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Clementine Gallium Arsenide/Germanium Solar Array

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The Clementine spacecraft was launched in January 1994 to demonstrate advanced, lightweight technologies for the Ballistic Missile Defense Organization. Clementine was the first U.S. spacecraft to orbit the moon since the 1970s. One of the key lightweight technologies was the solar cell array panels. The panels used thin gallium arsenide on germanium solar cells mounted on lightweight graphite epoxy/aluminum honeycomb core substrates. The combination provides an array specific power of 53 W/kg. Four deployable panels were fabricated to provide 380 W @ 30 V dc. The small conical-shaped solid rocket motor housing or interstage was also populated with solar cells to provide additional power during the low-Earth-orbit part of the mission. The design of the deployable and interstage arrays, qualification test results, and on-orbit performance data are presented.

Introduction

IN early 1992, a commitment was made to design, build, test, and launch a small spacecraft. The spacecraft was to be a test bed for new sensors and other lightweight components in development for future missile detection systems. Because the spacecraft sensors were to perform a complete geological mapping of the lunar surface, the mission was named Clementine after the miner's daughter, in this case a moon miner's daughter. Given only two years to accomplish the task, the engineers concentrated on designing a small, lightweight, efficient spacecraft. Initial spacecraft designs were small and light enough to use a Pegasus launch vehicle. However, as the spacecraft design matured, the size, weight, and power budget increased beyond the requirements of the Pegasus to the capabilities of a Titan II launch vehicle. Design trade studies showed that a gallium arsenide/germanium (GaAs/Ge) solar cell array had the highest power density and would be an enabling technology for the mission. This article discusses the design of the Clementine GaAs/Ge solar cell arrays, the results of qualification tests, and it presents on-orbit performance data.

Solar Array Design

Electrical Energy Requirements

Even though the Clementine solar cell arrays were to provide electrical power for the entire mission, the deployable solar arrays were sized to support the lunar mapping mission phase. Electrical energy requirements in units of watt-hours (W-h) for each mission phase are listed in Table 1.

In the lunar orbit, the solar array must support the load in sun plus recharge the battery before the next eclipse period. With 3.83 h of sunlight during the 5-h lunar orbit period, the array was required to generate 1318 W-h/3.83 h = 344 W.

Electrical Power Subsystem Design

The Clementine electrical power subsystem is a simple, lightweight, direct energy transfer design.[1] In Fig. 1, a nickel–hydrogen (NiH_2) common pressure vessel (CPV) battery is directly connected to the bus to regulate the bus voltage to 30 $\pm$ 6 volts direct current (V dc). Electrical power from the solar arrays charges the NiH_2 CPV battery and is distributed to the spacecraft and sensor loads. The battery charge control method is based on the linear increase of the NiH_2 battery's gas pressure with state of charge. Fully discharged, the battery's pressure is 0-Pa gauge (0 psig). Fully charged, the battery's pressure is 4.1-MPa gauge (600 psig). When the charge control electronics measured a battery pressure less than 4.1-MPa gauge (600 psig), then the electronics would regulate the battery charge current to 3.0 A. When the charge control electronics measured a battery pressure greater than 4.1 MPa (600 psi), then the charge current was regulated to a trickle charge of 0.150 A. Regulation of the charge current was achieved by pulse width modulating (PWM) a bank of field effect transistor (FET) switches wired in series between the solar array strings and the spacecraft bus. Each FET was connected to five parallel connected strings.

Solar Array Sizing Calculations

The solar array sizing calculations listed later use the electrical characteristics of a 4.0 $\times$ 4.0 $\times$ 0.0139 cm GaAs/Ge solar cell. Calculations using a comparable silicon (Si) solar cell were also performed, but are not listed.

Radiation Degradation

The Clementine spacecraft was designed to spend only one week in a low-Earth-orbit (LEO) before traveling to the moon. Therefore, degradation from the radiation belts was predicted to be very low. For 1-MeV equivalent electrons, a value of 3 $\times$ 10^{13} e/cm^2 was used in the solar array sizing calculations. A coverglass thickness of 3.5 mil was selected as a minimum form of protection and mass. Degradation from solar flare protons was not included.

On-Orbit Panel Temperatures

Panel temperatures in LEO were estimated to be 100°C. Panel temperatures in lunar orbit were estimated at a maximum of 60°C.

Minimum Array Voltage

From Eq. (1), the minimum array voltage V_A was calculated by summing the voltage losses between the solar array and the

Table 1 Electrical energy requirements

Mission phase	Period, min	Load in sun, W-h	Battery recharge, W-h
LEO	90	180	67
Lunar	300	1003	315
Asteroid encounter	Continuous	1003	n/a

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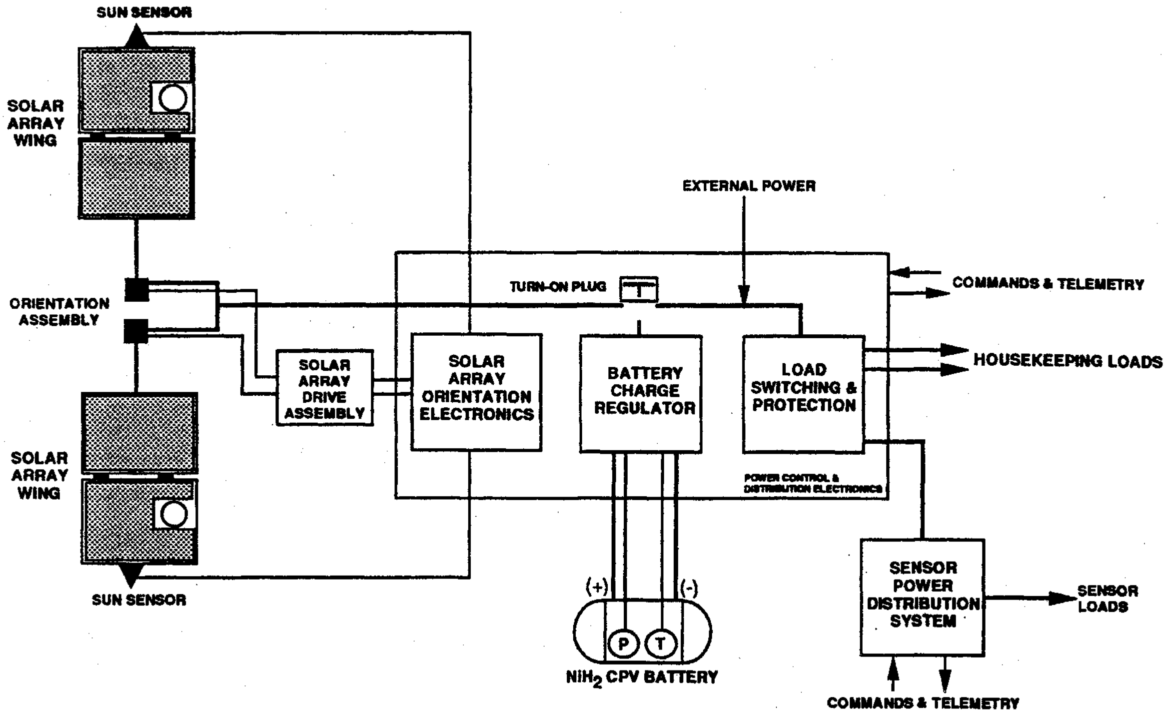

Fig. 1 Clementine electrical power system.

battery and adding that sum to the calculated voltage of a cold NiH_2 battery[2]:

$$V_A = V_{\text{Cold Battery}} + V_{\text{Diode}} + V_{\text{FET Switch}} + V_{\text{Wire Harness}} \quad (1)$$

$V_{\text{Cold Battery}}$ = voltage of cold NiH_2 battery, 22 cells × 1.64 V/cell, 36.08 V
$V_{\text{Blocking Diode}}$ = voltage drop across blocking diode, 0.80 V
$V_{\text{FET Switch}}$ = voltage drop across FET switch, (0.25 Ω) × (0.45 A), 0.113 V
$V_{\text{Wire Harness}}$ = voltage drop across array wire harness, 0.50 V
V_A = 37.41 V

Number of Solar Cells in Series

Using the minimum array voltage, the number of solar cells required in series N_s was calculated in Eqs. (2) and (3)[2]:

$$N_s = V_A / V_{\text{mp}\phi\text{HOT}} \quad (2)$$

$$V_{\text{mp}\phi\text{HOT}} = V_{\text{mp}\phi} + \Delta V_s' + B_{vp}(T_{\text{op}} - T_0) \quad (3)$$

$V_{\text{mp}\phi}$ = maximum power voltage after radiation, $V_{\text{mp}} \times \phi$, 0.890 V × 0.98
$\Delta V_s'$ = solar array off-pointing cosine factor, −0.062 (sun vector 30 deg from normal)
T_{op} = array operating temperature, 60°C
T_0 = calibration temperature, 28°C
B_{vp} = voltage–temperature coefficient, −0.0019 V/°C
$V_{\text{mp}\phi\text{HOT}}$ = 0.749 V
N_s = 37.41 V/0.749 V/cell
N_s = 50 cells/string

The number of cells in series was later revised to 52 to provide a margin for the hotter LEO temperatures.

Number of Strings in Parallel

From the required array power of 344 W, the number of solar cell strings in parallel Np was calculated as in Eqs. (4) and (5)[2]:

$$Np = I_L / I_{\text{mpav}} \quad (4)$$

Table 2 GaAs/Ge vs Si array area and mass comparison

Solar cell	Array area, m^2	Array mass, kg[a]
Si	4.28	5.60
GaAs/Ge	2.04	3.88

[a]Does not include substrate mass.

$$I_{\text{mpav}} = I_{\text{mp}\phi} \times S' \times [1 + B_{IP}(T_{\text{op}} - T_0)] \times F_m \times F_{\text{sh}} \quad (5)$$

$I_{\text{mp}\phi}$ = maximum power current after radiation, $I_{\text{mp}} \times \phi$, 0.490 A × 0.975, 0.478 A
S' = solar array off-pointing cosine factor, cos Γ (Γ = 30 deg), 0.866
B_{IP} = current temperature coefficient, 0.00056/°C (0.98)
F_m = cell mismatch (0.99) × minimum sun (0.965) × uv degradation (0.98) × micrometeorite damage (0.99)
F_m = 0.927
F_{sh} = array shadowing factor, 1.0
I_{mpav} = average solar cell current at maximum power, 0.400 A
I_L = P_A/V_{CB}, 344 W/36.08 V, 9.53 A
$\therefore Np$ = 9.53 A/0.400 A/string
Np = 24 strings

The total number of solar cells required was $Ns \times Np = 52 \times 24 = 1248$ cells. A packing factor of 95% allows for 603, 4 × 4 cm solar cells per square meter. The total solar array area is then: 1248 cells/603 cell/m^2 = 2.04 m^2. In Table 2, the GaAs/Ge array is compared to an array using 0.0203-cm-thick Si solar cells for total array area and mass.

Solar Array Manufacturing

From the sizing calculations, the number of solar cell strings was distributed onto two deployable wings. Each wing consisted of an inboard and an outboard panel. Each inboard panel is populated with five series strings of 52 solar cells/string. Each outboard panel is populated with seven strings of 52 solar cells/string. The wing configuration is depicted in Fig. 2. The procurement package, complete with drawings and specification, was released in November 1992. In the procurement package the contractor was required to populate five (four flight, one qualification) government furnished substrates with 4 × 4 cm GaAs/Ge solar cells to the specification. Features of the flight solar cells are listed in Table 3.

From LEO, the Clementine spacecraft was to be propelled to the moon with the firing of a small solid rocket motor (SRM). To achieve a stable attitude for the SRM firing, the spacecraft was spun up to 80 rpm around the spacecraft x axis. If the arrays had been deployed in LEO, the solar array drive motors and hinge hardware could not support the force applied from the 80-rpm spin. Therefore, the arrays wings remained stowed against the spacecraft in LEO before and during the SRM firing. As the spacecraft power requirements for LEO increased beyond the capability of the stowed arrays, it was

Table 3 Flight solar cell features

Cell type	GaAs/Ge
Process	MOCVD[a]
Area	4 × 4 cm
Thickness	5.5 mil
Coverglass	3.5 mil CMX
Antireflection coating	Yes
Coverglass adhesive	DC 93-500
Bond adhesive	RTV 560
Interconnects	Kovar soldered

[a]Metallic oxide chemical vapor deposition.

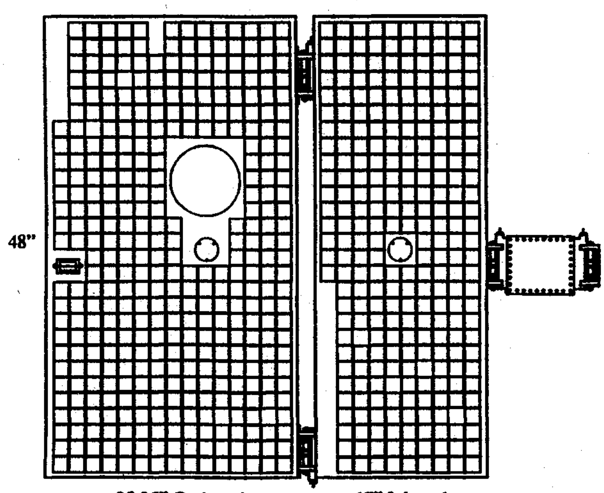

Fig. 2 Clementine solar array wing.

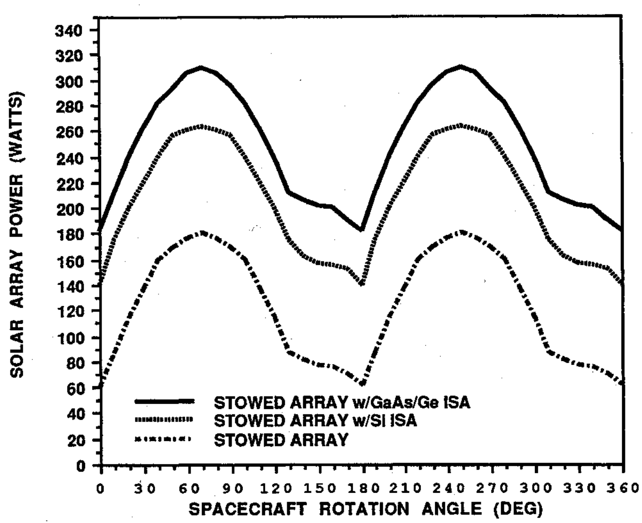

Fig. 3 Stowed solar array output.

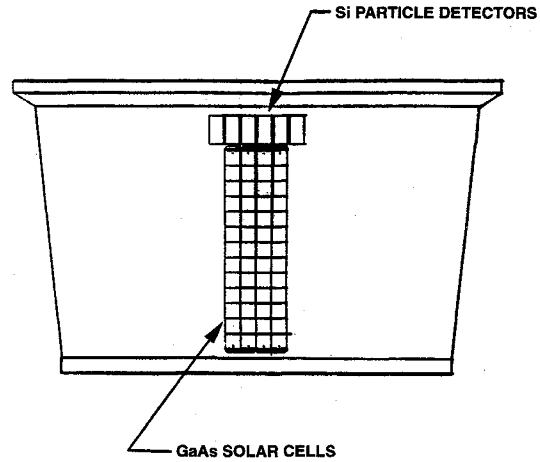

Fig. 4 Clementine qualification interstage solar array.

apparent that additional array area was required. However, with the arrays stowed against the body, no additional area was available. A large area was available on the SRM housing or interstage (ISA) that provided the interface between the spacecraft and the launch vehicle adapter. The ISA is a conical-shaped structure constructed of a graphite–epoxy conic section with machined aluminum end rings. The rings provide the mounting interface between the spacecraft and the launch vehicle adapter and the graphite conic section provided mounting points for the SRM. Two designs of an ISA populated with GaAs/Ge solar cells or Si were studied. The GaAs/Ge ISA array provided the most power. Figure 3 shows the combined stowed and ISA solar array output over one spacecraft revolution for both designs.

Eighteen strings of 52 solar cells in series were carefully adhered to the complex conical substrate. The qualification ISA solar array with just one string of solar cells is depicted in Fig. 4.

Qualification Test and Integration

For the purpose of qualification tests, one outboard panel substrate and the qualification ISA were populated with one string of solar cells. After the initial receiving and inspection, the qualification outboard panel was subjected to 13 thermal vacuum cycles over a temperature range from −180 to +100°C. After the test, the panel was inspected for anomalies and no discrepancies were discovered. A posttest current–voltage (I–V) curve was measured and showed negligible degradation from the thermal cycling. In Fig. 5, the beginning of life (BOL) I–V curve is compared to the posttest curve. The panel was then released to the mechanical integration team for assembly with the engineering model spacecraft. On the spacecraft, the panel was subjected to random vibration, acoustic vibration, and pyroshock tests. No discrepancies were observed during the posttest inspection. The qualification ISA was subjected to the similar qualification tests and completed all tests with no discrepancies.

Flight Solar Panels

The flight solar panels were shipped in September 1993, only seven months after the award of contract. Remarkably, the flight ISA array arrived in August 1993. This was even more outstanding given the difficult job of applying solar cells to the conical-shaped ISA. Moreover, in May, the sponsor decided to add a particle detection experiment to the ISA. A small area between the solar cell strings and the spacecraft mounting ring was available, and in that space, 54, 4 × 8 cm Si detectors were mounted. Despite being extremely sensitive to handling and difficult to work with, the detectors were

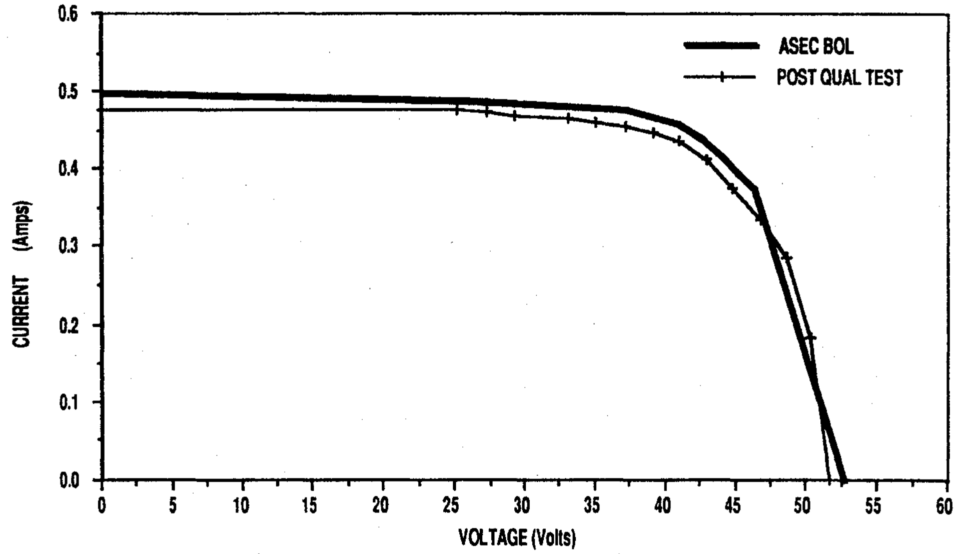

Fig. 5 Outboard qualification panel I–V curves.

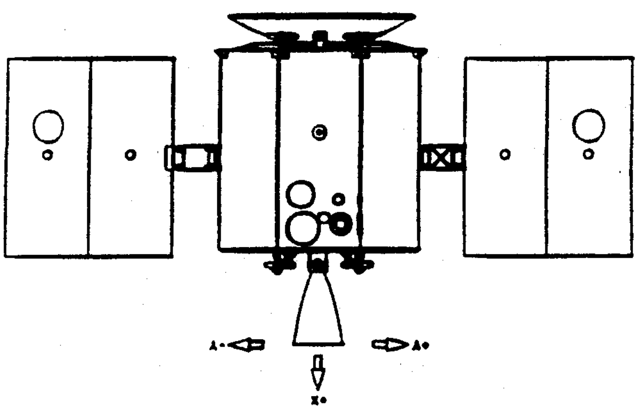

Fig. 6 Clementine spacecraft with deployed solar panels.

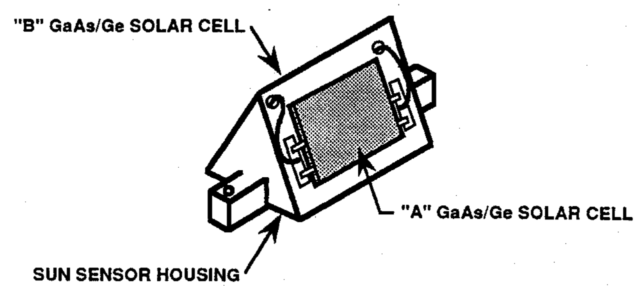

Fig. 7 Clementine sun sensor.

mounted and the array arrived ahead of schedule. The particle detector technicians and scientists spent one week wiring no. 30 gauge wires from the detectors to a data acquisition box mounted on the ISA.

The flight solar panels and ISA were individually thermal vacuum tested for four cycles over a temperature range from −100°C to +60°C. Next, each set of inboard and outboard panels was then assembled with hinge and deployment mechanisms and hardware and wiring from the outboard panel to the inboard panel was completed. The wings were temporarily installed on the flight spacecraft to complete the wiring from the inboard panel to the solar array drive motor. The spacecraft along with the two solar array wings is shown in Fig. 6.

Solar Array Orientation System

The solar panel wings were designed to rotate with one degree of freedom around the spacecraft *y* axis. The array orientation system used four sun sensors and a control electronics to command movement of the solar array drive motors. The sun sensors were mounted on the outboard solar panels with one sensor on each face of the panel. The sun sensor is depicted in Fig. 7. The two solar cells mounted on the front (solar side) were designated as the A and B sensors. The two cells on the rear sensor were designated as the C sensor. The orientation algorithm used the cosine characteristic of the solar cell current output. The current outputs of each solar cell were amplified and compared in the control electronics.[1] If the outputs of the A and B sensors were equal, then the sun vector was normal to the solar panel. If the outputs were not equal, then the control electronics rotated the panels in the direction of the sensor with the higher current output. The electronics were set up with a dead band such that the sun vector was required to move at least 10 deg in one direction before commanding the drive motors to orient the solar array. If the output from all four sun sensors was zero, then the electronics assumed that the array was not illuminated, and kept the array stationary. Verification of the orientation system was achieved by illuminating the sun sensors with a hand-held solar simulator.

On-Orbit Performance

The Clementine spacecraft was launched on Jan. 25, 1994. The spacecraft spent one week in LEO before leaving for the moon on Feb. 2, 1994. Output from the solar arrays varied for the first few days until the spacecraft attitude control system was able to orient the spacecraft normal to the sun. Unlike most spacecraft, the Clementine attitude control system did not have magnetometers or horizon sensors. For attitude control and navigation Clementine used lightweight inertial measurement units (IMU) and star tracker cameras. The IMUs provided spacecraft pitch, roll, and yaw rates and the star tracker cameras took pictures of the stars for processing by the spacecraft computer to compute the spacecraft position in inertial space. However, in LEO neither sensor could provide telemetry data on the location of the sun with respect to the spacecraft. Luckily, the output from the solar array sun sensors could do that. By plotting the output of the sensors on both wings, flight operations engineers were able to determine where the sun was and how the spacecraft was tumbling.

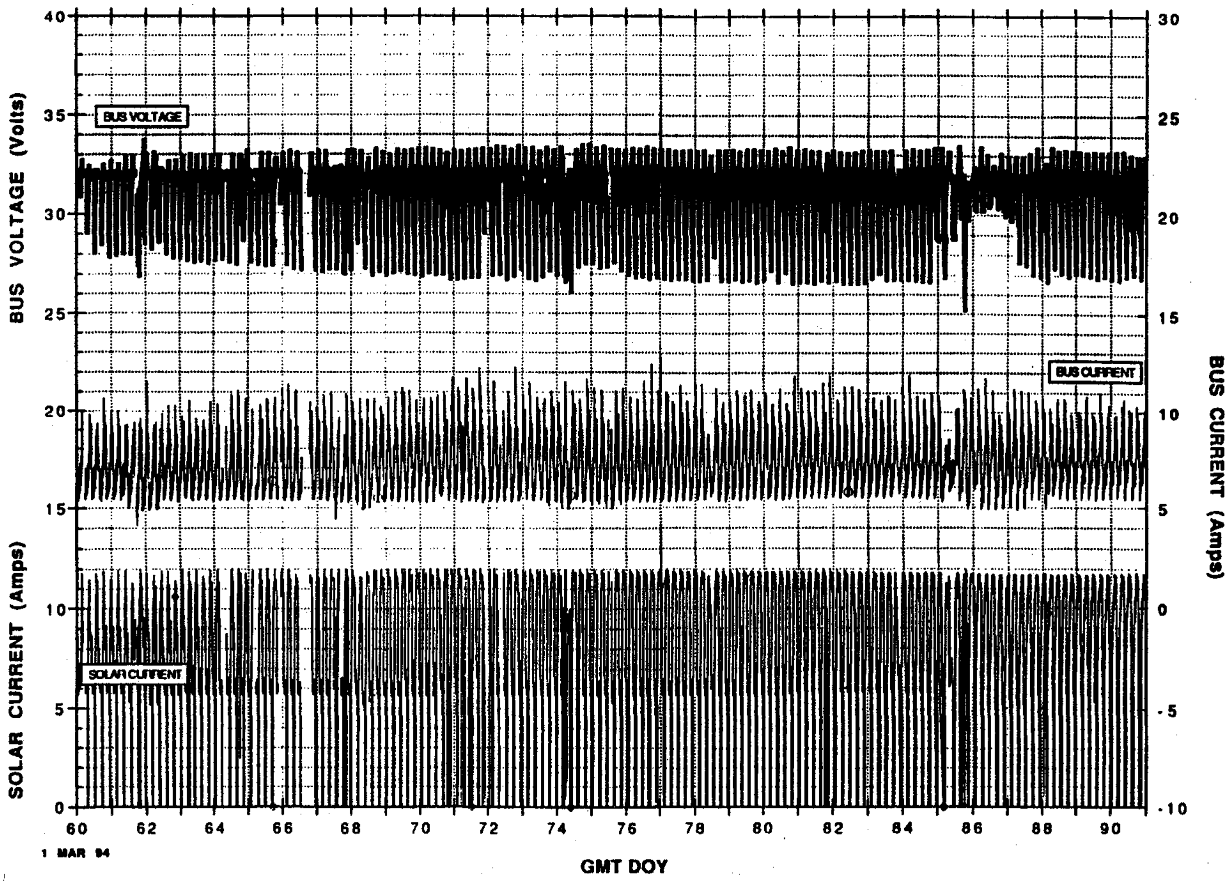

Fig. 8 Clementine lunar solar array performance.

Several days after the SRM firing, the solar arrays were deployed and the ISA detached from the spacecraft. As the spacecraft raced toward the moon, the ISA remained in a highly elliptical 72-h orbit around the Earth. Array performance at the moon was excellent, shown in Fig. 8, providing more than adequate power to complete a total mapping of the lunar surface in 68 days. Thermal cycling of the array in lunar orbit was severe. Ramp time of the lunar thermal cycle from +60°C to −160°C occurred in less than 25 min.

Conclusions

Thin, lightweight GaAs/Ge solar cell arrays were an enabling technology for the Clementine mission. Use of GaAs/Ge solar cells for both the deployable and the ISA solar arrays provided the Clementine mission with substantial power to accomplish the mission. The panels endured a rigorous qualification test program and severe conditions at the moon, including over 330 thermal cycles from −160°C to +60°C, and they performed successfully.

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[2]"Solar Cell Array Design Handbook, Volumes 1 & 2," Jet Propulsion Lab. SP 43-38, Pasadena, CA, 1976.