110807 (handwritten annotations)

Nickel Cadmium Battery Operations and Performance

Gopalakrishna Rao, Jill Prettyman-Lukoschek[1], Richard Calvin[2], Thomas Berry[3],
Robert Bote[4] and Mark Toft[5]
NASA/Goddard Space Flight Center
Greenbelt, Maryland 20771

The Earth Radiation Budget Satellite (ERBS), Compton Gamma Ray Observatory (CGRO), Upper Atmosphere Research Satellite (UARS), and Extreme Ultraviolet Explorer (EUVE) spacecraft are operated from NASA's Goddard Space Flight Center (GSFC) in Greenbelt, Maryland. On-board power subsystems for each satellite employ NASA Standard 50 Ampere-hour (Ah) nickel-cadmium batteries in a parallel configuration. To date, these batteries have exhibited degradation over periods from several months (anomalous behavior, UARS and CGRO {MPS-1}; to little if any, EUVE) to several years (old age, normal behavior, ERBS).

Since the onset of degraded performance, each mission's Flight Operations Team (FOT), under the direction of their cognizant GSFC Project Personnel and Space Power Application Branch's Engineers has closely monitored the battery performance and implemented several charge control schemes in an effort to extend battery life. Various software and hardware solutions have been developed to minimize battery overcharge. Each of the four sections of this paper covers a brief overview of each mission's operational battery management and its associated spacecraft battery performance. Also included are new operational procedures developed on-orbit that may be of special interest to future mission definition and development.

INTRODUCTION

The Earth Radiation Budget Satellite (ERBS), Compton Gamma Ray Observatory (CGRO), Upper Atmosphere Research Satellite (UARS), and Extreme Ultraviolet Explorer (EUVE) spacecraft are operated from NASA's Goddard Space Flight Center in Greenbelt, Maryland. Each satellite, except ERBS, employs the Multimission Modular Spacecraft (MMS) bus, which includes the Modular Attitude Control Subsystem (MACS), the Propulsion Module (PM), the Command and Data Handling Subsystem ((C&DH) - which incorporates the On-Board Computer (OBC), the Earth Sensor Assembly Module, and the Signal Conditioning and Control Unit (SC&CU)), and the Modular Power Subsystem (MPS). The ERBS spacecraft uses several, but not all, of the MMS bus features.


The Power Subsystem, in general, comprises all power control, power distribution and all other related hardware. It contains the McDonnell Douglas Electronics Systems Company (MDESC)-supplied MPS, the Solar Array (SA) equipment and three NASA Standard 50 Ah Nickel-Cadmium batteries. Figure 1 presents the power subsystem topology. Table 1 lists the major Power Subsystem components and their functions.
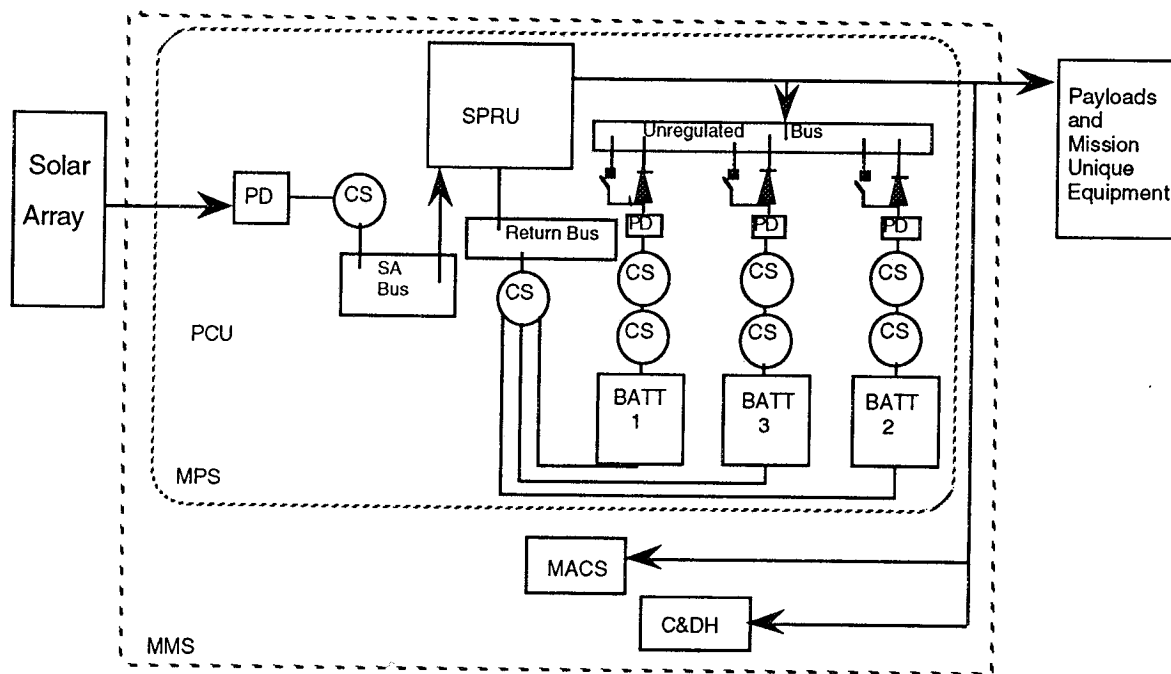
Figure 1. Power Subsystem Block Diagram

TABLE 1. Power Subsystem Components and their Functions

SPRU	Controls bus voltage and battery charging.
BPA	Provides redundant fusing for internal MPS loads.
Batteries	Provide power during spacecraft eclipse periods and supplement SA during peak loading.
Solar Array	Provides power for instrument loads and battery charging during spacecraft sunlit periods.

The MPS receives commands from the OBC which control its on-orbit battery charge modes. These modes and their operations are summarized in Table 2.

There are two to three NASA Standard 50 Ampere-hour (Ah) Nickel-Cadmium (NiCd) Spacecraft Batteries in each MPS. The NASA Standard 50 Ah NiCd batteries were manufactured and tested by McDonnell Douglas Corporation of St. Louis, using 22 serial-connected 50 Ah NiCd battery cells from Gates Aerospace Batteries (formerly General Electric Battery Business Division).

Table 2. MPS Charge Modes and their Operations

Peak Power Tracking (PPT)	While batteries are charging to the specified voltage limit, the SPRU draws maximum power from the SA so that the batteries charge on all current not required by the load bus.
Voltage Limited (Voltage/Temperature Mode, VT)	When battery voltages reaches the limit determined by one of eight user-selectable VT levels, battery charge current is tapered to maintain that voltage.
Current Limited (Constant Current Mode, CCM)	When selected by external command, battery charge current is controlled to one of three levels (0.75, 1.5 or 3.0 amps); the VT limit remains active as a backup.
Safehold	When an OBC fault is detected, the SPRU is commanded to PPT and a pre-selected VT level dependent on which VT was in effect at the time of the fault. CCM is inhibited. This mode remains in effect until reset by external command.

On all of the spacecraft, for each battery, there is hardwired circuitry that compares the sum of the first 11 series-connected cells to the sum of the remaining 11 series-connected cells. The result of this comparison is designated as the half-battery differential voltage. The first indication of possible problems with these batteries was a differential voltage exceeding 100 millivolts on charge, followed shortly thereafter by non-zero values on discharge. This initial divergence often arose after periods of increased overcharge and/or reduced discharge. It was then usually followed by a divergence in battery load-sharing (on charge and discharge), recharge (C/D) ratios, battery taper currents and battery temperatures.

Since the onset of degraded performance, each mission's Flight Operations Team (FOT), under the direction of their cognizant GSFC Project Personnel and Space Power Application Branch's Engineers has closely monitored the battery performance and implemented several charge control schemes in an effort to extend battery life. Various software and hardware solutions have been developed to minimize battery overcharge. Each of the four sections of this paper covers a brief overview of each mission's operational battery management and its associated spacecraft battery performance.

EARTH RADIATION BUDGET SATELLITE

Mission Summary

The Earth Radiation Budget Satellite (ERBS) was deployed into a 57-degree inclination LEO orbit from the Space Shuttle Challenger in October 5, 1984. The geometry of the orbit causes the angle between the orbit plane and the ecliptic plane (Beta angle) to change from 10 degrees to 170 degrees. At Beta angles equal to 90 degrees, the spacecraft executes a 180-degree yaw turn to keep its fixed SAs facing the sun.

Besides the two 50 Ah NiCd batteries and the SPRU, the Electrical Power Subsystem includes two redundant Ampere-hour Metering Units (AHMU). Batteries are charged via PPT until the VT limit is

reached, then the charge current tapers. When the AHMU's register 100% SOC, the SPRU fixes the charge current at 2 amps (approximately 1 amp per battery). Two other levels of CCM are available in the SPRU, a 7 amp level and a -5 amp level. At eclipse, the SPRU resets for the next orbit. VT 6 was initially selected for all Beta angles, and battery C/D ratio was used as the charge control parameter for switching from VT control to CCM.

Battery Management

For the first five years in orbit, battery performance was nominal. C/D ratios during this time were between 1.17 and 1.25. In September, 1989, half-battery differential voltages began to rise from the nominal value of 40 millivolts. By July, 1990, battery 1's half-battery differential voltage had risen to 220 millivolts, and battery 2's half-battery differential voltage had risen to 460 millivolts. The half-battery differential voltages reached a peak when the high power TDRSS (Tracking and Data Relay Satellite System) transponder was on during eclipse. The FOT reduced the duration of TDRSS events, and eventually turned off the TDRSS transponder during eclipses. The reduced power load decreased the half-battery differential voltages. Also at this time, it was discovered that battery 1 was carrying more of the spacecraft load. The current differential was .96 amps on discharge, and .72 amps on charge. Meanwhile, the CCM's 2 amp level had increased (apparently from component drift) from 1.00 amp to 1.50 amps for battery 1, and from 0.99 amp to 1.10 amps for battery 2.

The poor load-sharing and the increase in CCM current began to overcharge battery 1. The lowering VT level to 5 in January 1992 worsened the load-sharing. The VT level was further reduced to 4 in July 1992; but battery 2 was not fully charged at this VT level.

The first cell failure in battery 1 occurred in August 1992. The VT level was immediately lowered to 3, but the load-sharing continued to worsen. The C/D ratio for battery 1 was above 1.2, while the C/D ratio for battery 2 was less than 1. It became evident that the two batteries could not be charged together using straight V/T control. To reduce the overcharge on battery 1 (without undercharging battery 2), battery 1 was removed from the charge bus early in each charge period, then reconnected. This brought the C/D ratios to within 0.535 of each other. The VT level was raised one level after battery 1 was taken off-line so that the two batteries would be within 2 volts of each other at the time of reconnection. This required numerous commands from spacecraft memory, and became difficult when the frequency of memory bit upsets (a separate problem) increased.

The next approach was to disconnect battery 2 during eclipse and reconnect the battery at sunrise. This provided a 0.7 volt diode voltage differential in favor of battery 2 during eclipse. Battery C/D ratios were brought to within 0.307 of each other. This method was used until September 1992, when a second cell failed in battery 1. Battery 1 was immediately commanded off-charge. Battery 2 carried the entire spacecraft power load, with the charge control method reconfigured for a single battery. In attempts to stabilize battery 2, the SPRU was gradually raised to VT 5 to give battery 2 a C/D ratio of 1.10. The performance parameters of battery 2 returned to pre-1989 values. A charge scheme based on switching VT levels according to Beta angles was soon implemented. Spacecraft loads, namely heaters, were used to control the battery C/D ratio.

Eventually, however, one cell in battery 2 failed in June 1993. The previous charge control method was continued at lower VT levels. A second cell failed in July 1993. The SPRU was set to its lowest VT level and commanded into the 2 amp CCM (actual measured value: 2.7 amps) to achieve the desired C/D ratio of 1.10. The time to switch from VT control to CCM was manually calculated on the ground, and the

commands uplinked into spacecraft memory. In spite of continued high currents from PPT, this new method appeared to produce nominal battery performance.

During this time, it was found that the 7 amp CCM level had drifted to a value of 11.4 amps. The 11.4 amp level was preferred as the means to charge battery 2 without the high PPT charge currents.

Beginning in August 1993, the FOT began manually regulating battery charging by placing the battery in CCM and switching among the three CCM levels available within the SPRU. Battery management is accomplished entirely from spacecraft memory, and is based on ground calculations. All spacecraft charge controls and safeguards are disabled. Battery 2 is charged at the beginning and end of the sunlight period at the 2.7 amp level. During the middle of sunlight, the battery is charged at the 11.4 amp level. The duration at the 11.4 amp level is set to produce a C/D ratio of 1.02. Four commands per orbit are required to charge the battery in this way. The command times are manually calculated using predicted sunrise and sunset events and battery load. Since the battery load changes each orbit depending on Beta angle, spacecraft loads, and SA output, the duration at the 11.4 amp level can be adjusted every orbit (1.5 hours).

During full- sun periods, the -5 amp level is used to 'exercise' the battery. The negative current forces discharge on the battery, artificially increasing the power load. Reducing overcharge is seen as the best method to prevent another cell failure.

Operation of one battery containing two shorted cells continues in full support of the remaining scientific instruments. Battery temperature is between 2 and 5 degrees C, voltage is maintained between 24.00 and 29.66 volts, and the C/D ratio is kept between 1 and 1.02. The CCM charge control method, although labor intensive and totally reliant on down-linked data, is able to effectively control battery C/D ratio. The only difficulty in maintaining battery C/D ratio stems from the constantly changing power load (based on Beta angle and instrument status), and the time lag between real-time and off-line analysis of data. This method of charge control is expected to continue even in the event of a another cell failure in battery 2.

COMPTON GAMMA RAY OBSERVATORY

Mission Summary

The Compton Gamma Ray Observatory (CGRO) was launched aboard the Space Shuttle Atlantis on April 5, 1991 and was released into a circular orbit of 450 kilometers. The Observatory orbits the Earth with an inclination of 28 degrees and a nodal precession rate of about 3 degrees per day. The length of spacecraft daylight varies from 57 to 64 minutes per 93 minute orbit. CGRO is the heaviest civilian payload ever launched by a shuttle, weighing 35,000 pounds.

CGRO's Electrical Power Distribution Subsystem (EPDS) consists of two, rotatable (single-axis) SAs and two MPS' The MPS-1 and MPS-2 launch configuration was VT 5 with a load imbalance of less than 100 watts. Post-launch performance for both MPS-1 and MPS-2 was nominal.

To date, MPS-2 batteries have shown excellent well matched performance. Battery temperature, load-sharing, half-battery differential voltage, and C/D ratios are well within their expected operational ranges. Battery DOD has been 10 - 17%. Both VT 5 and VT 6 charging schemes have been used. For DOD'S in the 10 - 14% range, VT 5 is utilized. DOD'S in the 14 - 17% range require VT 6.

In the April - June 1992 time period, MPS-1 batteries 1 and 2 showed significant signs of divergence in half-battery differential voltage, temperature, net overcharge and load-sharing. Higher than nominal overcharge rates became the leading suspect for excessive battery degradation conditions. During this period, plans were developed to reduce consumption upon MPS-1 by switching mission critical components to MPS-2. By July of 1992, MPS-1 battery 2 was progressively degrading; half-battery differential voltage reached saturation (greater than 700 millivolts) on several occasions. Finally, battery 2's temperature soared to over 30 degrees Centigrade near the end of charge of one orbit, and the battery was commanded "off-charge" on July 16, 1992. MPS-1 was then operated with the two remaining on-line batteries. Attempts to reduce net overcharge by commanding MPS-1 to lower VT levels were ineffective. Consequently, a new plan was implemented to reduce battery net overcharge .

MPS-1 Battery Management

The use of CCM (the lowest SPRU-commandable value of 0.75 amps) and VT 3 was chosen to minimize overcharge. This battery management technique involved commanding the SPRU, through the OBC's Stored Command Processor (SCP), to CCM at the beginning of spacecraft sunrise. While in CCM, the two remaining on-line batteries trickle charge at the 0.75 amp rate until SA temperatures approach a maximum, which occurs approximately fifteen minutes after spacecraft sunrise. After this user-defined time interval, the SCP commands the SPRU to VT 3.

While in VT 3, MPS-1 enters PPT mode, drawing maximum power from the SA and providing power to recharge the batteries. The batteries taper charge until a user-defined instantaneous net overcharge threshold is achieved. Upon reaching this threshold, the SPRU is again commanded to CCM (0.75A) via a Relative Time Sequence (RTS) activation and remains in CCM until the end of spacecraft daylight. Net overcharge is calculated by the OBC's PMON Processor using battery 1's current. Accumulated battery charge and discharge values are subtracted to calculate the net overcharge for each battery. Net overcharge data are then reported to the OBC's TMON Processor which manages battery charge modes, based on a user-defined instantaneous net overcharge threshold (presently 0.4 Ah). Once that threshold is exceeded, TMON takes action by activating an RTS that commands MPS-1 to CCM. This battery management charging scheme has been in use since February of 1993 and has been highly effective in minimizing further battery degradation due to overcharge.

UPPER ATMOSPHERE RESEARCH SATELLITE

Mission Summary

The Upper Atmosphere Research Satellite (UARS), designed, built, integrated, tested and operated by NASA and Martin Marietta, is a Low-Earth orbit (LEO), Earth-observing spacecraft which was launched via the Space Shuttle Discovery on September 12, 1991 and deployed three days later. The mission orbit is a 96-minute, circular orbit inclined 57 degrees to the Equator with a 585 Km height. This allows stratospheric sensors to observe up to 80 degrees in latitude (North and South) and provides near total global coverage. The full range of local times at all geographic locations is viewed every 36 days. The spacecraft batteries were specified to operate in low-Earth orbit up to 20% Depth of Discharge (DOD), between 21 and 35 volts for a nominal 36 month mission. Thermal vacuum testing revealed nominal battery performance prior to launch. The spacecraft power system was designed for a maximum 1600

Watts (orbital average), 786 Watts of which was reserved for the instrument load. The spacecraft maximum load has been approximately 1350 Watts with instrument loads of approximately 450 Watts.

Nominal battery performance was observed over the first four months of spacecraft operation. The first evidence of anomalous battery performance was observed in January 1992, after the first maximum beta angle (low DOD or "Full Sun" period). Since then, the FOT) has monitored and managed battery performance by adjusting solar array offset angle, conducted periodic deep discharges, and controlled battery recharge ratios.

Battery Management

Due to the cyclical variation of the orbit Beta angle (Beta angle is defined as the angle between the orbital plane and the Earth-to-Sun line), caused by the 57 degree orbital inclination and orbital geometry, the SA power collection , the spacecraft loading, and the battery charge and discharge profiles are not constant. The Beta angle variation changes SA night periods (in addition to the normal seasonal changes) from a maximum eclipse of 36 minutes at zero degree Beta, to a minimum of zero minutes at Beta angles greater than 66 degrees. This has prompted the FOT to develop more aggressive Power Monitor (PMON) software to actively control battery performance. The use of Constant Current Mode (CCM) was employed as a means to minimize overcharging of battery 1 while ensuring battery 2 and 3 reach 100 percent State-of-Charge (SOC). When battery 1 reaches a preset charge to discharge (C/D) ratio in the PMON software, PMON configures to CCM and charges the batteries at 0.75 amps until spacecraft night. By adjusting solar array offset to maintain a constant peak charge current and employing CCM, the FOT has effectively limited overcharging of the battery and improved overall battery performance. Day to day battery operations require monitoring of the battery voltages (including half-battery differential), current sharing, SOC and the spacecraft Beta angle.

In addition to the aforementioned software enhancements, the FOT has also implemented a new charge control strategy to address constantly changing Beta angle. When the battery DOD is between 18 to 20 percent (low Beta angle) and the end-of-night (EON) Load Bus Voltage (LBV) approaches 24.8 volts, the MPS is operated at VT 5 with CCM to obtain C/D ratios between 1.04 to 1.05 on battery 1. When the battery DOD is between 15 to 18 percent and the EON LBV approaches 24.8 volts, the MPS is operated at VT 4 with CCM to obtain C/D ratios between 1.04 to 1.05 on battery 1. When the battery DOD is less than 10 percent (high Beta angle) and the temperature delta between battery 1 and 2 approaches 5 deg. C, the MPS is operated at VT 3 with CCM to obtain a C/D ratios between 1.04 to 1.05 on battery 1. Operating the CCM switch to maintain battery 1's C/D ratios between 1.04 and 1.05 has aided in improving charge acceptance and load sharing between all batteries.

Both battery load-sharing and battery temperatures are good indicators of battery performance which may identify the most efficient battery and the weakest battery. For example, battery 1 has had the greatest half-battery differential voltages and the highest temperatures. It frequently accepts the most charge current while providing the least discharge current, and hence is identified as the weakest performer.

In addition, the weakest performer has been the battery receiving the greatest overcharge. The battery charge method in place for the early part of the mission -- charging at VT 6 to a system C/D ratio of 1.00, then switching to VT 5 (resulting in total C/D ratios of 1.1 - 1.25) may have overcharged the batteries.

Aggressive management of overcharge has been the most effective method of stabilizing and improving battery performance. Battery temperatures, delta temperatures, and load-sharing during charge and discharge have all trended back to more nominal behavior.

Battery exercise also helps to limit overcharge during low load (high beta angle/minimum spacecraft night) periods. Adjusting the SA offset to achieve a power negative condition, allowing the batteries to "spiral down" in SOC for several orbits, exercises the batteries during those low load periods. The result is a DOD of 12-18% at least once per day over a week when the DOD's would normally be much smaller or zero.

Deep discharges have been performed during the bi-annual Full-Sun periods in an attempt to improve battery performance. UARS utilizes these very low load (0-6% DOD) intervals to condition the batteries through low rate, deep discharges (up to 40% DOD) followed by low-rate recharge. This activity is also aimed toward maintaining and/or boosting EON LBV.

EXPLORER PLATFORM/EXTREME ULTRAVIOLET EXPLORER

Mission Summary

The Explorer Platform/Extreme Ultraviolet Explorer (EP/EUVE) spacecraft is a LEO satellite launched by the United States Air Force on a McDonnell Douglas Delta rocket on June 7, 1992. The spacecraft orbits at an altitude of 517 kilometers with an inclination of 28 degrees. The spacecraft length of day varies from 58 to 68 minutes due to the spacecraft's orbital precession of -6.7 degrees/day with respect to the Earth. The Explorer Platform was designed to accommodate a variety of remote sensing, LEO missions requiring solar, stellar or earth pointing over its mission life of 10 years. The payload instruments and equipment can be exchanged during shuttle-based servicing missions. The current EP primary mission payload, EUVE, operates continuously, providing a consistent and stable loading profile on the spacecraft power subsystem.

EP/EUVE's power is provided by a modified MPS that is rendered unique by its inclusion of a heat pipe along the battery baseplate. Solar power is provided by 2 SA Wings, which are rotated by 2 mission-unique solar array drives (SAD). These drives are primarily commanded by flight software, with manual commanding available as required.

As a result of battery anomalies observed on the CGRO and UARS spacecraft in 1991 and 1992, the EP FOT began to implement new modes of operation to enhance the cycle life of the batteries.

Battery Management

The EP/EUVE spacecraft uses a combination of several battery controls to maintain a consistent battery performance. These controls include thermal regulation of the batteries, CCM at orbital sunrise and at battery full- charge, and SA offsets.

Battery temperature regulation was implemented to maintain a specific battery temperature operating range by TMON Processor Control. This can be performed efficiently on the EP spacecraft because the heat pipe maintains a stable thermal environment between all three batteries. TMON samples the battery

baseplate temperature and commands the battery heater thermostat bypass on and off to maintain the baseplate temperature between 5°C and 8°C. This method of operation was introduced five months after launch. In a trial period just prior to this, the baseplate was maintained at 2°C minimum. At launch, the batteries had been thermostatically maintained between -2°C and 0°C.

CCM for the cold array case was implemented to reduce the high current from the SA to the batteries when the arrays are cold (at the beginning of each orbital day). The current operational goals limit the inrush current to less than 20 amps per battery. This is implemented by an Orbit Time Processor (OTP) Command flag which trips approximately 2 minutes prior to the beginning of each orbital day. It commands the 3.0 Amp CCM for a user-defined duration, then resets the SPRU to VT control. The original operational implementation, for 10 minutes following orbital sunrise, was introduced on February 3, 1993. There were subsequent experimental implementations of 15, 20, 25 and 30 minutes following orbital sunrise. The present operational mode, for 10 minutes following orbital sunrise, was re-implemented on March 12, 1993.

CCM for the hot array case (at full charge) was implemented to maintain C/D ratios between 1.02 and 1.07 to minimize battery overcharge. When the batteries reach full charge, the SPRU commands 0.75 Amp constant current mode to maintain a trickle charge on the batteries. The C/D ratio goal is based on the assumption that when the battery reaches 100% SOC at a specified 0.98 PMON battery charge efficiency, the C/D ratio is approximately 1.02. TMON commands CCM when the state of charge on any battery reaches 100% for 2 consecutive counts of the TMON processor (≈ 32 seconds). The original operational implementation, based on just battery 3 reaching 100% SOC, was introduced on March 15, 1993. The present operational mode, based on any battery reaching 100% state of charge, was implemented on January 3, 1994.

VT changes have been performed twice thus far in the EP/EUVE mission. Both changes were made in an effort to minimize overcharging of the batteries. VT 5 was lowered to VT 4 on launch day when the C/D ratio was approximately 1.3, and further lowered to VT 3 on May 5, 1993, when the C/D ratio was approximately 1.1.

The SAs are maintained at a 40 degree effective offset to the sun. This offset was introduced to provide a thermally stable environment for the SA based on the specular reflection problem identified during the thermal envelop testing conducted in August of 1993. Prior to this, the solar array drives remained powered off and fixed at 90 degrees with respect to the $-X_{acs}$ axis on the spacecraft. Following the in-orbit-checkout of the SADs in July of 1993, the SAs were maintained manually at a 40 degree offset until a flight software patch could be developed to maintain the user-defined offset angle. The current flight software management of the offset angle was begun on November 29, 1993.

The EP/EUVE spacecraft management combines these techniques into a generic spacecraft orbit. The batteries remain in VT 3 control with the battery temperature regulation. The 3.0 Amp CCM occurs for 10 minutes at orbital sunrise, followed by VT control to full charge, then 0.75 Amp CCM until orbital night. This generic power management orbit successfully limits battery overcharge, thus extending the life of the EP/EUVE mission.

CONCLUSIONS

Degraded performance has been observed on several NASA missions employing 50 Ah NiCd spacecraft batteries. Each mission's Flight Operations Team (FOT), along with their respective GSFC Project Personnel and engineers from GSFC's Space Power Application Branch, has closely monitored the battery performance and implemented several charge control schemes in an effort to extend battery life. Various software and hardware solutions have been developed to minimize battery overcharge, and implemented with success. New operational procedures continue to be developed on-orbit. These new procedures may have application in the management of other spacecraft batteries, and may also serve as useful design considerations for future spacecraft power systems.

GLOSSARY

CCM	Constant Current Mode	PMON	Power Monitor
C/D	Charge/Discharge	PPT	Peak Power Tracking
DOD	Depth-of-Discharge	RTS	Relative Time Sequence
EON	End-of-Night	SA	Solar Array
FOT	Flight Operations Team	SAD	Solar Array Drive
LBV	Load Bus Voltage	SOC	State-of-charge
LEO	Low-Earth Orbit	SCP	Stored Command Processor
MPS	Modular Power System	SPRU	Standard Power Regulator Unit
OBC	On-board computer	TMON	Telemetry Monitor
		VT	Voltage-temperature