# SERT II: Solar Array Power System

G. A. D. Shaw* and J. D. Falconer†
*Lockheed Missiles & Space Co., Sunnyvale, Calif.*

**The solar array power system placed into a sun-synchronous orbit on the SERT II (Space Electric Rocket Test) Spacecraft was capable of providing over 6 × 10⁶ w-hr of electrical energy over a period of 6 months. The 56-v, 1100-w section output is provided directly to a power-conditioning unit with no batteries in the system. The 28-v, 180-w section provided the power to the spacecraft for housekeeping and auxiliary experiments. The deployment technique, which uses springs in each leaf joint to extend the folded array into a flat fixed position, and redundant electrical pyrotechnic circuits to enhance probability of deployment, is discussed.**

## General Mechanical Design

THE SERT II (Space Electric Rocket Test) solar array system is assembled as two independent array wings. Each wing consists of a mounting bed, actuator assembly, release mechanism, 45 solar panels with support structure, and pyrotechnic operated pin pullers. A unique release mechanism uses a simple clevis fitting in combination with two pin pullers to achieve active mechanical redundancy. Actuation of either pin puller releases a restraining tension element permitting a slider crank device to extend and angularly position the 45 solar panels into their final location. Each rotating or sliding joint has either the shaft or the journal coated with a baked-on, solid film lubricant to minimize friction and prevent electrochemical action. The exposed surfaces of the structure are finished with thermal control white paint to maintain temperature below 140°F throughout the flight.

The extendible arm (Fig. 1) is a "lazy-tong" type mechanism, consisting of a number of connecting links. Half the links (leaves) are hinged together in one continuous assembly, providing support for the collector panels; the others (inriggers) are hinged together and pivot about pins fixed within each leaf assembly. When the retaining torque bar/side clamps are released by the action of two squib-operated pin pullers, the arm automatically extends to a prealigned position. The extendible arm assembly is powered entirely by springs controlled by liquid damping. When fully extended, the leaf sections are in line, forming one planar surface; the inriggers remain at an angle of 150° with each other (15° with the leaves). With this linkage arrangement, the assembly is less flexible and the leaves are held in a flatter position than would be possible with the inriggers in the leaf plane. In the folded position, the array arm envelope is made as flat as possible by nesting the inriggers into their respective leaves.

The inboard leaf is rotated about a fixed pivot on the support assembly. The inriggers are attached through a single fitting to the slider. During the cycle, the actuator rotates a bell crank, which forces the slider along the support assembly tube until the full extension position is attained. The actuator controls the extension rate and complements the array arm forces when they are approaching their minimum output. A fluorosilicone fluid with stable temperature characteristics is the damping medium. Rate control is adjusted by throttling the fluid flow with a needle valve within the piston rod. A compression spring surrounding the actuator, aided by the mechanical advantage of the mechanism, complements the forces built into the arm assembly.

The mounting bed supplies the ascent protection for the arm assembly and deployment mechanism. It mounts on the Agena vehicle aft equipment rack prealigned to the geometric axes. It can be considered as the container for the array arm and deployment mechanism and consists of a frame and two retaining clamps. The weight of the completed array system (two complete wings) is 415 lb.

## Electrical Description

The solar array was designed to provide power to the Spacecraft Support Unit (SSU) and spacecraft for 6 months.[1] Panels on the two solar array arms were wired into a thruster section and a housekeeping section. The former provides power to the power conditioner in the spacecraft for ion engine operation, and the housekeeping section supplies power to the housekeeping equipment in the SSU and spacecraft. The techniques, concepts, and components of the array used to meet the electrical requirements for the SERT II mission were flight-proven and qualified.

The thruster section required 1100 w minimum at 56 vdc nominal (+28 vdc and −28 vdc referenced to the array center tap); the housekeeping section, 180 w minimum at 28 vdc nominal. This power was required throughout the minimum 6-month mission at air mass zero (AMO) under worst-case conditions of array temperature, radiation degradation, and a sunline incidence angle of 16°. However, a 12-month period was used in design to permit extended mission operation. Table 1 summarizes the SERT II array power capabilities for ideal 25°C, AMO conditions and for initial and six-month orbital average conditions (which would provide $6.1 \times 10^6$ w-hr). Predicted temperature, sunline angle, and radiation degradation values have been used as the basis for calculations. Spacecraft nominal requirements have been included for reference.

An existing Lockheed Missiles & Space Company (LMSC) designed solar collector panel as shown in Fig. 2 was used in design of the array arms comprised of 370 2-cm × 2-cm cells

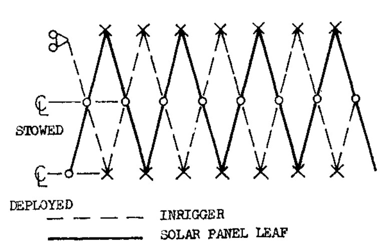


Fig. 1 Solar array geometry.

Presented as Paper 70-1159 at the AIAA 8th Electric Propulsion Conference, Stanford, Calif., August 31–September 2, 1970; submitted November 5, 1970; revision received November 30, 1970.

* Group Engineer. Member AIAA.
† Senior Design Engineer.

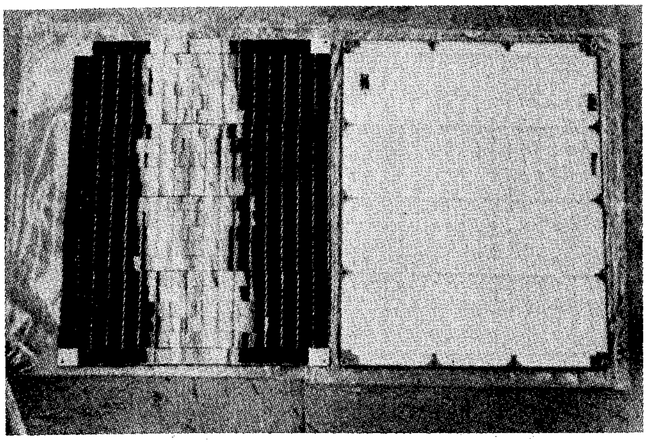

**Fig. 2 Solar photovoltaic collector panel.**

on 5-cell kovar submodule substrates and a die-cast magnesium structural substrate. Each of the two arms was three panels wide and 15 panels long for a total of 45 collector panels per array arm. This array size and configuration has successfully flown on several previous Agena missions. These orbits subjected the solar array to temperature extremes ranging from −200°F to +150°F and were also in the intense Van Allen radiation belt. The SERT II twilight (sun synchronous) orbit is very favorable in that array temperature will be less extreme even when eclipsing at certain orbit inclinations (see thermal design).

A careful evaluation of the parameters, time, altitude, inclination, radiation, and temperature showed all design factors to be encompassed by the existing panel design. Parameters not satisfied were those of power and voltage output.

## Solar Cells, Assemblies and Radiation Degradation

The original array design utilized 10-ohm-cm cells, which provided per-panel outputs of 23 v. The thruster system desired to operate at a nominal 63 v, which can be obtained from 2 panels in series. This increase in the operating point required a cell voltage increase from 389 to 470 mv at the maximum power operating point. This operating point requirement dictated a selection of solar cells with a nominal resistivity of 2 ohm-cm. The new cell characteristics are given in Table 2, which shows the nominal 2-ohm-cm cell together with panel output power at 25°C-AMO conditions.

The 20-mil-thick, fused-silica cover slide serves as an optical filter limiting the spectral transmission from about 0.4 to 1.0 $\mu$, which is the active conversion region for the solar cells. (Radiation outside this spectrum is reflected, thus removing infrared heat which would otherwise cause a degradation of power due to increased temperature.) It also protects the cell from electron and proton radiation fluxes as discussed later.

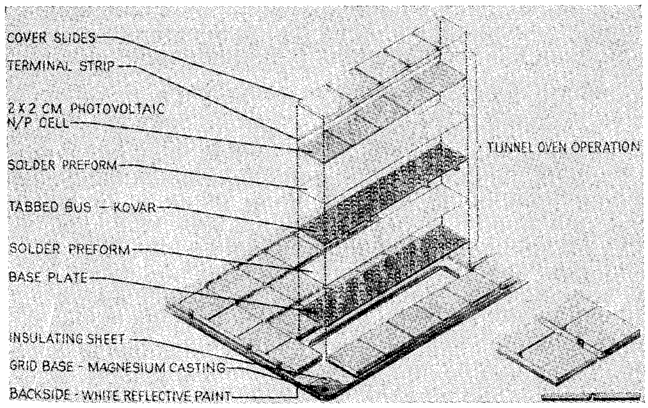


**Fig. 3 Solar photovoltaic collector panel assembly.**

**Table 1 SERT II solar array power capabilities, w**

| Condition[a] | Single panel | Thruster system | House-keeping system |
|---|---|---|---|
| 25°C AMO, no degradation | 20.4 | 1750 | 286 |
| Initial orbit; 45°C; $\beta = 0$ | 19.4 | 1474 | 272 |
| 6-mo orbit average, 45°C, $\beta = 16$, $\alpha = 15\%$ | 15.8 | 1201 | 221 |
| S/C requirements | — | 1100 | 180 |

[a] AMO = air mass zero; $\alpha$ = radiation degradation; $\beta$ = sunline angle.

Solar cells and cover slides are assembled on 5-cell substrates by the sequence illustrated in Fig. 3. The base plate provides a conductive support surface, while a tabbed bus provides the means of interconnecting submodules into series strings. A solar collector panel is comprised of 74 five-cell submodules wired in series and assembled onto an insulated, die-cast magnesium grid. The wiring is configured to avoid creation of current loops which would cause significant vehicle torques when interacting with the Earth's magnetic field (Fig. 4).

For a 1-yr mission from a 1970 launch, the combined radiation degradation factors in a 99° inclined circular polar orbit of 600-naut-mile altitude contributed by both the trapped proton and solar-flare proton fluences are estimated to be 0.89, 0.87, 0.855, and 0.838 after 3, 6, 9, and 12 months, respectively. The degradation contributed by the incident electrons with energies exceeding 0.35 and 0.5 Mev can be considered negligible in comparison to that for the trapped protons.

## Instrumentation

Instrumentation for the solar array system consists of a) electronic "tell-tales" to indicate events associated with deployment and b) sensors which provide temperature and power characteristics of the array assembly. Deployment monitoring consists simply of a switch whose normally-closed contacts were adjusted to open when the solar array arm deployed in excess of 95% of the full-open position. The switch position is signal-conditioned in the Agena and returned as multi-level voltage on the Agena telemeter. Both monitors

**Table 2 SERT II solar cell and panel power output**

| | |
|---|---|
| Solar collector | |
| Cell characteristics | 128 ma ave. at 470 mv, 77°F, 140 mw/cm²<br>124 ma min. at 470 mv, 77°F |
| Assembly test conditions | 77°F, 140 mw/cm² intensity, 34 vdc |
| Diode losses | 0.8-v drop |
| Voltage per cell | $\frac{34.0 + 0.8 \text{ (diode drop)}}{74 \text{ (cells in series)}} = 0.47$ vdc |
| Submodule current | 128 × 0.94 (assay loss) × 5 (cells in parallel) = 601 ma |
| Average power output | 0.601 × 34 = 20.4 w |
| Minimum power output | 20.4 (124/128) = 19.8 w |
| Collectors (numbers) | |
| Thruster | 38 × 2 = 76 |
| Housekeeping | 7 × 2 = 14 |
| Total | 90 |
| Submodules | 90 × 74 = 6660 |
| Cells (2 × 2 cm) | 6660 × 5 = 33,300 |

**Table 3 Control cell characteristics**

| Instrument | Predicted @25°C | Orbit actual | Predicted range |
|---|---|---|---|
| Open circuit voltage, $V_{oc}$, mv | 567 | 0 | 460–570 |
| Short-circuit current, $I_{sc}$, ma | 132 | 112–148 (125) | 112–148 |
| Maximum power, $I_{mn}$, mw | 58 | 47.5 | 45–56 |

performed flawlessly with extension times for the two arms of 32 and 35 sec.

The remaining instrumentation data are provided as raw data to the spacecraft telemeter, since the Agena telemeter operation is permitted to terminate with decay of Agena vehicle battery power. The raw data are conditioned in the spacecraft and comprise the following instrumentation points: 1) six thermistors, one each on the inboard, midpoint, and outboard sections of each array wing, and 2) a sequence of three control cells on the inboard end of the Agena plus $Y$ axis to measure short-circuit current ($I_{sc}$), open-circuit voltage ($V_{oc}$), and current at maximum power ($I_{mp}$). These cells are identical to production cells used in the manufacture of the array (2-ohm-cm/2-cm²). Nominal output of the control cells at 25°C and AMO conditions is $V_{oc}$ = 567 mv, $I_{sc}$ = 132 ma, and $I_{mp}$ = 128 ma (Table 3 and Fig. 5). Signal conditioning and telemeter processing are accomplished in the spacecraft and are adequate to monitor the control cell outputs over the predicted orbital operating ranges shown in Table 3. The thermistor accuracies are ±2% in the 0° to 350°F temperature range, with signal conditioner limits of 90° to 140°F. Actual operating temperatures were 122° to 126°F, compared to a predicted average value of 132°F for the first weeks of flight.

## Thermal Design

An ascent-phase thermal analysis confirmed that existing thermal protection design for the arrays was conservative. An orbital thermal analysis included an investigation of the change in orbit/solar incidence angles for the 6-month period and took into account changes in orbit altitude during the mission. The transient response of the solar arrays was determined for orbits which contain eclipse periods. The analysis showed that solar panel temperature would remain relatively constant during the total sun orbits with maximum temperatures approaching 140°F. Because of the relatively low thermal capacitance of the panels, temperatures will decrease and increase rapidly when entering and leaving the earth's shadow. The maximum rate of temperature change will occur when the panel first emerges from the shadow. However, no cell bonding problems are anticipated since these rates and temperature excursions are well within the thermal limits encountered by previous flight-proven arrays.

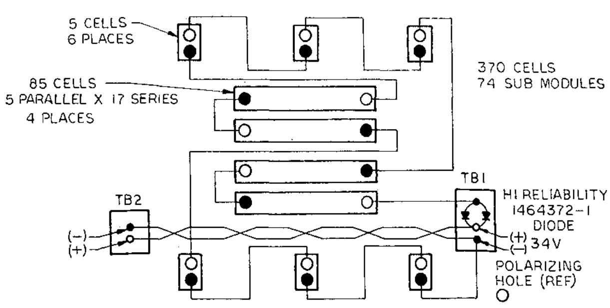


**Fig. 4 Solar voltaic collector wiring diagram.**

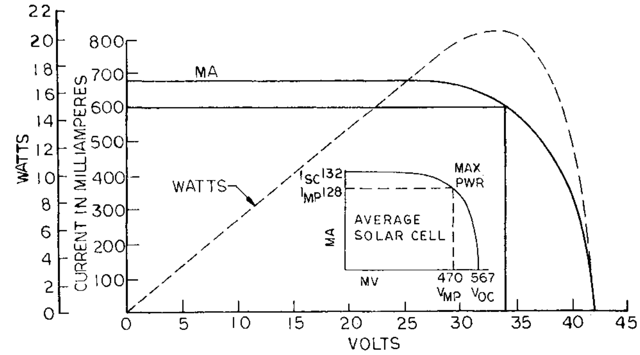


**Fig. 5 Solar collector and cell characteristics (25°C, AMO).**

## Ground Test Program and Flight Results

The tests required to qualify the basic array for the ascent phase and space environment were previously performed by LMSC. Thus, only those tests required to assure that the flightworthiness of the basic design had not been invalidated by design changes required by SERT II mission requirements were performed.

Ten deployments of a modified and refurbished test array were performed to assure proper deployment, repeatibility of deployment time, and panel flatness, and to determine the best location of the actuator adjustment within the mounting bed. Changes to the deployment mechanism design were required because the basic design provided for the arrays to deploy in a drooped and canted position, while the SERT II program required the array to deploy in a plane perpendicular to the sun.

The release mechanism was tested by firing live squibs in flight-type pin pullers. Six tests were performed in both the flight configuration and in failure mode configurations such as one pin puller and a single squib. These tests were required since a new and simpler release mechanism had been designed for the SERT II program which employed active mechanical redundancy.

The following tests were performed to verify that the solar array could withstand the higher dynamic loading anticipated from the Thorad launch vehicle: vibration in three axes, deployment before and after test, power output of a flight solar panel, and electrical continuity.

Flight acceptance testing was performed on each module. Numerous inspections and tests were performed throughout the assembly and installation period. The most important of these tests were: harness resistance/continuity measurements, extension limit switch electrical and mechanical functional tests, thermistor probe continuity tests, solar panel GO/NO GO illumination tests, alignment verifications, solar testing of panels at Table Mountain, and solar panel testing in artificial light. The last two items are worth further comment. All 90 of the flight solar panels were tested under an artificial light source to assure each was capable of delivery of 19.8 w (minimum) of electrical power (Fig. 5), and a number of panels were tested in natural sunlight to verify power capability. All sunlight testing was performed at the Jet Propulsion Laboratory test site at Table Mountain, Calif.

As of November 1970 the solar array had been providing the required power to operate the ion engine via the power conditioning unit,[1,2] and the output degradation was lower than predicted for the program based on preliminary data on the test cells being monitored by NASA.

## References

[1] Kerslake, W. R., Goldman, R. G., and Nieberding, W. C., "SERT II: Mission, Thruster Performance, and in-Flight Thrust Measurements," *Journal of Spacecraft and Rockets*, Vol. 8, No. 3, March 1971, pp. 213–224.

[2] Bagwell, J. W., "SERT II: Power Conditioning," *Journal of Spacecraft and Rockets*, Vol. 8, No. 3, March 1971, pp. 225–230.