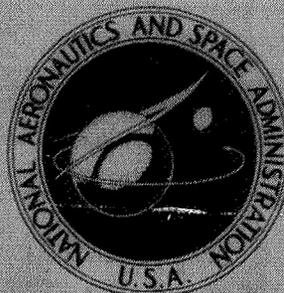


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**SERT II SPACECRAFT  
ELECTRICAL POWER SYSTEM**

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# SERT II SPACECRAFT ELECTRICAL POWER SYSTEM

by Richard J. Krawczyk

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## SUMMARY

The SERT II spacecraft utilizes a 1.5-kilowatt solar array as the primary source of electrical power. The spacecraft power system controls this power as required for housekeeping functions, experiments, and ion thruster system. To meet mission objectives, three separate systems were designed and implemented - regulated dc power, unregulated dc power, and ac power. Extensive ground command capability allows versatility of configuration for normal as well as abnormal flight conditions. Redundancy, fusing, diode isolation, and automatic switching were employed to minimize possibilities of mission critical component failures. Design objectives have been demonstrated in flight.

## INTRODUCTION

The basic objectives of any space mission impose definite requirements for the spacecraft electrical power system. The specific requirements and constraints of the SERT II mission defined the functions of the on-board power system. The translation of the particular mission requirements into specific hardware and the resultant spacecraft power system is the subject of this report.

The SERT II mission is unique in its overall concept; hence, the electrical power system is specially tailored to the specific requirements. Individual features or functions, however, may be similar to those required for other missions. The minimum mission length of 6 months in a sun synchronous orbit required a reliable design able to operate efficiently on solar cell power over a wide range of voltages. Possible shadowing of the orbit for short periods of time or loss of sun orientation required temporary internal power. A relatively large command capability allowed total ground control but inherent failure protection was still required.

This report covers the power system, one of the major portions of the spacecraft. Information of this type should be of interest to those involved in the design and evaluation of spacecraft power systems. The initial discussion concerns electrical power system design philosophy, applicable to SERT II or any other space mission for that matter. The general requirements for the power system are then presented as derived from the mission profile. Following this, the overall power system is broken down into its major subsystems and all individual functions explained with regard to meeting mission requirements. Special features or failure protection are discussed along with the normal operation of the subsystems. The report concludes with a summary of performance as encountered during ground testing and actual flight operation.

## BASIC DESIGN PHILOSOPHY

The general design philosophy of the SERT II program, as in most engineering programs, was to minimize cost and development effort while providing maximum reliability to insure a high probability for mission success. With this in mind, several basic ground rules were established as directly applicable to the design of the spacecraft electrical power system.

The use of previously flown, space-proven hardware was specified wherever possible. This included discrete components such as relays, transistors, and diodes as well as system components such as regulators and inverters. By taking advantage of previous spacecraft programs, new hardware development and qualification was minimized and demonstrated reliability was maintained.

The practice of derating component operating levels is a proven method of achieving maximum reliability. All components within the spacecraft power system were subject to derating as far as practical to insure that no overstresses in voltage, current, or power would occur under worst case operating conditions.

The use of fusing was specified for all components in order to isolate any individual component failures. Fuses were sized for a safety factor of at least two, with respect to the maximum operating currents or significant transients, while still remaining within the power source current capabilities. If a component draws an overload current not sufficient to open the fuse, ground command capability is available to disconnect that component.

Redundancy was specified wherever possible to insure maximum reliability of the power system. Dependent on the particular application, either active or standby redundancy was an integral factor in the power system design.

## GENERAL REQUIREMENTS

In order to fulfill the objectives of the SERT II mission, certain requirements were placed upon the spacecraft electrical power system. The spacecraft housekeeping functions were to be independent of the ion thruster prime experiment since they had completely different power requirements. This required two separate power systems, including solar array power source.

One characteristic of a solar array composed of photovoltaic cells is that its output voltage is dependent on immediate factors such as current drain, temperature, and solar incident angle as well as long term degradation due to radiation. Thus, the housekeeping power system must be capable of accepting a variable input voltage while maintaining proper and efficient operation of the downstream loads, hence, the requirement for voltage regulation. Since the SERT II mission is to be controlled by ground commands, a switching and distribution system was required to energize and deenergize the various loads operating on housekeeping power.

The ion thruster system, operating from the main portion of the solar array, would require relatively large amounts of power. This portion of the power system must, therefore, have the capability to switch and distribute high power levels reliably and efficiently.

As on most spacecraft, pyrotechnic devices would be utilized for actuation of various mechanical functions. The energy required to activate these devices upon ground command must be supplied by the spacecraft power system.

In order to observe the overall behavior of any power system, the values of voltage and current are measured at appropriate points. This capability must be provided in the spacecraft, via telemetry, in order for ground personnel to adequately evaluate the electrical performance.

A 6-month continuous endurance test of an ion thruster system dictated that the power system of the SERT II spacecraft have available continuous sunlight to provide solar power for housekeeping functions and the various experiments. Figure 1 indicates the orbital configuration of the spacecraft. Under certain conditions, however, such as a solar eclipse or loss of attitude orientation, it is possible to lose the solar power for short periods of time. This requires a backup energy source to provide corrective capability or standby power until full solar power is reacquired. If this situation arises within the range of ground stations, the spacecraft power system will be reconfigured for minimum power drain by ground command. Since it is quite likely that any loss of solar power will occur beyond the range of ground stations, the power system must be capable of automatically reconfiguring for minimum power operation. An "emergency only" tie-in to the thruster portion of the solar array was also required to cope with a possible situation in which the housekeeping portion of the solar array becomes inadequate through failure or degradation.

## HOUSEKEEPING POWER SYSTEM

The function of the housekeeping power system is to condition, distribute and control solar array power for spacecraft housekeeping loads such as attitude control, telemetry, command, and signal conditioning systems. This portion of the power system also provides power for the various secondary experiments such as the accelerometer and ambient plasma probe. The initial estimates of spacecraft peak power requirements are listed in table I. The actual load profile would be dependent on the sequence of ground commands for operating the various components.

### System Components

Solar array. - The housekeeping solar array was designed to provide reserve power capacity under all normal orbital conditions. However, being a "soft" supply, the output voltage is dependent upon the load current and also varies with temperature. Figure 2 indicates typical output characteristics of the housekeeping solar array.

Switching mode regulator. - A method of voltage regulation was necessary to simplify the design requirements of the various components operating on housekeeping power. The use of switching mode regulators (SMR) was decided upon since they accept an input voltage of 27.5 to 55 volts dc and had previous space flight use. The SMR efficiency of 87 to 96 percent under normal conditions surpasses that of a voltage limiter-converter system. The regulated output is maintained at 26.5 volts dc  $\pm 1$  percent over a current range of 0 to 7.5 amperes.

Battery. - The SERT II mission plan is based upon continuous sunlight incident upon the solar array to provide full spacecraft power. Certain conditions may occur, however, either as a natural course of events or through an unexpected spacecraft emergency, which would temporarily decrease or eliminate the available solar energy. A solar eclipse would be a natural event while loss of spacecraft attitude control would be an emergency situation. In any event an on-board battery is included in the housekeeping power system for temporary load support or emergency maneuvers to reacquire the sun.

The battery is an 18-cell hermetically sealed, silver oxide-zinc type weighing 34.5 pounds (15.65 kg). It has a minimum capacity of 40 ampere-hours with a useful voltage range of 25.8 to 33.3 volts dc. After reviewing the spacecraft power requirements during attitude reacquisition, this type of battery was chosen for its relatively high energy density and the fact that relatively few discharge-charge cycles were required. This type of battery has seen previous use on deep-space missions.

Battery charger. - The presence of a rechargeable battery in the power system defined the requirements for a battery charger. The characteristics of the charger were based upon the type of battery and the reacquisition requirements of the mission. This

battery charger is basically a previously qualified dc-dc converter with a 22- to 35-volt dc input and a modified output. When charging the battery, the on-board charger provides an initial 1.6-ampere current limit then automatically switches to a 34.65-volt dc constant potential tapered charge. After a 26-ampere-hour discharge, as anticipated for a spacecraft reacquisition sequence, the charger is capable of a 75 percent recharge within 30 hours.

Sensors. - In order to monitor the general performance of the spacecraft power system, voltage and current sensors were placed at critical locations within the system. Total power consumption of the housekeeping power system is monitored by a voltage and current sensor at the solar array input bus. The voltage monitors are simple resistive divider circuits since the telemetry system has a very high input impedance and accepts a 0- to 5-volt dc ground referenced signal. These voltage monitors are also used for SMR on-off status and output level, battery charger on-off status, charge-holdoff status, and battery voltage. The current sensors are of the transformer type, providing complete isolation between input and output. A dual range zero-center current sensor is used to monitor battery performance. The high range indicates battery charge or discharge current up to 2.5 amperes while the low range senses currents up to 100 milliamperes.

## AC SUBSYSTEM

The ac subsystem derives its input power from the regulated dc bus. The ac power source is a previously qualified dc to ac static inverter capable of accepting inputs of 22 to 35 volts dc. The output is a three-phase ungrounded delta connection at 115 volts rms, 400 hertz. This unit has a continuous output rating of 60 volt-amperes total with 30 volt-amperes from one phase and 15 volt-amperes from the other two.

The normal three-phase load on the inverter is about 9 watts at 0.92 power factor or better. This power is required for operating two control moment gyros (CMG's). When the two CMG's start, however, the inverter is required to supply approximately 20 volt-amperes peak per phase. This peak load decreases to the normal level within 20 minutes as the CMG's run up to speed. This short term peak load is within the intermittent duty capabilities of the inverter.

There are other ac loads within the spacecraft that require single-phase power for short periods of time. Motors for beam probes and gimbals require 11 volt-amperes each when operated by ground command. The horizon sensor system, when energized, requires 11.6 volt-amperes of ac power.

The ac power system is actually designed around two inverters where one is normally used and the other is standby redundant. Figure 3 is a simplified schematic of the ac power system. A latching relay switching network allows connecting any of the four

CMG's to either of the inverters. Any CMG switching is normally done with both inverters off; an interlock prevents switching if both inverters are on. Four pole, double-throw latching relays are used to switch the three-phase power to each CMG. The fourth contact pair provides regulated dc power to current regulators and phase demodulators for each CMG. These switching functions and fusing are contained within a central unit which controls the basic housekeeping switching. The single-phase power is taken from the inverter last turned on and routed to other switching units for the individual loads.

A separate switching unit contains relays and fusing for operating the gimbal motors. Power switching for beam probe motors and the horizon sensor is handled by relays within the individual subsystems. The switching circuits shown in figure 3 allow operating the hysteresis-synchronous motors in either direction by connecting the phasing capacitor to either side of the ac power line. The capacitor across the line corrects the power factor to better than 0.9 lagging as required by the inverter.

The CMG power input lines are not fused since the inverter is current limited to 0.8 ampere and adequate switching capability exists to isolate a failed CMG. The other ac loads of figure 3 are fused to prevent any failure from interfering with CMG operation. Since the ac power system is not grounded, a single short to ground has no effect and fusing one side of the ac line is adequate.

## System Operation

The housekeeping power system utilizes two SMR's in a standby-parallel redundant arrangement as shown in figure 4. Both SMR's are normally on. The main SMR supports the housekeeping bus directly and can also support the command bus through the coupling diode. The standby SMR supports only the command bus unless the main SMR is turned off, transferring the appropriate relay contacts. The standby SMR then supports both regulated buses. If the housekeeping array voltage were to fall and remain below the level for SMR regulation, as near the end of the mission, both SMR's can be commanded off. This would connect the solar array directly to the housekeeping and command buses and avoid the voltage drop through the SMR's.

The normal spacecraft power system configuration as shown in figure 4 has the battery charger on, the battery enabled and in the holdoff mode. This means that the charger or the diode coupled solar array input, whichever is at the higher voltage, supports the battery bus. This higher voltage on the battery bus essentially keeps the battery in a standby mode by virtue of the back biased diode pair.

All components operating from the command bus are essential loads, critical to spacecraft survival. This command system includes the command receivers and decoder and transmitters for beacon tracking. This command bus will always have power available from the solar array or battery.

All components and systems operating from the housekeeping bus are nonessential loads and not immediately mission critical. These loads include the instrumentation and data handling systems, inverters, horizon sensor, and various experiments.

If it is necessary to charge the battery, it can be commanded from holdoff to the charge mode. The charging current to the battery may be monitored on the high or low range of the current sensor depending on whether the battery is in the enable or off line mode, respectively. When the charging current decreases below 50 milliamperes, the recharge is essentially completed.

All the loads and subsystems operating from the regulated housekeeping bus utilize the extensive ground command capability for power switching. All ac and dc switching functions are performed by latching relays. These relays have been purchased to high reliability specifications and qualified for the long term environment of the SERT II mission. Consideration was given to vacuum degradation of materials and low leakage hermetic seals in the selection of the power switching relays.

The basic switching to achieve the various housekeeping power system configurations is in the central power control unit shown in figure 5. This includes all power control functions shown in figure 4 as well as power switching and fusing for the primary and backup units of the data handling system, and portions of the ac power system.

A second power switching unit which handles the remaining switching functions within the regulated dc power system is shown in figure 6. Command steering diodes, relays, and fuses for housekeeping loads such as instrumentation and signal conditioning, and experiment loads such as the accelerometer and plasma probes are located within this unit. Solenoid valve actuation and associated telemetry signals for the cold-gas attitude reacquisition system are also controlled by this unit.

## ION THRUSTER POWER SYSTEM

The primary function of the unregulated dc power system is to provide and control power to either of two ion thruster systems, the prime experiment of the SERT II mission. For nominal operation, a single-thruster system requires 1 kilowatt of power from the thruster solar array. The characteristics of this portion of the solar array are shown in figure 7.

The output of the thruster solar array is fed into the unregulated power system on three lines, positive, negative, and center tap. All functions for this portion of the power system are handled by a power switching unit similar to the unit shown in figure 6. The center tap is connected to spacecraft ground through a latching relay and a fuse. This means that the unregulated power system provides equal positive and negative voltages referenced to spacecraft ground. Figure 8 is a simplified schematic of the unregulated dc power system.

## Normal Operation

During normal operation the ion thruster system requires between 15 to 18 amperes at a potential of 55 to 65 volts dc. Under certain mission conditions, it may be necessary to shut down the ion thruster system by interrupting the input power. In order to do this reliably, a motor-driven switch (MDS) with 75-ampere DPST contacts was used. Actuation of this switch momentarily requires about 7 amperes at 22 to 40 volts dc, much more power than available from the standard command pulse. To meet this limitation and still maintain adequate contact derating for maximum contact life, a cascaded relay amplifier circuit was used. The initial command pulse to actuate either MDS momentarily transfers a nonlatching relay. This provides a pulse of regulated housekeeping power to transfer a heavy duty latching relay (K1 or K2, fig. 8). The contacts of this relay now connect the motor winding of the MDS across the positive half of the thruster solar array, causing switching of power to the thruster system. Successive actuation of the MDS causes alternate switching of the contacts.

An additional requirement placed on the unregulated power system was to control approximately 300 watts of power for thruster system heaters. Conveniently, a spare 5-ampere contact was available on each MDS although it did not allow for the usual contact derating. When both MDS, hence both thruster systems, are in the off state, both heaters are on. When either MDS is commanded to the on state, both heaters go off.

In order to evaluate the performance of the unregulated power system, thruster solar array and ion thruster system via telemetry, a voltage sensor and current sensor are used. The voltage sensor is connected across the positive and negative lines from the thruster solar array. The transformer-type current sensor is connected around the positive line.

## Alternate Operating Modes

No fusing is used in the positive or negative lines which supply thruster system power since the current rating of the wiring and MDS contacts is well beyond the short circuit current of the thruster solar array. No current normally flows through the center tap grounding fuse. Should a short to ground occur within the thruster system, however, the fuse will open. This allows thruster operation to continue although no longer referenced to spacecraft ground. If a positive to negative short circuit were to occur in the operating thruster system, the MDS can be commanded to open both sides of the power input. However, in this overloaded condition the array could not supply enough power to actuate the MDS. In order to interrupt the overload, the winding of the MDS can be connected by command (relay K4, fig. 8) to the regulated housekeeping bus to supply sufficient power for actuation.

Another major function of the unregulated dc power system is termed the emergency power mode. Normally, the regulated and unregulated power systems are electrically isolated although the negative side of the housekeeping solar array and the center tap of the thruster solar array are both grounded. Both sections of the solar array are capable of supporting their normal loads with adequate reserve. However, if the housekeeping array becomes incapable of supporting its normal loads through degradation or failure, the emergency power mode would be utilized. In this mode, the negative half of the thruster array is connected in parallel with the housekeeping array to aid in supporting housekeeping loads. To accomplish this the emergency power relay (K3, fig. 8) switches the thruster array center tap from its ground reference to the housekeeping array input bus while grounding the negative line from the thruster array. A blocking diode allows power to flow only from the thruster solar array to the housekeeping solar array.

## UNDERVOLTAGE OPERATION

As previously explained, the spacecraft is in a sun synchronous orbit to provide continuous operating power through the solar arrays. The inclusion of the battery-battery charger system within the spacecraft power system, however, allows for temporary loss of solar power. If the situation arose where the available solar power is below operational limits, the regulated power system will start to draw power from the battery. If the loss of solar array power is due to spacecraft tumbling, the battery must be conserved for the necessary solar reacquisition maneuvers. In order to prolong battery life, all spacecraft loads not immediately essential must be switched off.

If the spacecraft loses sun orientation, the output voltage of the solar array will fall off (neglecting temperature effects) as the solar offset angle increases and the available solar power decreases. Referring to figure 4 the main SMR will maintain the regulated bus at 26.5 volts until the housekeeping array voltage falls below 27.5 volts. As the housekeeping array voltage continues to drop, the regulated bus will now follow it, allowing a 1-volt drop across the SMR. As the regulated housekeeping bus drops below 23 volts dc, a voltage sensing circuit triggers a 200-millisecond time delay. When the 200-millisecond time delay is over and the low voltage condition still exists, the undervoltage circuitry emits a pulse of battery power. The function of this pulse is to disconnect the nonessential loads from the housekeeping bus by energizing the "off" coils of the appropriate power switching relays. The data handling system and inverters are shut down directly by this pulse. The shutdown of spacecraft instrumentation and the various secondary experiments is also triggered by this pulse and utilizes the positive half of the thruster array which would still have adequate power capability to transfer the necessary relays. The 200-millisecond time delay prevents false triggering of the undervoltage pulse by component turn-on transients.

A second voltage sensing circuit is connected to the housekeeping solar array input bus. This circuit is triggered when the array voltage drops to 23 volts dc. After a 1-second time delay, another battery pulse is emitted. This pulse enables the battery, if it is not already enabled, to support the command system through the standby SMR, and turns the battery charger off. With the battery charger on-off relay transferred, the battery can now also support the housekeeping bus through the main SMR. If the undervoltage condition was due to a failure of a component operating on housekeeping power, that component can now be determined and bypassed or left off by ground command. If the undervoltage condition was caused by spacecraft tumbling, the data handling system and inverter can be turned on as required and corrective action taken using the backup attitude control system.

The second undervoltage pulse is also used to remove power from the ion thruster system since spacecraft survival is the prime concern in this abnormal situation. As with the normal ground command, the actuator of the MDS is connected across the positive half of the thruster solar array to open the power circuit. The ion thruster system also has an internal undervoltage shutdown to unload the thruster solar array.

The previously described operations occur automatically and are essential if the spacecraft were to lose attitude orientation when beyond the range of ground stations. A solar eclipse would also cause a loss of solar power and trigger the undervoltage shutdown sequence. Since the eclipse can be predicted in advance and will only last for a short time, a major spacecraft reconfiguration is not necessary. This situation may be handled by commanding the battery charger off prior to the eclipse. The thruster system would also be commanded off since thruster solar array power would drop too low to support operation. With the charger off and the battery enabled, the battery will immediately support the battery bus and the housekeeping array input bus when the array voltage falls below the battery voltage. Thus, the undervoltage sensors will not be triggered and the normal housekeeping functions can be maintained during the eclipse.

## PYROTECHNIC POWER SYSTEM

The SERT II spacecraft utilizes several pyrotechnic devices for actuation of various mechanical functions. Squib actuated pin pullers are used for the release of gimbal and probe assemblies and a squib valve is used to activate the cold gas attitude control system. All the squibs require a minimum firing current of 3.5 amperes for a maximum of 10 milliseconds.

Since the solar array would be deployed prior to the firing of the pin pullers, it is available as the source of pyrotechnic power. The positive half of the thruster solar array is utilized for firing the pin pullers. Since the ion thruster is not operating at the time of pin puller initiation, the solar array has adequate power to fire a maximum of

three squibs simultaneously. The squib valve will be fired only if the backup attitude control system is required for spacecraft reorientation. The squib valve uses the house-keeping bus as the source of pyrotechnic power since the battery will be on line if the solar array voltage is low.

All squibs are fired by ground command to nonlatching double-pole, double-throw relays as shown in figure 9. Ground safety rules require that the individual bridgewires must be shorted up to the instant of squib initiation. Thus, all squib circuits have both lines grounded except during the 50-millisecond command pulse when one line is transferred to the positive power input. Each relay for pin puller actuation fires two squibs for two separate pin pullers. Thus, redundant squibs for the same pin puller are fired through separate relays. The redundant squibs for the squib valve are also fired through two separate relays. For the probe pin pullers, where three squibs are fired simultaneously, the three relay coils are connected in parallel to effect a six-pole relay. The redundancy within the pyrotechnic system insures that a single-relay failure will not prevent initiation of a pyrotechnic function.

Each squib circuit is provided with a fuse resistor in series with the bridgewire. Dependent on the current flowing through it, the fuse resistor will open, interrupting the current, after a specified time interval. By properly sizing the resistance this device limits the squib firing current to about twice the minimum value, allowing three squibs to be fired simultaneously. At this current level, the squibs will fire within 10 milliseconds, less than one third of the time required to open the fuse resistor. After a squib has been fired, there is a possibility that the "hot" side of the bridgewire may short to the case of the squib. If the relay contacts fail to open, the fuse resistor is a backup device to interrupt the current.

## PERFORMANCE

### Ground Testing

All units of the spacecraft electrical power system completed functional and environmental testing on a component basis. They were then integrated with the spacecraft and put through functional and environmental testing as a total system. Actually, three separate spacecraft were built up, each requiring an electrical power system. The testing of the experimental spacecraft verified the compatibility and proper operation of the power system with all other spacecraft systems. The prototype spacecraft was put through environmental qualification and endurance testing, verifying that all systems functioned properly under these conditions. The flight spacecraft was then subjected to the less severe acceptance testing to insure that all systems were ready for the actual flight.

The prototype spacecraft power system accumulated a total operating time of approximately 3500 hours. Of this time 3200 hours were under space simulated conditions of  $10^{-6}$  torr vacuum and the maximum expected temperature extremes. The power system on the flight spacecraft accumulated approximately 775 hours of ground testing, 544 of which were in the thermal vacuum chamber. Real-time flight simulations were performed during the many thermal vacuum tests simulating normal and abnormal modes of power system operation. This ground test phase provided opportunity to identify any minor problems that existed within the power system and make hardware or procedural modifications as necessary.

## Flight Performance

The SERT II spacecraft was launched on February 3, 1970. Spacecraft power system operation was directly initiated upon deployment of the solar array by the launch vehicle timer. The timer also enabled the battery and turned on the battery charger to put the power system in the normal configuration. All the subsequent power system operations were performed by ground command as the spacecraft passed over the tracking stations. The telemetry data from the solar array voltage sensors correlated well with the expected values after temperature effects were accounted for. The experiments were commanded on as specified in the flight sequence and power consumption was observed to be normal. The power drawn from the housekeeping solar array varied from 49 to 144 watts depending upon the loads being energized. The maximum power supplied by the thruster solar array was 1050 watts. The MDS have been cycled several times as both ion thruster systems have been commanded on and off.

The battery was put into charge shortly after launch and periodically thereafter to verify it is fully charged, and available as the backup power source. Because the battery thermal environment is  $80^{\circ}$  to  $86^{\circ}$  F (300 to 303 K) the nonrechargeable capacity loss is estimated at 1.5 percent per month. The ac power system is functioning normally on the main inverter as indicated by proper operation of the gyros, beam probes, horizon sensor, and gimbal motors.

## Eclipse Operation

A solar eclipse crossed the spacecraft orbit 31 days after launch. Since this event was anticipated, the spacecraft power system was reconfigured to provide minimum disturbance to spacecraft operations. The ion thruster system was shut down and the battery charger turned off to bypass the undervoltage shutdown feature of the regulated power system. During the first shadow period, both solar arrays dropped in voltage and

the battery came on line to support the housekeeping solar array bus. The deep shadow portion of the eclipse lasted only a few minutes and drained only 0.11 ampere-hours from the battery. The second time the eclipse crossed the orbit, the spacecraft passed through the penumbra. The decrease in solar intensity was not sufficient to cause the battery to come on line this time. After the eclipse period was over, the power system was returned to the normal configuration by turning the battery charger back on and transferring the MDS to restart the ion thruster. The battery was then charged for several hours to insure that it was returned to full capacity.

## CONCLUDING REMARKS

The electrical power system of a spacecraft is a major factor in determining the success or failure of the mission. The specific requirements of the mission are translated into subsystems and components to perform the necessary functions. The design of the SERT II electrical power system stressed the use of space-proven components to minimize cost and development while assuring maximum reliability for the duration of the mission.

The regulated dc power system conditions the power provided by the housekeeping portion of the solar array and switches it to the various housekeeping and experiment loads. The ac subsystem provides the dc to ac power conditioning and switching for loads such as gyros and motors for probe actuators and horizon sensors. The unregulated dc power system provides switching between the thruster portion of the solar array and the two ion thruster systems. The pyrotechnic system also utilizes the thruster array as the primary power source. A battery-battery charger system is provided for supporting various systems as required in the event that solar power is temporarily lost. An undervoltage shutdown system is available to conserve battery power but has not yet been used since no unexpected losses of solar power due to attitude misorientation have occurred.

The total electrical power system has responded to all commands and performed without failure for over 6800 hours of orbital operation. Since all portions of the power system are functioning properly, utilization of the redundancy and backup modes of operation designed into it have not been necessary.

Lewis Research Center,  
National Aeronautics and Space Administration,  
Cleveland, Ohio, November 18, 1970,  
704-13.

TABLE I. - INITIAL ESTIMATE OF SPACECRAFT  
POWER REQUIREMENTS

	Power, W
Horizon sensor	20
Experiments	43
Instrumentation and data handling	47
Command system	25
Attitude control system	20
Losses	31
Total	186

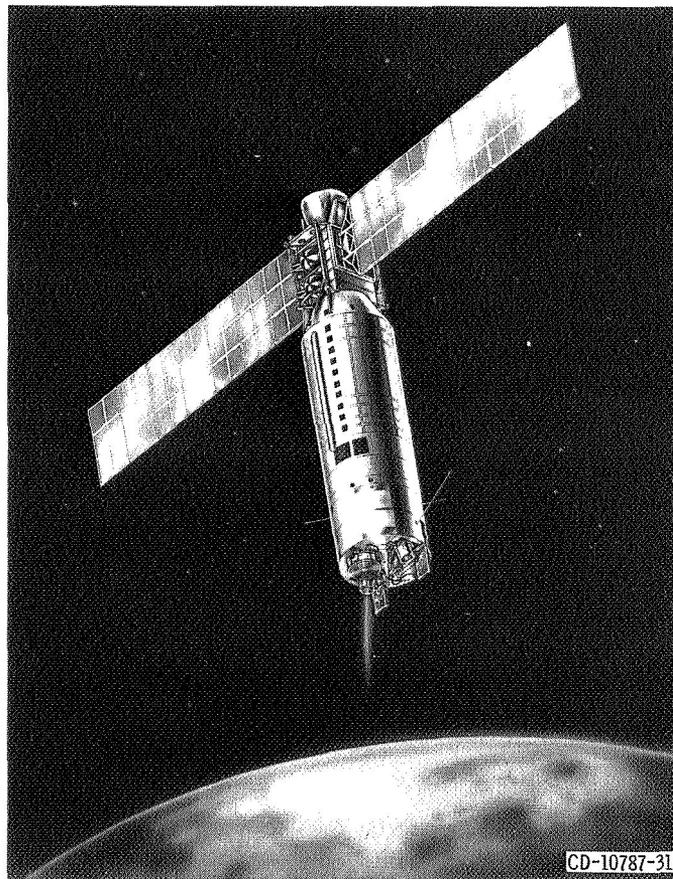


Figure 1. - Orbital configuration of SERT II spacecraft.

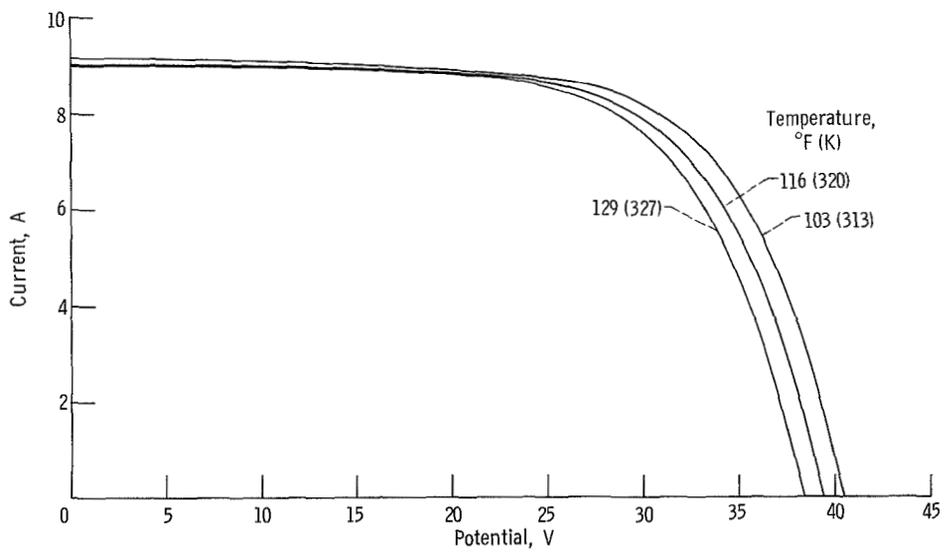


Figure 2. - Characteristics of housekeeping solar array.

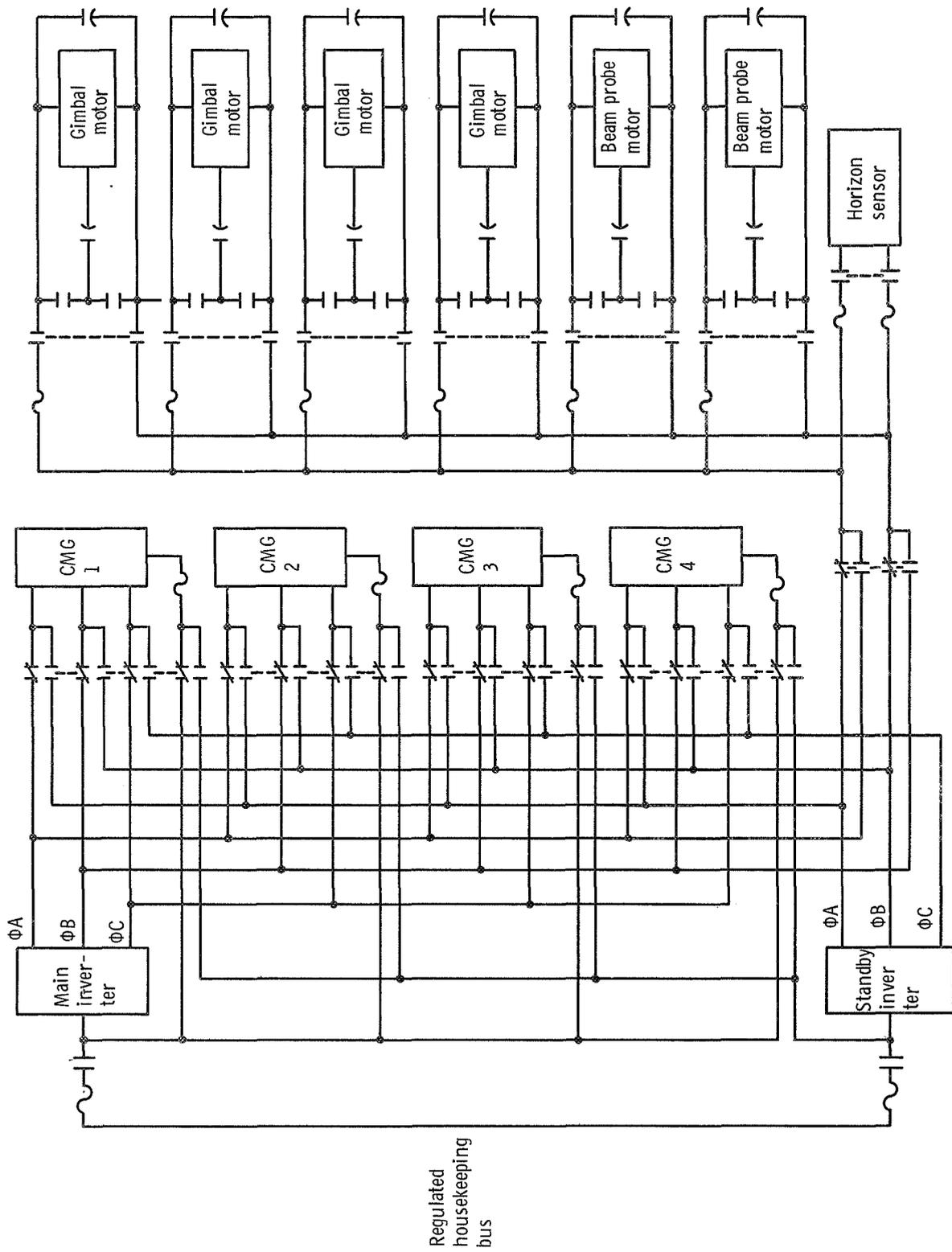


Figure 3. - AC power system.

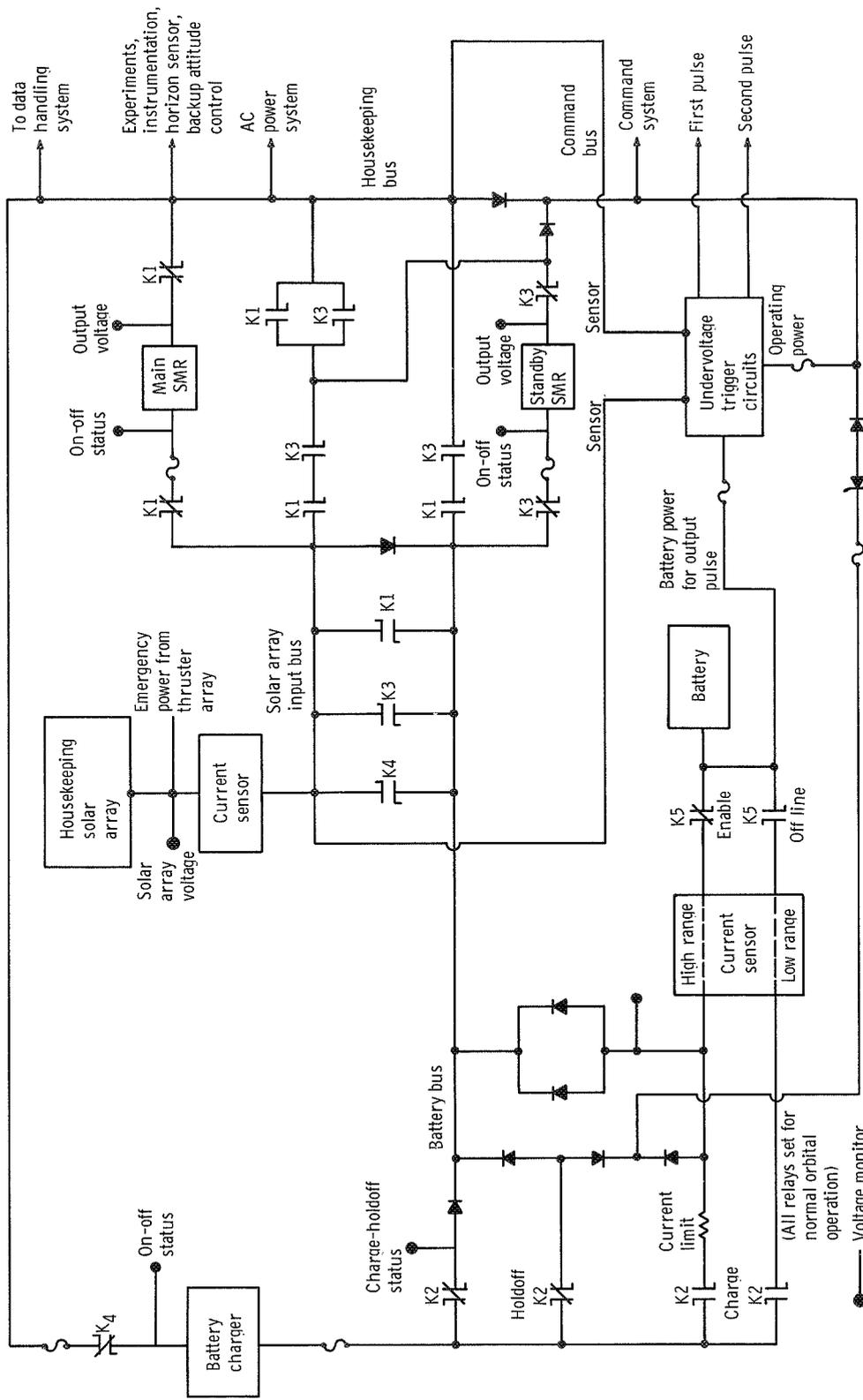
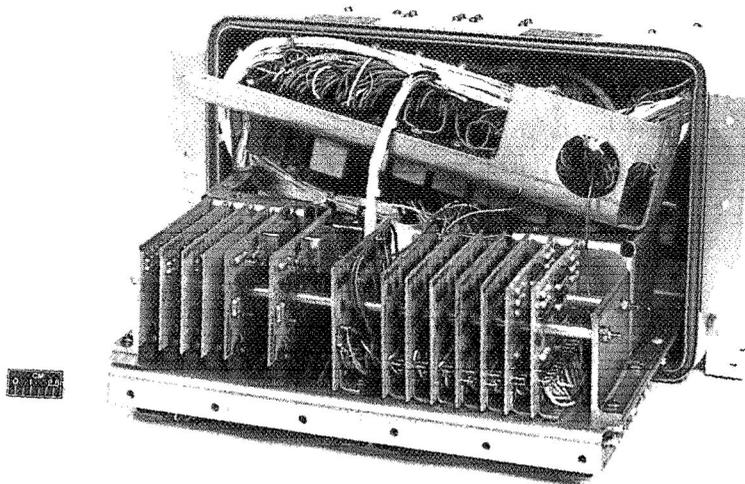
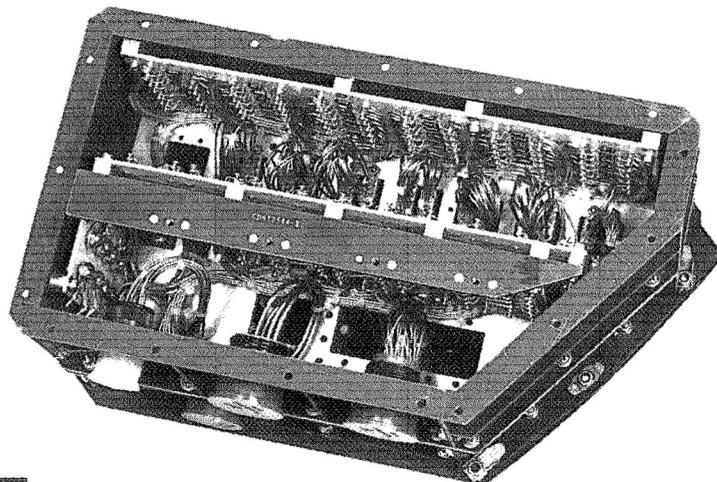


Figure 4. - Regulated dc power system.



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Figure 5. - Power control unit.



C-70-2511

Figure 6. - Regulated dc power switching unit.

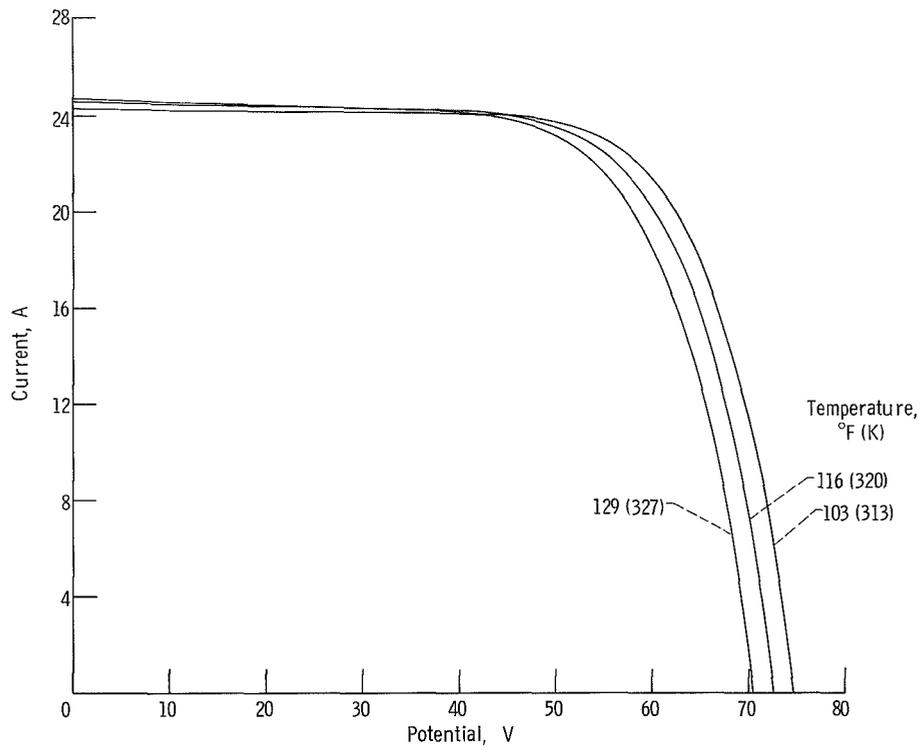


Figure 7. - Characteristics of thruster solar array.

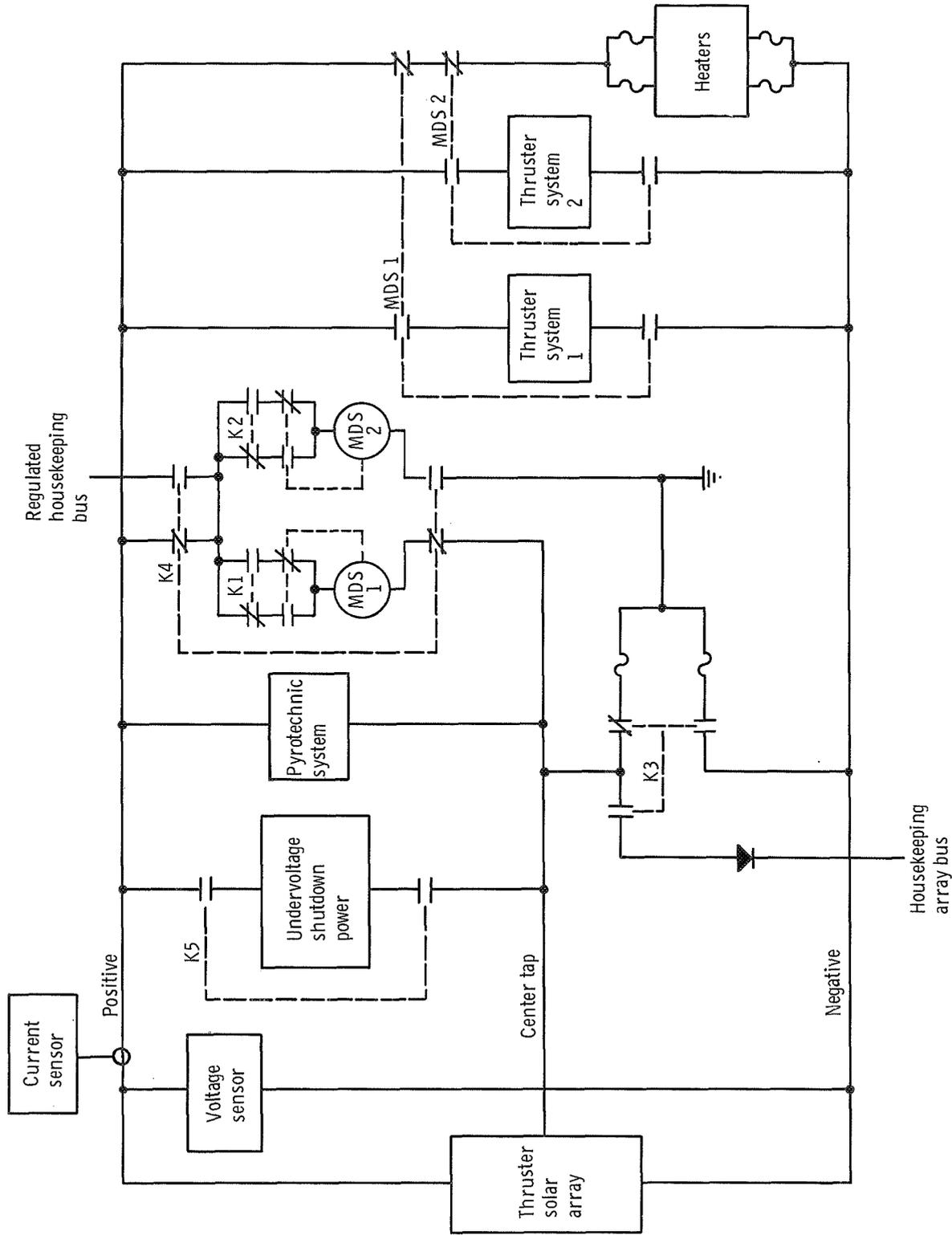


Figure 8. - Unregulated dc power system for ion thruster.

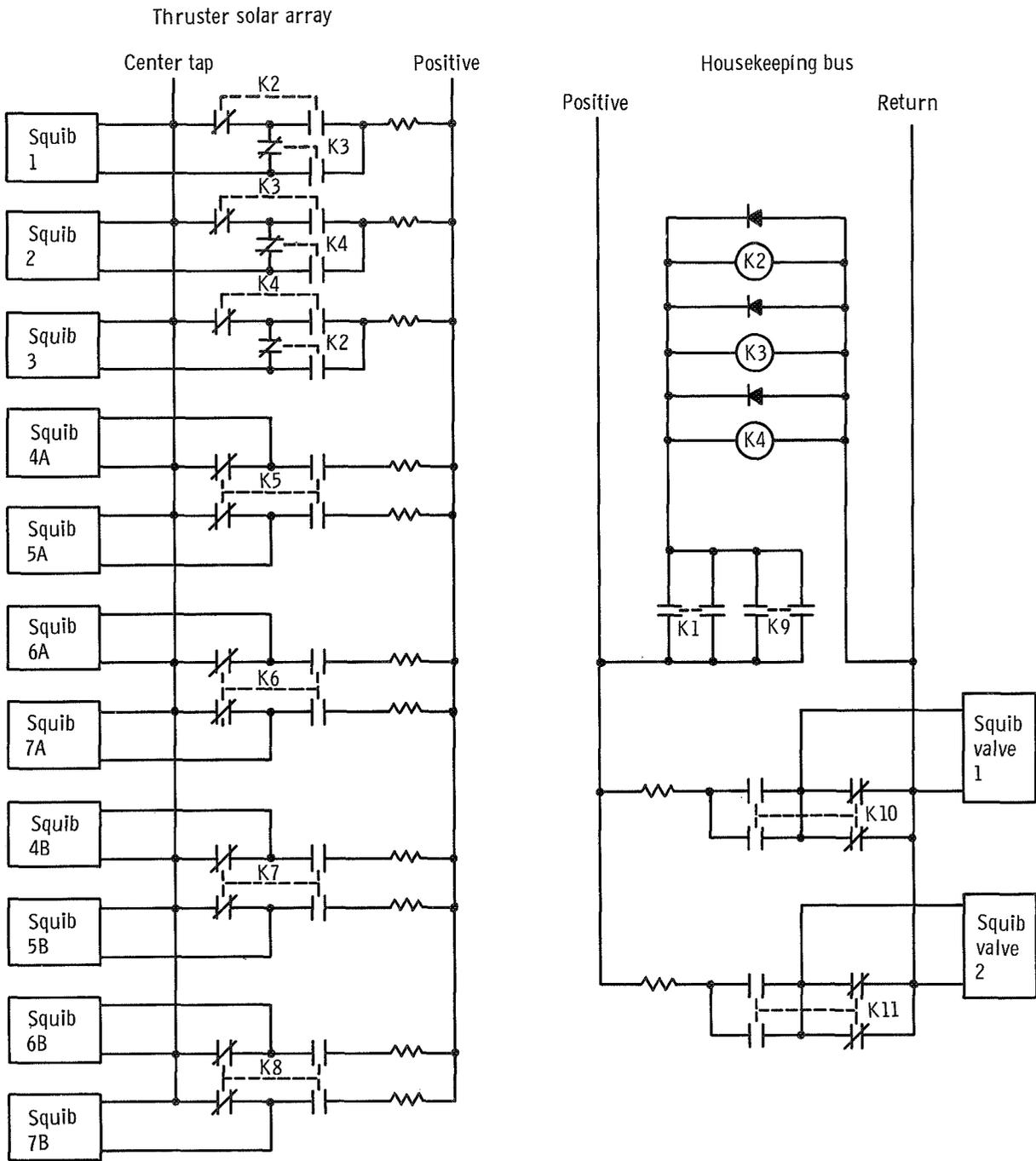


Figure 9. - Pyrotechnic power system.

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