1 kW average and 1.5 kW peak for 2 min.
	Main bus connector	27/32	1	1.5	
	Aft (bus B) Aft (bus C)	24/32 24/32	1 1	1.5 1.5	
On orbit payload operations	Dedicated fuel cell connector	27/Max. TBD	7	12	Peak power limited to 15 min. once every 3 hr.
	Main bus connector	27/32	5	8	
	Aft (bus B) Aft (bus C)	24/32 24/32	1.5 1.5	2 2	Power may be utili- from both inter- faces simultaneous- ly; buses must be isolated on the payload side of the interface.

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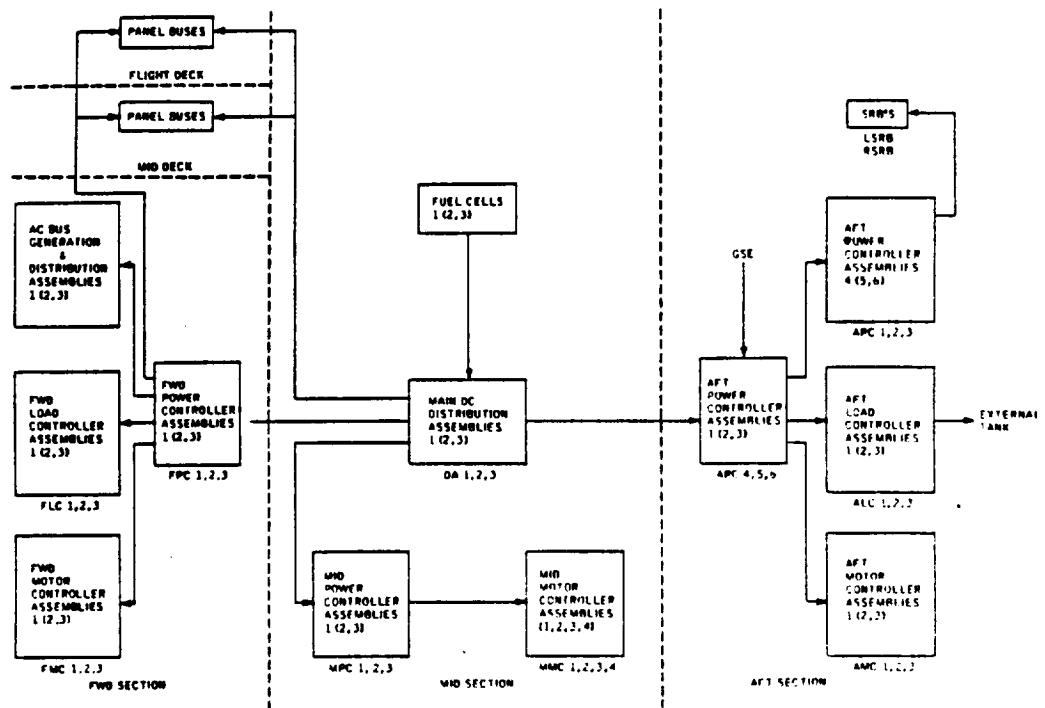
FIGURE 6-1. Electrical Power System of the Space Shuttle Orbiter

7. SPACELAB

In the mid 1980's Spacelab, a project developed by the European Satellite Association, will be placed in the payload bay of the Space Shuttle Orbiter and flown as an integral part of the Orbiter. The lifespan of Spacelab will be approximately 50 missions with a time duration from 1 to 4 weeks per mission [1].

Electrical Power System For Spacelab

The electrical power for the Spacelab and Space Shuttle Orbiter will be provided by the three fuel cells located in the Orbiter with any one of the three cells providing sufficient electrical power for a safe return of the Orbiter. The power output of any combination of two fuel cells would satisfy the Orbiter power demands leaving the third cell to supply power to Spacelab [2].

The output of the fuel cell supplying power to Spacelab will be 7 kilowatts average and 12 kilowatts peak for 15 minutes at a nomial 28 volts dc. At the interface with Spacelab, the voltage range is from 27 to 32 volts. Actually the fuel cell has a power capacity of 12 kilowatts continuous, but the time duration at this high power level is set by the heat rejection capability of the Orbiter.

A block diagram of the basic power requirements of Spacelab is shown in Figure 7-1 [2]. The design objective of the electrical distribution system are as follows:

- Deliver maximum power to the experiments

- Use known techniques to reduce risk and cost

- Design the system with flexibility in order to accommodate different payloads

In the initial stages of design, it appeared that the following choices were available:

- AC or DC distribution systems
- Distribution voltage levels
- Waveforms (sinusoidal or square wave) and frequency

These degrees of freedom were immediately constrained because of the electrical power supply of the Orbiter and the rigid requirements set by the users or experimenters. The users specified voltage levels, frequency, and power such that commercial equipment could be used on Spacelab with a minimal amount of modifications.

After careful analysis the ac segment was set at 115 volts and a frequency of 400 hertz. The compromise was based on inverter and filter weights. A sinusoidal waveform was preferred to the square wave because design of most commercial laboratory type equipment is based on a sinewave (single frequency). Square-wave signals are rich in harmonics and these harmonics can cause interference.

The choice of 28 volts dc or 115 volts ac depends on the importance of minimum conversion losses. Although the cable weight decreases with increased voltage, the decrease in cable weight may not compensate for the increase in inverter weight and the corresponding fuel losses. According to [2], if a load can use 28 volts dc, the voltage level should not be converted to 115 volts ac. The 50 hertz, 220 volts ac and 60 hertz, 115 volts ac are inverted directly from the 28 volts dc provided by the Orbiter.

Both the Common Support Subsystems and User Payload are supplied by 28

volts dc unregulated and 115 volts ac 400 hertz 3-phase sources. The user Payload has 50 hertz 220 volts and 60 hertz 115 volts single phase.

There are two battery supplies located on Spacelab. An emergency 10 ampere-hour, nickel-cadmium battery provides power to essential loads (emergency lights, computer, etc). The peak power battery, which is only flown on request by the user, is designed to supplement the power demand of experiments whose peak power may reach 10 to 20 kilowatts for brief periods. Silver-zinc, 500 ampere-hour batteries are used to supply peak power.

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OF POOR QUALITY

		REQUIREMENTS
	CDMS	28 V DC UNREG. ≈ 1 KW
	ECS	28 V DC UNREG. ≈ 1 KW 400 Hz AC
7 KW 28 V DC	CPSE	28 V DC UNREG. ≈ 100 W
	IPS	28 V DC UNREG. ≈ 400 W
COMMON SUPPORT 3-4KW	LIGHTS	28 V DC UNREG. ≈ 500 W

PAYLOAD
3-4KW

PAYLOAD

28 V DC UNREG. ±4 V
28 V DC REG. ± 2%
400 Hz AC 115 V
50 Hz AC 220 V
50 Hz AC 115 V
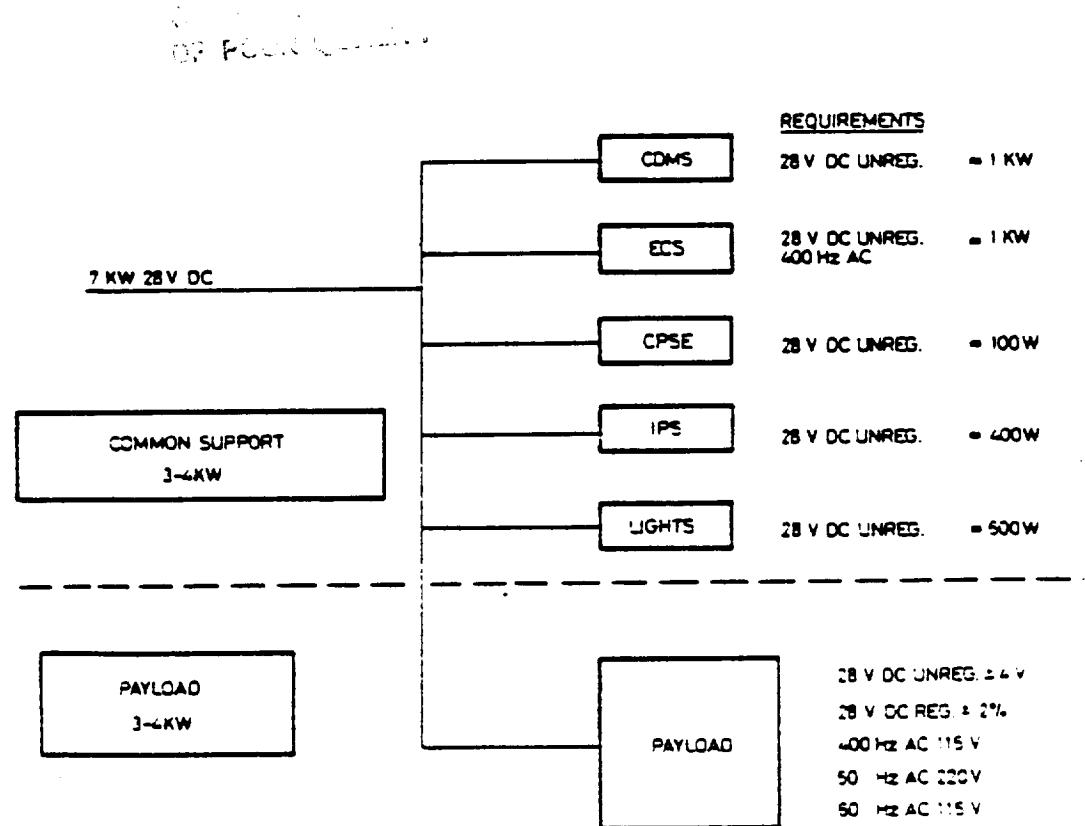
FIGURE 7-1. Basic Power Requirements of Spacelab

8. PIONEER MISSIONS

The Pioneer Project was initiated on October 11, 1958 [1] with the launch of Pioneer 1, the first man-made object known to escape the gravitational field of the Earth. Although it was labelled as a lunar probe, it never reached the Moon, but did travel a distance of approximately 73,000 miles from Earth. In its lifespan of 48 hours, it gathered enough data to indicate the extent of the Earth's radiation bands.

Pioneers 6 through 9 launched in 1965 through 1968 were designed to orbit the Sun. From the scientific data gathered from these missions, the Pioneer Program increased our knowledge of the interplanetary medium, especially giving a better understanding of solar wind, solar cosmic rays, and the structure of the Sun's plasma and magnetic fields. The last spacecraft in the series was launched August 27, 1969. Pioneers 6 through 9 were located at different angular positions around the Sun, but at approximately the same solar distance from the Sun as the Earth.

In 1969, a new group of Pioneer spacecraft was created having the following characteristics:

- Low cost
- Lightweight
- Spin-stabilized for fly-bys of other planets

Pioneers 10 and 11 were designed to fly by Jupiter with Pioneer 11 having the added task to fly by Saturn. These spacecrafts, launched in 1972 and 1973, passed through the radiation belts of Jupiter in 1973 and 1974 and Pioneer 11 continued on to fly by Saturn in 1979. After completing the fly by task, Pioneer 11 followed Pioneer 10 in the continuing exploration of interplanetary space.

The next step in the Pioneer Program was the development of two Pioneer Venus spacecrafts called the Orbiter and the Multiprobe [2]. The Multiprobe was a true planetary probe because it carried several spacecrafts into the Venusian atmosphere as opposed to orbiting the planet Venus like the Orbiter.

Because the Jupiter/Saturn mission was different from the Venus mission, the discussion will be divided into two parts. The first part will consider the Jupiter/Saturn mission (the outer planets) while the second part will focus on the Venus mission (inner planet).

A. PIONEER JUPITER/SATURN MISSION

Three complete Pioneer spacecrafts were built, a test vehicle and two flight versions to be launched to Jupiter. Pioneer 10, launched on March 2, 1972, was the first object launched with sufficient energy to escape the solar system. On April 5, 1973, Pioneer 11 was launched. The primary objectives of these missions were [3]:

- To investigate the interplanetary medium beyond Mars

- To investigate the nature of the asteroid belt

- To explore the environment of Jupiter

Scientists determined that, if a spacecraft flies past Jupiter in just the correct trajectory, the spacecraft will gain sufficient energy to allow it to move toward Saturn. Using this approach, the launch vehicle requirements and travel times to distant objects were greatly reduced [4]. The primary objectives were extended [5]:

- If the first fly-by to Jupiter accomplished its scientific objectives, a second spacecraft would be launched to fly-by Jupiter along a trajectory that would enable it to reach Saturn.

. The second spacecraft would then investigate the
  Saturnian environment.

Pioneer 10 was launched March 2, 1972. The data collected from this
mission was used to select a trajectory for Pioneer 11's Jupiter fly-by.
Pioneer 11 was launched on April 5, 1973.

Electrical Power System of Pioneer Jupiter/Saturn Mission

Solar cells were considered as the main source of electrical power.
Since the light intensity at Jupiter is only 1/27 the intensity at Earth,
arrays with large surface areas would be required in order to meet the
spacecraft power demand. Another serious problem would be the potential
damage that could occur to the solar cells when the spacecraft passes through
the Jovian radiation belts. Hence, radioisotope thermoelectric generators
(RTG) were specified for Pioneers 10 and 11 [5].

Nuclear-fueled electric power was derived from SNAP-19-type RTGs, similar
to the power source that had been used successfully on the Nimbus-3
meteorological satellite [5].

In order to reduce the effects of neutron radiation onto the scientific
instruments, the RTGs were mounted in pairs at the end of each of two extended
booms. The four RTGs developed approximately 155 watts of power at
launch-time and when the spacecraft reached Jupiter, the power output was
about 140 watts. The power output continued to decrease, but at a lesser
rate, after Pioneers 10 and 11 passed Jupiter. The decrease in power level
was attributed to the deterioration in the thermocouple junctions rather than
the radioactive decay of plutonium-238.

All systems and experiments on the spacecraft required only 100 watts, allowing a margin from 40 to 55 watts. The scientific instruments required 25% of the total power or 25 watts.

The excess power not used by the spacecraft was dissipated into space as heat by a shunt resistor radiator or was used to charge a battery that supplied additional power when the power demand exceeded the power output of the RTGs.

The electrical power system was designed with high redundancy such that any single component failure, even power from one RTG, would not degrade the performance or cause a mission failure [6].

The 4.2 volt dc output of the four RTGs was inverted to a higher ac voltage by four inverters whose outputs were connected in parallel. The inverter output had a 2.5 kHz trapezoidal waveform with a rms value of 30.5 volts. Each RTG was connected to its own inverter in order to simplify fault isolations.

The power control unit (PCU), which contains a rectifier filter, shunt regulator, charge-discharge control for the battery, undervoltage sensing circuit, and telemetry conditioning circuit, supplied 28 volts dc regulated ($\pm$ 2%) to the electrical loads such as the scientific instruments.

If the power output of the RTGs exceeded the electrical demand by the spacecraft, the shunt regulator and thermal radiator dissipated the excess power. When the system electrical power demand exceeded the RTG power output, the PCU controlled the discharge of the battery and provided regulation of the main dc bus to 28 volts $\pm$ 2%.

The subsystem design requirements are presented in Table 8.1, the major power subsystem parameters in Table 8.2, and the RTG characteristics in Table 8.3 [6].

Figures 8-1 and 8-2 show the typical RTG power and I-V characteristics and electrical system block diagram for the Pioneer Jupiter/Saturn spacecraft, respectively.

B. PIONEER VENUS MISSION

The Orbiter and Multiprobe used the same basic Pioneer Bus in order to reduce the cost of the mission. The Multiprobe carried four probes (one large probe and three identical small probes). After approaching Venus, the Multiprobe released its four probes along with the Bus toward different target areas on the surface of the planet.

The Orbiter was designed to explore Venus in four ways [7]:

- To investigate the clouds of the entire planet

- To measure the characteristics of upper atmosphere and the ionosphere over the entire planet

- To penetrate the Venusian cloud layers using a radar instrument

- To determine the general shape of the gravitational field of Venus

Likewise, the Multiprobe spacecraft was designed to investigate Venus in four ways [7]:

- To study the nature and composition of the clouds surrounding Venus by direct sampling of the clouds

- To determine the profile composition, structure, and heat balance of the atmosphere of Venus as a function of altitude by direct sampling and measurements of radiation

- To determine the atmospheric circulation behavior around the planet

. To investigate how the planet interacts with the
solar wind

To accomplish the above objectives, the Orbiter and Multiprobe carried a
complement of 12 and 18 scientific experiments, respectively. For the
Multiprobe spacecraft, the 18 experiments were located as follows: 2 aboard
the bus, 7 on the large probe, and 3 on each of the three identical small
probes [7].

Electrical Power System of the Pioneer Venus Spacecraft

The Multiprobe power system was essentially the same as that for the
Orbiter spacecraft. However, the Multiprobe had a power interface unit that
allowed probes to be powered from the Bus without depleting their own
batteries during the interplanetary flight to Venus.

The Orbiter and Multiprobe power systems were designed with 432 watt-hour
of stored battery energy to support launch, eclipse, and periapsis modes [8].

The power electronics of the Orbiter spacecraft provided a semiregulated,
$28 \pm 10\%$ volts dc to all the electrical loads of the spacecraft which included
its science instruments.

The main source of power was the solar array which provided 226 watts at
Earth's orbit and 312 watts when the spacecraft orbited Venus. Two
nickel-cadmium batteries, 24 cells each, acted as the secondary source when
the output of the solar array was insufficient, as for example when the Sun
was not shining directly enough on the array or when the spacecraft was in the
shadow of Venus. These batteries were connected to the bus when the voltage
dropped below 27.8 volts [7]. Recharging the batteries was accomplished via a
small solar array.

The solar panel main and charge array characteristics are shown in Table 8.4 [2] along with other Orbiter spacecraft and Multiprobe Bus power subsystem characteristics.

Seven and five voltage limiters on the Orbiter spacecraft and Bus spacecraft, respectively, limited the maximum bus voltage to 30.8 volts dc (28 + 10% volts dc). Each limiter along with its load resistors dissipated 66 watts minimum at 30.8 volts dc. On command, any limiter that failed could have been disconnected.

The two nickel-cadmium batteries on both the Orbiter and Multiprobe were discharged through dissipative-type regulators. When the voltage was less than approximately 29.05 volts, the output voltage of the regulator tracked the terminal voltage of the battery to its minimum discharge level. In case the primary regulator failed, a redundant regulator could be turned on via a ground command.

Power was transmitted through the spacecraft via four isolated power buses. If the current level exceeded its safe level, loads were removed with the following priority:

. First, the scientific instruments were disconnected

. Second, the switched loads such as control and data-handling units and finally the transmitter were disconnected

Only the loads that were necessary for survival (command units, heaters, receivers, and power conditioning units) were left connected to the buses.

The Orbiter bus power subsystems and thermal subsystem were interfaced with each other in order to stabilize spacecraft unit temperature by dissipating excess solar panel power.

The Orbiter and Multiprobe main solar arrays were designed to supply electrical power to the subsystem within the voltage range of 28 volts dc $\pm$ 10% under varying conditions of sun angle, temperature, and solar intensity. Each solar panel had two smaller battery charge arrays that were connected in series with the main array to supply 36 volts to the battery chargers.

The probe power subsystem of the Multiprobe system was a silver-zinc battery that was located within each probe. Before the probes were separated from the bus, they received power from the bus spacecraft solar panel and/or nickel-cadmium batteries. Once the power was switched from the bus to the probes, a 40 ampere-hour battery in the large probe and an 11 ampere-hour battery supplied all probe power. The probe battery characteristics are shown in Table 8.5. Figure 8-3 is a schematic of the power subsystem [2].

TABLE 8.1. SUBSYSTEM DESIGN REQUIREMENTS

. Current output from RTGs shall be approximately equal to the value of the maximum power point current.

. The maximum value of the total RTG power be 174 watts with not more than 46 watts from one RTG.

. Built-in system redundancy such that a single component failure shall not cause mission failure.

. Battery protection by fail-safe circuitry.

. Initial battery capacity of 35 watt hours.

. Minimize magnetic fields.

. Subsystem shall have the following specifications:

      i. 24 watts of 28 volts dc regulated power to the scientific instruments.

      ii. 73.8 watts of power to the spacecraft subsystems.

. Subsystem reliability to exceed 0.9542 for 900 days.

TABLE 8.2. MAJOR POWER SUBSYSTEM PARAMETERS

- Spacecraft Load

  Cruise load — 97.8 watts

  Loads when commanding — 105.5 watts

- Inverter Output Voltage (line to center tap) — 30.5 volts rms ± 3%

- Power Control Unit

  Output voltage — 28.0 volts dc

  ± 1% (± 2% Including long-term drift) and 61.0 volts rms ± 3% line to line

  Undervoltage level — 26.0 ± 0.5 volt dc

  Shunt radiator power dissipation — 103 watts (maximum)

  Shunt regulator power dissipation inside PCU — 40 watts (maximum)

  Shunt power capability — 118.5 watts (maximum)

- Battery

  Discharge current at 28 volt bus — 1.0 (maximum)

  Discharge current at battery — 10.0 amps

  Charge current — 300 milliamps maximum

  Capacity at beginning of mission — 5 amp hour

  40 watt hour

- Control Transformer Rectifier and Filter Unit

  Separate output voltages — 33 outputs

  Output voltages —
  + 5   volts
  5.3 volts
  + 12   volts
  + 16   volts

- Reliability

  Reliability of subsystem for 900 days — 0.9860

TABLE 8.3. RTG CHARACTERISTICS

• Maximum power at liftoff from all four RTGs with pre-launch shroud air-conditioning unit on	174	watts
• Power at BOL	158	watts
• Predicted power at encounter	134	watts
• RTG maximum power voltage	4.2	volts
• Open circuit voltage (instantaneous)	6.9	volts

TABLE 8.4. ORBITER SPACECRAFT POWER AND MULTIPROBE
BUS POWER SUBSYSTEM CHARACTERISTICS

	CHARACTERISTICS
• Solar Panel Main Array	Orbiter Spacecraft: 305.1 watts (EOL) Bus Spacecraft: 212 watts (EOL)
• Charge Array	Orbiter Spacecraft: 0.25 ampere at L+48 and 0.54 ampere at Venus Bus Spacecraft: 0.35 ampere at earth and 0.33 ampere at Venus
• Battery	24, 7.5 ampere-hour nickel-cadmium cell/battery, two 12 cell packs/battery, two batteries/spacecraft
• Bus Voltage Limiter	Shunt Regulator: dissipating 66 watts/regulator

TABLE 8.5. PROBE BATTERY ELECTRICAL CHARACTERISTICS

	LARGE PROBE	SMALL PROBE
• Battery Capacity (ampere-hours)	40	11
• Steady-state Bus Voltage (volts)	25.2 to 30.8	25.2 to 30.8
• Maximum Steady-state Current (amperes)	17	3.5

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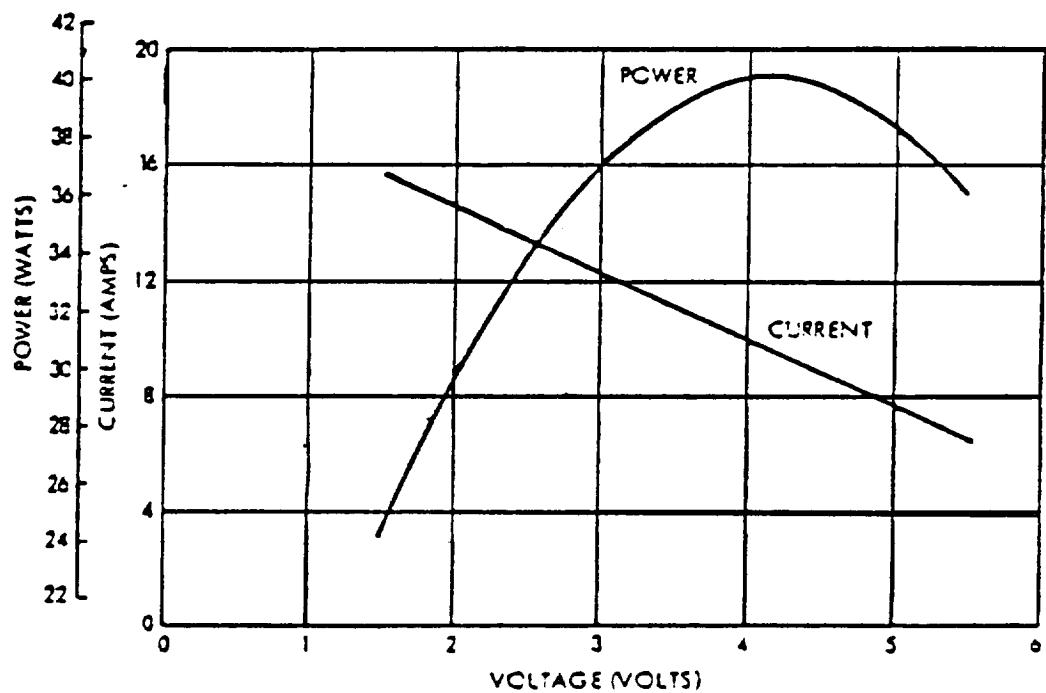
FIGURE 8-1. Typical Radioisotope Thermoelectric Generator
Characteristics
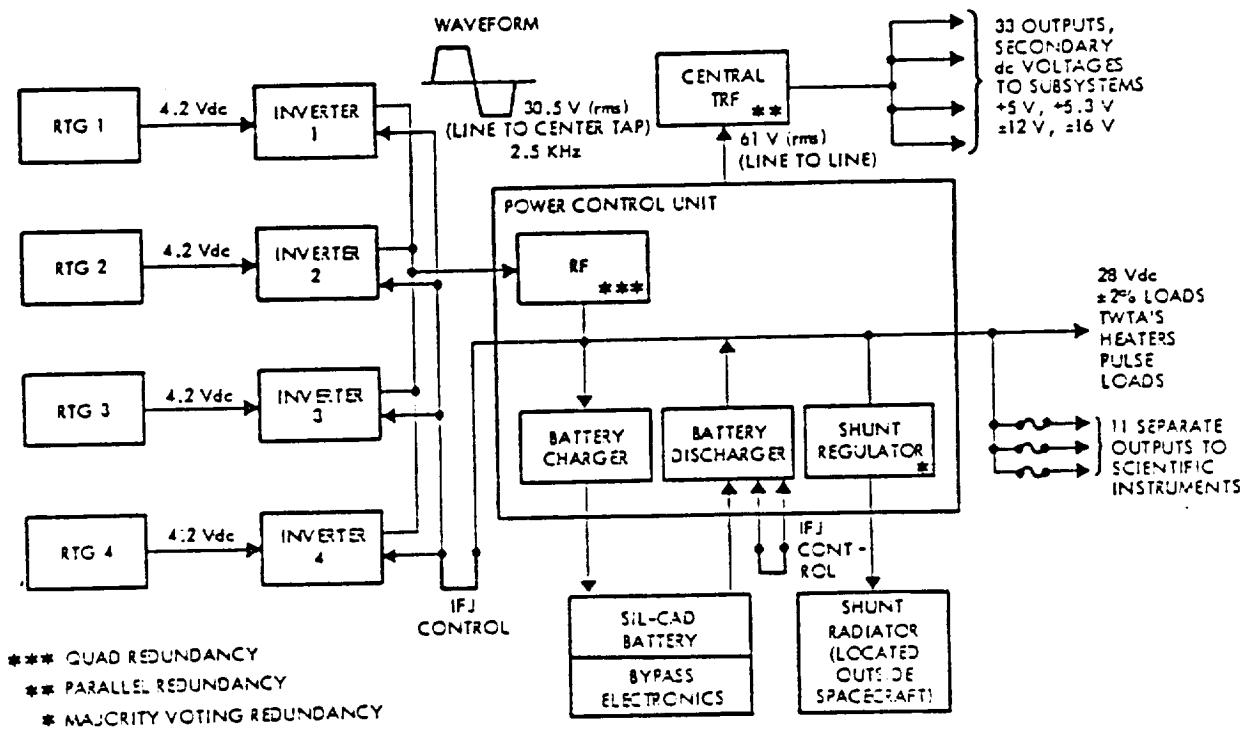
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FIGURE 8-2. Electrical System for the Pioneer Jupiter/Saturn Spacecraft
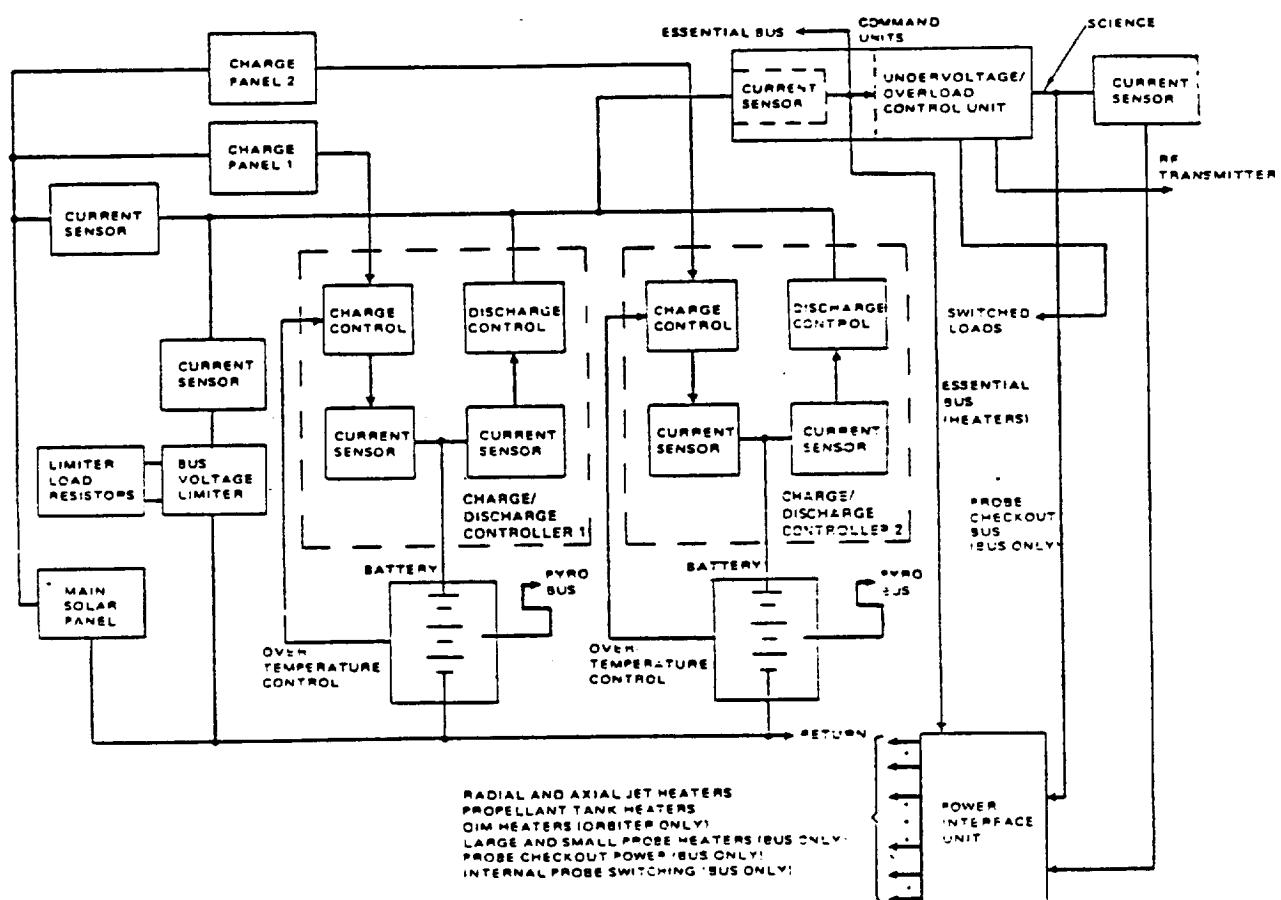
FIGURE 8-3.   Electrical Power System of the Multiprobe

9. MARINER PROGRAM

The Mariner Program consisted of 10 launches with the initial launch in 1962 and the final launch more than ten years later (1973). These missions were space probes that involved the exploration of other planets in our solar system. Table 9.1 shows the highlights of the Mariner Missions [1]. Missions 1, 3, and 8 failed for various reasons. The spacecrafts that will be discussed are: Mariners 2 and 5, Mariners 6 and 7, and Mariner 10 because these missions explored Venus, Mars, and Venus/Mercury, respectively.

A block diagram of the basic electrical power system of a Mariner is shown in Figure 9-1 [2]. The main source of power was generated by the solar panels, while the secondary power source was an 18-cell, silver-zinc battery. A pulse-width-modulated switching regulator increased the source voltage and regulated it at a nominal value of 52 volts dc. The output of the boost-regulator was converted to 50-volt, 2400 Hz square wave in the inverter for distribution to the electrical subsystems. Also, there was three-phase, 400 Hz quasi-square-wave power for the altitude control gyros.

The power-conditioning system would accept a source voltage between 25 and 50 volts. The solar panel voltage operated between 40 and 50 volts. This was above the terminal voltage of the battery which varied between 27 and 33 volts. The maximum solar panel voltage was set by a series string of six 50-watt zener diodes connected across each panel. The battery was connected to the input bus via an isolation diode which prevented the battery from continuously sharing with the solar arrays the power input to the power-conditioning system. If the main source (solar arrays) could not supply the power demand of the spacecraft, the battery would be connected through the

isolation diode to the bus because the bus voltage would be less than the battery voltage.

A. Mariner 2 Electrical Power System

The Mariner 2 power system consisted of a self-sufficient control supply of electrical power with the main power developed by two solar panels and a secondary supply composed of a rechargeable 1000 watt-hour sealed silver-zinc battery [3]. Figure 9-2 is a functional block diagram of the power system.

The solar array and battery supplied power to a switching and logic circuit whose output drove a booster regulator. The regulator excited a 3-phase, 400-Hz sinusoidal power amplifier and 2400-Hz square-wave power amplifier. Transformer-rectifier units (TRU) converted the 2400-Hz square-wave power to the appropriate dc levels. The TRUs were provided by the user. The 3-phase, 400 Hz power amplifier source supplied the necessary power to the ac motors contained in the gyros, antenna and update servos.

The spacecraft systems depended on the battery from launch until the solar arrays were positioned to face the sun. After the solar panels were sun-oriented, the solar arrays were the main source of power for all electrical loads and for recharging the battery. Approximately one hour after launch, sun acquisition was established and the solar arrays supplied power to the spacecraft during the rest of the mission which was approximately 17 months.

The solar arrays contained approximately 9800 solar cells with each cell producing about 0.23 volt. The arrays were designed to generate between 148 and 222 watts of electrical power [4]. The power demand during cruise-mode was approximately 150 watts.

B. Mariner 4 and 5 Electrical Power Systems

Because these missions were to different planets, general observations about the power systems will be made between Mariner 4 and 5.

The solar panels from the Mariner 5 Mission had to be redesigned because the spacecraft was to fly toward the Sun on its way to Venus rather than away from the Sun as in the case of Mariner 4 Mission to Mars.

For the Mariner 4 Mission, the solar flux varied between 135 mW/cm^2 at Earth and 58 mW/cm^2 at Mars with the solar arrays providing approximately 320 watts at Mars. Eighty-four series-connected cells generated 42.9 volts at 55°C near Earth. At Mars, the temperature decreased to 10°C. This would have caused the panel voltage to exceed 50 volts if it were not for the voltage-limiting zener diodes. Each panel consisted of four sections connected in parallel via isolation diodes for increased reliability.

For the Mariner 5 Mission the solar flux varied between 135 mW/cm^2 at Earth and 270 mW/cm^2 at Venus. One-hundred and five series-connected cells provided 43 volts at Venus. At Earth the panel voltage was limited to 50 volts by the six zener diodes. Because the number of cells was increased from 84 to 105, each panel contained three rather than four sections as used on Mariner 4.

The battery used in the Mariner 5 Mission was essentially the same type that was used in Mariner 4. The battery capacity was adequate for both the launch and midcourse maneuver on Mariners 4 and 5.

Table 9.2A and Table 9.2B show the significant changes in the Mariner 5 electrical power system because of different power requirements and the necessary modifications to convert the Mariner 4 power conditioning equipment [5].

C. Mariners 6 and 7 Electrical Power Systems

The primary source of electrical power for Mariners 6 and 7 was an array of photovoltaic cells which were directed toward the Sun for most of the flight to Mars and supplied 800 watts near Earth and 450 watts at Mars. There was a 2.32 times increase in power output when compared to the array power output at Mars for Mariner 4 [6]. The 450 watt power output provided a margin above the maximum power requirement of 380 watts at encounter to account for any solar cell degradation due to solar flares. A rechargeable silver-zinc battery on each spacecraft supplied power during launch, midcourse maneuver, and when the solar arrays were not directed toward the Sun. The battery, which was maintained at full charge throughout the mission, was available as an emergency power source during encounter. This was true in the case of Mariner 6, but for Mariner 7, the battery failed a few days before encounter.

D. Mariner 9 Electrical Power System

Mariner 9 or Mariner-Mars 1971 was the first spacecraft to orbit another planet. Its mission was to make scientific observations of the surface of Mars.

The basic difference between this mission and the previous Mariner missions to Mars was the fact that Mariner 9 has existed in a Mars orbit environmental mode for a long period of time. After 45,960 commands to Mariner 9, its radio transmitter was turned off and it is expected to orbit Mars for at least 50 years [1].

It was noted during the flights of Mariners 6 and 7 that there was an unexpected 3 to 5% degradation in the solar array current output as the spacecrafts traveled to Mars and that the degradation could not be attributed to electron or proton bombardment on the surface of the solar cells.

Mariner 9 mission furnished an opportunity to obtain more information about this phenomenon [9]. Data from this mission suggested that degrading environment was not due to electron or proton bombardment, but more probably due to the solar cell exposure to ultra-violet effects.

The Mariner 9 electrical power system, Figure 9-3, has the following primary functions [10]:

- Provide a central supply of electrical power to the spacecraft electrical equipment

- Provide switching and control functions for management and distribution of power

- Provide a control timing function for the spacecraft

The main source of power was supplied by four solar panels and a secondary battery source. Power from the main and secondary source was processed and distributed in the following forms:

- Regulated 30 volts dc for gimbal and engine valve actuators

- 2400 Hz, single-phase, square-wave power (main and standby inverter) for science and engineering systems and for the cone actuator and propulsion module heaters (as required)

- 400 Hz, single-phase, square-wave power (inverter) to scan control subsystem

- 400 Hz, three-phase, quasi-square-wave power (inverter) to the altitude control subsystem for gyro motors

- Unregulated dc power to heaters, battery charger, and radio frequency subsystem

E. Mariner 10 Electrical Power System

The basic design of Mariner 10 was an octagonal main structure with eight equipment bays similar to the earlier Mariner spacecrafts.

Appropriate modifications were required because no previous Mariner spacecraft was designed to travel into the inner solar system. The solar intensity near Mercury would be almost five times the intensity at Earth, and the solar array must be designed so that it would not overheat. In order to maintain the temperature of the solar panels within design specifications, their mountings could be tilted away from the Sun [7]. To maintain a solar panel array temperature at approximately 100°C, provide a reasonable constant power output from the arrays into the electrical system, and meet the weight constraints, the rotatable configuration or sail configuration was adopted. Although the configuration was more complicated, weighed more, and would be more expensive than the earlier V-tilt design, the design allowed the mounting of the roll/yaw cold gas jets at the tips of the sails thereby increasing the leverage of their thrust [8].

A functional block diagram of Mariner 10 is shown in Figure 9-4 [10]. The main source of power was derived from a set of solar panels. The cells on each panel were arranged into 46 groups with groups connected in parallel. Each group was composed of 131 cells in series. The groups were arranged into six separate sections that were connected in parallel through blocking diodes to the dc unregulated bus. The maximum voltage of a section was limited to 51 volts dc by a zener diode that was connected across the section.

During the spacecraft maneuver periods when the arrays were not directed toward the Sun, the secondary power was supplied by a rechargeable, 26-cell, nickel-cadmium battery with a capacity of 20 ampere-hours. The battery was fully charged at 39 volts dc. When the unregulated dc bus voltage was less than the battery voltage, the diode automatically connected the bus to the battery thereby allowing the battery to either share with the solar panel or supply all the power to the load. The battery could be charged either automatically or by ground command at a high or low rate (1 or 0.65 amperes).

The main booster regulator converted the solar array voltage to 56 ± 1% volts dc while the standby booster regulator, which was identical to the main booster regulator, provided redundancy. The standby booster regulator was activated by onboard detection of over or under voltage at the main booster regulator output.

The booster regulator, in turn, drove a 2400 Hz, single-phase (main) and two 400 Hz, single and three-phase inverters. The main inverter was designed to generate a square-wave signal at 50 volts rms, the single-phase 400 Hz inverter supplied 28 volts ac (rms), and the three-phase 400 Hz inverter provided 27.2 volts ac (rms). Table 9.3 lists the inverter specifications [9]. The 400 Hz single-phase and three-phase inverters provided power to the scan actuators and gyros, respectively.

Unregulated power was supplied to the heaters and the radio subsystem, which provided its own high voltages required by the traveling wave tube power amplifiers.

TABLE 9.1. HIGHLIGHTS OF MARINER MISSIONS

Mission	Launch Date	Remarks
Mariner 2	8/62	Take microwave and infrared measurements in order to determine the characteristics and temperature of Venus atmosphere, conduct interplanetary fields and particles measurements, and verify interplanetary communications.
Mariner 4	11/64	To investigate the Martian atmosphere and its surface and conduct fields and particles measurements between the orbits of Mars and Earth.
Mariner 5	6/67	To measure the surface temperatures, magnetic field and ionosphere of Venus.
Mariner 6 & 7	2/69 & 3/69	Explore Mars using two spacecrafts in the flyby mode in order to study the atmosphere and surface of Mars and to extend technology for future Mars missions.
Mariner 9	5/71	Map 70% of Mars.
Mariner 10	11/73	First dual-planet mission (Venus/Mercury) and the first mission to use the gravity assist of one planet (Venus) to achieve a Mercury encounter.

TABLE 9.2.A.   SIGNIFICANT CHANGES IN THE MARINER 5
ELECTRICAL POWER SYSTEM

- The main 2.4 kHz inverter maximum output
  power was increased from 80 to 105 watts

- No single-phase 400 Hz power was required
  for Mariner 5

- Unregulated power was necessary to keep the
  magnetometer warm

TABLE 9.2.B.   REQUIRED MODIFICATIONS TO CONVERT THE MARINER 4
POWER CONDITIONING EQUIPMENT

- Modify the power regulator by the addition
  of some small circuits

- Redesign of the logic on the power distri-
  bution assembly

- Addition of a sensor to ascertain the state
  of the battery charger for the 28-V dc
  command toggling

- Employ the main 2.4 kHz inverters for
  Mariner 4 as spares for the Mariner 5
  maneuver and construction of new modules
  for the main inverter

- Design the maneuver 2.4 kHz inverter such
  that power could not energize it if it
  were accidentally used as a main inverter

- Delete the 400 Hz single-phase inverter
  from the power system

TABLE 9.3. INVERTER SPECIFICATIONS

	Frequency	Voltage Output (rms)
Main Inverter	2400 Hz ± 0.01%	50, + 2 or -3%
Inverter (400 Hz single-phase)	400 Hz ± 0.01%	28, ± 5%
Inverter (400 Hz three-phase)	400 Hz ± 0.01%	27.2 ± 5%

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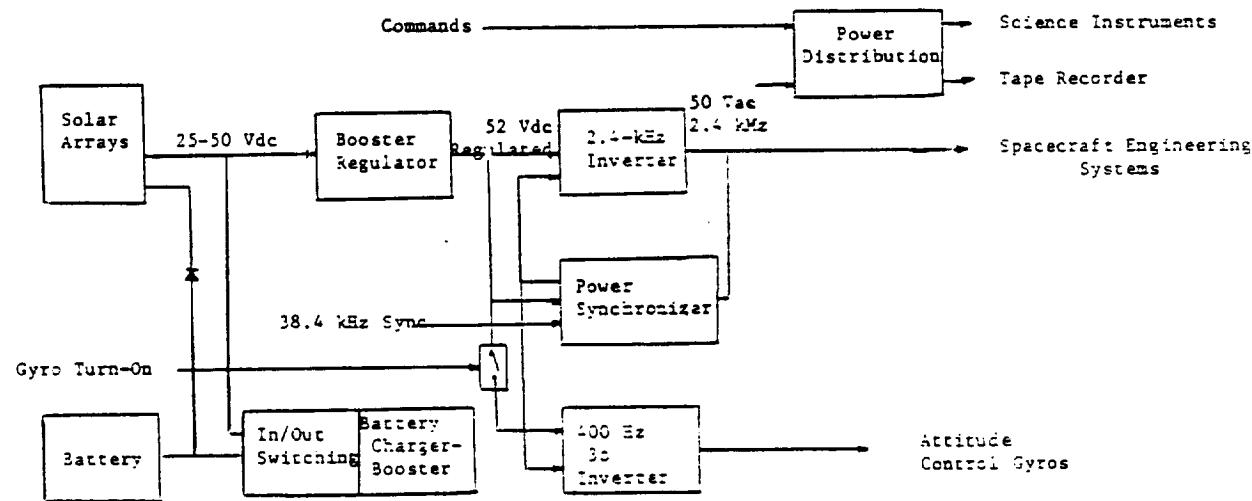
FIGURE 9-1. Electrical Block Diagram of the Mariner Power System
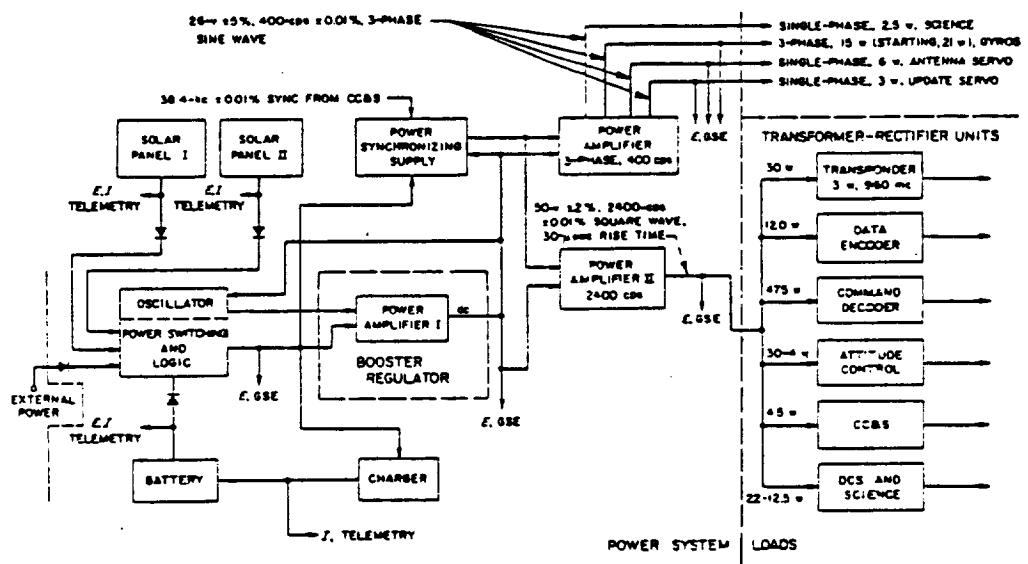
FIGURE 9-2. Mariner 2 Electrical Power System
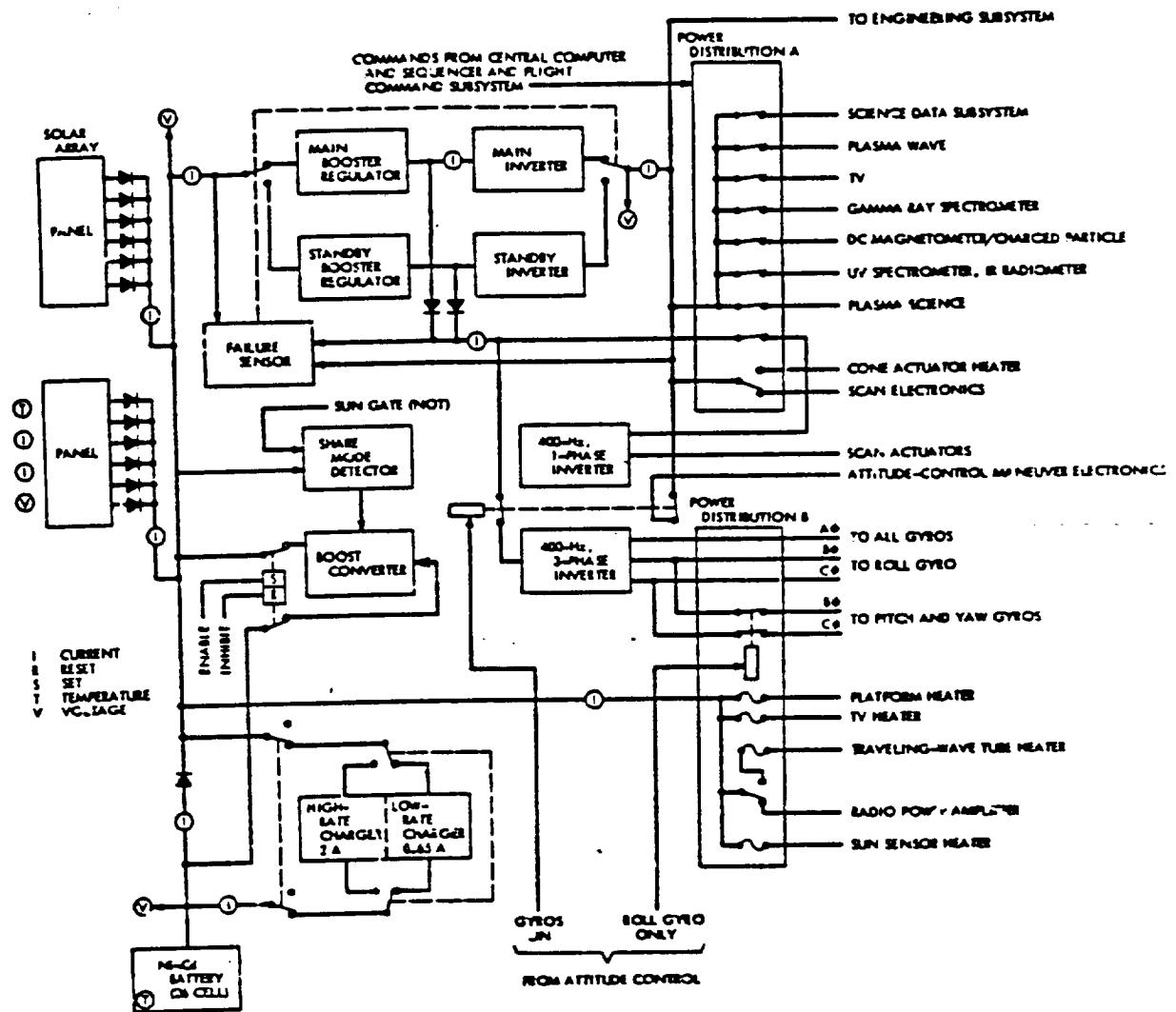
FIGURE 9-3. Mariner 9 Electrical Power System
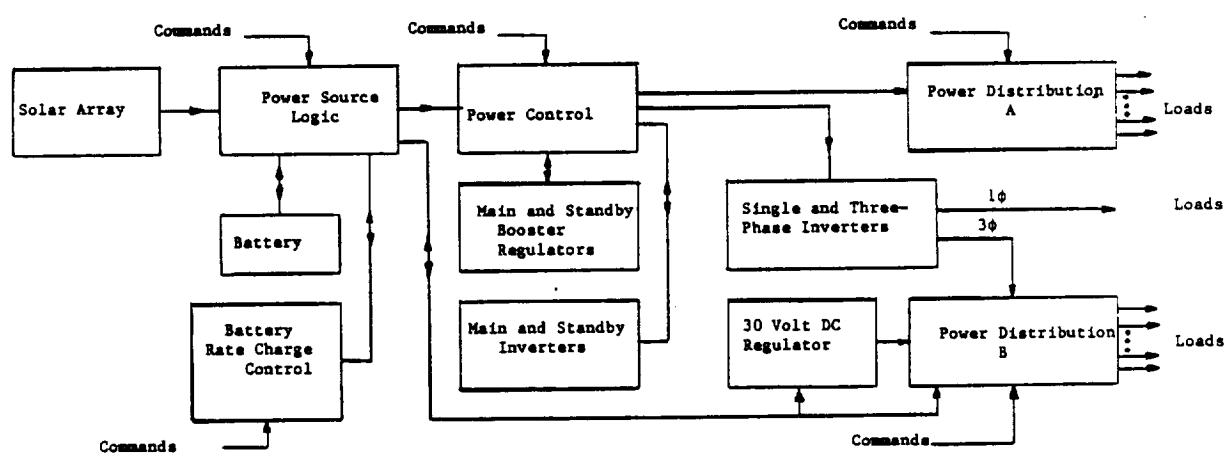
FIGURE 9-4. A Functional Electrical Block Diagram of Mariner 10.

10. RANGER PROJECT

There were three space projects which were designed to gather knowledge about the lunar surface in order to determine if manned Apollo landings were possible. These three were Ranger, Lunar Orbiter, and Surveyor. This segment will discuss the Ranger project.

The Ranger project consisted of nine missions of which the first 6 missions ended in failure [1]. After a 68-hour flight, Ranger 7, which was launched in July 1964, returned 4308 close-up pictures of the lunar surface before it impacted in the Sea of Clouds on the surface of the Moon. Ranger 9, launched in May 1965, was the last in the series. Table 10.1 lists the successful mission flights [2].

The objectives of Ranger spacecraft were to deliver its cargo to a point on the surface of the Moon within specified tolerances, position the experiments, perform the scientific experiments, and transmit the data back to Earth. Table 10.2 lists the electrical power specifications of the Ranger spacecraft.

Ranger Electrical Power System

The power system in Rangers 7, 8, and 9 were essentially similar in design which consisted of two solar-cell panels, two silver-zinc batteries, and switching control equipment. A block diagram of the power system is shown in Figure 10-1 [3].

External power was supplied at 25.5 volts dc until 5 minutes before launch, when the two 42 ampere-hour batteries assumed the load at 25.5 volts dc [4]. During a normal Ranger mission, less than 20% of the battery capacity

was used, and thus no provision was provided for recharging the batteries when the solar panels were generating power. The solar panels supplied power over the entire mission after Sun acquisition except during the midcourse maneuver phase.

Each solar panel consisted of 4896 silicon solar cells. During the midcourse maneuver, the solar panels supplied the raw power load of approximately 145 watts out to a pitch angle of 48 degrees. The electrical load was shared between the solar panel and battery during this period. After the midcourse motor burn and Sun acquisition, the solar panels supplied the total raw power load of approximately 120 watts at a pitch angle of 58 degrees [3].

TABLE 10.1. SUCCESSFUL RANGER MISSIONS

	LAUNCH DATE	RESULTS
Ranger 7	7/64	After lunar impact in the Sea of Clouds, returned televised picture of the lunar terrain.
Ranger 8	3/65	After lunar impact in the Sea of Tranquility, returned televised pictures of the lunar terrain.
Ranger 9	3/65	After lunar impact in the Crater Alphonsus, returned televised pictures of highland crater.

TABLE 10.2. ELECTRICAL POWER SPECIFICATIONS OF RANGER SPACECRAFT

Solar Panel: 4896 Silicon Solar Cells [2]

Panel Total Area: 2.3 square meters

Panel Power Output: 100 watts

Batteries: Silver-zinc, 1000 watt-hours [2] S/C

Silver-zinc, 1200 watt-hours [2] TV

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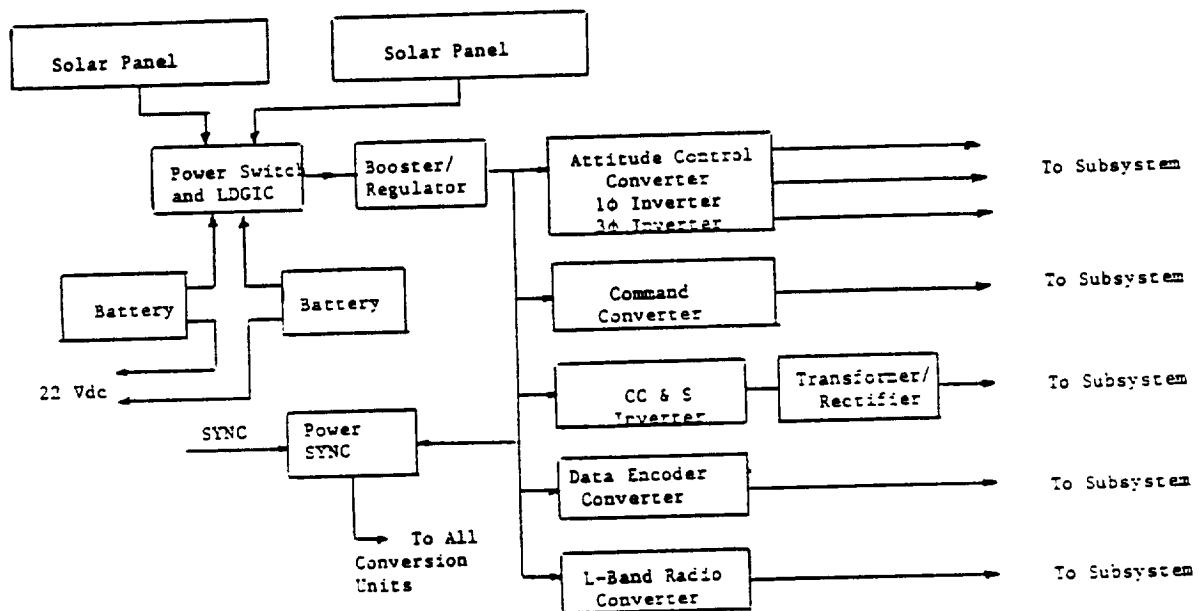
FIGURE 10-1. Ranger Electrical Power System

11. LUNAR ORBITER PROGRAM

The Lunar Orbiter program consisted of five spacecraft missions launched between August, 1966 and August, 1967 and was the second of the three unmanned projects (Ranger, Lunar Orbiter, and Surveyor Programs). The main focus of the program was to aid in the selection of Apollo landing sites in the equatorial regions of the Moon [1,2]. During the first three missions, 20 potential lunar landing sites were photographed from low-inclination and relatively low-altitude orbits. Missions four and five, however, were assigned broader scientific tasks and were placed in polar orbits.

For example, Lunar Orbiter I included instrumentation to sample certain lunar environmental conditions, to monitor the performance of the spacecraft subsystems and to further define the exact size and shape of the Moon [3].

All Lunar Orbiter spacecraft were deliberately crashed on to the surface of the Moon in order to make sure that there would be no radio frequency interference with later missions.

Electrical Power System of Lunar Orbiter

A power system block diagram of Lunar Orbiter I is shown in Figure 11-1 [3] and is composed of the following subunits:

- Four identical solar arrays

- Two 10-cell nickel-cadmium batteries with a total capacity of 12 ampere-hours

- Charge controller with a maximum charging of 2.85 amperes and a maximum trickle charging current of 0.3 ampere

- Shunt regulator provided a 20-volt dc bus

There were 2714 silicon solar cells per panel with the cells arranged in five circuits that were isolated via five diodes. Three circuits consisted of 104 series connected modules with each module composed of a six-cell group, one circuit consisted of 104 series connected modules with each module composed of an eight-cell group, and one auxiliary 10-cell series connected solar patch. Cells within the six and eight-cell modules were connected in parallel. The four circuits consisting of six and eight-cell modules on each panel were connected in parallel via isolation diodes to a common bus. The resultant spacecraft solar array was composed of 10,816 cells. The auxiliary 40-cell circuit provided base voltage and current that was required to saturate the main pass transistor in the charge controller unit.

The two 10-cell nickel-cadmium, 6 ampere-hour batteries received electrical energy from the solar array and supplied all the electrical energy from 6 minutes before launch to Sun acquisition and during Sun occultations. When the load demand exceeded the power output of the solar array, the batteries and solar array shared the load.

Protection from battery overvoltage and overtemperature were controlled by the battery charging rate that was regulated by the battery-charge controller whose maximum charging and trickle current was 2.85 and 0.3 amperes, respectively.

The shunt regulator was functionally active when there was excess power from the solar array and it fixed the upper solar array bus level to 30.56 volts dc by dissipating the excess power in heat dissipating elements located outside the heat shield. The spacecraft bus voltage was closely regulated at 20 volts.

Lunar Orbiters III and V electrical power systems, like their predecessor Lunar Orbiter I, were similar in configuration [4,5] except for the addition of the booster regulator for the photo subsystem in Lunar Orbiter V spacecraft. See Figure 11-2. The solar array for both Lunar Orbiters III and V functioned normally during the extended mission, providing sufficient power to maintain a constant bus voltage of 30.56 volts dc when the solar panels were directed toward the sunlight. The total solar panel power for Lunar Orbiter III and V at launch was 13.30 and 12.49 amperes at 30.56 volts, respectively. The battery performance was as predicted throughout the mission.

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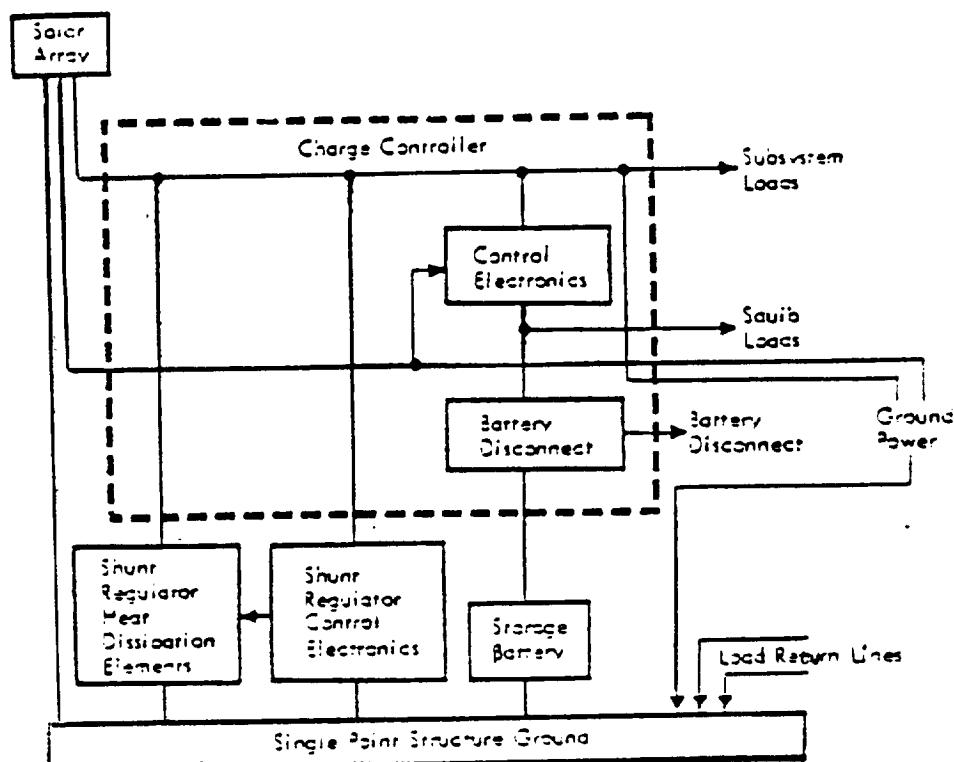
FIGURE 11-1. Power System Block Diagram of Lunar Orbiter I
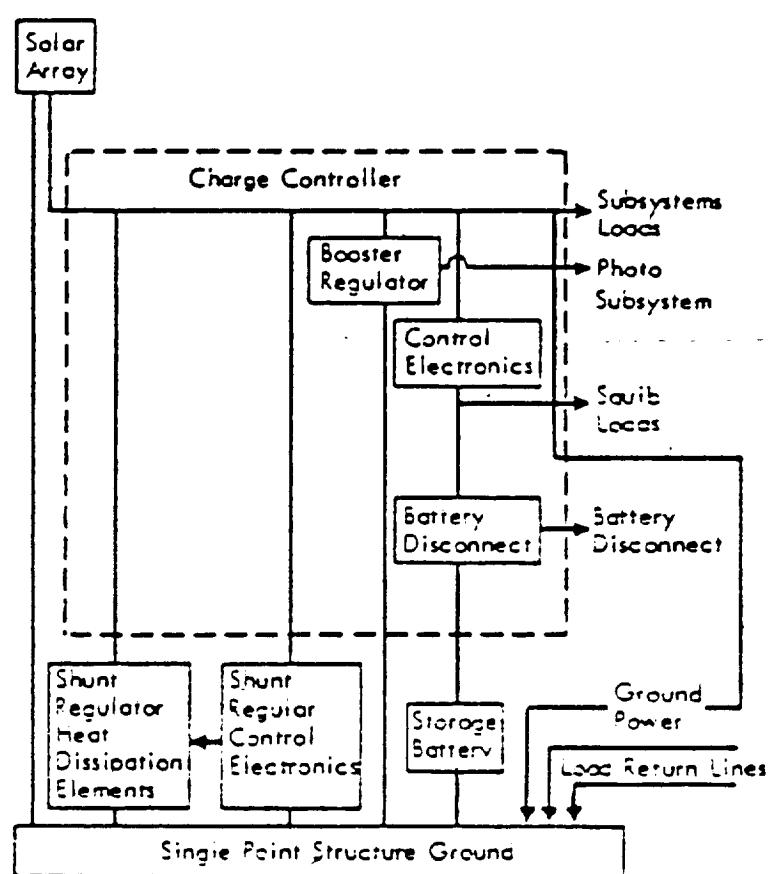
FIGURE 11.2. Power System Block Diagram of Lunar Orbiter V

12. SURVEYOR PROJECT

The Surveyor spacecraft was designed to perform a soft lunar landing, transmit to Earth engineering and scientific data about the physical characteristics of the Moon, and accomplish the following principle objectives [1]:

* To land on selected areas of the surface of the Moon.

* To perform experiments on the surface of the Moon.

* To gather data on the functional behavior of the spacecraft for future space programs.

* To be able to execute the following items as a minimum:

    i.   Operate during lunar day.
    ii. Operate for 3 and 20 hours before dawn for SC-I through SC-IV and SC-V through SC-VII missions, respectively.
    iii. Provide 150 hours of postsunset operations.

Surveyor I, launched in May, 1966 approached the Moon approximately 63 hours later and landed on the surface of the Moon on 6/66 [2]. The last mission, Surveyor 7, was launched in January, 1968.

Surveyor Spacecraft Electrical Power System

The electrical power system design for Surveyor I-IV was different from Surveyor V-VII missions because of the extension of the pre-dawn activity from 3 to 20 hours. Hence, the discussion of the electrical power system will be divided into two parts: Surveyor I-IV and Surveyor V-VII missions.

The electrical power system for Surveyor I-IV, shown in Figure 12-1, received energy from the solar panel and the spacecraft battery system [3]. The power output of the solar array varied from 55 to 90 watts depending on the temperature of the panel and its orientation with respect to the Sun.

The respective current capacity for the main and auxiliary battery was 165 and 45 ampere-hours. Both the main and auxiliary batteries were of silver-zinc type.

Power from the solar arrays and batteries was controlled, regulated, and distributed via power conditioning units prior to delivery to the various loads. Table 12.1 presents the important performance characteristics of the electrical power system for Surveyor I-IV [3].

The electrical power system for Surveyor V-VII, shown in Figure 12-2, was designed to provide sufficient power to be sure the transit and touchdown demands were fulfilled and to land with ample electrical capacity to maintain the operation of the spacecraft in the event that the spacecraft solar power is not available for about 20 hours. Table 12.2 presents the important performance characteristics of the electrical power system for Surveyor V-VII.

In the case of Surveyor I-IV, the solar panel maximum power point during transit was approximately 47 volts. Using an optimum charge regulator, maximum power was supplied by the solar array at a nominal voltage of 22 volts which was detrmined by the battery system. This voltage was boosted to 30.5 $\pm$ 1 volts. In contrast, the new design of solar panel on Surveyor V-VII eliminated the need for the optimum charge regulator because the power transfer could be accomplished directly from the solar array to the preregulated bus. This approach had two advantages: (1) it eliminated the electrical losses in the optimum charge regulator (approximately 18 watts was lost during the operation of optimum charge regulator in Surveyor I-IV) and (2) the internal losses in the booster-regulator preregulator circuitry were decreased by a sizeable amount because most of the preregulated output current was supplied directly from the solar array.

Surveyor V-VII power system had the following important improvements over the previous Surveyor electrical system [3]:

- Increased power system efficiency by eliminating the losses in the optimum charge regulator and the boost-regulator

- Spacecraft power requirements were decreased to a level such that the auxiliary battery could be omitted from the system

- A more reliable solar panel could be employed at a lower power level

- During the cruise mode, the power demand was such that no battery discharge occurred

- Reliability was improved because of simplification and a decrease in the number of parts

Table 12.3 lists the units contained in the power system of Surveyor I-IV and Surveyor V-VII [3].

TABLE 12.1. SURVEYORS I-IV ELECTRICAL POWER SYSTEM
PERFORMANCE CHARACTERISTICS

Solar Panel	89 ± 5 watts at $122 \pm 5.4°F$ at mean solar intensity
Main Battery Voltage (Rechargeable)	22, + 4.0, - 4.5 volts dc Temperature: 40-125°F
Auxiliary Battery Voltage (Non-rechargeable)	22, + 4.0, - 3.5 volts dc Temperature: 40-130°F
Battery Charge Regulator	Input voltage: 30-90 volts Input current: 0-2.3 amperes Output voltage: 17.5-27.5 volts (unregulated) Output current: 0-5 amperes
Booster-Regulator	Unregulated input Voltage: 17.0-27.3 volts Current: 15.3 amperes at 18 volts maximum for 7 ampere output Regulated output Voltage: 29 volts Continuous maximum current Essential bus: 0-0.06 amperes Non-essential bus: 0-6.0 amperes Flight control bus: 0-3.0 amperes
Booster-Regulator, Flight Control Regulator	Output voltage: 29.0 ± 0.29 volts Maximum output current: 3.0 amperes

TABLE 12.2. SURVEYORS V-VII ELECTRICAL POWER SYSTEM
PERFORMANCE CHARACTERISTICS

Solar panel   81 watts at 143°F
                                          60.5 watts at 250°F

Battery charge regulator   Input voltage: 17.5-70 volts
                                          Input current: 0-3.5 amperes
                                          Output voltage
                                              Unregulated bus: 18-27.3 volts
                                              Preregulated bus: 30 ± 0.3 volts

Booster-Regulator   Unregulated input
                                              Voltage: 16.75-27.3 volts
                                              Current: 15.3 amperes at 18 volts
                                                       maximum for 7 ampere output

                                          Regulated input
                                              Voltage: 29.0 ± 0.29 volts
                                              Current: 25 milliamperes

                                          Regulated outputs
                                              Essential bus: 0-0.6 ampere continuous
                                                          at 29.3 ± 0.586 volts
                                              Nonessential bus: 0-0.6 amperes at
                                                          29.0 ± 0.29 volts or
                                                          29.0 ± 0.87 volts
                                                          for overload trip
                                                          circuit enabled or
                                                          bypassed, respectively
                                              Flight control bus: 0-3 ampere at
                                                          29.0 ± 0.29 volts

TABLE 12.3. UNITS CONTAINED IN THE POWER SYSTEM OF
SURVEYOR I-IV AND SURVEYOR V-VII

Surveyor I-IV	Surveyor V-VI
Solar panel	Solar panel (new)
Main battery	Main battery
Auxiliary battery	
Main power switch	Main power switch
Auxiliary battery control	
Battery charge regulator	Battery charge regulator (new)
Booster-regulator	Booster-regulator (new)
Engineering mechanismn auxiliary	Engineering mechanism auxiliary (modified)
Thermal control and heater assembly	Thermal control and heater assembly
Electrical conversion units	Electrical conversion units

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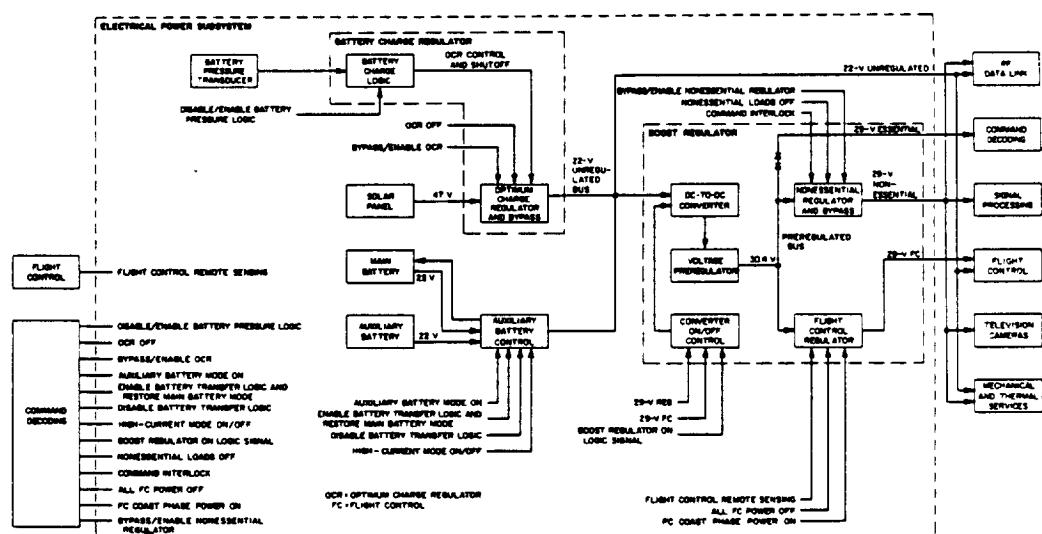
FIGURE 12-1. Electrical Power System Block Diagram
(Surveyors I through IV)
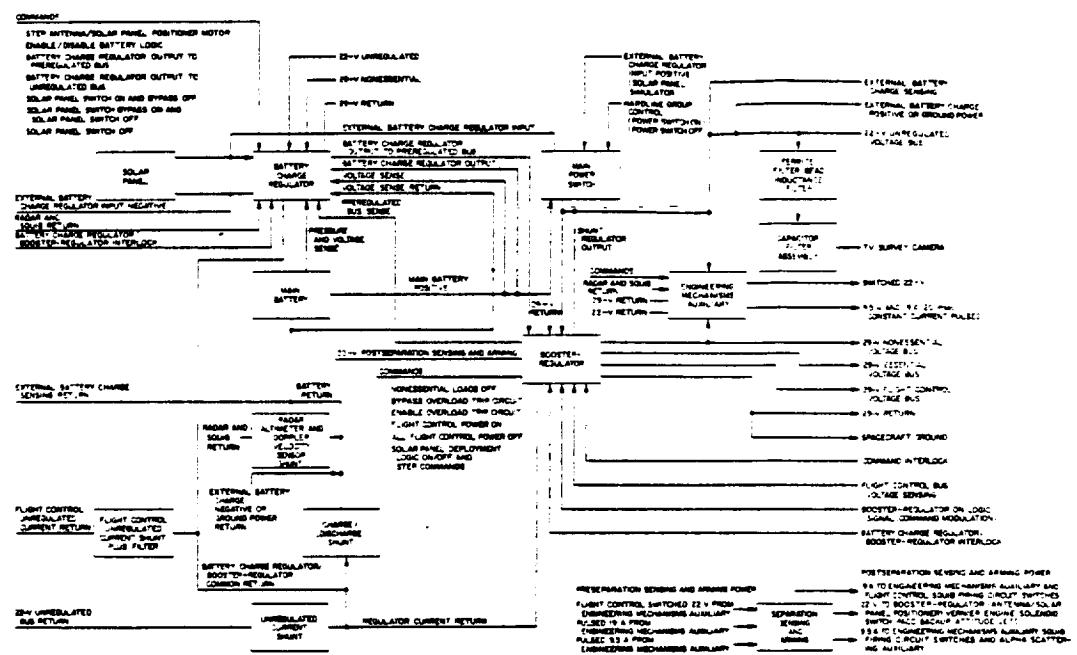
FIGURE 12-2.   Electrical Power System Block Diagram
(Surveyors V through VII)

13. VIKING PROJECT

The principle objective of the Viking Project was to send two vehicles, each consisting of an Orbiter System and a Lander System, to Mars to perform scientific experiments to enlarge our present knowledge about the physical characteristics of Mars, especially its ability for supporting life [1]. Viking 1 and 2 were launched in August and September, 1975, respectively.

The design of the Viking Orbiter was essentially based on the Mariner spacecraft series. The design philosophy was to scale-up the Mariner 9 spacecraft. There were differences between the two spacecraft because the Orbiter was designed to perform more complex tasks that were not required by Mariner 9. The physical size of the Orbiter was greatly influenced by the size of the propellant tanks because the Orbiter and Lander had to be decelerated in order to be captured by the gravitational field of Mars.

The function of the Orbiter System was to transport and deploy the Lander System to a selected landing site on Mars. After achieving a Mars orbit, the Viking mission depended on reliable, careful communications among the Orbiter, Lander and Earth with a one-way transmission up to 20 minutes between the Orbiter/Lander and Earth. The Orbiter transmitted data to and received instructions from Earth. Likewise, the Lander transmitted data and received commands from Earth and transmitted data daily to the Orbiter.

The instruments on the Orbiter measured atmospheric and surface parameters as a function of position and time in order to determine the dynamic characteristics of Mars. With knowledge gained from the topography of the planet, the proposed landing site was investigated prior to deorbit of the Lander System.

During descent and after landing, the instruments on the Lander measured the atmospheric composition, temperature, pressure, and density profiles as a function of height above the surface of Mars. After landing, the landing site was mapped and the planet's surface composition, temperature, pressure, humidity, and wind speed were measured.

A. Power System of the Viking Orbiter

Orbiter power was provided by four solar panels that supplied 620 watts of power at Mars [2]. During the periods when the peak load demand exceeded the power supplied by the solar panels or when the solar panels were not facing the Sun, which occurred during the braking maneuver at Mars, the power difference was provided by two 30 ampere-hour nickel-cadmium storage batteries.

A block diagram of the Viking Orbiter power system is shown in Figure 13-1 along with the voltage and current levels or power ratings of the various subassemblies [3]. The Orbiter Power system is divided into three subsystems as follows:

- Power Source

    Four panel solar array
    Array Zener diodes
    Array blocking diodes

- Energy Storage

    Orbiter batteries
    Battery chargers
    Boost converter
    Share mode detector
    Battery blocking diodes
    Battery test loads

. Power Conditioning and Distribution

> Boost regulator
> 2400 Hz inverter
> 3-phase, 400 Hz inverter
> 30 volt dc converter
> Power control
> Power distribution

B. Power System of Viking Lander

Because the amount of sunlight at Mars is approximately one-half that at Earth and the sunlight vanishes during the cold Martian night (night temperature may reach -120°C), a radioisotope thermoelectric generator (RTG) was used on the Lander. It provided a long-lived source of electrical and heat energy where the heat energy was conveyed through a thermal switch to a temperature-controlled instrument compartment.

An electrical block diagram of the Viking Lander power system is shown in Figure 13-2 [4]. The electrical system consisted of two series-connected 35 watt RTGs, power conditioning and distribution unit, shunt regulators, and four 24-cell, 8 ampere-hour nickel cadmium batteries. Originally, silver-zinc batteries were chosen for the 1973 mission. They had a cell life of 14 to 18 months and were capable of 200 charge-discharge cycles [5]. The launch date was changed from 1973 to 1975 causing the cruise period to be extended from 7 to 12 months because of the position of Mars relative to Earth. This constraint increased the risk factor because it was not certain that the silver-zinc battery could meet all the specifications with an extended life of 5 months. Hence, nickel cadmium batteries were selected for the 1975 mission.

The batteries on the Lander were designed to be heat sterilizable because

the Lander had to be biologically clean in order not to contaminate the surface of Mars. The nickel-cadmium cells were qualified to withstand up to 200 hours of heat at 125°C in a discharged open circuit state [6].

The power control and distribution unit provided the interface between the batteries and the bioshield power assembly which in turn provided the power interface between the Lander and Orbiter [7]. The major functions of the power control and distribution assembly were:

- Processing RTG power for Lander loads and battery charging

- Sense and switch batteries as a function of battery voltage and temperature for purposes of charging

- Power transfer from Orbiter (external) to Lander (internal) power

- Provide load switching under the direction of the Guidance Control and Sequencing Computer

- Detect faults and overloads for selected Lander electrical loads

- Undervoltage sensing and protection

- Provide a sequencer to take appropriate switching steps in case of measuring an undervoltage or a Guidance Control and Sequencing Computer failure or overload

The redundant battery chargers, which were part of the power control and distribution assembly, were a 34.8 volt constant voltage design. The power supplied by the RTGs varied from 70 to 85 watts.

The equipment bus voltage was maintained between 35.25 and 37 volts dc by redundant shunt regulators which were comprised of four differential amplifier circuits that monitor the voltage difference between the bus voltage and independent zener diode references. Each differential amplifier excited one of four quad redundant power transistors. When in the "on" mode, the power

transistors diverted the excess power to resistor load banks that were mounted on the Lander legs.

In order to prevent damage to the batteries as well as optimize the utilization of power, the charge control, which was part of the power control and distribution assembly electronics, monitored the bus voltage, battery temperature, and set the state of the charge enable and discharge enable relays. Charge control failure, which could abort the mission, was circumvented by using redundancy. During the cruise segment of Lander 2, one of the redundant battery chargers failed requiring another redundant charger to be switched into the system. After this encounter, one battery on each Lander was maintained in a full-charge-state in order to assure an electrical source of power on the Lander when the Lander was separated from the Orbiter.

The RTG converted thermal energy directly to electrical power using Plutonium-238 fuel and a thermoelectric couple array. The nominal thermal power of the RTG was equivalent to 682 watts and had an electrical power output of 35 watts. When the RTG was in the "on" mode, the power control and distribution assembly maintained the RTG output voltage at 4.4 $\pm$ 0.1 volt dc.

Some of the excess thermal energy from the RTGs was used to maintain the temperature of the Lander. The heat flow was controlled by a thermal switch which, when closed, directed the heat from the RTG to the Lander body; in the open position, the heat flowed to a thermal panel.

In each Lander, the electrical energy was stored in two battery units each containing two 24 series connected nickel-cadmium cells rated at 8 ampere-hours. Prior to launch, the Lander batteries were heat sterilized for 54 hours at 112°C. The 8 ampere-hour capacity of the four batteries was

selected based on a maximum depth of discharge of 75% during insertion which was an overestimate. Data from both Landers indicated only a 46 to 50% depth of discharge.

Because of the failure of one of the redundant chargers in Lander 2, a combination of three batteries was connected to the equipment bus while the fourth battery was charged.

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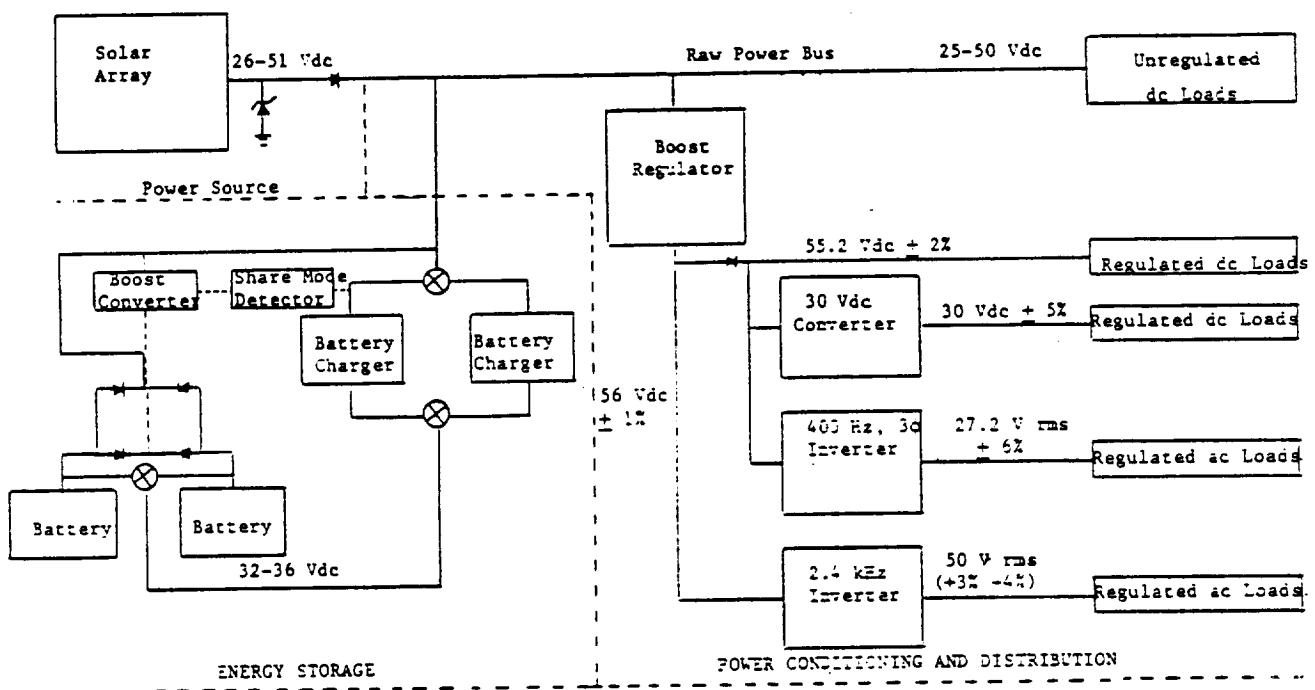
FIGURE 13-1. Viking Orbiter Electrical Power System
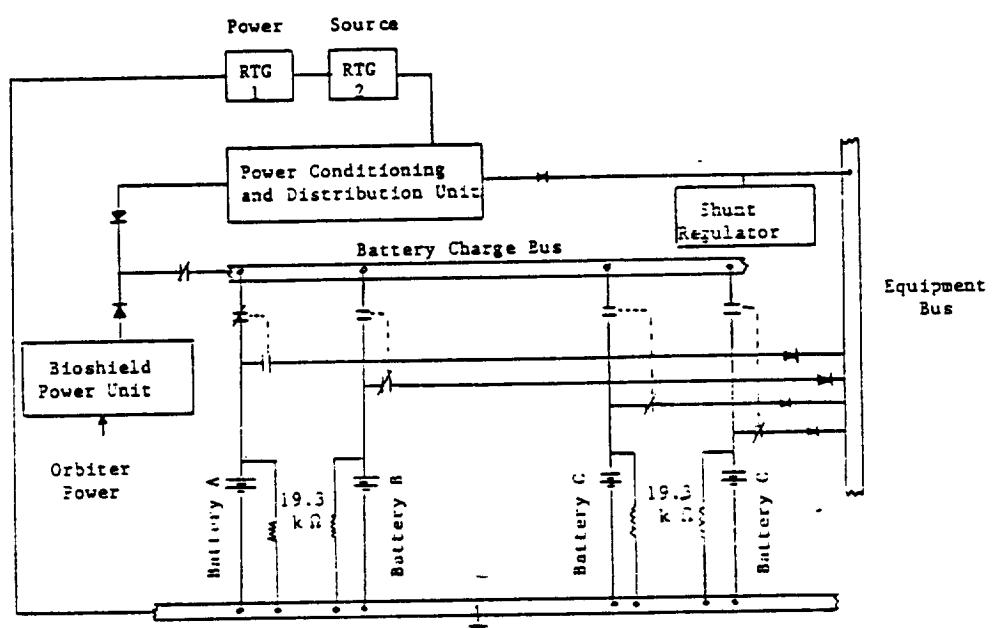
FIGURE 13-2. Viking Lander Electrical Power System

14. VOYAGER MISSION

Initially this mission was named Mariner/Jupiter-Saturn. Because the Grand Tour, exploration of the outer three planets—Uranus, Neptune, and Pluto—was cancelled due to budgetary reasons, the mission title was changed to Voyager in 1977.

Two identical spacecraft, Voyager 1 and 2, were launched in 1977 to perform a similar study of the giant planets of the outer solar system, namely Jupiter and Saturn. After successfully encountering the Jovian systems in 1979 and using the gravity-assist boost of Jupiter, Voyagers 1 and 2 reached Saturn in late 1980 and August 1981, respectively. Using Saturnian gravity-assist boost, Voyager 2 should reach Uranus in 1986 [1].

Comparing the Voyager spacecraft with the spacecraft used in the Pioneer Missions, the Voyager spacecrafts were more independent of ground commands from Earth. This autonomy was important because of the large distances, between the giant planets and Earth. Any ground command instruction to correct a malfunction in the spacecraft would take hours. For example, it takes eighty minutes for a radio signal to travel round trip between Earth and Jupiter. With Saturn about twice as far as Jupiter and Uranus about twice as far as Saturn, the communication link, timewise, would become unwieldy.

Each Voyager spacecraft was designed to perform eleven science investigations. See Table 14.1 [1].

The objectives of Voyager 1 were to investigate:

- Jupiter

     large satellites of Jupiter

          Io

          Ganymede

          Callisto

     small satellite of Jupiter

          Amalthea

- Saturn

     its rings

     several satellites including Titan

The objectives of Voyager 2 were to investigate:

- Jupiter

     Europa

     Ganymede

     Callisto

- Saturn

     several of its satellites

     encounter with Uranian system

Voyager Spacecraft Power System

The design of the spacecraft electrical power system was based on a primary mission lifetime of five years with the ability to be extended.

An electrical system block diagram of the Voyager is shown in Figure 14-1 [2]. The voltage output of the three radioisotope thermoelectric generators (RTGs) was regulated to 30 volts dc by a shunt regulator. The 30 volt dc

source supplied directly the rf subsystem, some of the heaters and a 2.4 kHz, 50-volt inverter. The inverter output supplied almost all the engineering and science subsystems. Since the lifetime of the mission was approximately five years and mass constraint on long missions was tight, batteries were replaced by a charge capacitor energy storage to supply energy during short duration transient overloads of approximately 7 joules and less than 4 amperes.

Each RTG contains 2400 W_t plutonium dioxide heat source and 312 SiGe unicouples which were connected in a series-parallel ladder configuration. The shunt regulator regulated the output of the RTG at 30 volts dc which was the RTG's maximum power operating voltage.

The Power Conditioning Unit consists of the following items:

- Power Control
- Power Distribution
- Shunt Regulator
- Discharge Controllers
- Inverters

The power control had triple-redundant undervoltage detectors on both the dc and ac buses which would disconnect faulty noncritical loads if either of the bus voltages were out of tolerance. Isolation diodes were switched in series with the three RTGs by the power control unit in case there was an undervoltage due to a short within the unicouples. The power control unit was also switched, after a reasonable time delay, from the main to standby inverter if there were an undervoltage on the 2.4 kHz bus.

The distribution of dc and 2.4 kHz power was controlled by the power distribution subsystem via binary coded commands. Single or dual relays were used depending on the criticality of the particular load. For critical loads,

the dual relays, wired as three-way switches, furnished the required redundancy.

Majority-voted error amplifiers and majority quad shunt stages constituted the shunt regulator. The shunt stages were excited sequentially in order to maintain 30 volt bus voltage. When the shunt stage reached its designed capacity of 120 watts, another stage was activated. A total of 480 watts could be controlled by the four shunt stages. Four stages versus a single stage distributed the power dissipation more evenly within the electronic bay. Eventually, this heat was radiated into space by a shunt radiator mounted externally to the spacecraft. Controlling the rate of radiated power controlled the temperature of the spacecraft.

A bank of charged capacitors constituted the main component of the discharge controller. The capacitor bank supplied power to the electrical system when transient demands on the regulated dc bus exceeded the RTG output. The shunt regulator and discharge controller complemented each other to maintain the 30 volt dc bus within the regulation specifications. The combination of the two subsystems allowed a current up to four amperes in excess of the RTG capability and for a maximum total energy of 7 joules at 25°C.

Each single phase inverter was rated at 250 watts and provided a 50-volt rms, 2.4 kHz regulated square wave to the science and engineering subsystems. A malfunction in the main inverters caused the redundant inverter to be switched on by the power control unit.

Information transmitted to Earth indicated that the three RTGs power

levels on both spacecraft were exceeding the prelaunch predictions. The RTG power output on Voyager 1 was 470 watts just after launch and decreased to 430 watts at Saturn (3.2 years later).

As of 1981 on Voyager 1 and 2, the power systems had performed without failure, a span of 3.5 years. Power margin predictions were accurate.

Several faults to chassis in user loads were determined during system level testing of the bus voltage balance with respect to chassis. This type of fault in one of the instruments on Voyager 2 might be the reason the receiver was inoperative on that spacecraft.

TABLE 14.1 SCIENCE EXPERIMENTS ON VOYAGER MISSIONS

- Spectrum: visual, infrared, and ultraviolet

- Remote Sensing Studies: planets and satellites

- Studies of radio emissions, magnetic fields, cosmic rays, and low energy particles

- Studies using the spacecraft radios

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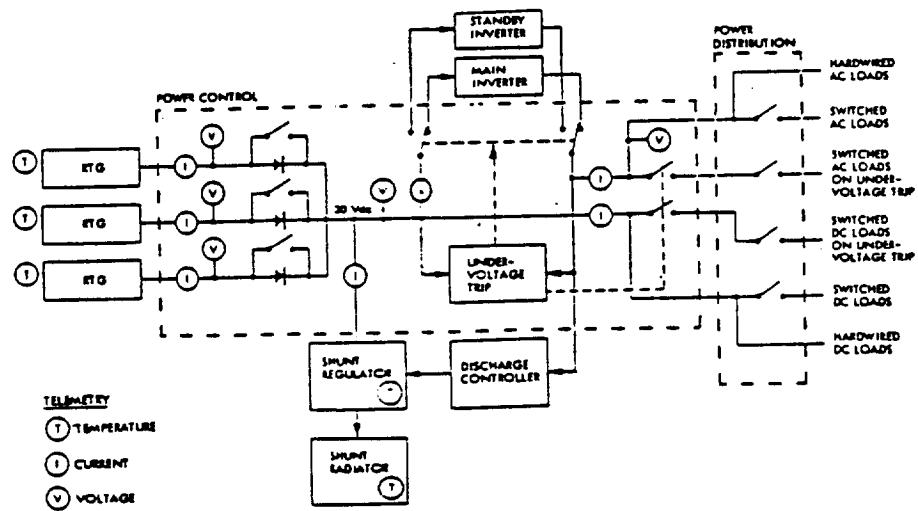
FIGURE 14-1. Voyager Electrical Power System

15. SUMMARY

The electrical power systems of fourteen U.S. manned and unmanned missions or projects have been studied in this investigation. Obviously, not all U.S. power systems used in the space programs could be investigated because this would involve hundreds of unmanned missions. The unmanned space missions in this investigation were chosen because of the importance of their electrical power systems to the manned space program. This is not to say, however, that other U.S. unmanned missions did not contribute to the overall manned space program. Each space mission expanded our knowledge of the universe and permitted technological barriers to be crossed.

In order to bring the results of this study into focus, a tabular summary is presented at the end of this section of the report. Details of the electrical power system structure of U.S. manned and unmanned spacecraft are presented elsewhere in this report by program title. The summary lists the U.S. manned space program first because it will probably have the greatest impact on the design of future space stations.

Project Mercury had silver-zinc batteries that were non-rechargeable as the prime source of power. This configuration was very expedient at a time when it was of paramount importance to have a spacecraft reach and maintain an orbital path. Because the flight time duration was short and the power demand was low, it was not necessary to have a secondary source of power on board the spacecraft. Conversion from dc to ac was necessary in order to operate the gyro motors and other motors aboard the spacecraft.

In the early stages of Project Gemini, the main power source was silver-zinc batteries because fuel cells were not reliable at that time. In later missions, however, fuel cells replaced the batteries allowing the flight duration to be increased from hours to days. Besides providing

-15.1-

electrical power, the fuel cells supplied in the form of a by-product the
required water needed to sustain the crew during the mission.

Fuel cells were the main source of electrical energy for the Apollo
Program with silver oxide-zinc batteries serving as the secondary power
source. Again, inverters were necessary for electrical motors in the
spacecraft. The combination of fuel cells and batteries on the Apollo
spacecraft provided an ample amount of electrical power for the round trip to
the Moon.

Projects Mercury and Gemini with the Apollo Program were the
underpinnings for the Skylab Project. This spacecraft used the fuel cell
technology from the two previous programs and solar arrays for the main source
of electrical power with nickel-cadmium batteries serving as the secondary
power source. This combination of electrical sources allowed for extended
flight duration, well over 150 days.

Finally, the Space Shuttle Program plus Spacelab should move the progress
of electrical power space technology one step closer to a space station.
Because the Space Shuttle was designed to have long flight durations, fuel
cells were the main source of power.

Two of the three fuel cells on board the Shuttle had sufficient power
capacity to supply all the required power. The third fuel cell will supply dc
power to Spacelab. Using a set of inverters, Spacelab provides 220 and 115
volts at 50 and 60 hertz, respectively, as well as 115 volts at 400 hertz.

The culmination of the manned space program may not have moved as rapidly
in time if it were not for the parallel unmanned space program. These
spacecraft, acting as space probes, permitted measurements to be conducted on
the kind of environment that the astronauts would face in the manned program.
Since human life was not a primary concern in the design of an unmanned

spacecraft, such items as life support during flight were not

necessary. Also, the power level demand in the unmanned spacecraft was

considerably lower as compared to a manned spacecraft.

The Pioneer Missions could be divided into two destinations: outer

planetary, such as Jupiter/Saturn, and Venus Missions. Each had different

primary power sources. For the case of the outer planetary missions, a

radioisotope thermoelectric generator was used because the amount of sunlight

at Jupiter and Saturn was far less than near Earth. Therefore, solar arrays

would be too large to supply the necessary power. Besides supplying

electrical power, the radioisotope thermoelectric generator also provided a

source of heat to control the temperature of the spacecraft.

The main source of power in the Mariner Venus Program was the solar array

with silver-zinc or nickel-cadmium batteries as the secondary source depending

on the particular Mariner Mission. Also, the number of solar array panels

varied. For example, Mariner 9 and 10 had 4 and 2 panels, respectively.

Ranger, Lunar Orbiter, and Surveyor Projects formed a group of missions

to investigate the environment and surface characteristics of the Moon. These

three missions laid the foundation for the Apollo Program. Because of the

ample amount of sunlight near the Moon, solar arrays were used in all three

missions with the secondary source being silver-zinc and nickel-cadmium

batteries for the Ranger/Surveyor Projects and Lunar Orbiter Program,

respectively.

The spacecraft used in the Viking Project consisted of the Viking Orbiter

and Lander. The main and secondary power sources for the Orbiter were,

respectively, solar panels and nickel-cadmium batteries. The main source of

power for the Viking Lander was the radioisotope thermoelectric generator

which supplied the necessary electrical and thermal power. Thermal power was necessary to control the temperature of the Lander.

Likewise, the Voyager spacecraft received its main power from a radioisotope thermoelectric source. Instead of using batteries as a secondary source, charged capacitors served as energy storage. This increased the life span of the secondary source almost indefinitely.

TABULAR RESULTS OF THE PROGRAMS STUDIED IN THIS REPORT

Project Mercury

Main Power Source: silver-zinc batteries (non-rechargeable)
D.C. Bus Voltage: 24 volts
Inverters: 115 volt, 400 hertz, single phase, 250 and 150 volt-amperes

Project Gemini

Main Power Source:
    A. silver-zinc batteries (non-rechargeable)
       D.C. Bus Voltage: 22/30 unregulated

    B. fuel cells
       D.C. Bus Voltage: 22/30 unregulated

Apollo Program

Main Power Source: fuel cells
Secondary Power Source: silver-oxide-zinc batteries (rechargeable)
D.C. Bus Voltage: 28 volts (nominal)
Inverters: 115/200 volt, 400 hertz, three-phase, 1250 volt-amperes

Space Shuttle Program

Main Power Source: fuel cells
D.C. Bus Voltage: 28 volts (unregulated)
Inverters: 117 volts at 400 hertz

Spacelab

Main Power Source: one fuel cell from Space Shuttle Orbiter
Secondary Power Source: Peak power battery (flown on request)
D.C. Bus Voltage: 27/32 volts dc (unregulated)
Inverters: 220 volts at 50 hertz, 115 volts at 60 hertz, 115 volts at 400 hertz

Pioneer Missions

A. Pioneer Jupiter/Saturn Mission
   Main Power Source: radioisotope thermoelectric generator
   Secondary Power Source: silver-cadmium batteries (rechargeable)
   D.C. Bus Voltage: 28±2% volts
   Inverters: 30.5 volts, 2500 hertz, trapezoidal waveform

B. Pioneer Venus Mission
   Main Power Source: solar array
   Secondary Power Source: nickel-cadmium batteries rechargeable via a small
                           solar array
   D.C. Bus Voltage: 28±10% volts (semiregulated)

Mariner Program

Main Power Source: solar array
Secondary Power Source: silver-zinc or nickel-cadmium batteries depending on
                        particular mission
D.C. Bus Voltage: 30 or 56 volts depending on particular mission
Inverters: 50-volt, 2400 hertz, single-phase, square-wave
           28-volt, 400 hertz, single-phase
           27.2 volt, 400 hertz, three-phase depending on particular mission

Ranger Project

Main Power Source: solar arrays
Secondary Power Source: silver-zinc batteries
D.C. Bus Voltage: 25.5 volt regulated

Lunar Orbiter Program

Main Power Source: solar arrays
Secondary Power Source: nickel-cadmium batteries rechargeable
D.C. Bus Voltage: 20-volt regulated

Surveyor Project

Main Power Source: solar array
Secondary Power Source: silver-zinc batteries rechargeable
D.C. Bus Voltage: 29 volts±0.29 volts regulated; 17-27.3 unregulated

Viking Project

A. Viking Orbiter
    Main Power Source: solar arrays
    Secondary Power Source: nickel-cadmium batteries
    D.C. Bus Voltages: 55.2±2% regulated
                       25 - 50 unregulated
                       30±5% regulated
    A.C. Bus Voltages: 27.2±6% regulated
                       50+3% or -4% regulated
    Inverters: 27.2-volt, 400 hertz, three-phase, 12 watts
               50-volt, 2400 hertz, single-phase, 350 watts
    Converter: 30-volt, 90 watts

B. Viking Lander
    Main Power Source: radioisotope thermoelectric generator
    Secondary Power Source: silver-zinc batteries
    D.C. Bus Voltage: 35.25 - 37 regulated

Voyager Mission

Main Power Source: radioisotope thermoelectric generators
Secondary Power Source: charge capacitor energy
D.C. Bus Voltage: 30 volts regulated
Inverter: 50-volt, 2400 hertz, regulated square-wave

16. CONCLUDING REMARKS

Results of this investigation on the spacecraft electrical power systems of the U. S. manned and some of the U.S. unmanned programs have shown that the electrical power systems have operated at mostly low voltage (~ 28 volts dc). In general, the 28V dc was supplied by batteries, fuel cells, solar arrays or a combination of them with either close regulation at about ± 2% or coarse regulation at battery voltage of approximately ± 10%. Inverters were used to boost voltage levels primarily in two cases:

1) To provide 115V ac, 400 Hz power for motor loads in certain manned and unmanned spacecraft, and

2) To step up the low output voltage (at 4.2V) of the Radioisotope Thermoelectric Generators to the desired bus voltage levels.

In all cases, however, the spacecraft were designed with reasonably well-defined electrical load demands and for relatively short duration missions. Future space station electrical power systems, on the other hand, will need to evolve and expand with time in order to meet the increased electrical power demands as the station grows in size, complexity and versatility. Reliability, safety and autonomy will be increasingly important issues for the space station operating lifetime that is projected to exceed 10 years.

As the power levels increase beyond the 10-kilowatt level by one and eventually two or more orders of magnitude into the megawatt level, the corresponding bus voltages must also increase. Distribution voltages will rise from the 28V level to the 100 to 400 volt levels in the first generation space station. Requirements for future multimegawatt space platforms

obviously could push distribution voltage requirements above the 400V level.

New technologies are under development by NASA and others to provide the required electrical power system options for the future. New types of semiconductors, advanced distribution components, new inverter/converter topologies, and improved power management and distribution techniques for operation at high voltage, high frequency and higher temperature offer great promise for the realization of the United States' space goals into the next century.

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AUTH: A/KRUG, A. B.   PAN: (AA/CALIFORNIA INST. OF
TECH., JET PROPULSION LAB., PASADENA, CALIF./.)
68A37805   68/00/00

UTTL: Mariner Venus 67 guidance and control system
UNOC: Subsystems of Mariner Venus 67 spacecraft
guidance and control system   AUTH: A/PACE, G.   CORP:
Jet Propulsion Lab., California Inst. of Tech.,
Pasadena.   68N29444   RPT#: NASA-CR-95687
JPL-TR-32-1258   CNT#: NAS7-100   68/07/01

UTTL: Mars spacecraft power system development Final
report   UNOC: Development of optimum Mariner
spacecraft power system for application to future
flyby and orbiter missions   AUTH: A/BERKERY, E.;
B/KIRPICH, A.; C/KUTNER, N.; D/PELLMAN, R.; E/READ,
M.; F/SHEPARD, N.; G/TASCA, D.; H/WEIN, D.   CORP:
General Electric Co., Philadelphia, Pa.; Jet
Propulsion lab., California Inst. of Tech., Pasadena.
CSS: (MISSILE AND SPACE DIV.)   69N11450   RPT#:
NASA-CR-97770 DOC-68SD4305   CNT#: NAS7-100 JPL-952150
69/07/26

UTTL: Mariner-Mars 1969 A preliminary report   UNOC:
Scientific data from Mariners 6 and 7 Mars flyby
missions during 1969   CORP: National Aeronautics and
Space Administration, Washington, D. C.   70N18343
RPT#: NASA-SP-225   69/00/00

UTTL: The Voyage of Mariner 10: Missions to Venus and
Mercury AUTH: A/DUNNE, J. A.; B/BURGESS, E. CORP:
Jet Propulsion Lab., California Inst. of Tech.,
Pasadena. 78N27146 RPT#: NASA-SP-424 LC-77-18956
78/00/00

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UTTL: Mariner-Venus 1962 Final project report   UNOC:
Mariner Venus probe of 1962 - final project report
CORP: Jet Propulsion Lab., California Inst. of Tech..
Pasadena.   66N14702   RPT#: NASA-SP-59   CNT#:
NAS7-100   65/00/00

PRINT 09/4/1-6   TERMINAL=73
UTTL: Lunar Impact: A history of Project Ranger
AUTH: A/HALL, R. C.   CORP: National Aeronautics and
Space Administration. Washington, D. C.   78N15149
RPT#: NASA-SP-4210   77/00/00

UTTL: The Luna and Zond programs   CORP: National
Aeronautics and Space Administration. Goddard Space
Flight Center. Greenbelt, Md.   77N32184   77/07/00

UTTL: The Apollo program   CORP: National Aeronautics
and Space Administration. Goddard Space Flight Center.
Greenbelt, Md.   77N32183   77/07/00

UTTL: The Lunar Orbiter program   CORP: National
Aeronautics and Space Administration. Goddard Space
Flight Center. Greenbelt, Md.   77N32182   77/07/00

UTTL: The Ranger program   CORP: National Aeronautics
and Space Administration. Goddard Space Flight Center.
Greenbelt, Md.   77N32180   77/07/00

UTTL: Catalog of lunar mission data   AUTH: A/MANTEL,
E. J.; B/MILLER, E. R.   PAT: A/ed.; B/ed.   CORP:
National Aeronautics and Space Administration. Goddard
Space Flight Center. Greenbelt, Md.   77N32179   RPT#:
NASA-TM-X-72612 NSSDC/WDC-A-R/S-77-02   77/07/00

PRINT 01/4/1   TERMINAL*73

UTTL: Ranger vii- mission description and
performance, part I   UNOC: System performance
evaluation of Ranger   VII LUNAR probe and analysis of
photographs from mission   AUTH: A/SCHURMEIER, H. M.
CORP: Jet Propulsion Lab..   California Inst. of Tech..
Pasadena.   65N13280   RPT#: NASA-CR-59904
JPL-TR-32-700, PT. I   CNT#: NAS7-100   64/12/15

UTTL: Lunar Orbiter program guidance and control system design survey Final report UNOC: Design criteria for Lunar Orbiter program guidance and control system CORP: Boeing Co., Seattle, Wash. CSS: (AEROSPACE SYSTEMS DIV.) 69N36724 RPT#: NASA-CR-86228 D2-114277-2 CNT#: NAS12-2027 69/08/00

UTTL: Lunar Orbiter Spacecraft Guidance and Control System - An appraisal UNOC: Lunar Orbiter Spacecraft Guidance and Control, discussing design and flight results AUTH: A/MONTGOMERY, J. E. PAN: (AA/BOEING CO., SEATTLE, WASH./.) 70A30455 CNT#: NAS1-22027 70/00/00

UTTL: Lunar Orbiter 4 - Mission system performance UNOC: Lunar Orbiter 4 mission system performance CORP: Boeing Co., Seattle, Wash. 68N10947 RPT#: NASA-CR-66498 D2-100754-3 CNT#: NAS1-3800 67/10/18

UTTL: Lunar Orbiter 5 - Mission system performance UNOC: Performance analysis for Lunar Orbiter 5 mission CORP: Boeing Co., Seattle, Wash. 68N24923 RPT#: NASA-CR-66608 D2-100755-3 CNT#: NAS1-3800 68/01/04

UTTL: Lunar Orbiter 3 - Extended-mission spacecraft operations and subsystem performance UNOC: Spacecraft performance, control, and flight path analysis of Lunar Orbiter 3 extended mission CORP: Boeing Co., Seattle, Wash. 68N29995 RPT#: NASA-CR-1109 D2-100753-4, V. 3 CNT#: NAS1-3800 68/07/00

UTTL: Lunar Orbiter 5. Extended-mission spacecraft operations and subsystem performance UNOC: Control, performance, and tracking Lunar Orbiter 5 in obtaining selenodetic and lunar environment data CORP: Boeing Co., Seattle, Wash. 68N30631 RPT#: NASA-CR-1142 D2-100755-4, V. 4 CNT#: NAS1-3800 68/08/00

UTTL: Lunar Orbiter program guidance and control system - Design survey. NASA/ERC design criteria program - Guidance and control UNOC: Design and tests of Lunar Orbiter guidance and control systems CORP: Boeing Co., Seattle, Wash. CSS: (AEROSPACE SYSTEMS DIV.) 70N27310 RPT#: NASA-CR-86372 D2-114277-3 CNT#: NAS12-2027 69/05/09

UTTL: Photovoltaic power systems on flight spacecraft Lunar Orbiter 3 UNOC: Design and performance data of power subsystem in flight spacecraft Lunar Orbiter 3 AUTH: A/DRAZDAUSKAS, C. J.; B/READ, M. D. CORP: . General Electric Co., Philadelphia, Pa. CSS: (MISSILES AND SPACE DIV.) 69N29374 RPT#: NASA-CR-100700 DOC-69SD4225 CNT#: NAS7-547 69/05/26

PRINT 06/4/1-4   TERMINAL=73

UTTL: Lunar orbiter- its mission and capability
UNOC: Mission and capabilities of proposed lunar
spacecraft - Lunar Orbiter   AUTH: A/TABACK. I.
CORP: National Aeronautics and Space Administration.
Langley Research Center, Hampton, Va.   65N35236.
RPT#: NASA-TM-X-51596   64/00/00

UTTL: Second Orbiter launch scheduled in Nov. 6-11
period   UNOC: Lunar orbiter B spacecraft. spacecraft
configuration, tasks. Atlas Agena D launch vehicle.
deep space network. mission. and flight events   CORP:
NE368373   66N39950   RPT#: NASA NEWS RELEASE-66-286
66/10/31

UTTL: Lunar Orbiter Mission I.   Volume III - Mission
operational performance   Final report   UNOC: Systems
performance analyses. launch vehicle configuration.
and launch and flight operations of Lunar Orbiter 1
CORP: Boeing Co., Seattle. Wash.   CSS: (AEROSPACE
GROUP.) 67N27419   RPT#: NASA-CR-66324 D2-100727-3
CNT#: NAS1-3800   67/02/03

UTTL: Lunar Orbiter III.   Mission system performance
Final report   UNOC: Performance of photographic.
communications. power. attitude control. and velocity
control subsystems of Lunar Orbiter III   CORP: Boeing
Co., Seattle. Wash.   67N37275   RPT#: NASA-CR-66461
D2-100753-3   CNT#: NAS1-3800   67/08/11

PRINT 06/4/1-4   TERMINAL=73

UTTL: Surveyor 7 Preliminary report   UNOC: Principal scientific results on Surveyor 7 mission   CORP: National Aeronautics and Space Administration. Washington, D. C.   68N33553   RPT#: NASA-SP-173   68/05/00

UTTL: Surveyor program results   Final report   UNOC: Summary of lunar data and scientific observations generated by Surveyor landings on Moon   CORP: National Aeronautics and Space Administration. Washington, D. C.   69N36451   RPT#: NASA-SP-184   69/00/00

UTTL: Surveyor project. Part 1 - Project description and performance. volume 2   Final report   UNOC: Designs of systems for Surveyor spacecraft   CORP: Jet Propulsion Lab., California Inst. of Tech., Pasadena. 69N35753   RPT#: NASA-CR-105302 JPL-TR-32-1265   CNT#: NAS7-100   69/07/01

UTTL: Unmanned space project management: Surveyor and lunar orbiter   UNOC: Analysis of management procedures applied to Surveyor and Lunar Orbiter projects   AUTH: A/KLOMAN. E. H.   CORP: National Academy of Public Administration. Washington, D. C. 73N13973   RPT#: NASA-SP-4901 LC-72-600225   72/00/00

PRINT 04/4/1-2   TERMINAL=73
UTTL: Surveyor spacecraft A-21A model description
UNOC: Surveyor spacecraft A-21A design and performance
CORP: Hughes Aircraft Co., El Segundo, Calif.; Jet
Propulsion Lab., California Inst. of Tech., Pasadena.
CSS: (SPACE SYSTEMS DIV.)   67N39240   RPT#:
NASA-CR-04186 REPT.-224040   CN1#: NAS7-100 JPL-950056
64/09/01

UTTL: Surveyor I - A preliminary report   UNOC:
Surveyor I preliminary data on lunar surface
mechanical, thermal, and electrical properties.
topography, and geology   CORP: NE36B373   66N29481
RPT#: NASA-SP-126   66/06/00

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UTTL: Impactable power subsystems for Mars landers.
UNOC: Mars landers impactable power subsystems.
considering thermoelectric generators, batteries,
conversion equipment, Mars environment, etc   AUTH:
A/SWERDLING, M.   PAN: (AA/CALIFORNIA INST. OF TECH..
JET PROPULSION LAB., PASADENA, CALIF./.)   69A42253
69/00/00

UTTL: Space programs summary no. 37-56, volume 1 for
the period 1 January to 20 February 1969. Flight
projects   UNOC: Mariner Mars 1969 project. Mariner
Mars 1971 Project, and Viking Project descriptions
CORP: Jet Propulsion Lab., California Inst. of Tech..
Pasadena.   70N19163   RPT#: NASA-CR-108093
JPL-SPS-37-56-VOL-1   CNT#: NAS7-100   69/03/31

UTTL: Space programs summary no. 37-61, volume 1 for
the period 1 November - 31 December 1969. Flight
projects   UNOC: Mariner Mars 1969 project. Mariner
Mars 1971 project, and Viking 1973 project - research
and advancement development   CORP: Jet Propulsion
Lab., California Inst. of Tech.. Pasadena.   70N26078
RPT#: NASA-CR-109605 JPL-SPS-37-61-VOL-1   CNT#:
NAS7-100   70/01/31

UTTL: Viking project, orbiter system and project
support   UNOC: Viking Orbiter 1975 RF and relay
systems, attitude control, Mars atmospheric water
detection spectrometer   CORP: Jet Propulsion Lab..
California Inst. of Tech.. Pasadena.   70N36382
70/05/31

UTTL: Space programs summary no. 37-63, volume 1 for
the period 1 March - 30 April 1970. Flight projects
UNOC: Mariner Mars 1971, Mariner Venus-Mercury 1973
and Viking Orbiter 1975 status reports   CORP: Jet
Propulsion Lab., California Inst. of Tech.. Pasadena.
70N36379   RPT#: NASA-CR-112336 JPL-SPS-37-63-VOL-1
CNT#: NAS7-100   70/05/31

UTTL: The Viking Lander Power System   UNOC: Viking
Lander power system design, discussing functional
requirements by mission and science objectives and
reliability features   AUTH: A/FARLEY, K.; B/SEALS,
D. R.; C/SMITH, R. D.   PAN: (AC/NASA, LANGLEY
RESEARCH CENTER, HAMPTON, VA./. AA/MARTIN MARIETTA
CORP., FRIENDSHIP INTERNATIONAL AIRPORT, MD./.)
71A30940   71/00/00

UTTL: Experience applicable to the Viking lander from
a study of related space flight projects   UNOC:
Related space mission data for baseline evaluation of
Viking Lander design and operation   AUTH:
A/BUTTERFIELD, A. J.; B/HERZ, J. N.   CORP: General
Electric Co.. Hampton, Va.   71N33565   RPT#:
NASA-CR-111076   CNT#: NAS1-9100   71/03/25

UTTL: The Viking mission to Mars   CORP: National
Aeronautics and Space Administration. Langley Research
Center. Hampton, Va.   74N34207   RPT#: NASA-SP-334
74/00/00

UTTL: Viking 1:   early results   --- Mars atmosphere
and surface examinations   CORP: National Aeronautics
and Space Administration. Washington, D. C.   76N28296
RPT#: NASA-SP-408   76/08/21
76/00/00

UTTL: Viking lander spacecraft battery   TISP: Final
Report   AUTH: A/NEWELL, D. R.   CORP: Martin Marietta
Corp.. Denver, Colo.; Jet Propulsion Lab., California
Inst. of Tech.. Pasadena.   77N20557   RPT#:
NASA-CR-152618 MCR-76-321   CNT#: NAS7-100 JPL-649000
76/09/00

UTTL: Synopsis of Viking flight data   AUTH:
A/BRITTING, A. O.   CORP: Jet Propulsion Lab..
California Inst. of Tech.. Pasadena.   79N28692
78/00/00

UTTL: Viking flight data after 24 months on Mars
surface   AUTH: A/LEAR, J.   CORP: Martin Marietta
Corp.. Denver, Colo.   79N28691   78/00/00

UTTL: Automated management of power systems   ... for
future planetary spacecraft   AUTH: A/IMAMURA, M. S.;
B/BRIDGEFORTH, A.   PAA: A/(Martin Marietta Aerospace.
Denver, Colo.); B/(California Institute of
Technology, Jet Propulsion Laboratory, Pasadena.
Calif.)   CORP: Martin Marietta Aerospace. Denver.
Colo.; Jet Propulsion Lab., California Inst. of
Tech.. Pasadena.   79A34769   RPT#: AIAA 79-0886
79/00/00

UTTL: Viking orbiter views of Mars   AUTH: A/CARR, M.
H.; D/BAUM, W. A.; C/BLASIUS, K. R.; D/BRIGGS, G.
A.; F/CUTTS, J. A.; F/DUXBURY, T. C.; G/GREELEY, R.
; H/GUEST, J.; I/MASURSKY, H.; J/SMITH, B. A.
CORP: National Aeronautics and Space Administration.
Washington, D. C.   81N11966   RPT#: NASA-SP-441
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UTTL: Viking lander five year summary   AUTH:
A/BRITTING, A.   CORP: Jet Propulsion Lab., California
Inst. of Tech., Pasadena.   81N21524   81/03/00

UTTL: A new approach to implementing spacecraft power
system autonomy   AUTH: A/BRIDGEFORTH, A. O.   PAA:
A/(California Institute of Technology, Jet Propulsion
Laboratory, Pasadena, CA)   CORP: Jet Propulsion Lab.,
California Inst. of Tech., Pasadena.   84A10057
RPT#: AIAA PAPER 83-2419   83/00/00

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UTTL: Summary, volume 1   Final report   UNOC: Voyager
project study summaries on spacecraft powered by
radioisotope thermoelectric generators, planetary
quarantine, redundancy, and data management and
control   AUTH: A/OLEARY, K. C.   CORP: General
Electric Co., Philadelphia, Pa.;   Jet Propulsion Lab..
California Inst. of Tech.. Pasadena.   CSS: (MISSILE
AND SPACE DIV.)   68N10370   RPT#: NASA-CR-89994
VOY-CO-FR, V. 1   CNT#: NAS7-100 JPL-951112   67/07/28

UTTL: Voyager spacecraft system study phase B, task D.
Volume 3 - Spacecraft functional description   Final
technical report, 16 Jun. - 16 Oct. 1967   UNOC:
Standard trajectories and parameters for design of
Voyager spacecraft, and performance prediction.
Interfaces, and reliability of spacecraft subsystems
CORP: Boeing Co., Seattle, Wash.   CSS: (SPACE DIV.)
68N19097   RPT#: NASA-CR-93559 D2-115002-3, V. 3
CNT#: NAS8-22602   67/10/00

UTTL: Voyager spacecraft phase B, task D. Volume 2 -
System description.   Book 3 - Guidance and control,
computer and sequencer, power subsystem, electrical
system   Final report   UNOC: Baseline design and
functional requirements of Voyager spacecraft guidance
and control, computer and sequencer, and power
subsystems, and electrical system   CORP: General
Electric Co., Philadelphia, Pa.   CSS: (MISSILE AND
SPACE DIV.)   68N19103   RPT#: NASA-CR-93539
DIN-67SD4379, V. 2, BK. 3   CNT#: NAS8-22603
67/10/16

UTTL: Voyage to Jupiter   AUTH: A/MORRISON, D.;
B/SANZ, J.   CORP: National Aeronautics and Space
Administration. Washington, D. C.   81N10910   RPT#:
NASA-SP-439 LC-80-600126   80/00/00

UTTL: Flight performance of the Voyager electrical
power system   AUTH: A/PACKARD, L. A.   PAA:
A/(California Institute of Technology, Jet Propulsion
Laboratory, Pasadena, CA)   CORP: Jet Propulsion Lab..
California Inst. of Tech.. Pasadena.   82A11771
CNT#: NAS7-100   81/00/00

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UTTL: Voyager capsule phase B. Volume III - Surface
laboratory system. Part C1 - Subsystem functional
description Final report UNOC: Voyager capsule
surface laboratory system - subsystem functional
descriptions CORP: Jet Propulsion Lab.. California
Inst. of Tech.. Pasadena.: McDonnell Aircraft Co..
St. Louis, Mo. 67N40585 RPT#: NASA-CR-89695 F694.
VOL. III, PT. C1 CNT#: NAS7-100 JPL-952000
67/08/31

UTTL: SNAP 19, phase 3, Volume 1. Power supply system Final report UNOC: Performance and reliability testing of power supply system for SNAP 19 with interfaces for Nimbus B CORP: Martin Co., Baltimore, Md. CSS: (NUCLEAR DIV.) 68N38006 RPT#: MND-3607-239-1, V. 1 CNT#: AT/30-1/-3607 68/05/00

UTTL: Spacecraft power UNOC: Spacecraft power research and development CORP: Jet Propulsion Lab., California Inst. of Tech., Pasadena. 68N37401 68/06/30

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UTTL: Proceedings of the sixth ESRO summer school. Volume 6 - Space power systems - Introduction UNOC: Space power systems lectures on sources and requirements CORP: European Space Agency, Paris (France). 70N11301 RPT#: ESRO-SP-45 69/07/00

UTTL: Power systems in ESRO satellites UNOC: Design of ESRO 1, ESRO 2, and HEOS A power systems and control equipment AUTH: A/PREUKSCHAT, A. W. CORP: European Space Research and Technology Center, Noordwijk (Netherlands). 70N17621 RPT#: ESRO-TN-83 69/07/00

UTTL: Space power systems, part 2 lecture series UNOC: Turbomachinery, thermoelectric and thermolonic energy converters, and electrochemical and photovoltaic devices for spacecraft power supplies - conference CORP: Advisory Group for Aerospace Research and Development, Paris (France). 70N16222 RPT#: AGARDOGRAPH-123-PT-2 69/11/00

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UTTL: Electrical power system configuration study AUTH: A/MAIORANA, V.; B/SIEGEL, H. CORP: Grumman Aircraft Engineering Corp., Bethpage, N. Y. 79N76575 RPT#: NASA-CR-117581 LED-390-2 CNT#: NAS9-1100 63/04/01

UTTL: Space programs summary no. 37-47, volume 1 for the period July 1 to August 31, 1967. Flight projects UNOC: Engineering and scientific work on Mariner 4, Mariner Venus, Mariner Mars 1969, Voyager, and Surveyor projects CORP: Jet Propulsion Lab., California Inst. of Tech., Pasadena. 68N11643 RPT#: NASA-CR-90544 JPL-SPS-37-47, V. 1 CNT#: NAS7-100 67/09/30

UTTL: Space programs summary no. 37-48, volume 1, for the period September 1 to October 31, 1967. Flight projects UNOC: Mariner, Mariner, and Voyager planetary-interplanetary flight projects, and Surveyor lunar program CORP: Jet Propulsion Lab., California Inst. of Tech., Pasadena. 68N15004 RPT#: NASA-CR-91775 JPL-SPS-37-48, V. 1 CNT#: NAS7-100 67/11/30

UTTL: Energy in space - Program planning for space power system technology. UNOC: Space power generation systems converting heat or light energy to electricity, noting energy storage AUTH: A/WOODWARD, W. H. PAN: (AA/NASA, WASHINGTON, D.C./.) 68A40071 68/00/00

UTTL: Nuclear power supplies for space. UNOC: Nuclear power supplies for space noting trends. potential and SNAP systems AUTH: A/POLAK, H. PAN: (AA/NORTH AMERICAN ROCKWELL CORP.; NORTH AMERICAN AVIATION INTERNATIONAL INC., BRUSSELS, BELGIUM/.) 68A37252 68/00/00

UTTL: Research at the Stanford Center for Radar Astronomy Semiannual status report, 1 Jul. - 31 Dec. 1967 UNOC: Mariner 5 space probe radio occultation analysis, Pioneer 8 space probe data, and bistatic radar data reduction from Lunar Orbiter and Mariner 5 AUTH: A/ESHLEMAN, V. R. CORP: Stanford Univ., Calif. CSS: (RADIOSCIENCE LAB.) 68N16085 RPT#: NASA-CR-92563 SASR-10 CNT#: NSG-377 68/01/00

UTTL: Astrionics system designers handbook, volume 1
CORP: International Business Machines Corp.,
Huntsville, Ala.   CSS: (Federal Systems Div.)
74N27365   RPT#: NASA-CR-120229 IBM-73W-00315-VOL-1
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UTTL: Outlook for space   TLSP: Report to the NASA
Administrator by the Outlook for Space Study Group
CORP: National Aeronautics and Space Administration.
Washington, D. C.   76N18004   RPT#: NASA-SP-386
76/01/00

UTTL: Spacecraft power conditioning   --- conference
proceedings   AUTH: A/HOGSHOLM, A.;   B/GUYENNE, T. D.
PAT: A/ed.; B/ed.   CORP: European Space Agency.
Paris (France).   78N15097   RPT#: ESA-SP-126
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UTTL: Advanced space power requirements and
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requirements.   Volume 1:   Technical report   AUTH:
A/WOLFE, M. G.   CORP: Aerospace Corp., El Segundo,
Calif.   CSS: (Systems Engineering Operations.)
78N28597   RPT#: NASA-CR-157344 ATR-78(7667)-1-VOL-1
CNT#: NASW-3078   78/03/01

UTTL: Advanced space power requirements and
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requirements.   Volume 3:   Appendices   --- cost
estimates and computer programs   AUTH: A/WOLFE, M. G.
CORP: Aerospace Corp., El Segundo, Calif.   CSS: (
Advanced Orbital Systems Div.)   79N20147   RPT#:
NASA-CR-157266 ATR-78(7667)-1-VOL-3   CNT#: NASW-3078
78/03/01

UTTL: Future Orbital Power Systems Technology
Requirements   CORP: National Aeronautics and Space
Administration. Lewis Research Center. Cleveland,
Ohio.   79N10122   RPT#: NASA-CP-2058 E-9713
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UTTL: Effects of long life requirements on spacecraft
design and technology   TLSP: Final Report.   AUTH:
A/COUGNET, C.; B/FLOCH, C.; C/ARNOULT, J. F.;
D/BERGER, G.; E/DUBOIS, Y.; F/HORBLIN, M.;
G/SCHIETECATTE, B.; H/VIALET, C.   CORP: MATRA
Espace, Paris-Velizy (France).   83N30512   RPT#:
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