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DEVELOPMENT HISTORY AND FLIGHT PERFORMANCE OF SERT II SOLAR ARRAY

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16. Abstract

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SUMMARY

The SERT II solar array consists of 33 000 2- by 2-centimeter N on P silicon cells configured to supply a nominal electrical power at end of mission of 1100 watts at 56 volts for an ion thruster and 180 watts at 28 volts for the spacecraft's command system, telemetry system, attitude control system, and secondary experiments. The design and testing of the array was limited to modifications necessary for the SERT II mission since similarly designed arrays were flown successfully on several previous missions. The flight performance of the solar array indicated that all requirements for the mission have been met.

INTRODUCTION

A variety of missions have been performed in the NASA space program using solar cells as the source of electrical power. The designers for missions in the past missions and missions in the future have had a common concern, the selection of a solar array that will furnish sufficient power for the mission objectives to be accomplished. Future designers can exercise greater care in their selection of solar arrays if information is available about the design, development, testing, and flight performance of previously flown arrays.

The major objective of the SERT II mission was to endurance test an ion thruster for a minimum of 6 months in a 1000-kilometer polar orbit. The solar array selected to supply 1280 watts for the operation of the ion thruster experiment and spacecraft functions was an array that had been designed and flown successfully on previous Air Force missions. The number of square feet of solar cells and the power produced in orbit makes the SERT II solar array the largest array flown by NASA to date. The

SERT II solar array is the first array flown by NASA that was configured to supply more than one potential. A section of 1100 watts for thruster operation was configured for a nominal potential of 56 volts (±28 V), and the 180-watt section for spacecraft operation was configured for a nominal potential of 28 volts.

The purpose of this report is to present the requirements of the solar array for the SERT II mission, the overall design philosophy of the program, the mechanical and the electrical modifications necessary to adapt the existing designed array to the SERT II mission, the history of the limited development and testing programs, and the data of the solar arrays's flight performance. The presentation of flight data will include the flight voltage and current characteristics, the operating temperatures of the array and the influence of orbit position and geometric location, and an evaluation of the radiation-caused degradation of short circuit current.

DESIGN REQUIREMENTS

Mission Requirements

The SERT II spacecraft was to be launched by a Thorad/Agena D vehicle into a nominal 1000-kilometer Sun synchronous circular orbit, inclined at 99.38°. A mercury bombardment ion thruster system was to operate for a minimum period of 6 months and cause an orbit altitude variation of 100 kilometers. After a minimum period of 6 months sunlight, the spacecraft would enter the Earth's shadow each orbit. This eclipsing would last for 2 months and then normal spacecraft operation and thruster operation would be resumed.

Figure 1 illustrates the orbit of the spacecraft and the orientation of the Agena and spacecraft. The ion thrusters are located on the Earth-facing end of the vehicle.

Mechanical Requirements

The solar array was to consist of two deployable solar array wings. Each wing was to be mounted to the aft rack of the Agena. For launch, the two arrays are folded inside the envelope of the Agena booster adapter structure. Once in orbit the release mechanism would activate and the deployment mechanism would extend the array. Figure 2 shows the fully extended solar array.

Electrical Requirements

The power requirement for the solar array when in orbit was a minimum of 1100 watts for the operation of the thruster and a minimum of 180 watts for the operation of the spacecraft. These on-orbit power requirements, when converted to a standard of air mass zero (AMO) and 25° C, require the thruster section of the array to furnish 1425 watts at a minimum of 62 volts (±31V), and the housekeeping section to furnish 286 watts at a minimum of 34 volts.

The SERT II solar array consisted of 90 panels. Figure 3 illustrates the arrangement of these panels. The fourteen panels that are shown shaded (seven per wing) were the panels that supply the electrical power for the operation of the spacecraft. The remaining 76 panels (38 per wing) supply the electrical power for the operation of the thruster. Note the polarity distribution of the panels used to supply to the ion thrusters. This configuration was selected to provide minimum magnetic torques.

Also shown in figure 3 is an individual panel with the arrangement of the 74 submodules. Each submodule consists of five 2- by 2-centimeter cells connected in parallel. The submodules are series connected to form a panel or solar collector assembly (SCA). For 25° C and AMO, open circuit voltage of the SCA was 43.2 volts, short circut current was 682 milliamperes and maximum power was 21.2 watts. Figure 4 shows the construction of a five-cell submodule.

Instrumentation Requirements

The instrumentation requirements for the solar array were six temperature sensors, an open circuit voltage monitor, a short circuit current monitor, and a peak power monitor. The open circuit voltage monitor was a single 2- by 2-centimeter cell connected to a 2.5-kilohm load resistor. The peak power monitor was a single 2- by 2-centimeter cell connected to a 4-ohm load resistor. The short circuit current monitor consisted of two cells. One cell was located on the active side of the array and a second cell was located on the backside almost directly behind the front cell. The cell on the backside of the array would aid in determining Sun-spacecraft angles if a reacquistion of the spacecraft were necessary. The two cells were connected in parallel to a 1-ohm load resistor.

Figure 5 illustrates the location of the six temperature sensors and the three monitor cells. All of the sensors except temperature sensor 3 were located on the backside of a solar collector assembly. Temperature sensor 3 was located on the backside of the plate to which the monitor cells were mounted. The temperature sensor is almost directly behind the peak power monitor.

DEVELOPMENT HISTORY

Base Resistivity Selection

The solar cell base resistivity used on the similarly designed arrays previously flown was 10 ohm-centimeter. When the flight data for previous programs were reviewed and compared to the SERT II mission requirements, it was found that the power and voltage characteristics of the 10 ohm-centimeter solar array were lower than the minimum requirements for the SERT II mission. The cell configuration for the 90 panels of the array could have been redesigned into a different series-parallel combination; however, this approach would cause an increase in development effort. A second approach was to investigate the voltage and power characteristics of cells other than 10 ohm-centimeter. The result of the investigation was the selection of a 2-ohm-centimeter cell which would meet mission requirements without changing the design of the panels.

Deployment Mechanism Design

The deployment mechanism consists of a mounting bed, an adjustable liquid-damped spring actuator, a slider, and a support assembly. The design of the deployment mechanism for previous flown missions canted and rotated the solar array so that electrical power was available for a large variation in Sun angles. The SERT II solar array required a deployment mechanism which orientated the solar array with minimum incidence angle to the Sun and located the center of mass of the solar array along the centerline of the spacecraft. The center of mass alinement was important because the spacecraft attitude was gravity gradient stabilized. Three different configurations of the solar array and spacecraft with the associated location of center of mass are shown in figure 6.

The top two illustrations in figure 6 show configurations in which the center of mass would be located along the longitudinal axis of the vehicle and the lower configuration illustrates the offset center of mass location used for the SERT II mission. The change in deployment mechanism design to achieve the configuration in the top illustration was very extensive. The change in the deployment mechanism design to achieve the middle configuration was a moderate one; however, it required the manufacturing of a left and right wing. The configuration shown in the bottom illustration was achieved by a minor change to the existing deployment mechanism. This was the configuration chosen and the resulting mechanism is shown in figure 7.

Release Mechanism

The release mechanism for previous programs is shown in figure 8. The tie bar is fastened to the outboard panel by a bracket and the tie rod is held at the ends by pin pullers. This type of configuration required both pin pullers to operate in order to obtain a successful release. A study was conducted to redesign the release mechanism to increase the reliability of the system. The design concepts A, B and C in figures 9 and 10 use the concept of attaching one end of the tie rod to the release mechanism clamp and securing the tie rod to the other clamp by two pin pullers. These designs A, B, and C require the operation of both pin pullers for a successful release.

The final design illustrated in the bottom of figure 10 has the capability of a successful release with only one pin puller operating. The operation of only one pin puller will allow the swivel section on the tie rod end to rotate free from the second pin puller.

Electrical Modifications to Thruster Array

The thruster array supplied power to a power conditioner. One of the specifications imposed on the thruster array by the power conditioner design was a limitation of open circuit voltage to less than 75 volts during periods of thruster operation. When the thruster is operating at 80 percent or greater thrust level, the operating voltage of the thruster array was calculated to be between 55 and 65 volts. However, during initial startup of the thruster, the thruster draws approximately 1 ampere and, as a result, the thruster array is near open circuit voltage.

This condition of operating near open circuit voltage is not by itself a problem if the array temperature is 59° C or above. During the initial selection of the 2 ohm-centimeter cells, the predicted temperature was at 59° C and open circuit voltage was less than 75 volts. As thermal studies were completed, a revised temperature as low as 45° C was predicted, and as the actual current-voltage characteristics of the flight hardware were obtained, it became apparent that the open circuit voltage early in the mission could exceed the 75-volt requirement.

The method selected to correct the open circuit voltage problem based on minimum design change, revised temperature predictions and final current and voltage predictions was to short out six of the five-cell submodules. A typical panel of 74 submodules is shown in figure 11. All 74 submodules are connected in series. The six submodules that are shaded were the ones that were shorted out. Only the 76 panels for the thruster array were modified. The 14 panels for the housekeeping array were not modified because the output of this section was regulated by switching mode regulators, so the higher open circuit voltage was not a problem.

The predicted open circuit voltage at 45° C for the thruster section before shorting of submodules was 79.4 volts; the open circuit voltage after the submodules were shorted was 73.0 volts. The voltage at maximum power was changed from 61.2 to 56.2 volts; the short circuit current and current at maximum power were not effected.

Degaussing of the Solar Array

When orbital stability studies were performed a maximum magnetic moment of 3000 gauss-centimeter^3 was assumed for the solar array. Measurements performed on several test panels revealed an average magnetic intensity sufficient to produce torques for the entire array above 3000 gauss-centimeter^3. The source of the magnetic intensity was traced to the use of a magnetic test fixture to hold the five-cell submodules during their electrical check out after assembly.

The solar array was degaussed by introducing each wing into a 10-gauss field which was created in a 7- by 7- by 3-foot volume by 300 turns of 12-gage wire carrying 8.14 amperes at 60 hertz. The array was in a stowed configuration. The magnetic field was gradually increased by increasing the current to 8.14 amperes in a 2-minute period; the field was held constant for 2 minutes, and then gradually decreased in a 2-minute period. Measurements were taken, and the degaussing procedure was repeated with different orientations of the array until minimum readings were obtained. Final magnetic measurements were taken when the array was fully extended.

The magnitude of the residual magnetic moment of the solar array was 2900 gauss-centimeter^3. This amount was determined to be within limits for orbit stability.

TEST HISTORY

Deployment Test

The purpose of the deployment test was to verify the new design of the deployment mechanism. The structure and deployment mechanism are designed to be operated in a zero-gravity environment and not in a 1-g field, so fixturing to deploy the array in a 1-g field was important. The test fixture for this test, shown in figure 12, supported each section of wing by cable in a low friction roller assembly and provided a simulated zero-g environment for the solar array. The solar array structure shown is not the flight structure but a model. The tests were recorded by high-speed motion picture cameras. The film was used to analyze the motion of the solar array as it deployed.

The actuator assembly was instrumented with strain gages which provided information about the loads being applied to the assembly during deployment.

Ten deployment tests were performed with different adjustments on the actuator arm assembly. For each test the deployment times were recorded and the flatness between panels on the solar array was measured. The film data, strain gage information, deployment times, and panel flatness measurements were used in the selection of the best adjustment for the actuator arm assembly.

The test results showed deployment times from 46.8 to 54 seconds and the angle between panels varied from 179.0° to 180.0°. The fact that the deployment times and the panel flatness were reproducible established confidence in the deployment mechanism.

Release Mechanism Test

The purpose of the release mechanism tests was to integrate it with the other components of the system, verify normal operation, and demonstrate mechanical redundancy. To verify normal operation, two pin pullers are actuated by four squibs. To demonstrate mechanical redundancy, only one pin puller would be operated by one and two squibs.

The zero-g environment for the release mechanism test was provided by a different fixture than was used for the deployment tests. For the release mechanism tests, the solar array was mounted to a pedestal as shown in figure 13. The pedestal allowed a rotational freedom of movement during deployment. When the wing was deployed in space, it would not necessarily follow a straight line, as it was restrained to do during ground testing in the zero-g test fixture. The movement of the pedestal would compensate for the restraint applied to the wing.

The release mechanism was actuated seven times, six times by squibs and once pneumatically. For two of the release tests, both pin pullers were operated by four squibs, two squibs for each pin puller. Two tests were performed with pin puller 1 not operating and pin puller 2 being activated first by two squibs and then by a single squib. Then two tests were performed with pin puller 2 not operating and pin puller 1 being activated first by two squibs and then by a single squib.

The results of the tests verified the operation of the release mechanism with two pin pullers and four squibs, with one pin puller and two squibs, and with one pin puller and one squib.

Solar Array Vibration Confidence Test

The solar array had previously been qualified for the launch environment of an Atlas/Agena. The purpose of this test was to establish confidence that the solar array assembly could withstand the higher dynamic loading from a Thorad/Agena D launch. The test solar array was not the flight structure but a structure very similar to flight hardware.

The solar array confidence test consisted of a vibration test in the x, y, and z-axes, a deployment test before and after the vibration test, electrical continuity checks, harness insulation measurements, visual inspections, and post vibration check of electrical characteristics of a single solar panel.

It was recognized that an overtest can occur on a shaker when a greater number of cycles are introduced into the structure at resonant frequencies than were actually experienced in flight. Two possible methods to prevent such an overtest are to program the shaker with amplitudes equal to the sinusoidal components and vary the sweep rate, thereby duplicating the number of cycles, or program the shaker with a constant sweep rate and reduce the amplitudes of the sinusoidal components proportional to the damping ratio of the structure. The method of constant sweep rate was selected for the vibration testing of the solar array.

The procedure for obtaining vibration test amplitudes from actual flight data was accomplished in three steps. The first step was to transform the actual flight data into sinusoidal components which are reproducible on a shaker. Next, the damping ratio of the solar array was experimentally determined. The third step was to modify the amplitudes of the sinusoidal components proportional to the experimentally determined damping ratio.

The expected flight levels and the actual test levels for the x-, y-, and z-axes are shown in figures 14, 15, and 16, respectively. Figure 17 shows a photograph of the solar array mounted to the shaker.

The solar array assembly satisfactorily passed the visual inspection, the deployment test, the electrical continuity check, and the harness insulation measurements. The postvibration illumination test performed on the single panel indicated no serious degradation due to the higher vibration levels.

Flight Acceptance Test

The purpose of the flight acceptance test was to perform the following on the flight solar array:

(1) Visual check of the mechanical structure

(2) Harness continuity and insulation checks

(3) Power output check on each of the 90 panels

(4) Assembly of solar array structure to mounting bed

(5) Actuation of release mechanism

(6) Three deployment tests on each wing

The flight solar array assembly completed the flight acceptance test without any anomalies.

Mechanical Fit Check of Solar Array to Agena

The purpose of the mechanical fit check of the solar array to the Agena was to properly aline the solar array mounting bed to the Agena aft rack, check for proper clearances on the assembled unit, mechanically fit and secure the solar array harness, and check for proper continuity from the solar array terminal board through the Agena vehicle. In figure 18 the solar array is shown in its stowed configuration mounted to the aft rack of the Agena vehicle. The solar array fit check was completed with only one minor modification, the Agena booster adapter connector was relocated because of its interference with the Agena heat shield.

Final Deployment Test

The final deployment test was performed in conjunction with the thermal taping modification necessary for a spring launch and the degaussing of the entire solar array. Thermal taping of the tie rod, which is on the shade side of the vehicle during ascent, is necessary to prevent excess tension due to temperature differentials. The thermal taping had been applied in anticipation of a fall launch. When the launch was rescheduled to a spring launch, the thermal tape was removed from the tie rod of one wing and applied to the tie rod of the other wing.

Two months prior to the final deployment test, the solar array had been electrically modified by shorting out six of the submodules on each of the thruster array panels, and a shade shield was installed on the short circuit current monitor. It was necessary to disassemble the solar array structure from the mounting bed and remove the associated harness in order to accomplish the electrical modification. The final deployment test revealed that a service loop in the power harness was not correct, and, as a result, the solar array would not have fully extended in flight. The harnessing was reworked while the array was in a fully extended configuration to ensure proper formation of the service loop.

Prelaunch Test of Solar Array at Western Test Range

At the Western Test Range, Vandenburg Air Force Base, Lompoc, California, the following tests were performed prior to launch:

(Before solar array was mated to vehicle)

(1) Solar array continuity and insulation checks

(2) Electrical compatibility check with vehicle

(3) Electric checkout of the squib circuits

(After solar array was mated to vehicle)

(4) Continuity and insulation checks, and installation of squibs

(5) Polarity checks on power harnesses

(6) Connection of solar array power harness to spacecraft support unit and to spacecraft

After successfully completing these tests, the solar array was ready to support the SERT II mission.

FLIGHT RESULTS

Thruster Array Performance

The predicted current voltage characteristics for the thruster array are presented in figure 19. The I-V curves were drawn for the predicted temperature of 116° F (47° C) with a $\pm 13^{\circ}$ F ($\pm 7^{\circ}$ C) deviation. An effective solar flux, which includes spacecraft angle and solar flux variation due to the Earth's orbit, of 410 Btu per square foot - hour (129 $\mathrm{mW/cm^2}$) was used to calculate values for these curves.

There are nine flight data points plotted which represent the performance of the thruster array for the first 6 months. The two data points for orbits 95 on 2/10/70 and 436 on 3/7/70 represent the thruster array open circuit voltage. Three data points represent the voltage and current characteristics of the thruster array during the operation of thruster 2. The data point for orbit 100 on 2/11/70 is the 30 percent thrust level of the thruster, the data point for orbit 109 on 2/11/70 is the 80 percent thrust level, and the data point for orbit 113 on 2/12/70 is the 100 percent thrust level. The remaining four data points represent the voltage and current characteristics of the thruster array during the operation of thruster 1. The data point for orbit 140 on 2/14/70 represents 30 percent thrust level, the point for orbit 146 on 2/14/70 represents 80 percent thrust level, the data point for orbit 171 on 2/16/70 represents 100 percent level, and the point for orbit 2200 on 7/15/70 represents 100 percent thrust level 149 days later than orbit 171.

Housekeeping Array Performance

The predicted current-voltage (I-V) characteristics for the houskeeping array are presented in figure 20. The I-V curves were drawn for a predicted temperature of 116° F (47° C) with a $\pm 13^{\circ}$ F ($\pm 7^{\circ}$ C) deviation. An effective solar flux of 410 Btu per square foot - hour (129 mW/cm^2) was used to calculate the values for these curves.

The four data points plotted on the curves in figure 20 are for orbits 2, 8, 556, and 2200. The electrical loads for orbit 2 on 2/3/70 were the command system, data handling, spacecraft instrumentation, and two tape recorders. The electrical loads for orbit 8 on 2/3/70 were the electrical loads of orbit 2 and two control moment gyros. The electrical loads for orbit 556 on 3/16/70 were the electrical loads of orbit 8, as well as a miniature electrostatic accelerometer (MESA), radiofrequency interference (RFI), and space probe experiment. The electrical loads for orbit 2200 on 7/15/70 were the electrical loads of orbit 8 and a space probe experiment.

Temperature Variation of Solar Array

The curves shown in figures 21 and 22 represent the temperature variation of one wing of the solar array as a function of its position around the Earth. The time span covered is the first 90 days of the mission. As shown in figure 5, temperature sensor 4 was located at the outside tip of the solar array, sensor 5 was located in the middle of the array, and sensor 6 was located on the inside section near the Agena vehicle. Also shown in the figures are the thruster array voltage that are associated with the temperature variation for the various orbits.

The temperature curves show that the temperature of the array nearest the Agena is the warmest. The temperature gradient from sensor 4 to sensor 5 is smaller than the temperature gradient from sensor 5 to sensor 6. This indicates that the thermal effect of the Agena vehicle is limited to the half of the solar array wing nearest the vehicle.

An examination of the curves will reveal the effect of spacecraft angle and Earth's albedo on the temperature variation of the solar array. In figure 21 the spacecraft angle was 21° and temperature variation of sensor 4 was 10.7° F (5.9° C). In figure 22 the spacecraft angle was 6° and sensor 4 temperature variation was 4.5° F (2.5° C). A similar reduction of temperature variation can be noted for the other two sensors.

The thruster array voltage was selected for purposes of showing the voltage variation of the array because the current drawn from this array was nearly constant. For figures 21 and 22, the current being drawn was 16.4 amperes. The telemetry resolution for the thruster array voltage is larger than the variation of the thruster array voltage, hence the variations are represented by one of two different voltages for a particular orbit.

Solar Array Monitors

The three solar array monitors were open circuit voltage, short circuit current, and peak power. The open circuit voltage monitor failed; however, this data could be obtained from the thruster section of the array whenever the thruster experiment is turned off.

Short circuit current. - The flight data for the short circuit current monitor are plotted in figure 23 for the beginning of mission, 30 days, 60 days, and 103 days after launch.

The large variation in the value of the short circuit current is caused partially by spacecraft angle, Earth's albedo, and a shade shield. The position of the shade shield was such that data, as shown, from the equator to the South Pole were affected because the short circuit current monitor was being shadowed. Data from the South Pole to the North Pole were affected by the spacecraft angle and Earth's albedo. The curve for February 4 has an abrupt change as the spacecraft was at the equator traveling toward the North Pole. This change was caused when the shadow cast by the tie rod crossed the short circuit current monitor.

The variation of the short circuit current which was not affected by the shade shields has a similar trend in variation as the solar array temperatures. The larger the spacecraft angle, the larger the amplitude variation in short circuit current. This amplitude variation resulted from added flux of the Earth's albedo. Generally, the larger the spacecraft angle, the larger the deviation of the spacecraft orbit from the Earth's terminator. From the vernal equinox to autumnal equinox the spacecraft is on the sun side of the terminator at the North Pole and on the shade side of the terminator at the South Pole. From autumnal to vernal equinox, the spacecraft's orbit is on the shade side of terminator at the North Pole and on the sun side of terminator at the South Pole.

Peak power monitor. - The flight data for a span of 60 days of the mission for the peak power monitor are shown in figure 24. The data shown for the peak power monitor are taken from the same orbits as the short circuit current monitor. The abrupt change in value of peak power for the orbit on February 4 is caused by the shadow of the solar array tie rod crossing the cell.

The peak power data show a trend similar to that of the short circuit current. The larger the spacecraft angle, the larger the variation in the amplitude of the value for peak power. It can be noted from the curves that the peak power value for February 4 is higher than the value for April 2 despite the fact that on April 2 the spacecraft angle was less and the solar flux was greater. This lower value is attributed to increased temperatures of the cell mounting plate and radiation degradation.

At the beginning of the mission, the temperature of the plate to which the cells were mounted was below the telemetry range and is estimated to be 90° F (32° C). As the mission progressed and the paint on the mounting plate degraded, the temperature of the mounting plate increased into the telemetry range. The telemetry reading obtained from the peak power monitor was the voltage reading across a 4-ohm resistor. This voltage is affected by the temperature of the cell, as well as radiation degradation effects on both current and voltage.

Radiation degradation. - The data presented in figure 25 show the relative percentage of radiation degradation of the short circuit current monitor. The previous data presented in figures 21, 22, 23, and 24 give an indication of the difficulty involved in comparing data points of the solar array monitors over a 6-month period because of the variation of solar flux, spacecraft angle, Earth's albedo, and temperatures. The technique used for comparing data over a 6-month span was to use the temperature of the solar array as an indicator of the amount of effective solar flux falling on the array.

The thruster array had nearly a constant current load. Any degradation of the thruster array would be indicated by a decrease in thruster array voltage. Several data points taken on the thruster array during open circuit voltage conditions showed no voltage degradation.

During the first 6 months it was noted that, for the same array temperature, the thruster array voltage decreased twice. The first voltage decrease occurred 36 days after launch; the second occurred 161 days after launch. The selection of these two data points allows the thruster voltage to be determined more accurately than telemetry resolution normally permits. The decrease in thruster array voltage represents a 11 percent degradation in current. The curve in figure 27 for short circuit current shows that the percentage change in degradation from day 36 to day 161 is 10 percent. The estimated degradation from launch to 161 days after launch was 14 percent.

SUMMARY OF RESULTS

The results of the design, development, testing, and performance of the solar array are as follows:

1. A previously designed solar array was successfully modified to meet the requirements of the SERT II mission, and flight qualified status was maintained with a minimum of confidence testing.

2. The existence of two different potentials on the solar array has not presented any in-flight problems.

3. The flight temperatures for the solar array were nearer the upper limit of the predicted temperature range.

4. The degradation of the short circuit current followed the curve of the predicted degradation of the peak power.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, December 23, 1970,
704-13.

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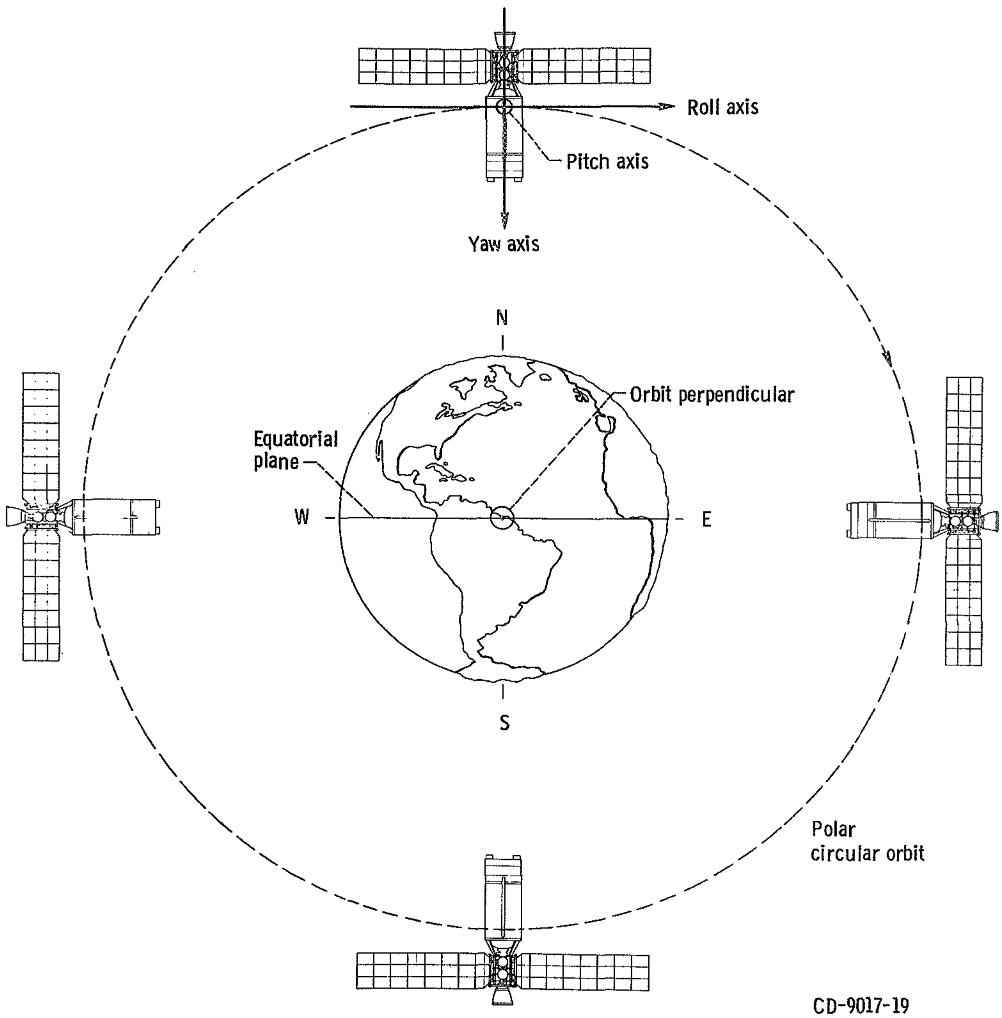


Figure 1. - SERT II in orbit viewed from Sun.

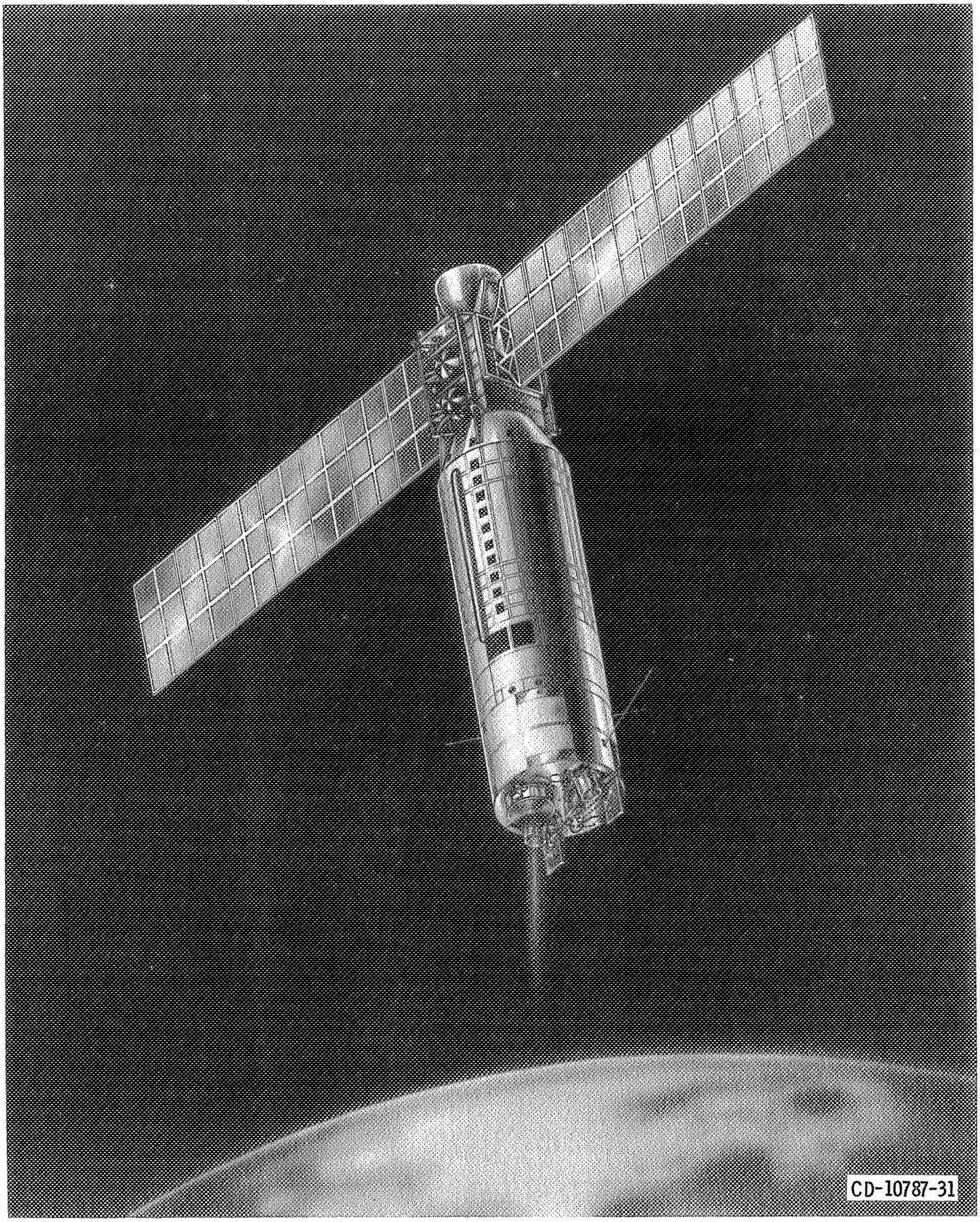
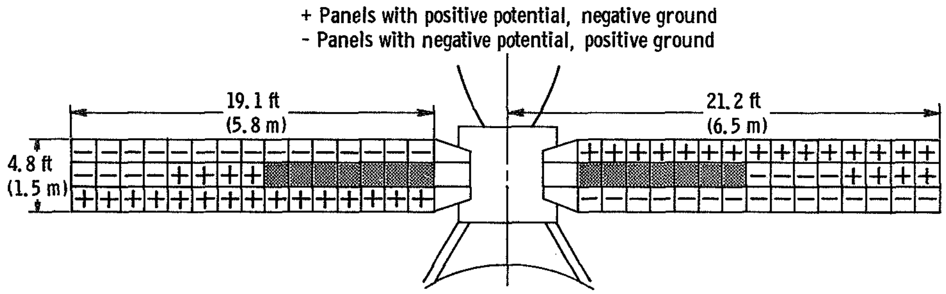
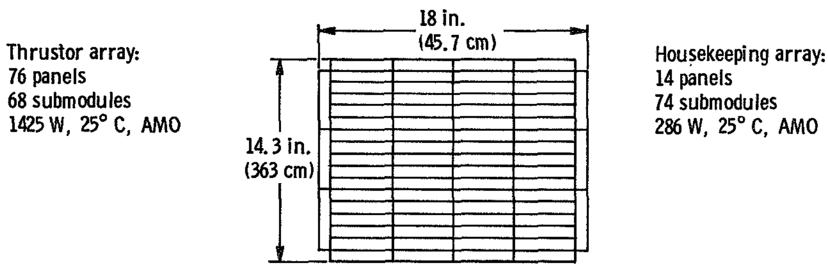


Figure 2. - Solar array in extended position mounted to Agena in Earth orbit.



(a) 45 panels per wing arranged in 15 by 3 configuration.



(b) One of 90 panels for SERT II.

Figure 3. - Panel arrangement for solar array and submodule arrangement for a panel.

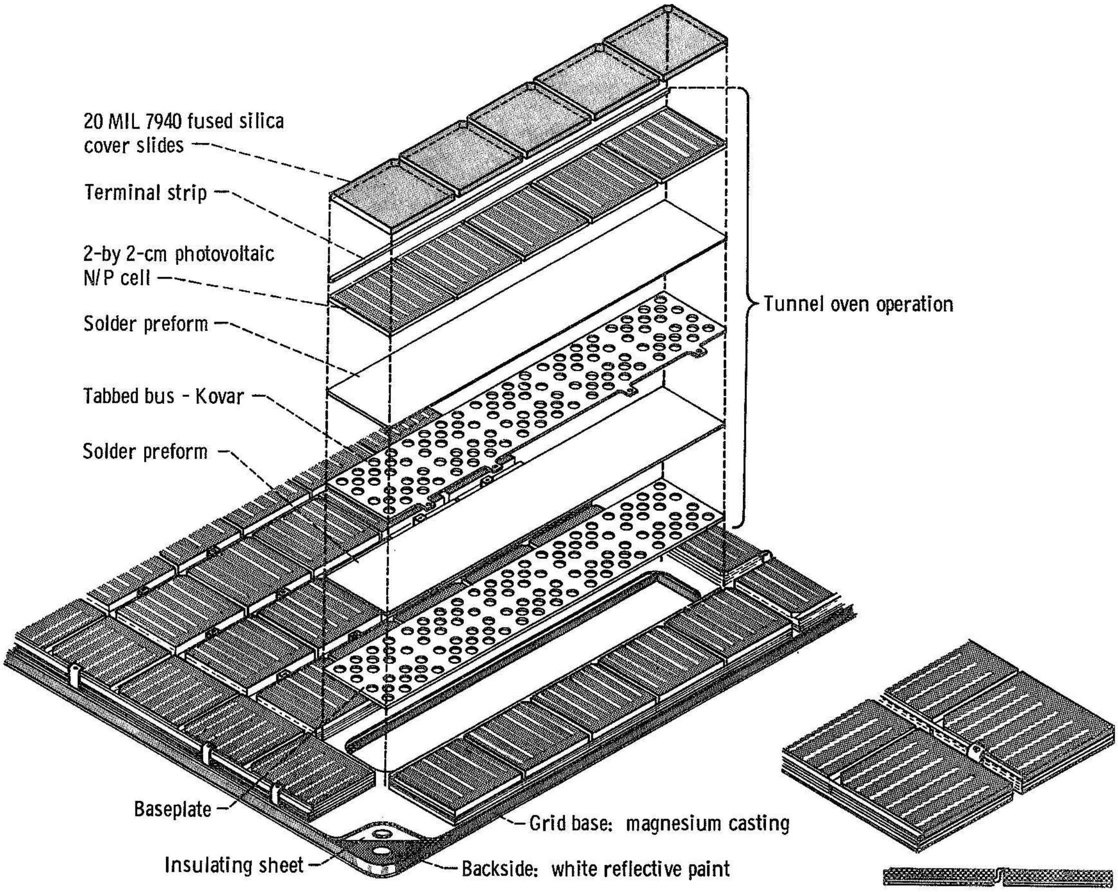


Figure 4. - Assembly of five-cell submodule.

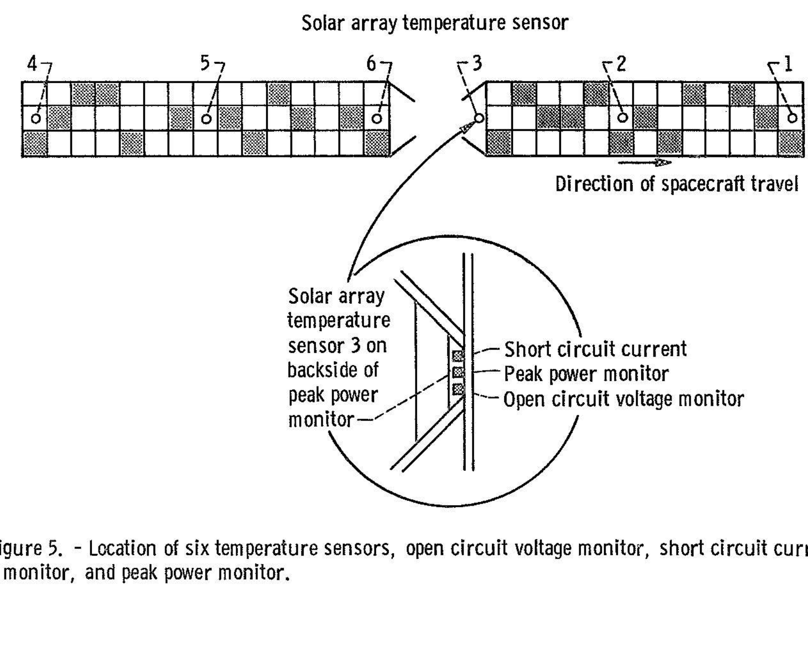


Figure 5. - Location of six temperature sensors, open circuit voltage monitor, short circuit current monitor, and peak power monitor.

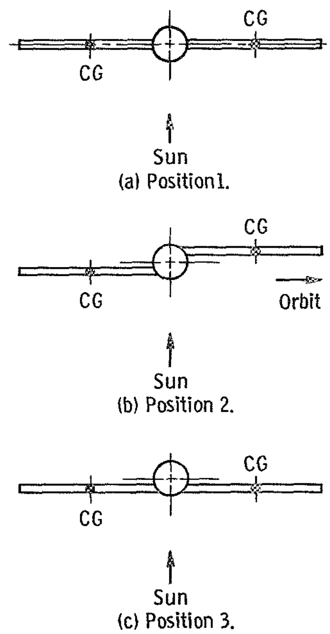


Figure 6. - Alternate solar array configuration considered for SERT II mission.

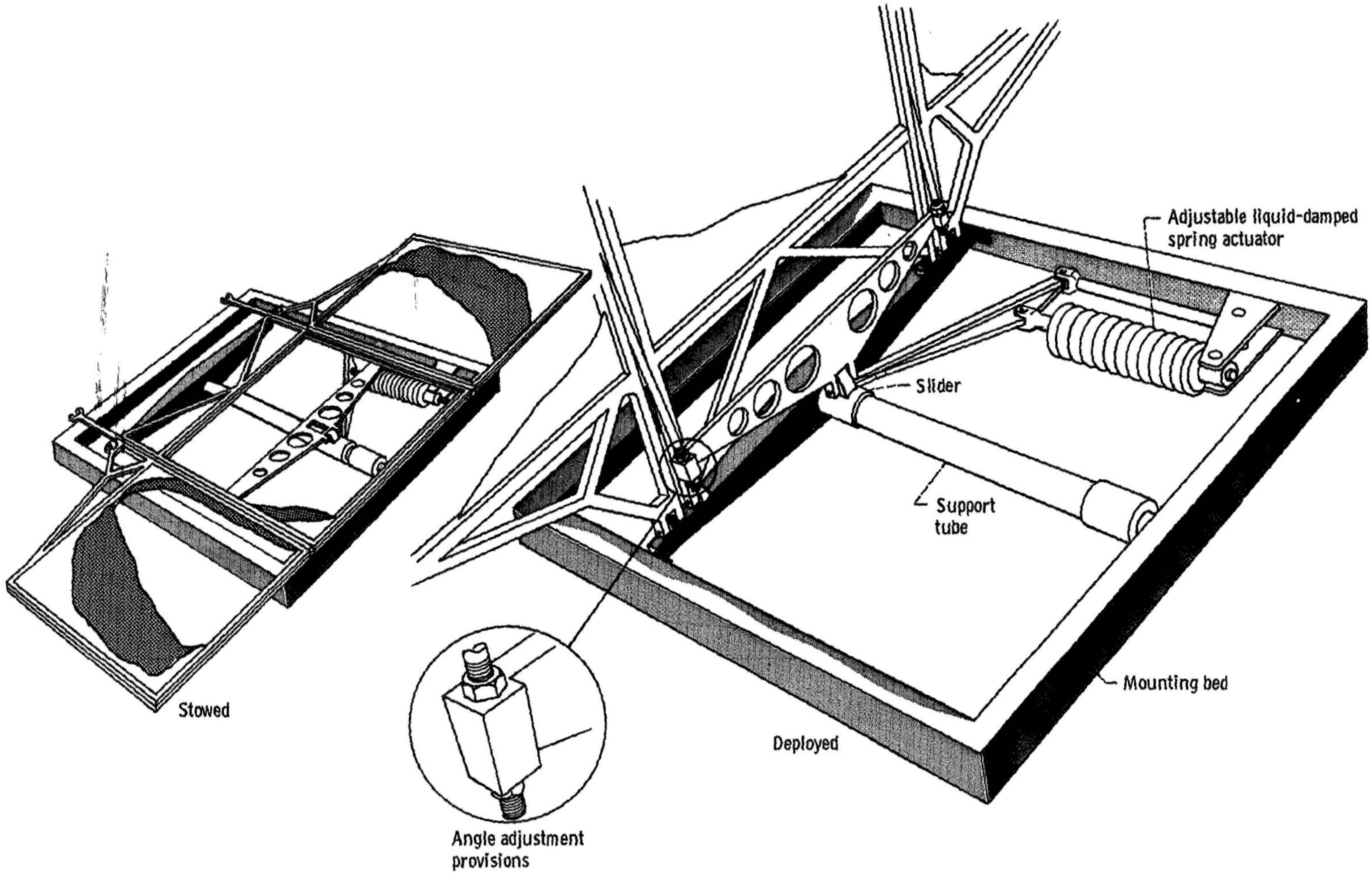


Figure 7. - Deployment mechanism of SERT II array in stowed and fully extended position.

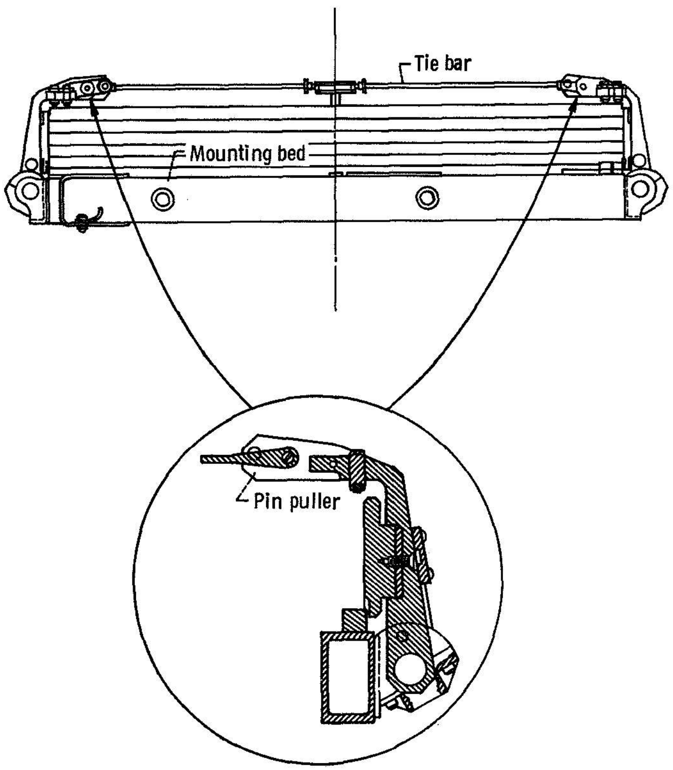
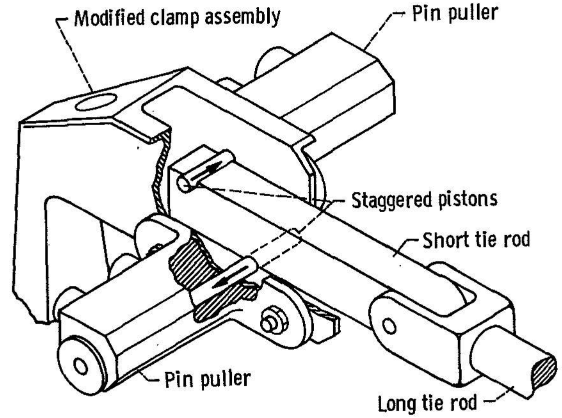
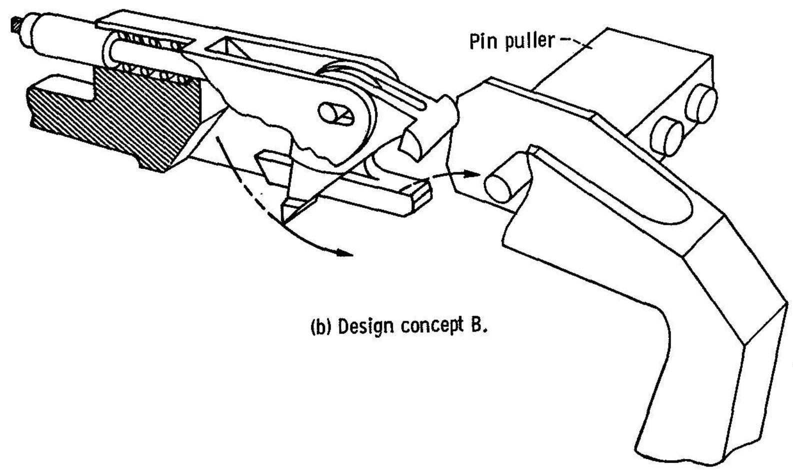


Figure 8. - Original release mechanism design.

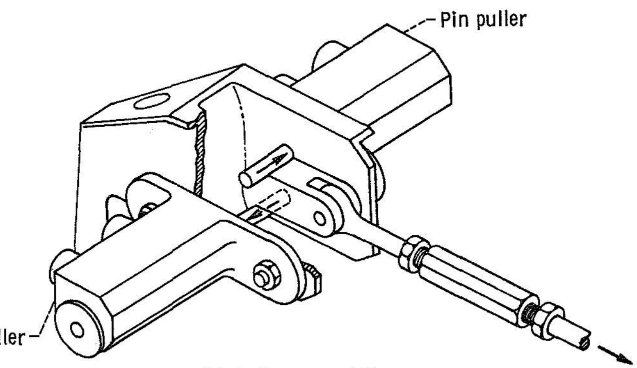


(a) Design concept A.

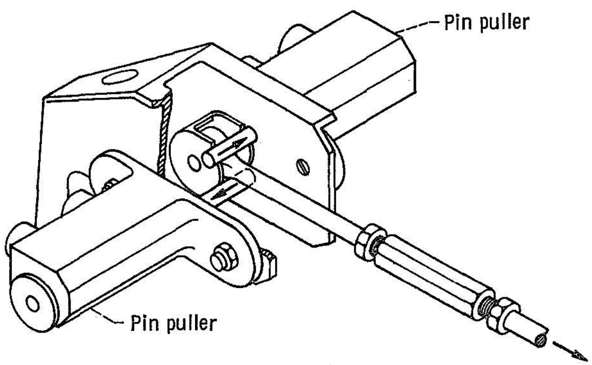


(b) Design concept B.

Figure 9. - Proposed design changes for release mechanism.



(a) Design concept C.



(b) Final design.

Figure 10. - Proposed and final designs for release mechanism.

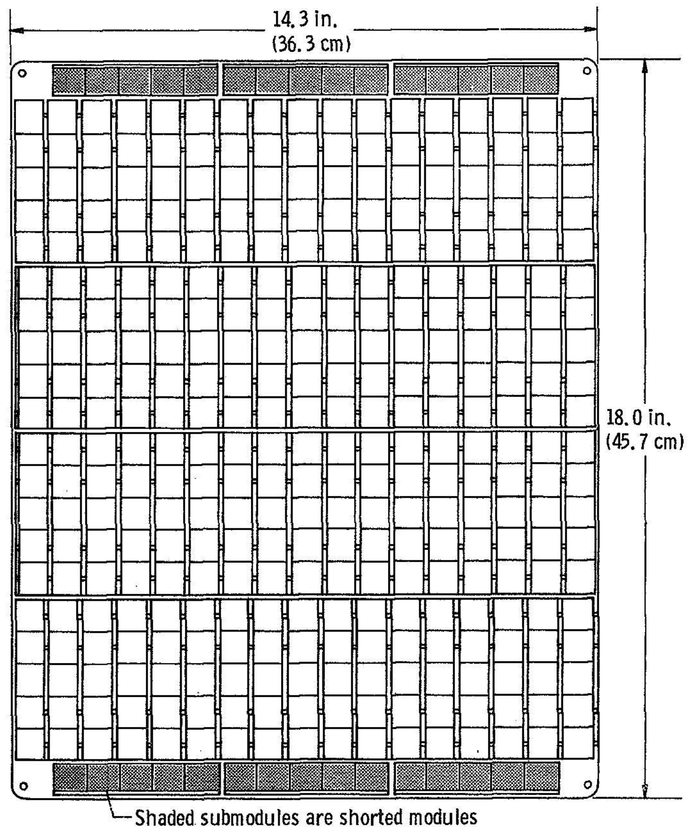


Figure 11. - Typical solar array panel with six of 74 submodules shorted out for 56-volt section.

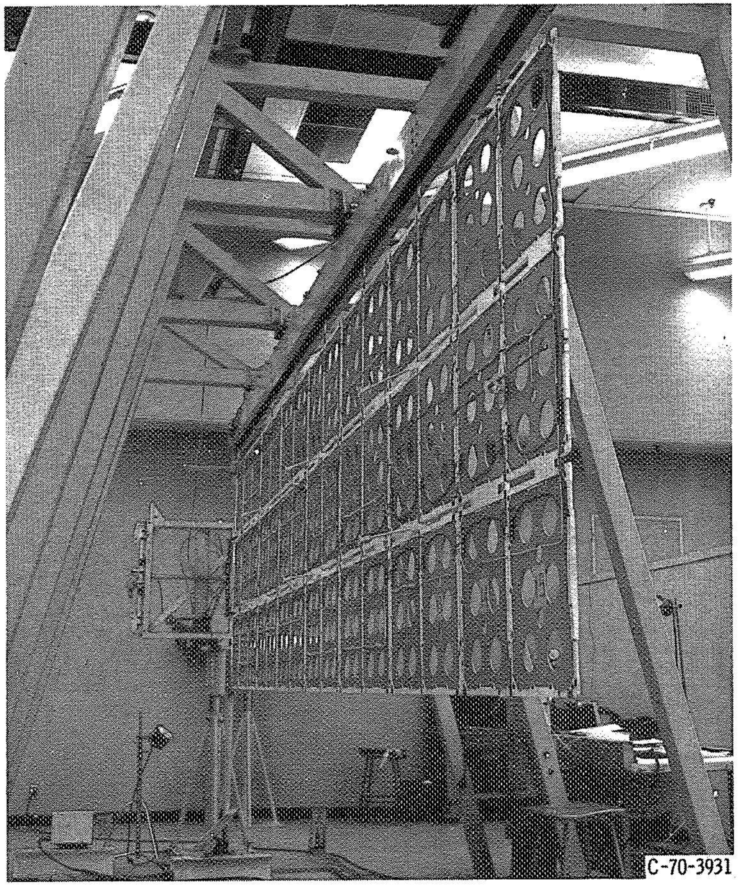


Figure 12. - Test fixture for zero-gravity simulation for deployment mechanism testing.

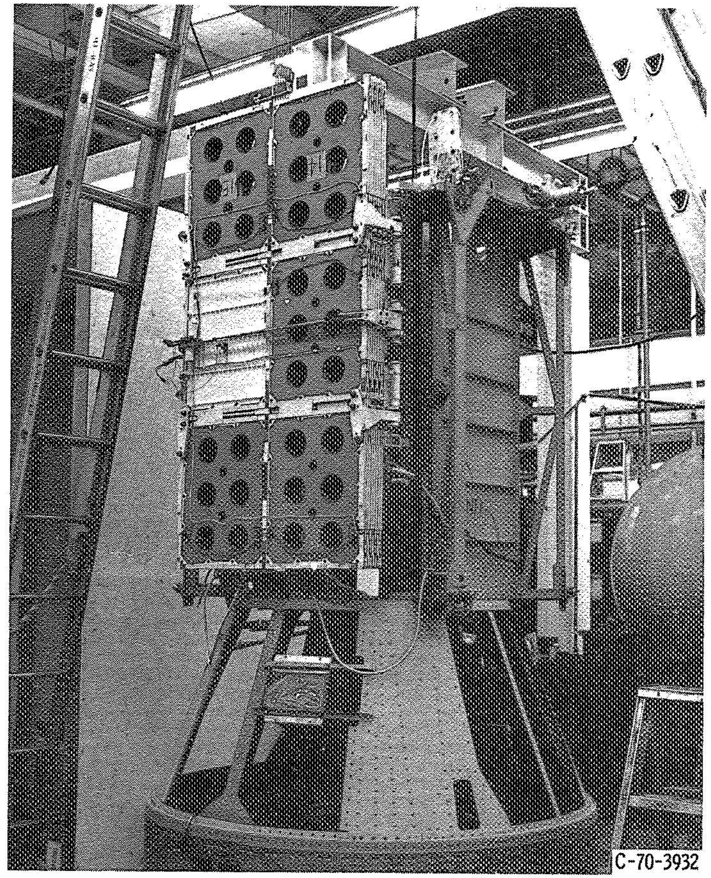


Figure 13. - Solar array mounted to pedestal for release mechanism testing.

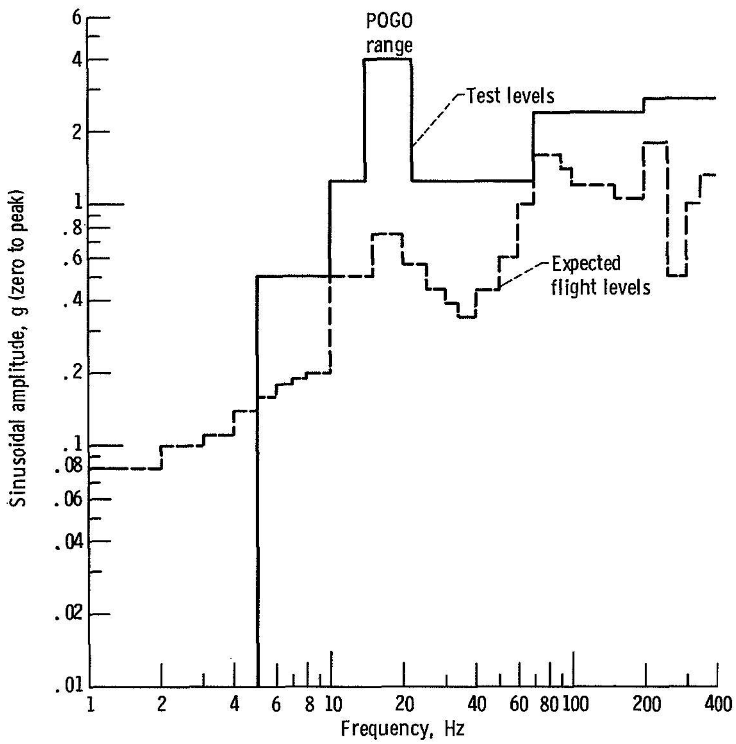


Figure 14. - Sinusoidal test levels for X-axis excitation.

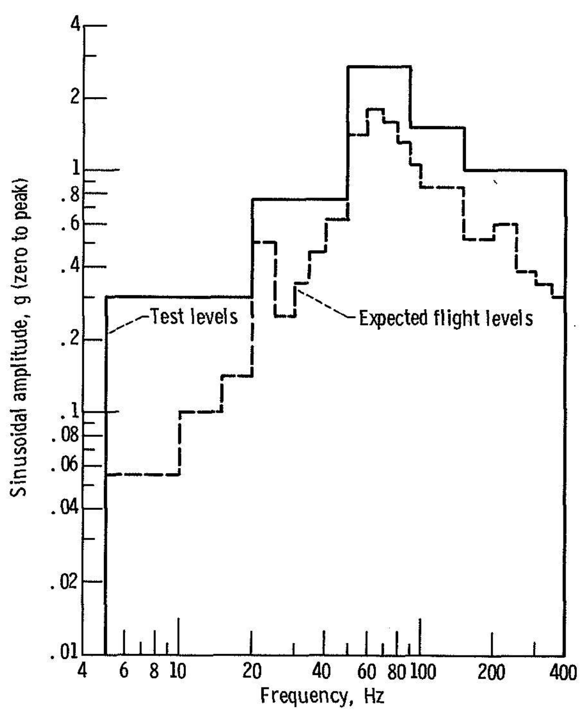


Figure 15. - Sinusoidal test levels for Y-axis excitation.

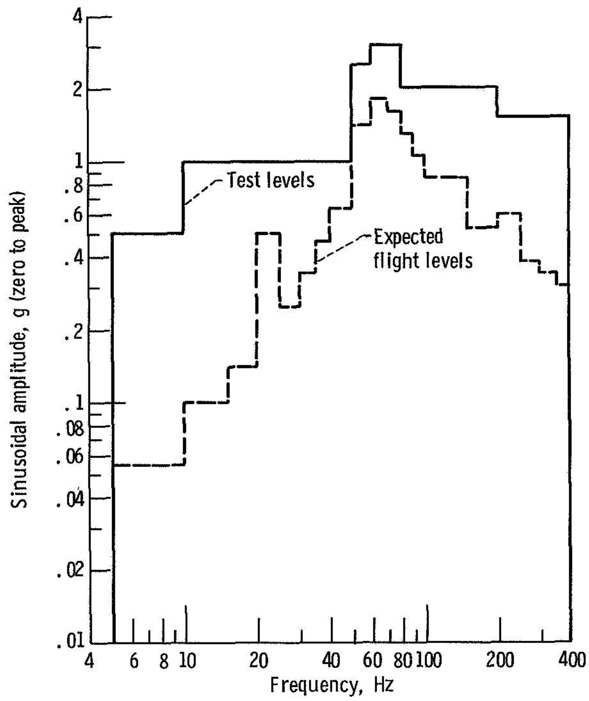


Figure 16. - Sinusoidal test levels for Z-axis excitation.

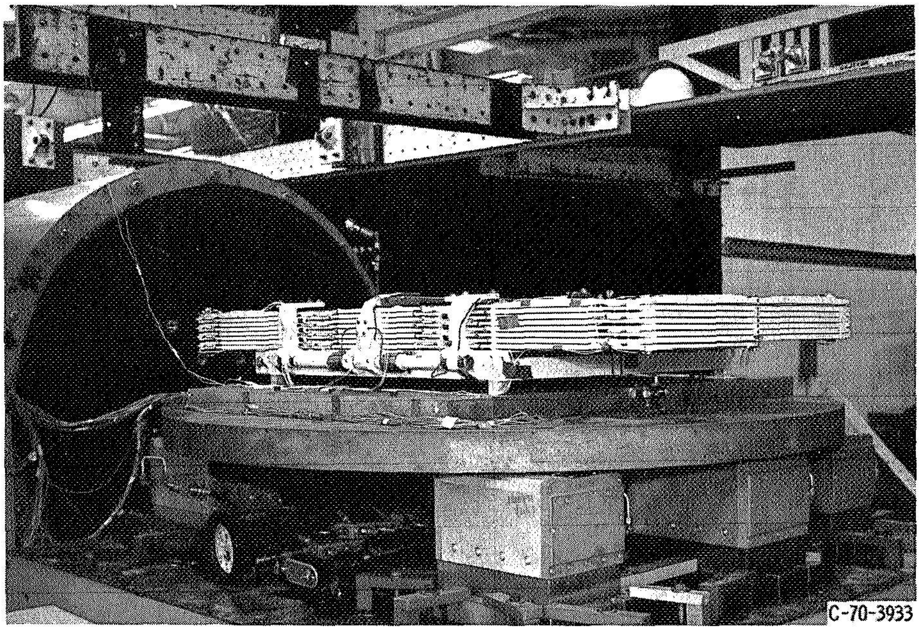


Figure 17. - Solar array in stowed configuration mounted to shake table.

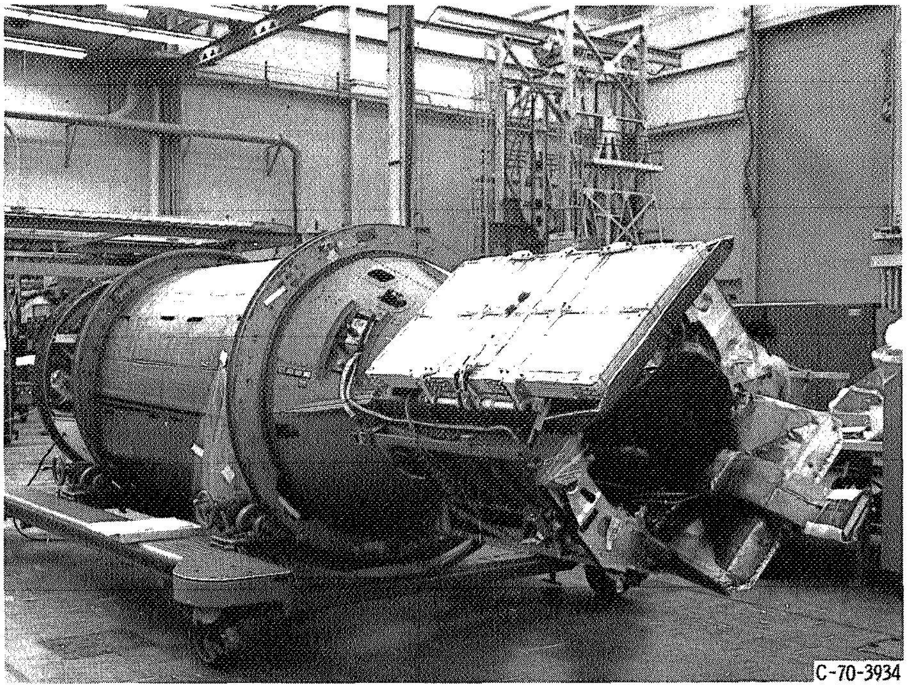


Figure 18. - Solar array in stowed position mounted to Agena in horizontal position.

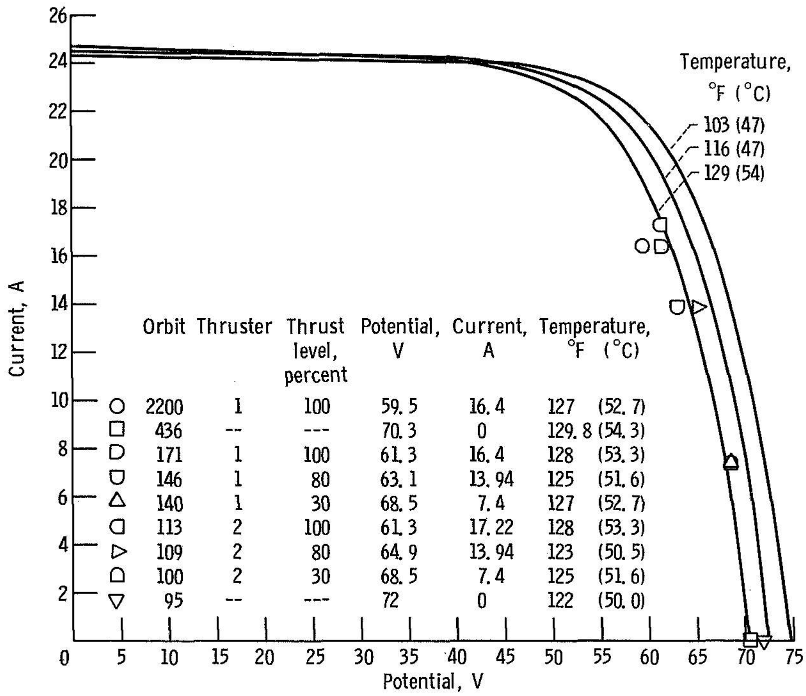


Figure 19. - Thruster array characteristics for spring launch. Solar input, 410 Btu per square foot - hour (129.2 mW/cm^2); array temperature, 116±13° F (47±7° C).

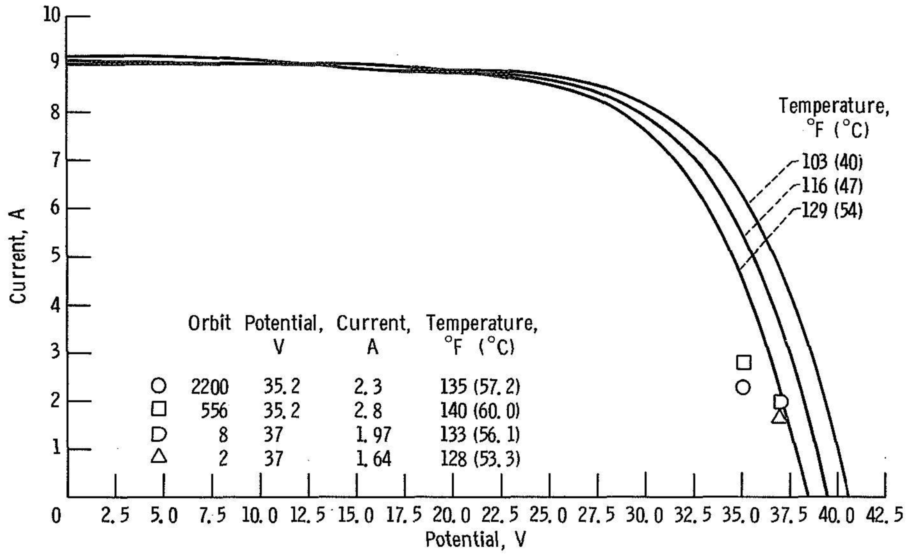


Figure 20. - Housekeeping array characteristics for spring launch. Solar input, 410 Btu per square foot - hour (129.2 mW/cm^2); array temperature, 116±13° F (47±7° C).

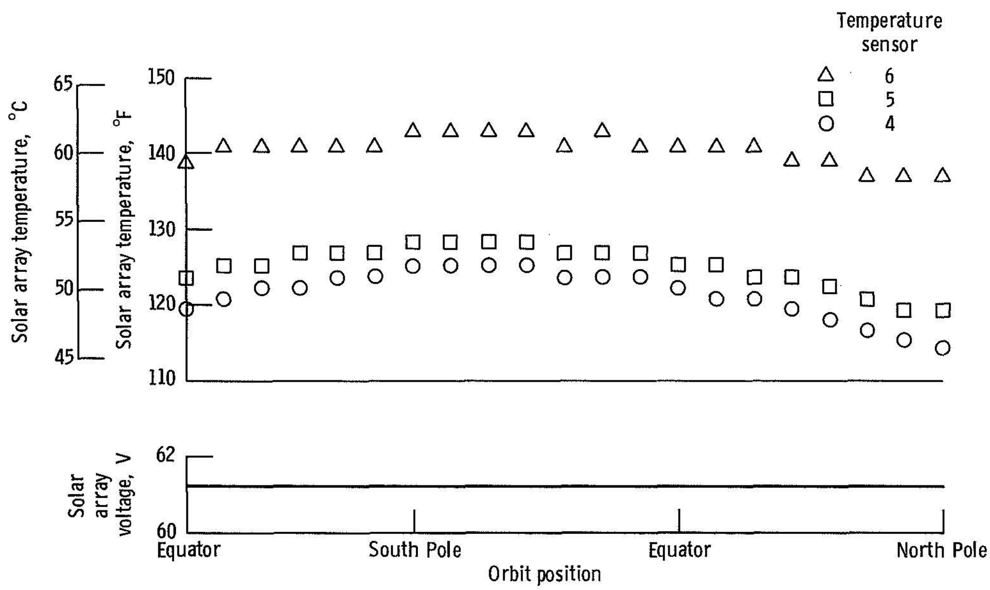


Figure 21. - Solar array temperatures and thruster array voltage for February 18, 1970. Spacecraft angle, 21°; solar flux, 418.5 Btu per square foot - hour (131.9 mW/cm^2).

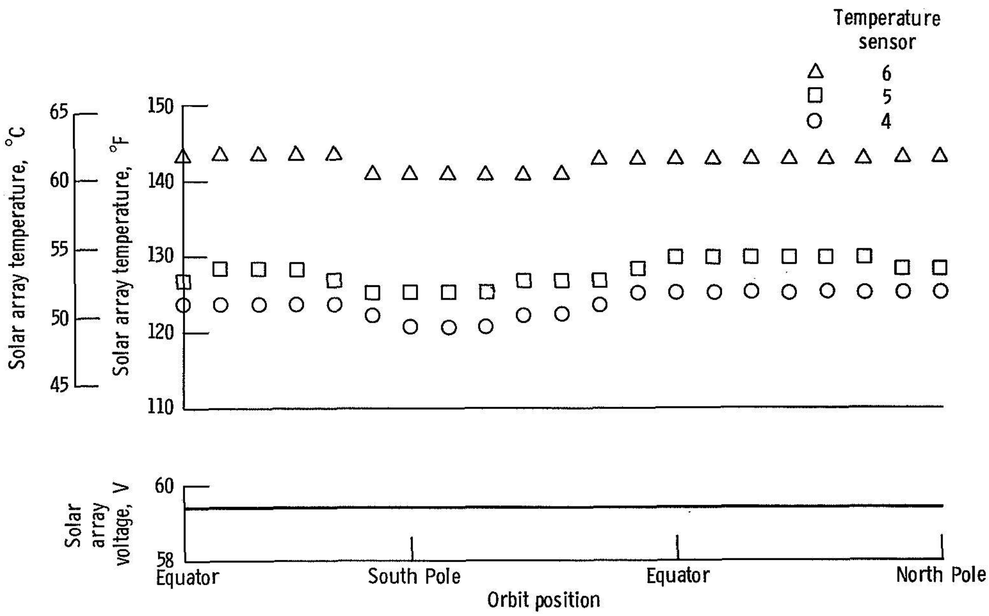


Figure 22. - Solar array temperatures and thruster array voltage for April 28, 1970. Spacecraft angle, 6°; solar flux, 430 Btu per square foot - hour (135.5 mW/cm²).

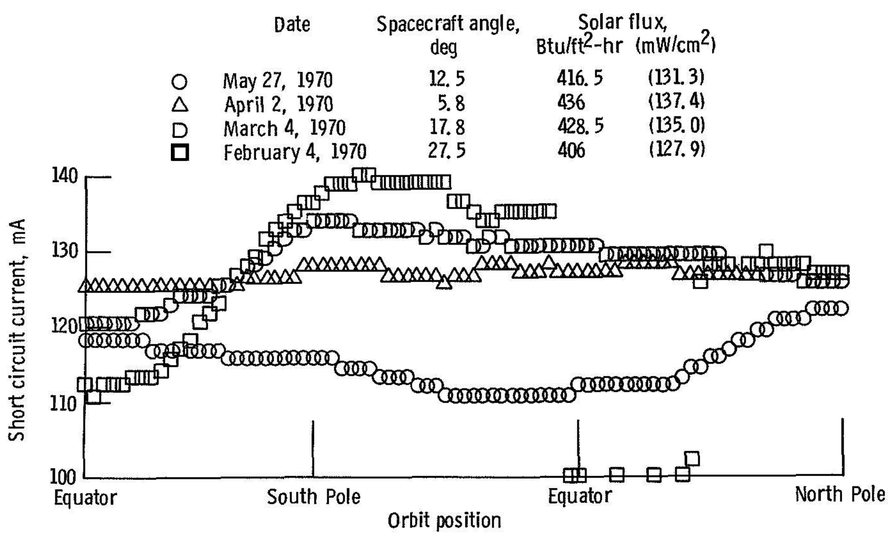


Figure 23. - Short circuit current plotted against orbit position for periodic orbits throughout mission.

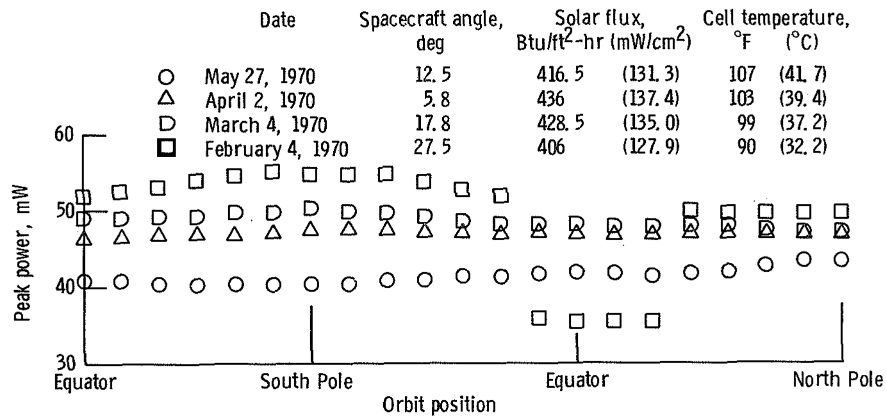


Figure 24. - Peak power plotted against orbit position for periodic orbits throughout mission.

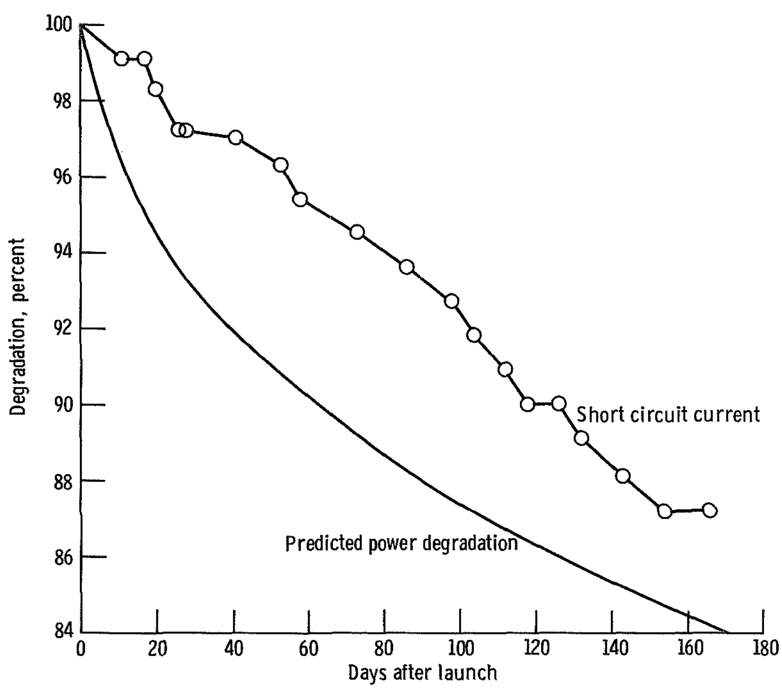


Figure 25. - Percentage of degradation for short circuit current monitor for SERT II mission.

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