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# Orbital performance of AMPTE/CCE nickel–cadmium batteries with a large number of charger-induced, very small, charge/discharge cycles

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**Abstract**

The AMPTE/CCE spacecraft used switching regulators to short-circuit the excess solar array current and control the charge to each of its two 28 V, 4 Ah nickel–cadmium batteries. This digital regulation of the battery overcharge current caused approximately $10 \times 10^6$, very small battery charge/discharge cycles during the long sunlight periods of the highly elliptical, 24 h orbit; a unique system influence that did not significantly degrade battery performance during the mission life. The occurrence of three very long eclipses at the beginning, middle and end of the mission, provided a rare opportunity to analyze the performance of the batteries for the mission life. This paper presents the power system design, and the battery performance for about five years of mission life. © 1997 Elsevier Science S.A.

*Keywords:* Orbital performance; Nickel-cadmium batteries

## 1. Introduction

The flight segment of the Active Magnetospheric Particle Tracer Explorers (AMPTE) Program consisted of three spacecrafts, which studied the solar wind and magnetosphere by the release and monitoring of lithium and barium tracer ions. One of these spacecrafts, the Charge Composition Explorer (CCE), developed by the Johns Hopkins University's Applied Physics Laboratory (JHU/APL) for NASA's Goddard Space Flight Center, operated in a 24 h elliptical orbit for about five years: from shortly after its launch on 16 August 1984 until mid-June 1989, when radiation-induced damage to the command system would no longer permit normal operation of the spacecraft, a period of approximately 1765 days.

The power system, designed to support the spacecraft during the transient phase of orbit acquisition as well as the on-orbit operations, used a solar array and two 4 Ah, 28 V, nickel–cadmium batteries as its primary power supply. To minimize spacecraft internal heat dissipation, a simple, switching battery charge regulator was employed for each battery. During overcharge these regulators caused the batteries to alternately charge and discharge to very shallow depths. Although the precise number of these regulator-induced, shallow discharges is not known, it is estimated to have been on the order of $10 \times 10^6$, without any apparent deleterious effects on battery performance.

## 2. Battery power requirements

The batteries were designed to support the spacecraft electrical loads whenever the load demand exceeded solar array capability throughout launch, the orbit-acquisition and adjustment phase, and during the primary mission. Due to the high (56 000 km) apogee of the elliptical orbit, there were periods of 100% sunlight, alternating with extended periods of eclipsing orbits. Fig. 1 is a plot of the eclipse duration experienced as a function of time after launch [1]. Although most of the shadow periods were less than 30 min, there were three periods with eclipses exceeding 1.5 h. The first such period, 237 days after launch, contained a 1.53 h eclipse. The second long-eclipse, of 2.73 h, occurred approximately half way through the planned four-year mission (768 days after launch). The longest eclipse of 2.95 h came 1677 days after launch, just before the end of the nearly five-year actual mission. The battery voltage–time curves, resulting from the deep discharges during these three eclipses, provide snapshots of the batteries' performance throughout the mission. The batteries were sized for the maximum depth-of-discharge (DOD): analyzed as the minimum load for the 3 h long

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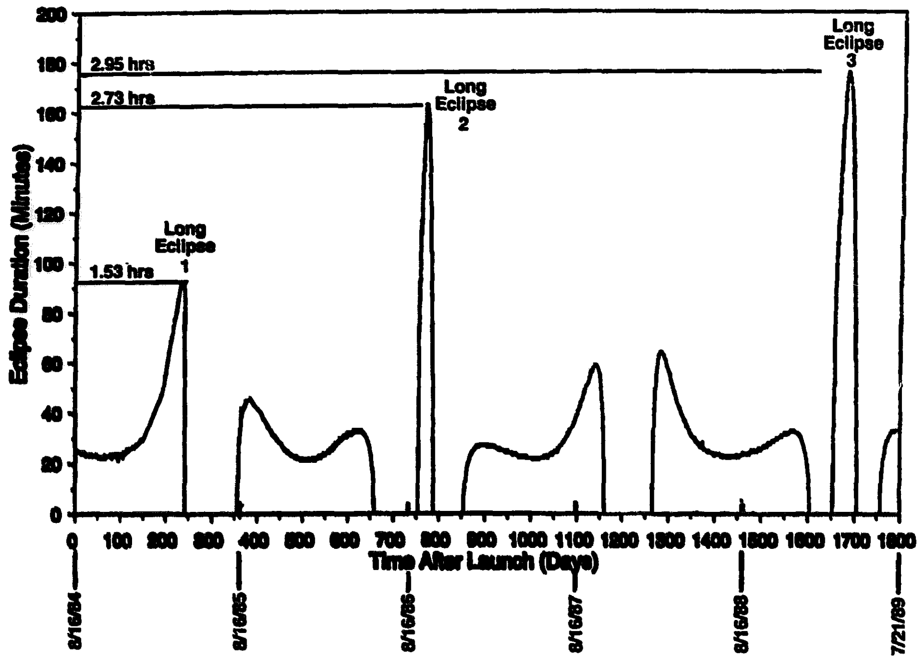


Fig. 1. Eclipse duration vs. time after launch for AMPTE/CCE.

eclipse of the primary mission. Table 1 provides a summary of the battery design drivers [2–4].

## 3. Power system design

The major power system design features are summarized in Table 2. The system, shown in Fig. 2, was an unregulated, Direct Energy Transfer (DET) system, consisting of two redundant subsystems, both sharing a common load through 'oring' diodes. Each subsystem had a solar array, battery, and Battery Charge Regulator (BCR), including an electronic coulometer [1]. The low voltage sensing system and d.c./d.c. converters (not shown) were also redundant.

The four solar panels were arranged into two equal arrays, each of which typically provided current to one battery as shown, but could be switched to the other battery by ground-command transfer of switch S1 or S2 as required. A BCR and its Field Effect Transistors (FETs), connected across each of the sixteen sections of the solar array, limited the array power generation to that required for spacecraft power balance by shunting the excess array current.

The eight circuits in each of these arrays consisted of a single string of 10 Ω cm, n/p doped, single-crystal silicon, 2 cm × 4 cm solar cells connected in series. Each of these eight solar cell circuits, connected in parallel with a BCR-controlled FET switch, was capable of supplying current to the bus. When the FET was controlled to high resistance, full current from the circuit was provided to the bus. Whereas, when the FET was driven to low resistance by the BCR, it shunted all of the solar cell circuit current at low voltage, preventing any current from reaching the bus. This minimized the circuit's power generation and consequent dissipation. Put another way, the FET switch dissipated very little power,

Table 1
Battery design drivers

| | |
|---|---|
| Configuration | Two 4 Ah, 22-cell batteries |
| System | Redundant, 28 V, direct energy transfer (DET) |
| Orbit [2,3] | |
| Initially | 16 h |
| | 28° inclination |
| | 633 × 56850 km |
| Main mission | 24 h |
| | 4° inclination |
| | 1111 × 56880 km |
| Design life [2,3] | 4 years |
| Temperature [4] | |
| Operating | 5 to 15 °C |
| Survival | − 10 to 30 °C |
| Eclipse load | |
| Normal | 97.4 W (3.48 A) |
| Reduced [a] | 51.5 W (1.84 A) |
| Alt. reduced [b] | 30.5 W (1.09 A) |
| Eclipse Period | Depth-of-discharge |
| 30 min | 22% (1.74 Ah at 3.48 A) |
| 60 min | 44% (3.44 Ah at 3.48 A) |
| 2 h | 46% (3.68 Ah at 1.84 A) |
| 2.5 h | 58% (4.60 Ah at 1.84 A) |
| 3 h | 69% (5.52 Ah at 1.84 A) |
| | or |
| | 41% (3.27 Ah at 1.09 A) |

[a] Reduced load: most science experiments off to reduce battery discharge to acceptable level during long eclipse periods.
[b] Alternate reduced load: for further load reduction, telemetry system power drain was minimized, further reducing battery discharge during long eclipses.

[1] The coulometers were designed to be used for battery charge control and/or monitoring, but due to their inaccuracy in measuring the low battery recharge rates, they were not used.

Table 2
Summary of power system characteristics

| | |
|---|---|
| System type | nominal 28 V unregulated direct energy transfer (DET) |
| Bus voltage range | $28 \pm 6$ V |
| Low voltage trip | 22 V |
| Solar array (sum of two arrays) | Four panels, 4 $m^2$ each, 16 solar cell circuits in parallel |
| Solar cell circuits | Single string of solar cells in series |
| Solar cells (total) | 1776 $2 \times 4$ cm silicon, n/p cells |
| Batteries | Two 22 cell, 4 Ah Ni–Cd |
| Charge control | Voltage limiting, non-dissipative by switching solar cell circuits on and off at 1 Hz rate |
| Charge monitor | 2 coulometers |

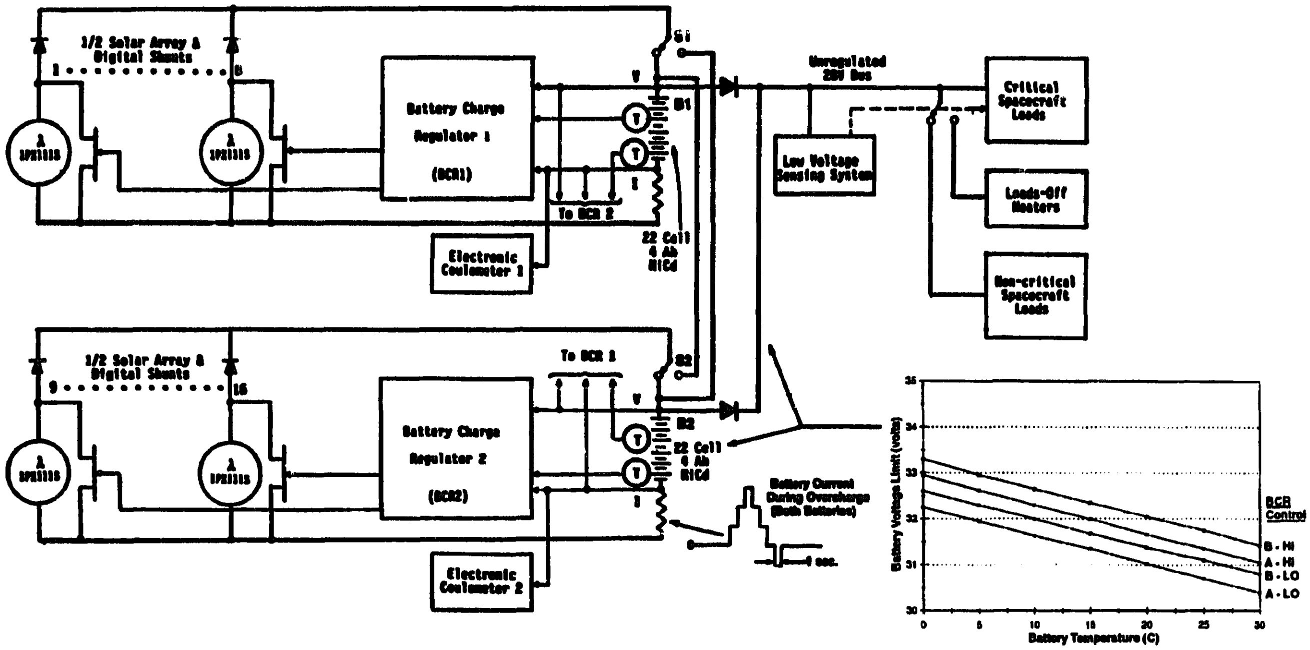


Fig. 2. AMPTE/CCE power system block diagram.

since it was either drawing no current, or it drained the full circuit current at a very low voltage.

Each BCR monitored the current, voltage and temperature of both batteries, but controlled only the battery that was selected by ground command [2]. The BCR then compared the battery's voltage and temperature, to one of four ground-selectable voltage–temperature (*V–T*) curves shown in lower-right of Fig. 2. Although the four voltage levels were provided in the event they would be required to accommodate a possible change in battery limit voltage later in life, the B-LO level was selected at the beginning and used for the entire mission.

To describe the system operation, it is convenient to explain an orbital recharge sequence starting at the end of eclipse. After powering the spacecraft through the eclipse, the batteries are discharged. Upon entering sunlight, the full array current is available to power the load and recharge the battery because the battery recharge voltage is below its BCR voltage limit and all of the FETs are open circuit. After the battery reaches full charge, its charge voltage increases to the voltage limit, signaling the BCR to sequentially switch the FETs on (increment the FETs), decreasing the available charge current, until the charge voltage drops below the voltage limit. In response, the BCR sequentially switches the FETs off (decrements the FETs) until the charge voltage again reaches the voltage limit. That is, throughout the overcharge period (the long period in sunlight following the recharge period), the BCR satisfies its voltage limit. Due to a lag in the system response, the battery voltage overshoots that limit on charge, and undershoots it on discharge. The same is true of the battery current. The digital control of the solar array current results in alternately too little current followed by too much current. An approximate battery current–time profile is shown at the bottom of Fig. 2, which shows a random sample of one cycle during overcharge current. Each digital step was 1 s wide, with an amplitude equal to the solar cell circuit current, which was nominally 300 mA at the beginning-of-life (BOL), decreasing to 220 mA at the end-of-life (EOL) [5]. Assuming this to be a typical cycle, its DOD range was 0.0021% BOL and 0.0015% EOL. Also, it

[2] The same ground command that steered relays S1 and S2. The arrangement was such that either array–BCR pair could be selected to charge either battery, or both could be used to charge one battery. But the typical operation was as shown in Fig. 2.

is estimated that there were about $10 \times 10^6$ such cycles during AMPTE/CCE's almost five-year mission life. The actual shape of the overcharge current and voltage profile was measured only a few times at the subsystem level, at room temperature. Also, the time resolution of the telemetry system did not allow such measurements during system testing or in-orbit. However, the amplitude of the charge and discharge current can easily be estimated, from orbital plots of Fig. 8, discussed at the end of this paper.

Battery reconditioning is a system design feature that was planned but never used in-orbit. The reconditioning plan was to switch both solar arrays so that they would recharge one battery; for example, solar array 1 and its BCR would be switched to battery 2 by commanding switch S1 to the alternate position. Battery 1 would then discharge, first to the load, then through its cell letdown resistors, a procedure that was shown by ground test to take about three weeks. Since each cell letdown resistor was permanently wired across each battery cell, it was selected to be 118 Ω, to minimize its continual (300 mW) load on each battery. Either battery could be reconditioned in this manner, however, relay S1 and S2 were arranged so that the solar arrays could never be removed from both batteries simultaneously.

## 4. Battery design

The battery cell characteristics are summarized in Table 3. The cell was a General Electric [3], rectangular, Ah, nickel–cadmium type with stainless-steel case and cover. Both terminals were insulated by means of dual, nickel–braze, ceramic seals. Nylon separator material, Pellon 2505 ML, was used between the plates. The positive plates were passivated and there was no treatment of the negative plates. The cells were activated in August 1982, twenty-four months before the launch in August 1984.

The cells were tested, matched for charge voltage and capacity, and assembled into batteries, at JHU/APL. A summary of the battery elements is listed in Table 4 with weights and dimensions.

Fig. 3 shows the assembly of both batteries mounted on their common baseplate. Each battery had two 'end plates', held in place by connecting rods. The configuration was shaped by the requirement to keep the batteries and their cells near the same temperature and between 5 and 15 °C: requirements which drove their thermal and mechanical design. Aluminum thermal fins were inserted between cell pairs, to conduct the heat from the broad face of each cell to the battery baseplate. Each cell was wrapped with Kapton tape to electrically isolate it from the others and from the battery case with minimum thermal insulation. Each battery had its own flat-wire strip-heater (not shown), mounted underneath it on the baseplate.

[3] General Electric's Battery Division in Gainesville, FL was later purchased by Gates Energy Products which later sold this division to SAFT in Poitiers, France.

Table 3
Summary of cell characteristics

| | |
|---|---|
| JHU/APL Specification No. 7254–9017 A | |
| Manufacturer | General Electric |
| Catalog no. | 42B004AB37 |
| Capacity | Manufacturer's rating: 4 Ah<br>Actual at 5.0–5.5 Ah |
| Construction | Low profile, rectangular, 304L stainless steel |
| Size | $0.819(L) \times 2.128(W) \times 2.330(H)$<br>(2.79 $H$ including terminals) |
| Weight | 200 g actual<br>225 grams specified maximum |
| Seals | Dual nickel–braze ceramic |
| Terminals | Dual ceramic |
| Separator | Pellon 2505 ML nylon |
| Positive plate treatment | Passivated |
| Negative plate treatment | None |
| Positive plate loading | 12.47 $g/dm^2$ |
| Negative plate loading | 15.46 $g/dm^2$ |
| Final KOH quantity | 15 to 16 $cm^3$ of 31% KOH |
| Precharge setting | 1.26 Ah |
| Average ECT (electrochemical capacity test) | 5.88 Ah positive<br>10.63 Ah negative |
| Activation date | August 1982 |

Table 4
Battery summary

| | |
|---|---|
| Manufacturer | JHU/APL |
| Number of batteries | 2 |
| No. of cells per battery | 22 |
| Battery dimensions | $11.5 \times 5.5 \times 3.5$ $in^3$<br>(Not included connector bracket) |
| Battery weight | Battery 1: 5325 g<br>Battery 2: 5334 g<br>Baseplate: 484 g<br>Hardware: 155 g<br>Total: 11298 g |
| Battery-to-baseplate thermal-grease | TC-4 |
| Letdown resistors | 118.5 Ω mounted on top of each cell |
| Each of 2 battery heaters | 500 Ω, 1.6 W flat-wire, mounted between cells |

The entire battery assembly was then covered with a thermal blanket and mounted to the spacecraft structure with thermally isolating spacers, to minimize the heat transfer between it and the spacecraft, effectively creating a battery compartment with its own temperature control. When mounted, the underside of the common baseplate was then radiating to space through 0.07 $m^2$ of temperature-controlled louvers on the anti-sun-facing side of the spacecraft [6]. The baseplate surface was coated with a black Chemglaze paint, to facilitate the rejection of heat to space.

The batteries were formed by first matching the cell charge voltages so that the cells would all be protected by the selected battery voltage limit during recharge and then matching their capacity to insure that the batteries would share the load evenly on discharge. For the cells of both batteries, capacities

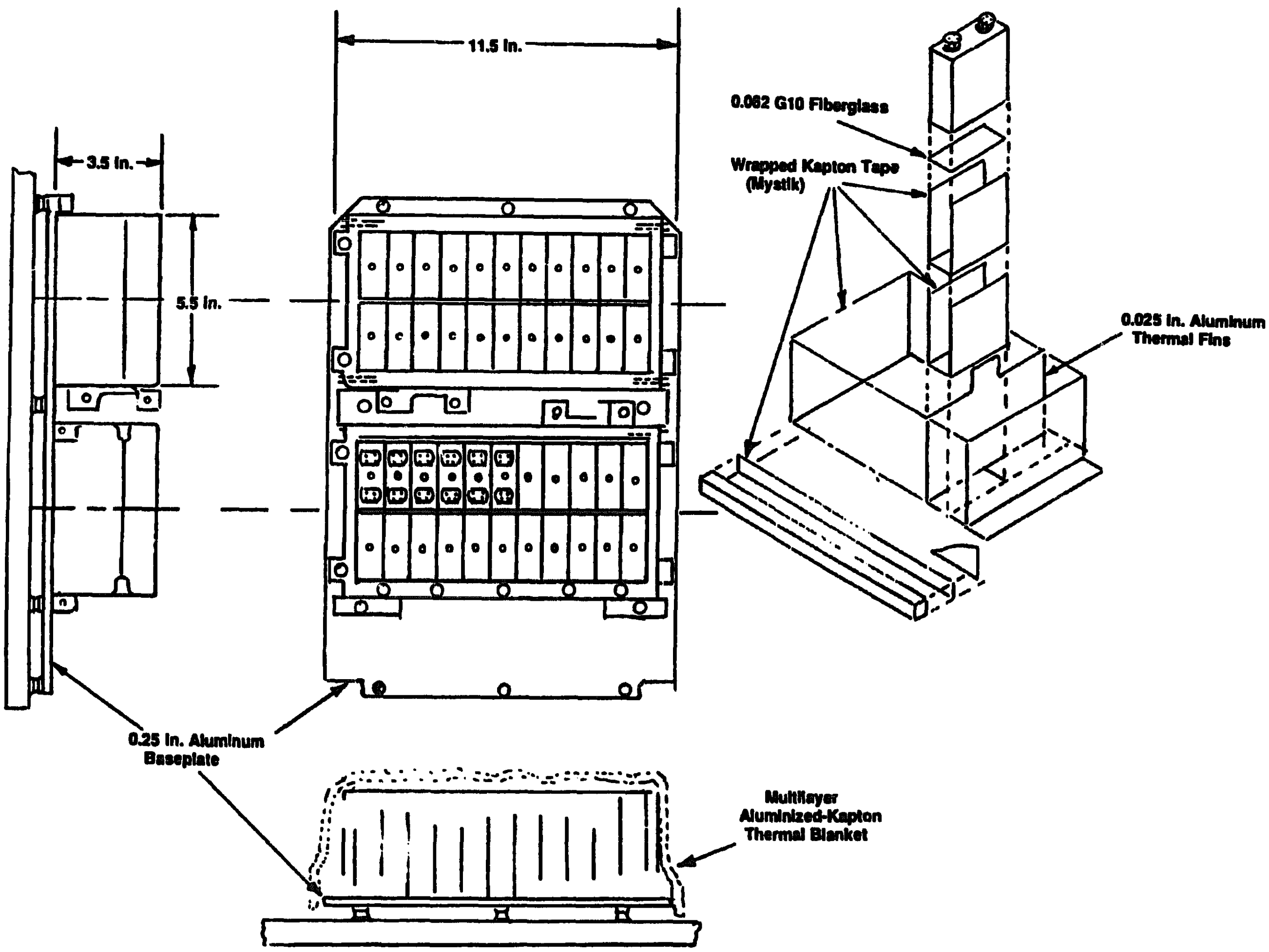


Fig. 3. AMPTE/CCE battery assembly.

were matched to within ±3% and cell end-of-charge (EOC) voltages to within ±7 mV of their averages at test temperatures of 0 and 25 °C.

## 5. Ground test performance

The Naval Surface Warfare Center (NSWC), Crane Division assembled a test pack (no. 4 H) of five battery cells from the AMPTE/CCE flight lot. Prior to the start of the test, the cells were reconditioned as part of their acceptance testing.

Throughout all tests, a 16 h total cycle time was maintained and the pack was held at an approximately constant temperature of 15 °C. No attempt was made to simulate the digital overcharge scheme that was used in the AMPTE/CCE flight power system. Rather, a typical d.c. power supply and voltage-limiting shunt was used for recharge. The recharge rate was 1.8 A until reaching 1.439 V per cell, followed by an approximately 200 mA trickle charge for the remainder of the recharge time.

To simulate the flight battery load, a 118 Ω reconditioning resistor was permanently placed across each cell. The orbital period was kept at 16 h throughout the test. However, two separate discharge scenarios were used. The first (variable discharge) portion of the ground tests was reasonably representative of the first 40% of the AMPTE/CCE orbital experience. The second part was indicative of the generic performance of these cells, but, at a constant discharge to 40% depth, was much more severe than the actual AMPTE/CCE experience.

### 5.1. Variable discharge

This part of the life cycle ground test was started in December 1983 and continued until completion of cycle no. 791 in November 1985. It was an approximate simulation of the actual AMPTE charge/discharge profile with the full-sun periods eliminated, resulting in a slightly accelerated test. A variable discharge schedule was imposed to approximate that of the actual AMPTE/CCE mission by frequently varying the discharge time and occasionally changing the load. The discharge time was varied in accordance with an annually varying shadow time, approximating that of the AMPTE spacecraft. For a period of about 250 days (0.7 year) the discharge time was varied according to the schedule shown at the top of Table 5. To limit the battery DOD for the longer eclipses, the load was reduced according to the mission plan. For eclipse times up to 0.48 h, the normal spacecraft load of

Table 5
Ground test summary — pack 4H

| | | | |
|---|---|---|---|
| Nos. cells in pack | | | 5 |
| Temperature | | | 15 °C |
| Cycle (orbit) period | | | 16 h |
| Variable discharge regime | | | |
| **Fraction of year** | **Discharge time (h)** | **Discharge rate (A)** | **DOD (%)** |
| 0.45 | 0.35–0.48 | 3.3 | 29–40 |
| 0.11 | 0.48–0.92 | 1.8 | 22–41 |
| 0.13 | 0.92–1.64 | 1.0 | 23–41 |
| 1. | Start acceptance testing (1st reconditioning) | | 9/23/83 |
| 2. | Start life cycle test, varying DOD 23 to 41% for ~250 days | | 12/26/83 (cycle no. 36) |
| 3. | First float (trickle charge) period of 115 days | | 9/6 to 10/23/84 |
| 4. | 2nd reconditioning (GSFC expected to discontinue the test) | | 10/23/84 (cycle no. 412) |
| 5. | End 1st float and 1st year | | 11/9/84 (cycle no. 416) |
| 6. | Repeat during 1984 (no reconditioning) | | |
| 7. | Repeat during 1985 (no reconditioning) | | |
| 8. | Terminate life-cycle test with varying DOD | | 11/23/85 (cycle no. 791) |
| 9. | Start 4th year with fixed 41% DOD (1.6 h eclipse at 1.0 A, 14.4 h recharge at 1.8 A to 1.439 V limit) | | 1/8/86 (cycle no. 792) |
| 10. | Repeat during 1987 | | |
| 11. | Repeat during 1988 | | |
| 12. | ADACS computer system failed | | (cycle no. 2826) |
| 13. | Pack left open circuit for 4.5 months | | |
| 14. | 3rd reconditioning | | 5/25/90 (cycle no. 2827) |
| 14. | Repeat during 1991 | | |
| 15. | Repeat during 1992 | | |
| 16. | 4th reconditioning and discontinue test | | 7/20/93 (cycle no. 4526) |

3.30 A was used. Whereas, for eclipse times between 0.48 and 0.92 h, the load was reduced to 1.8 A, and for the very long eclipse times, the current was reduced to 1.0 A. The resulting DOD varied from 23 to 41%. The voltage limit was set at 1.439 V per cell, to simulate the JHU/APL B-LO voltage temperature level.

The pack was reconditioned on 10/23/84 (cycle no. 412) in expectation of ending the test. However, after further consideration, the test was resumed and continued through cycle no. 791, with no significant signs of performance degradation.

### 5.2. *Constant discharge*

After 791 cycles, the life test was continued with the same 16 h orbital cycle, but with a constant 40% DOD (1.0 A discharge for 1.6 h) for over 3000 additional cycles. This was much more severe than the actual AMPTE charge/discharge profile. Fig. 4 shows the life-cycle data [7,8]. The end-of-discharge (EOD) voltage showed significant dispersion starting around cycle no. 2182. The pack was left open circuit for 4.5 months following a computer system failure on cycle no. 2826. The pack was then reconditioned, returned to test, and subsequently, exhibited an EOC voltage dispersion with a near doubling of the percent-of-charge return about cycle 4000.

## 6. Orbital performance

Although the variable discharge portion of the ground tests, described above, was not conducted exactly like the flight scenario, it was reasonably representative of the AMPTE/CCE orbital experience, and correctly predicted the success of the cells.

Table 6 is a breakdown of the AMPTE/CCE eclipse periods shown in Fig. 1. The probable DOD was determined based on the assumption that the load was 3.48 A, for eclipses of 25 min or less, and 1.84 A for longer eclipses. It indicates a total of 1888 eclipses, only 98 of which would have resulted in a DOD of 38% or higher.

Fig. 5 provides an overview of battery 1 performance data over the entire mission. Fig. 6 shows the same information for battery 2. There is no apparent long-term drift in any of the parameters on either battery. Load sharing of the batteries was excellent throughout the mission, as indicated by their nearly identical EOD current. The temperature differences

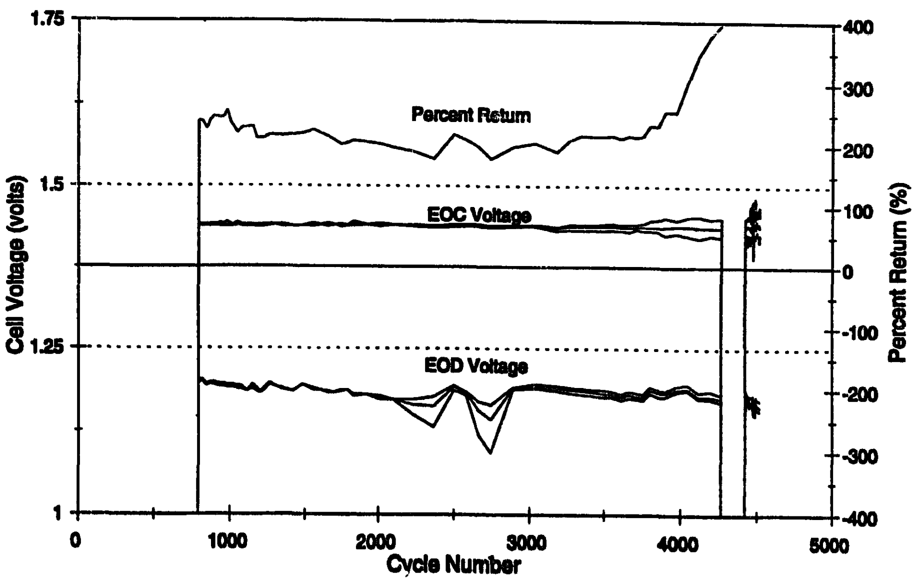


Fig. 4. NSWC ground test results for pack 4 h.

Table 6
Eclipse durations

| Range of eclipse durations (min) | Probable DOD range (%) [a] | Nos. of eclipses |
|---|---|---|
| 0–25 | 0–18 | 813 |
| 25–50 | 18–19 | 821 |
| 50–75 | 19–29 | 127 |
| 75–100 | 29–38 | 29 |
| 100–125 | 38–48 | 22 |
| 125–150 | 48–58 | 31 |
| 150–175 | 58–67 | 43 |
| 175–200 | 67–77 | 2 |
| Total number of eclipses | | 1888 |

[a] Assumed load of 3.48 A up to eclipse times of 25 min and 1.84 A for longer eclipses.

between the two batteries were typically less than 2 °C. However, the difference between the EOC and EOD temperatures, were as much as 7 °C on each battery. These relatively wide variations in battery temperature, as shown in Figs. 5 and 6, were influenced by occasional changes in spacecraft-sun orientation, periodic changes in spacecraft operations, and variations in the length of the eclipse period. Further evidence of the stability of the batteries' performance characteristics can be seen by a comparison of the battery discharge voltage, measured during the relatively deep discharges of the three long eclipses defined in Fig. 1. Fig. 7 shows these three discharge voltages versus capacity for both batteries, superimposed on their BOL discharge curves. The results are quite similar for both batteries. The discharges, during eclipses 2 and 3, were at nearly the same discharge rate, temperature

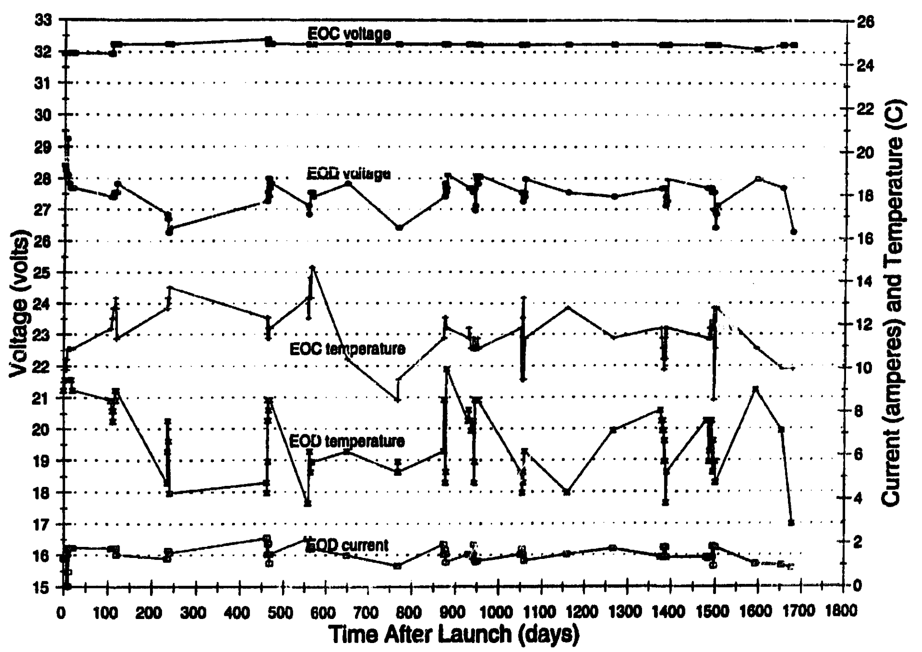


Fig. 5. Selected battery 1 performance characteristics vs. time.

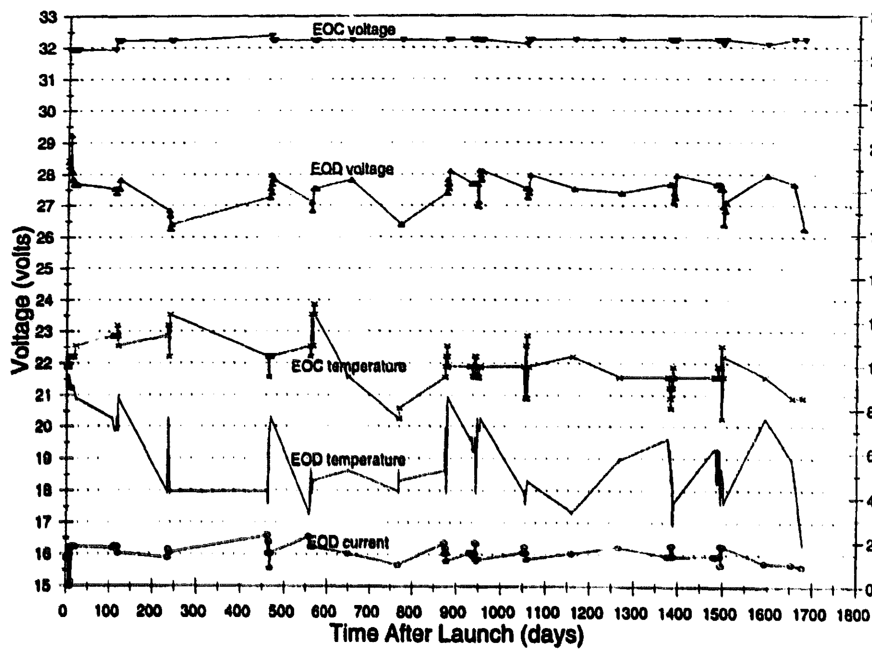


Fig. 6. Selected battery 2 performance characteristics vs. time.

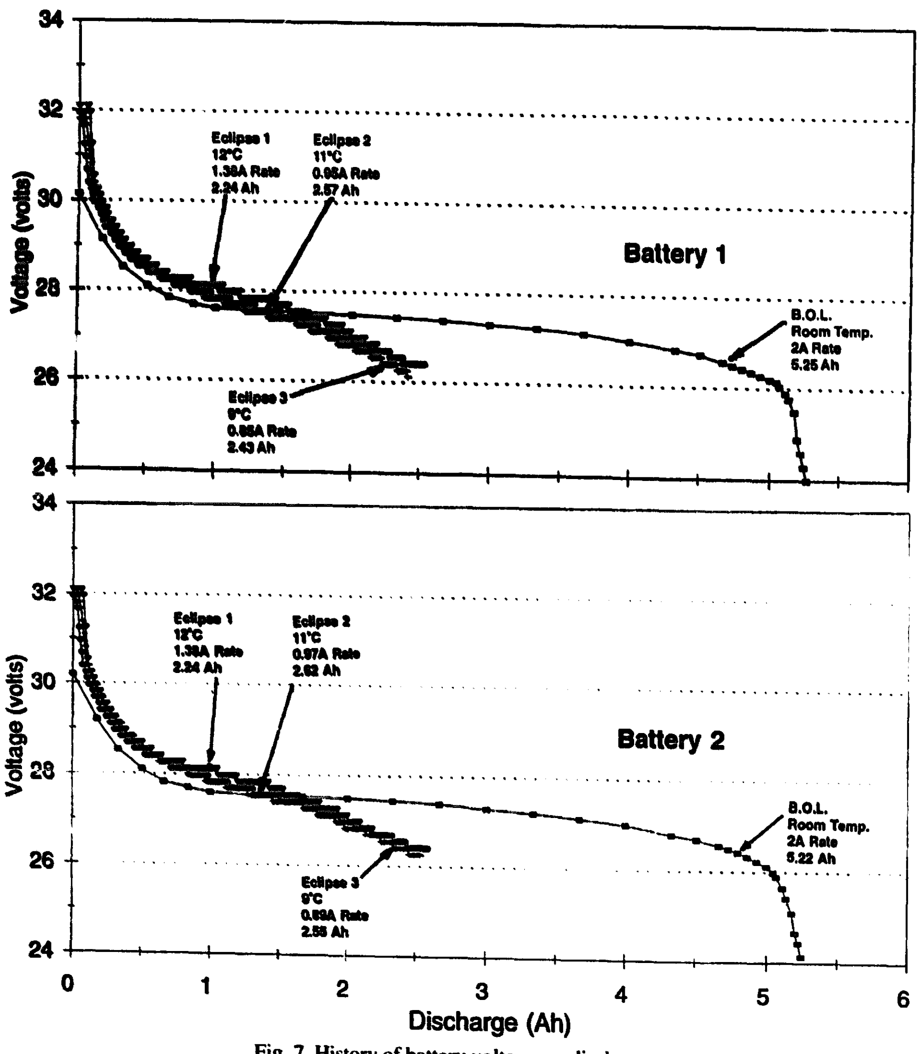


Fig. 7. History of battery voltage vs. discharge.

and DOD, but occurred nearly 2.5 years apart. Yet, they show almost no change in performance over this time period. Their performance during the first eclipse is more difficult to compare since the load was approximately 50% higher.

As a result of the spacecraft-load reduction to prevent excessive discharge, the net discharges were nearly the same during all three long eclipse periods, 2.5 Ah or 63% DOD. The higher beginning-of-discharge voltages of the three post-launch curves, compared to the pre-launch, BOL curve, is due to their lower temperature. The typical nickel–cadmium voltage degradation, is apparent. All three of the post-launch discharge curves dropped significantly below the BOL curve at discharges exceeding 1.5 Ah ( ~40% DOD). But, although these three discharge voltages are successively lower, for increasing time on-orbit, the voltage differences between them are not significant. That is, nearly all of the deterioration of the battery discharge voltage relative to the BOL voltage, occurred before the first long eclipse (within the first 237 days). Subsequent performance during the second and third long eclipses at 768 and 1677 days into the mission, showed very little additional change.

One possible explanation for the early drop in discharge voltage is that the selected voltage limit may have been lower than optimum. However, it is clear that the system constraints, including the voltage limit, battery operating temperature, and the many, very small charge/discharge cycles that occurred during the 1440 days (nearly 4 years), between long eclipses 1 and 2, did not measurably affect the performance of either battery.

Fig. 8 shows full 24 h orbit profiles of battery voltage and current, corresponding to the three long eclipses. Batteries 1 and 2 voltage matching and current sharing are excellent. However, the discharge current for long-eclipse #3 shows a slight (7%) divergence between the two batteries after 1677 days (4.6 years) in-orbit.

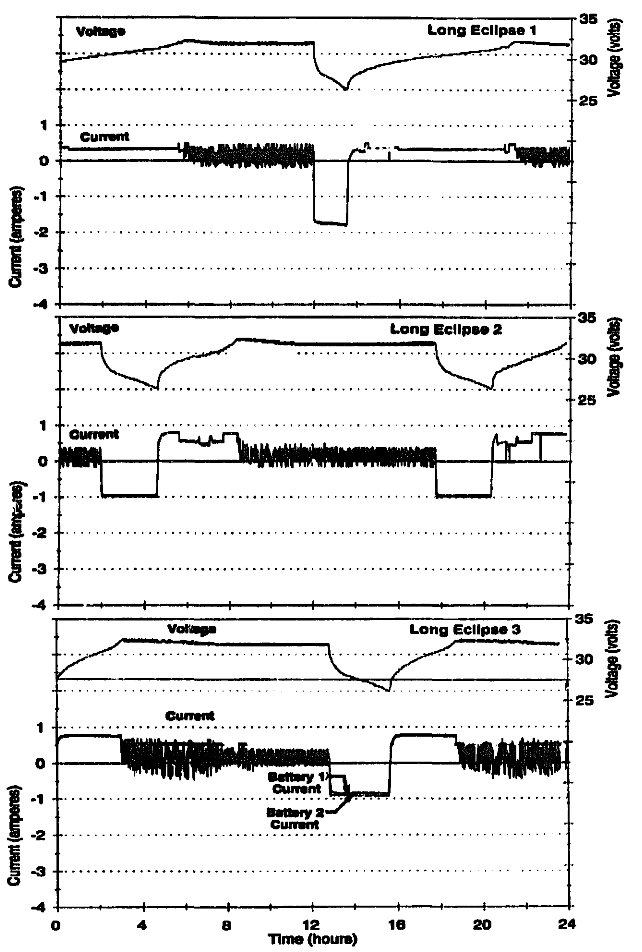


Fig. 8. AMPTE/CCE battery 1 & 2 current and voltage.

## 7. Conclusions

Both the ground tests on one five-cell pack and the consistent in-orbit performance of two 22-cell batteries showed these nickel–cadmium cells to be well matched, and remain so for the life of the mission. The first 791 orbits of the ground test data, using variable discharge times, produced performance results that were reasonably representative of those realized in-orbit, even though the actual 24 h orbit was tested as a 16 h orbit, and the digital aspect of the battery charge regulator, with its many, very small charge/discharge cycles during overcharge, was not simulated.

The in-orbit data showed the battery to perform well with the system constraints that had been selected, except that the voltage limit selection, while certainly adequate, may not have been optimum. However, these results did demonstrate that the system imposed constraints such as the limit voltage, operating temperature and the many, very small charge/discharge cycles imposed by the battery charge regulator, did not degrade the performance of either battery.

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