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PERFORMANCE OF NICKEL-CADMIUM BATTERIES ON THE GOES I-K SERIES OF WEATHER SATELLITES

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Abstract

The US National Oceanic and Atmospheric Administration (NOAA) operates the Geostationary Operational Environmental Satellite (GOES) spacecraft (among others) to support weather forecasting, severe storm tracking, and meteorological research by the National Weather Service (NWS). The latest in the GOES series consists of five spacecraft (originally named GOES I-M), three of which are in orbit and two more in development. Each of five spacecraft carries two Nickel-Cadmium batteries, with batteries designed and manufactured by Space Systems Loral (SS/L) and cells manufactured by Gates Aerospace Batteries (sold to SAFT in 1993). The battery, which consists of 28 cells with a 12 Ah capacity, provides the spacecraft power needs during the ascent phase and during the semi-annual eclipse seasons lasting for approximately 45 days each. The maximum duration eclipses are 72 minutes long which result in a 60 percent depth of discharge (DOD) of the batteries. This paper provides a description of the batteries, reconditioning setup, DOD profile during a typical eclipse season, and flight performance from the three launched spacecraft (now GOES 8, 9, and 10) in orbit.

INTRODUCTION

The US National Oceanic and Atmospheric Administration (NOAA) operates the Geostationary Operational Environmental Satellite (GOES) spacecraft (among others) to support weather forecasting, severe storm tracking, and meteorological research by the National Weather Service (NWS). The latest in the GOES series consists of five spacecraft (originally named GOES I-M), three of which (GOES 8, 9, and 10) are now in orbit. National Aeronautics and Space Administration (NASA) is responsible for procurement, launch, and checkout of these spacecraft before turning them over to NOAA for operational use. They were built by Space Systems Loral (SS/L) with components from many vendors.

A key element in overall mission success is the successful operation of the battery system. These spacecraft carry two Nickel-Cadmium (Ni-Cd) batteries designed and manufactured by SS/L with cells manufactured by Gates Aerospace Batteries (sold to SAFT in 1993). The battery, which consists of 28 cells with a 12 ampere-hour (Ah) capacity each, provides the spacecraft power needs during the ascent phase and during the semi-annual eclipse seasons lasting for approximately 45 days. The maximum duration eclipses are 72 minutes long which result in a 60 percent depth of discharge (DOD) of the batteries.

This paper provides a description of the batteries including the design and manufacturing process through acceptance testing and pre-launch preparation (References 1-4). Results are also provided from life cycle testing at the Naval Surface Warfare Center, Crane (Reference 5). The tests used battery packs prepared from cells out of the batch manufactured for spacecraft use. We also provide on orbit performance data from ascent phase, reconditioning, and from the eclipse seasons for the three spacecraft in orbit.

SPACECRAFT

The GOES 8, 9 and 10 spacecraft are three-axis stabilized geostationary satellites. Their dry mass is 977 kg and the nominal main bus power is 1150 W. They carry two main scientific instruments (Imager and Sounder) and a number of other measuring devices. During sunlight, they are powered by a single wing, two-panel solar array (SA). The SA has an output of 1300 W at beginning of life (BOL) Equinox and 1050 W at end of life (EOL) Summer Solstice. During eclipse, two 12 Ah Ni-Cd batteries with 28 cells each sustain the spacecraft. Power transfer from the batteries to the spacecraft bus is achieved through diode coupling. The power control unit (PCU) is the principal element for management and control of spacecraft primary power. The power control electronics (PCE) consists of the PCU, one sequential shunt unit, and four electro-explosive device (EED) extension units. Key features of the PCE are as follows:

a. Provides functions required for the primary bus.
b. Provides direct energy transfer of SA power to the power distribution bus.
c. Regulates and limits the voltage of the primary power bus to 42 ± 0.5 V dc during sunlight operation.
d. Minimizes primary bus ripple and voltage transients by use of SA regulation and primary bus filtering.
e. Allows flexibility of battery-charge control to optimize battery energy balance, thermal control.
f. Maintains minimum battery temperature control through use of thermistors and heaters in the associated batteries.
g Includes provisions for individual battery reconditioning.
h. Provides fail-safe and redundant actuation control of spacecraft EED (pyrotechnics) functions.
i. Eliminates single-part failure criticality by use of circuit redundancy, protective functions, fault protection of spacecraft heater loads, and alternate mode operation selectable by command.
j. Permits operational flexibility and status monitoring of key subsystem parameters by command and telemetry functions.

Redundancy Provisions. Power conditioning functions use both active and commandable redundancies. Commandable redundancy is provided for the following PCE functions:

a. Battery discharge control
b. Battery temperature control
c. Voltage telemetry monitors

d. Battery relay commanding
e. Battery charge control
f. EED ignition control (3 levels)
g. Battery reconditioning control

Some important functions for the battery control are listed below.

Battery Charge Control. The battery charge control configuration provides capability of up to eight discrete battery charge rates. Battery charge current is supplied by six charge control arrays located on the SA wing. Within the PCU, charge control array sections A, B, and C are connected through relays and isolating diodes to battery 1. A similar arrangement is provided for charge control arrays D, E, and F for battery 2.

Battery Temperature Control. The battery temperature control circuitry provides dual modes to connect and disconnect the heater. A total of four battery temperature controls are provided, one for each half of each battery. Automatic control of minimum battery temperature* at 5 (2) ±1 °C is provided by heaters integrated with each battery assembly and separate temperature controllers in the power control unit. A precision thermistor on each battery provides temperature feedback to these controllers. Manual override allows the heaters to be switched on or off by command.

* Value in parenthesis for GOES 8 only.

Battery Reconditioning. Individual reconditioning of each battery is provided by command. A parallel group of resistors mounted external to the power control unit provide the reconditioning load.

Commanded Load Control. Application of power to individual loads is through an on/off control input to each load dc/dc converter and by direct power bus switching of non-electronic loads (heaters).

Eclipse Load Controller. The GOES PCU contains redundant load controller functions. Each load controller monitors the SA current and the shunt current. Each primary power bus load (excepting command functions) is connected to and disconnected from the primary bus by command. In addition, power is automatically removed from sunlight loads upon eclipse entrance and automatically restored upon eclipse exit. Command override of this automatic function is provided.

BATTERY DESIGN DESCRIPTION

The battery subsystem of the GOES spacecraft consists of two assemblies, each containing 28 series-connected cells with a nominal capacity of 12 Ah and weighing approximately 12.9 kg each. Battery cell design has been guided by the life and reliability requirements of a 5-year geo-synchronous satellite application and incorporates features that reduce the effects of unavoidable Ni-Cd cell degradation modes. Figure 1 shows one of the two batteries (S/N 206) used on GOES-9.

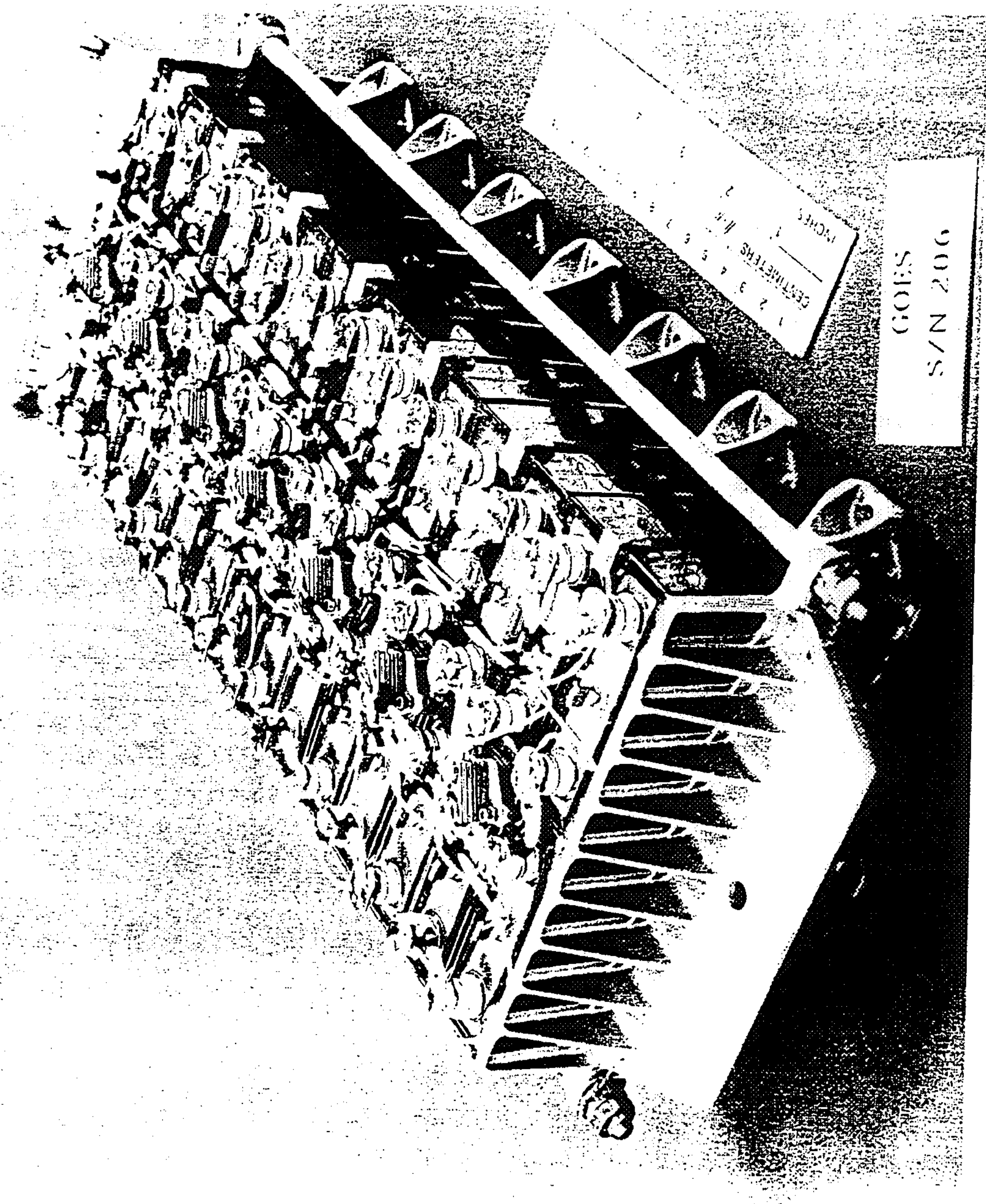


Figure 1. 12 Amp-hour NiCd Battery (S/N 206) used on the GOES-9 Spacecraft

Electrical Design. The nominal capacity of each battery assembly is 12 Ah. The 60% DOD design limit for maximum duration eclipse operation represents load capabilities of 409 W at BOL and 398 W at five years, based on average cell discharge voltages of 1.245 and 1.22 V, respectively, and a 1.0 V loss in the PCU. The 28 cells in series provide compatibility with charge voltage limits and deliver typical discharge voltages of 32.4 to 36.0 V.

Reconditioning. Reconditioning of the batteries during the one-month period preceding each eclipse season is required to maximize subsequent voltage performance and consequently minimize DOD.

Circuit Reliability. Circuit reliability within the battery is achieved by redundancy in wiring. Series connection between cells is provided by two parallel-connected, stranded copper wires soldered to the cell terminal lugs. Battery power connections are made to the terminals at the end of the cell series string by four redundant wires leading to the battery power connector.

Telemetry. The battery wiring harness design includes provision for telemetry of 28 individual cell voltages and battery temperatures. Additionally, overall battery voltage and current signals can be sensed.

Mechanical Design. The structural and mechanical design of the GOES battery is optimized to efficiently perform two important functions:
a. Cell support and restraint
b. Battery to equipment platform mounting.

Cell Support and Restraint. The basic mechanical design approach involves the restraint of two rows of 14 cells between two endplates, with each group of four cells supported by a rib structure. The key component to the battery design is the concept of supporting four cells on one rib, and compressing seven of these subassemblies between two endplate/tie-rod assemblies. The design of the rib positively holds the corners of the prismatic cells in place. The compression force exerted by the endplates on the cells is controlled by the torque applied by the two tie rods. The cells are pre-loaded to 40 lbf/in^2 in this manner. The end plates are designed to withstand a nominal cell overcharge pressure of 75 lbf/in^2 with minimal deflection. The endplates also provide mounting flanges for supporting the two connectors. The endplates are machined from heat-treated 7075-T73 aluminum, which alleviates stress corrosion issues, and the tie rods are titanium. The cell support ribs are cast from A357-T61 aluminum.

Thermal Design. The thermal design of the GOES battery is such that temperature control is both passive and active. The passive control is primarily conductive through the cell support ribs, and the active control is via resistive heaters mounted on the same ribs. The seven aluminum ribs are sized to effectively conduct the heat dissipated by the four adjacent battery cells to the equipment platform of the spacecraft. The design is such that temperature gradients are only 3 °C. The ribs also have a mounting flange for the resistive heater elements. These flanges are centrally located such that an even distribution of heat throughout the battery assembly is achieved at times when the heaters are activated. The seven heaters are sized to dissipate the required heat for thermal control. Temperature sensing for each battery is accomplished by four precision wafer thermistors located on the battery assembly. Battery flatness is maintained within 0.020 inch during assembly to allow for uniform thermal contact with the spacecraft. A silicone thermal grease is used between the battery and equipment platform on the GOES spacecraft to further reduce gradients and contact resistance.

Cell Design Summary. The Ni-Cd cell is designed to maximum margins for a greater than 5-year orbital lifetime. The prismatic Ni-Cd cells are packaged in a thin wall (0.012 in) 304L stainless steel container. A low profile terminal seal/cover configuration is used with double ceramic alumina-to-metal seals. Eleven positive and 12 negative plates are used. The specified negative-to-positive electrochemical capacity ratio is 1.70:1. This ratio ensures that the cells will remain positive limited during the 5-year mission. The positive electrode group has a theoretical electrochemical capacity of 15.6 ±1.2 Ah and the negative group 26.4 ±3.6 Ah. Active material loading is 1.25 ±0.06 and 1.55 ±0.065 Kg/m^2 for the positive and negative plates, respectively. The negative plates are treated with Teflon to reduce cadmium migration and increase the quantity of electrolyte used within the cell. The balance of positive and negative loading, electrochemical capacity ratio, and negative plate treatment enhance long-term cell performance required for the 5-year GOES mission. Non-woven Pellon 2536 nylon filament material approximately 0.007-0.009 inch thick is used for the separator. Weight concentration of 31% potassium hydroxide electrolyte provides for the transport of ions between the electrodes. This combination is being used for sealed Ni-Cd cells operated over the temperature range of 0 to 30 °C. Carbonate and nitrate levels are maintained at less than 2.0 g/l and 1.0 mg/l, respectively.

Interfaces. The battery interfaces are rigidly controlled during battery assembly. The battery mounting surface and mounting hole pattern are controlled within close tolerances for an assembly of this type. Flatness of the battery is maintained within 0.020 inch, and the mounting hole pattern is drilled within 0.020 inch. Two electrical connectors are provided on the battery assembly. The power connector is an eight-socket connector through which the battery is charged and discharged. Four bus wires each are connected to the positive and negative terminals. The second connector is a 50-socket connector. Through this connector the interface is provided for individual cells voltage sensing, thermistor measurement, and heater bus.

Battery and Cell History. The following designations refer to the flight batteries.

Spacecraft	Battery Serial Number
GOES-I	204, 203
GOES-J	205, 206
GOES-K	207, 208

Life-test. As with most NASA spacecraft, life-testing was started at Crane Naval Surface Warfare Center. A pack of cells for each satellite is undergoing real time or accelerated life-test. The test conditions are close to those in orbit with a 6.0 A discharge rate (equivalent to C/2) and 0.9 A charge rate followed by a trickle charge period. Before entering a shadow period a reconditioning cycle is run. The cycles shown are (Figures 2-4):

GOES I. Shadow Periods 2, 12, and 25;
GOES J. Shadow Periods 2, 9, and 25;
GOES K. Shadow Periods 1 and 10.

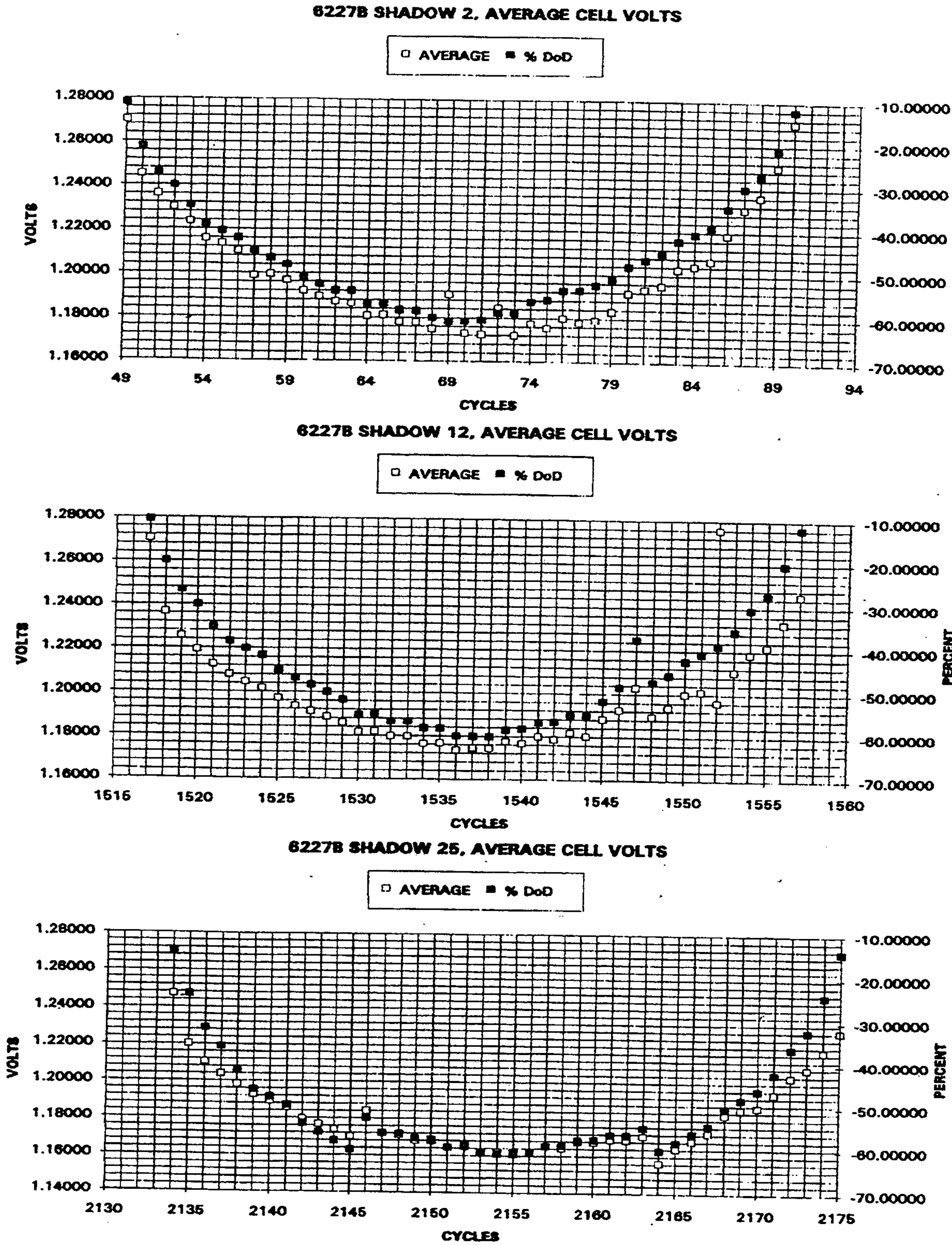


Figure 2. Battery DOD and Average Cell Voltages for Battery Pack 6227B (GOES-8) during shadow Periods 2, 12, and 25

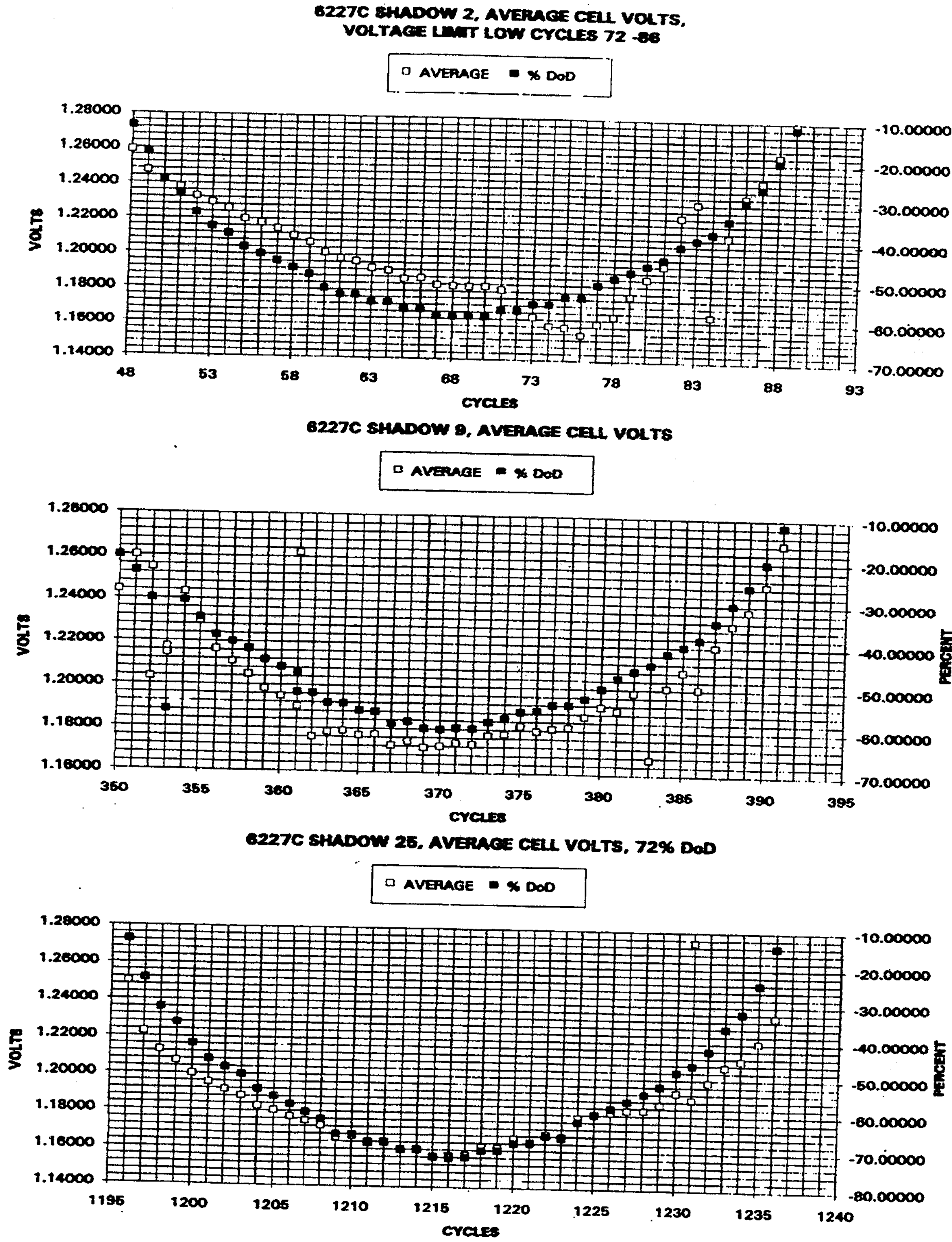
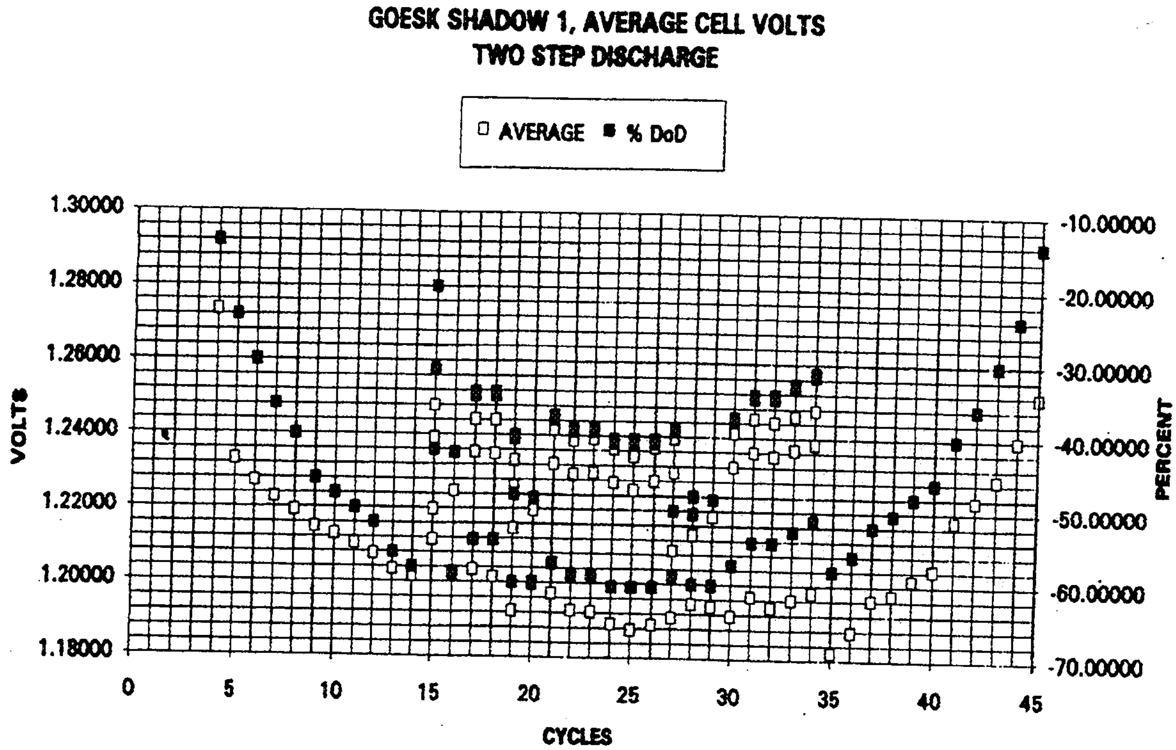


Figure 3. Battery DOD and Average Cell Voltages for Battery Pack 6227C (GOES-9) during Shadow Periods 2, 9, and 25



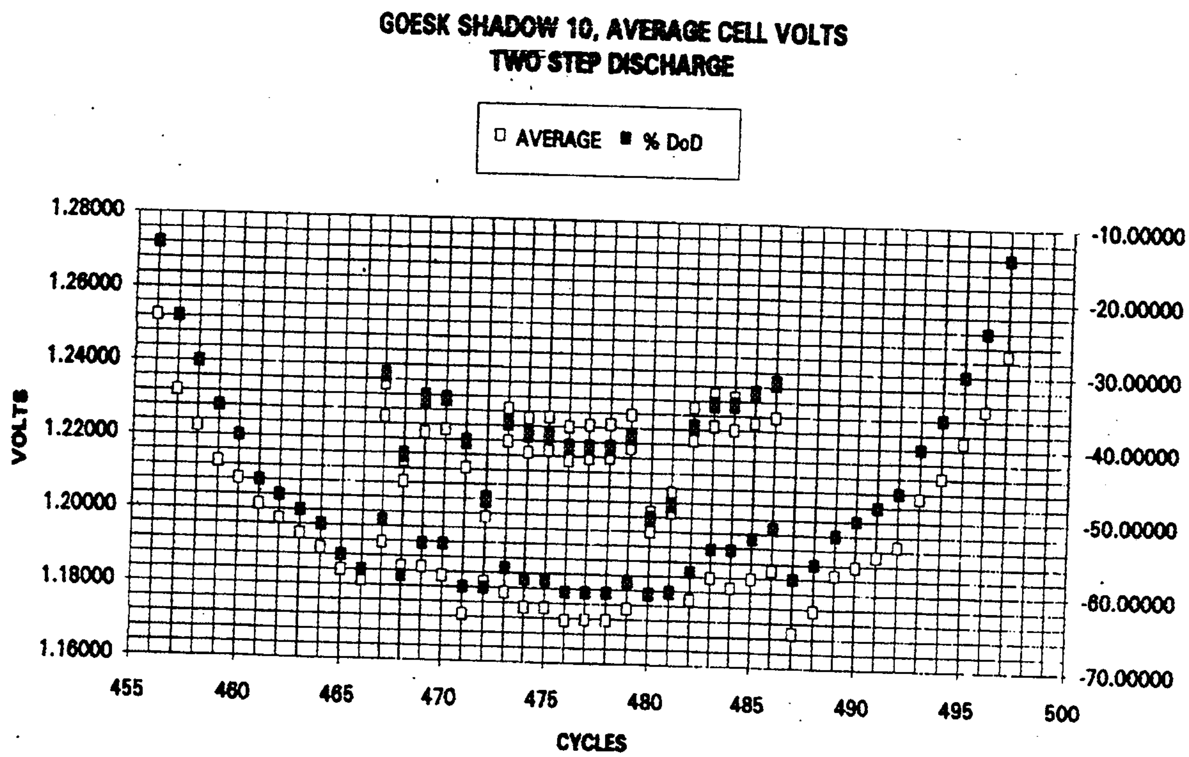


Figure 4. Battery DOD and Average Cell Voltages for Battery Pack GOESK (GOES-10) during Shadow Periods 1, and 10

All curves show nominal behavior and are within specification. There is also good agreement with the recorded flight data. Please note that shadow period 25 corresponds to 12.5 years of mission, which is more than twice the specified lifetime.

Manufacturing. The cells used for this battery were manufactured by Gates from 1989 to 1993, designated as lot # 5 with serial numbers 12AB31 and 12AB35. Based on an original 9 Ah design, they were name-plated as 12 Ah cells with specified acceptance capacity at 12.6 Ah. Out of the 500 originally manufactured cells 20 were taken into a 500 cycle test to address problems in two Aerospace Alerts. One alert from 1985, reported problems of negative electrode failure; and other, from 1988, reported problems in the hot gas sinter process and with the Pellon 2536 separator.

Storage. The cells were put in dry storage in a controlled environment

Acceptance Testing. The basic acceptance testing consists of seven steps (listed below), some of which were repeated until required results were achieved.

1. Reconditioning
2. Voltage Recovery
3. Capacity(20 °C): 12.8 Ah at 1.51 V max
4. Capacity(10 °C): 12.0 Ah at 1.52 V max
5. Capacity(0 °C): 11.5 Ah at 1.54 V max
6. Conditioning (20 °C): 12.8 Ah at 1.51 V
7. Voltage Recovery

Specified key values and acceptance test results for the GOES-I, -J, and -K batteries are listed in Table 1.

Table 1. Cell Acceptance Data for GOES-I, -J, -K Batteries.

Step	Requirement	GOES-I 9/20/88 - 5/20/90	GOES-J 7/16/90 - 7/6/92	GOES-K 6/7/93 - 5/11/94
3. Capacity at 20 °C				
Max Voltage	< 1.51 V	< 1.485 V	< 1.490 V	< 1.479 V
Capacity	> 12.8 Ah	12.8 Ah - 13.5 Ah	12.8 Ah - 13.2 Ah	11.9 Ah - 12.9 Ah
4. Capacity at 10 °C				
Max Voltage	< 1.52 V	< 1.510 V	< 1.514 V	< 1.52 V
Capacity	> 12.0 Ah	13.0 Ah - 14.3 Ah	13.4 Ah - 14.5 Ah	12.0 Ah - 13.1 Ah
5. Capacity at 0 °C				
Max Voltage	< 1.54 V	< 1.530 V	< 1.515 V	< 1.53 V
Capacity	> 11.5 Ah	12.7 Ah - 13.8 Ah	12.7 Ah - 14.3 Ah	12.1 Ah - 13.0 Ah
6. Conditioning at 20 °C				
Max Voltage	< 1.51 V	< 1.485 V	< 1.485 V	< 1.469 V
Capacity	> 12.8 Ah	12.8 Ah - 13.5 Ah	12.8 Ah - 13.5 Ah	11.8 Ah - 13.2 Ah
7. Voltage Recovery	> 1.145 V	N/A	N/A	> 1.190 V

BATTERY OPERATION

The two 12 Ah rated Ni-Cd batteries connected to the primary bus via redundant diodes and relays are designed for operation at a 60% maximum DOD for the mission. Criteria for battery management, including power subsystem design features to implement the same, are discussed below.

Functions provided for each battery by the power subsystem to monitor, control, and protect the batteries are as follows:

a. Battery voltage monitors
b. Individual cell voltage monitors
c. Battery charge current monitors
d. Battery discharge current monitors
e. Battery temperature monitors
f Battery heater status monitors
g. Battery relay status monitors
h. Battery reverse current monitors
i. Battery relay control commands
j. Battery charge rate control commands
k. Battery reconditioning controls
l. Battery charge voltage limiting
m. Thermostatically controlled battery heaters

Pre-launch and Launch. During the pre-launch and launch phases of the mission, the spacecraft is kept on external power until approximately 4 minutes before launch. Should a launch hold of more than 10 minutes be encountered while the spacecraft is on internal power, consideration should be given to recharging the batteries before continuing with the launch operation.

Battery Charge Control. Batteries can be charged in either a continuous or sequenced mode. The primary consideration on the following recommended battery charge control implementation is to minimize battery stress while providing adequate charge return to ensure energy balance. This objective is satisfied using the lowest practical charge rate in the sequenced mode to minimize average battery temperature. The methodology recommended has been flight proven on previous SS/L programs and verified by life cycle testing.

General Configuration Control. The electrical power subsystem permits considerable flexibility in battery charge management. Charge power is derived from the SA using two identical groups of three charge control arrays. Group 1 consists of arrays A, B, and C; group 2 of arrays D, E, and F. The typical current output of these arrays is summarized in Table 2 for various conditions.

Table 2. Battery Charge Array Current

	Charge Control Array Module		
Season/Life	A, D	B, E	C, F
Vernal Equinox			
BOL	0.99	0.33	0.33
EOL	0.91	0.30	0.30
Autumnal Equinox			
BOL	0.97	0.32	0.32
EOL	0.89	0.30	0.30
Summer Solstice			
BOL	0.86	0.29	0.29
EOL	0.80	0.27	0.27
Winter Solstice			
BOL	0.93	0.31	0.31
EOL	0.87	0.29	0.29

Two basic charge modes are available. First, with the normal recommended synchronous orbit sequenced mode, current is applied to the two batteries in an alternating fashion in an approximately 10-minute cycle. Any combination of the six charge arrays can be switched between the batteries. Alternately, any combination of arrays A, B, and C can be used for battery 1 and any combination of arrays D, E, and F for battery 2. Second is the continuous charge mode. In this mode only the latter arrangement is practical.

For one time only, a maximum DOD of 70% would be allowed for a battery temperature maintained between 5 °C and 25 °C. This maximum DOD, if approached, allows no margin for contingencies. To provide such a margin, a mission goal for maximum DOD should not exceed 60%.

Normal Eclipsed Orbits Charge Control. Battery charge control performance is evaluated by use of the following telemetry provided for each battery:

a. Battery voltage
b. Individual cell voltages
C. Charge current
d. Battery temperature
e. Battery heater status

The 110% charge return at the full rate, followed by trickle charge until the next eclipse, ensures sufficient recharge for all eclipse cycles.

Battery Temperature Control. Battery temperature profiles are largely governed by the battery charge profiles. Thermostatically controlled 19.5 watt battery heaters are provided to maintain the minimum battery temperature* above +4 (1) °C. These heaters are designed to cycle on at +5 (2) ±1 °C and off at +9 (5) ±1 °C. Heater operation is controlled by a precision thermistor in each battery in conjunction with level detectors and relay drivers within the PCU. This function can be overridden by command to either turn the battery heater on or off regardless of battery temperatures. The intended use of this override function is to provide manual heater control in the event of a failure in the automatic heater control.

* Values in parenthesis for GOES 8 only.

Battery Reconditioning. To maximize battery discharge voltage during each eclipse, both batteries are reconditioned prior to the start of each eclipse season. These reconditioning cycles are performed as close as possible to the next solar eclipse to obtain the maximum benefit.

ON-ORBIT PERFORMANCE

First of the GOES series I-M spacecraft was launched on April 13, 1994, and was renamed GOES 8 after achieving nominal operational orbit. The next two were launched on May 23, 1995 (GOES 9) and April 25, 1997 (GOES 10). The batteries provide spacecraft power needs from just before launch to the time of partial SA deployment soon after launch (the outer panel is deployed approximately ten days later providing full power). The batteries also shoulder the power needs during eclipses, any maneuvers causing loss of solar power, and whenever the power produced falls short of the spacecraft needs. The sections below discuss battery performance during these various phases. Specific data is generally provided for GOES 10. However, data from GOES 8 and GOES 9 is also included whenever this data was useful to establish trends.

Ascent phase Battery Performance

The batteries on the GOES spacecraft provide power to the primary bus starting a few minutes before launch when the spacecraft is switched to internal power. This first round of battery support is completed when the SA is partially deployed approximately 1.5 hours after launch. During the next few days, the batteries provide the spacecraft power needs during special events such as the magnetometer boom and full SA deployment phase and the dipole estimation phase. Also depending upon the time in orbit before the Apogee Maneuver Firing No. 1 (AMF #1), the batteries provide support during several eclipses lasting approximately 15 minutes each.

For all cases when the batteries experienced a non-zero discharge current, data was retrieved from the archives at 5.12 second intervals. The charge removed during discharge (D) from the batteries was then calculated as an integral of the discharge current over time, i.e.,

$$D\ (Ah) = \int I\, dt.$$

The integral is evaluated numerically by using current values at 5.12 sec intervals. The DOD is then given as a percentage fraction of the nameplate capacity (C_0) of the batteries (12 Ah each for a total of 24 Ah for the system). Thus

$$DOD = 100 * D\ (Ah) / C_0$$

gives the DOD value for the individual battery or the system as a whole depending on the values of the current and capacity used above.

Table 3 lists all times during the ascent phase when the batteries on the GOES-K spacecraft were subjected to discharge. The maximum DOD was recorded during the Magnetometer boom and SA deployment phase; corresponding battery discharge currents are shown in Figure 5.

Table 3. GOES-10 Battery Discharge History from Launch to Early On-orbit

Date	DOY	Time Range	Event	Max Dis I Battery 1	Max Dis I Battery 2	Max DOD (%)
4/25/97	115	05:43 - 07:08	Launch to Partial SA deploy	4.0	4.1	20.6
4/25/97	115	18:33 - 18:47	Transfer Orbit Eclipse #1	4.4	5.2	9.5
4/26/97	116	07:11 - 07:25	Transfer Orbit Eclipse #2	5.4	5.9	10.2
4/26/97	116	19:48 - 20:03	Transfer Orbit Eclipse #3	5.7	6.0	12.7
4/29/97	119	16:50 - 17:30*	Apogee Maneuver Firing #2	1.6	1.3	4.0
5/04/97	124	23:14 - 00:01	Mag Boom & SA Deploy	4.5	4.7	26.0
5/05/97	125	18:03 - 18:30	Dipole Estimation	7.1	7.4	22.7

* Intermittent discharge currents (up to 0.6 A) from 17:30 to 18:00.

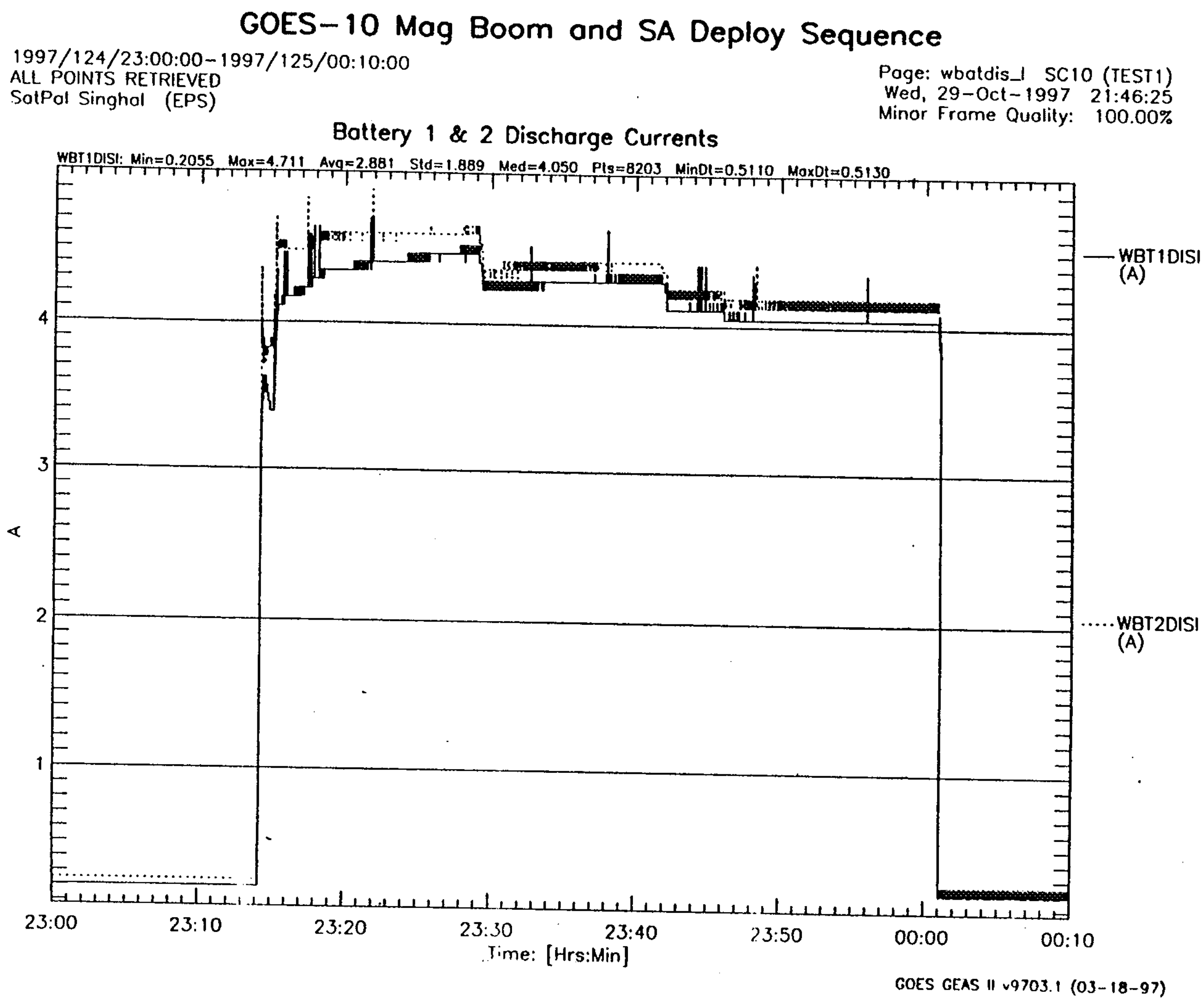


Figure 5. Battery Discharge Currents During the Magnetometer Boom and full Solar Array Deployment Phase for the GOES-10 Spacecraft.

Battery Reconditioning

Spacecraft Batteries are reconditioned prior to the start of each eclipse season. The batteries are individually reconditioned by use of the following sequence after verifying that the other battery is connected to the spacecraft bus.

a. Turn off battery charging
b. Open battery discharge relay number 2
c. Inhibit the battery under voltage protection
d. Turn on battery reconditioning

The 139.6 ohm resistive load is connected across the battery, resulting in an initial C/48 (0.25 A) reconditioning discharge rate. The individual cell voltages of the selected battery are monitored throughout the reconditioning discharge period. When the first cell voltage reaches 0.5 ± 0.1 V, the reconditioning discharge is terminated. Figure 6 shows the battery reconditioning circuitry.

On-orbit reconditioning has been performed prior to 7 eclipse seasons for GOES-8, and 5 eclipse seasons for GOES-9. The batteries on GOES-10 were not reconditioned prior to the fall 1997 eclipse season.

Charge removed from the batteries during reconditioning was calculated using a different approach than that described earlier. During reconditioning, the nominal discharge current has a value of 0.25 A (C/48 rate). However, the step size for the discharge current telemetry is 0.06 A, too coarse to show discharge current changes as the battery voltage changes. Since the battery is being discharged by connecting it to a constant resistor (139.6 Ohms), the discharge current is given by the use of Ohm's law

$$I = V/R$$

and the charge removed as an integral of battery voltage, i.e.,

$$D\ (Ah) = (1/R) \int V\, dt$$

In addition, since the reconditioning process continues for 60 - 65 hours, the voltage data for integration is sampled at 1-minute intervals at the beginning and end of the process (where the voltage is changing comparatively rapidly) and at 5-minute intervals during the middle 48 hour period.

Figure 7 shows the performance of GOES-8 battery 1 during its first reconditioning cycle (Fall 1994). The reconditioning was terminated when cell 12 voltage dropped to a value of 0.5 V. Corresponding data for battery 2 is shown in Figure 8.

Table 4 compares the results from all reconditioning cycles to date: seven for GOES-8 and five for GOES-9. The data show that the battery capacity has improved with time. The table also shows the end of discharge (EOD) battery voltage for each case.

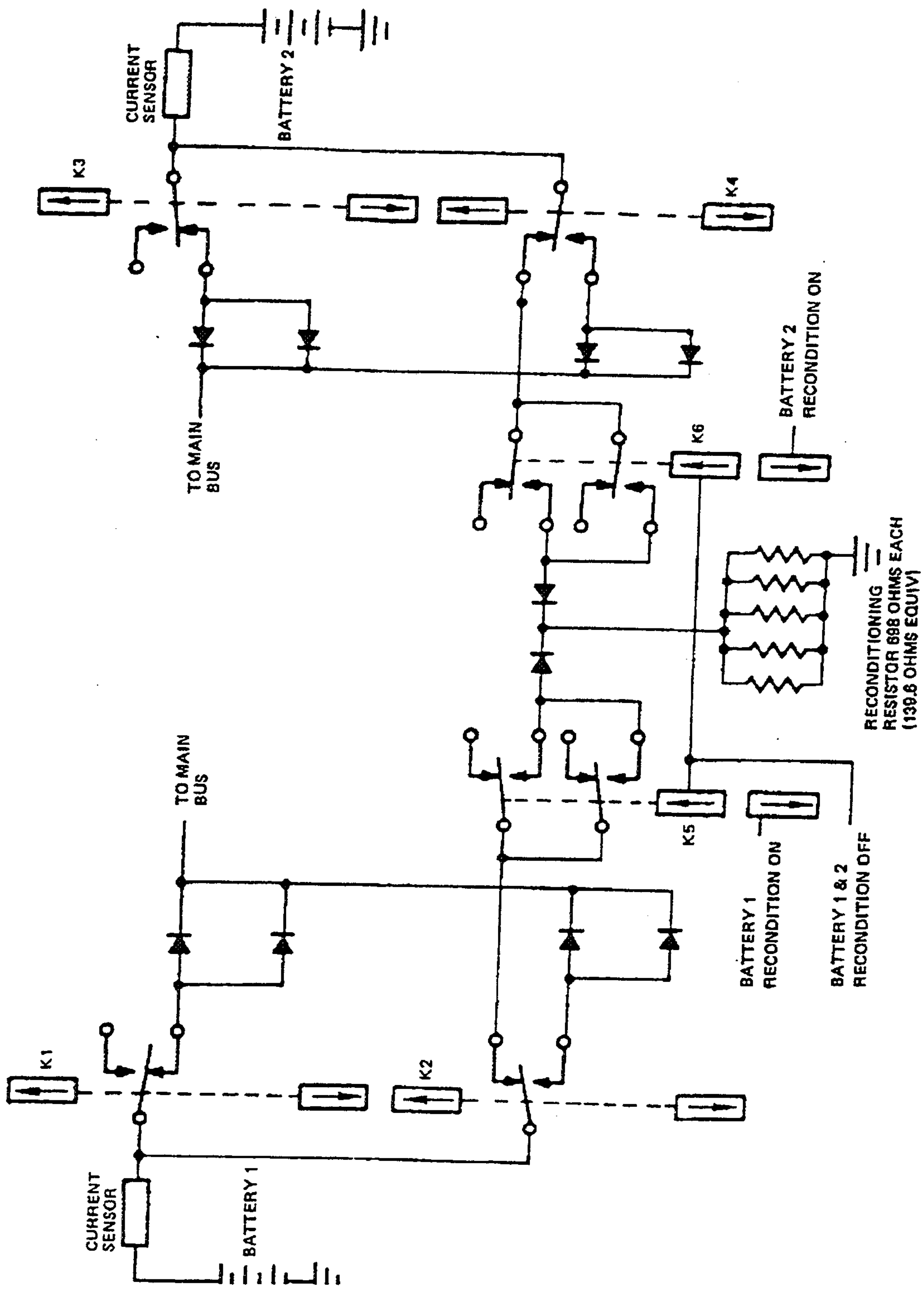


Figure 6. Battery Reconditioning Circuitry

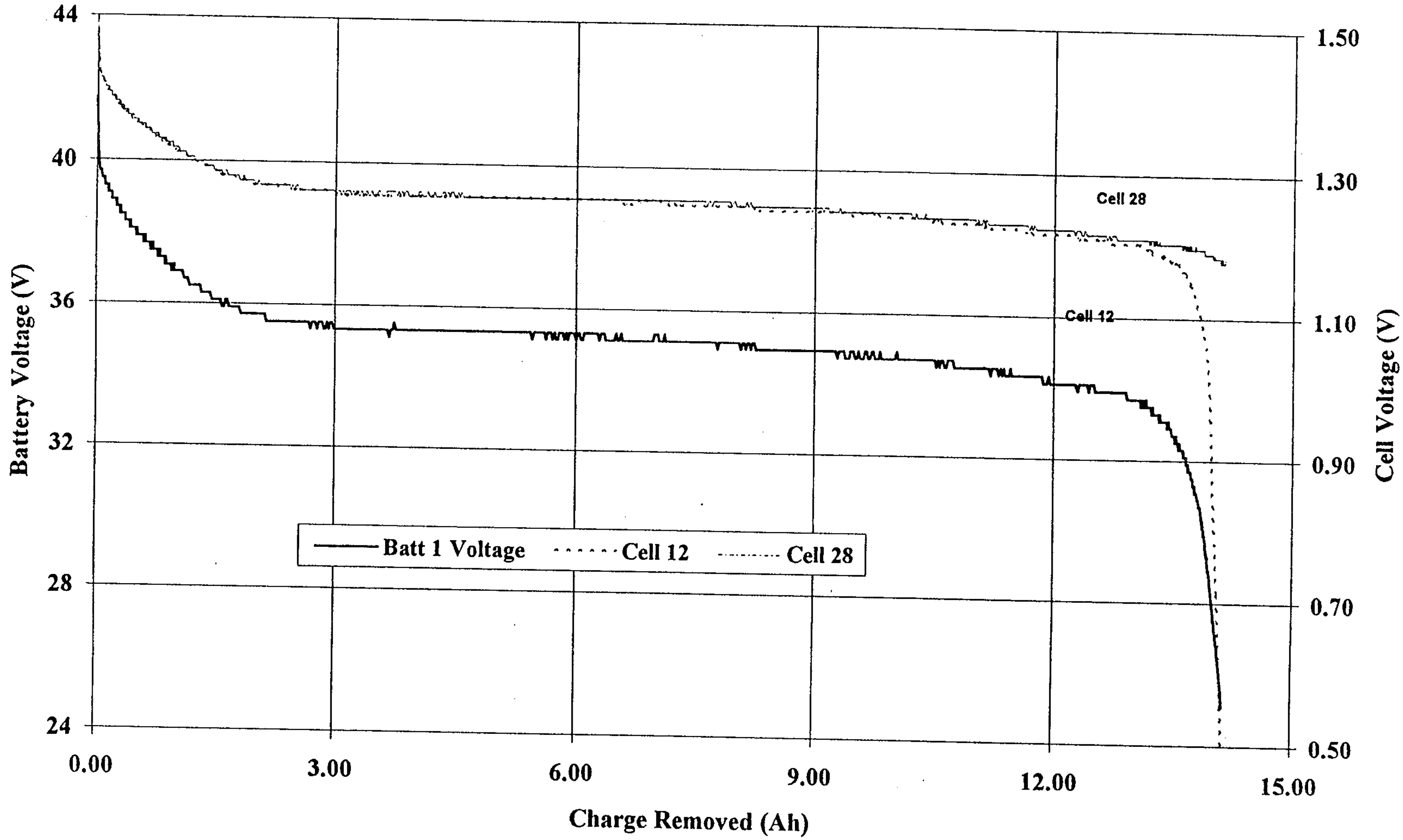


Figure 7. GOES-8 Fall 1994 Battery 1 Reconditioning: Battery and Cell Voltages for the Weakest (Cell 12) and Strongest (Cell 28) Cells as a Function of Charge Removed from the Battery.

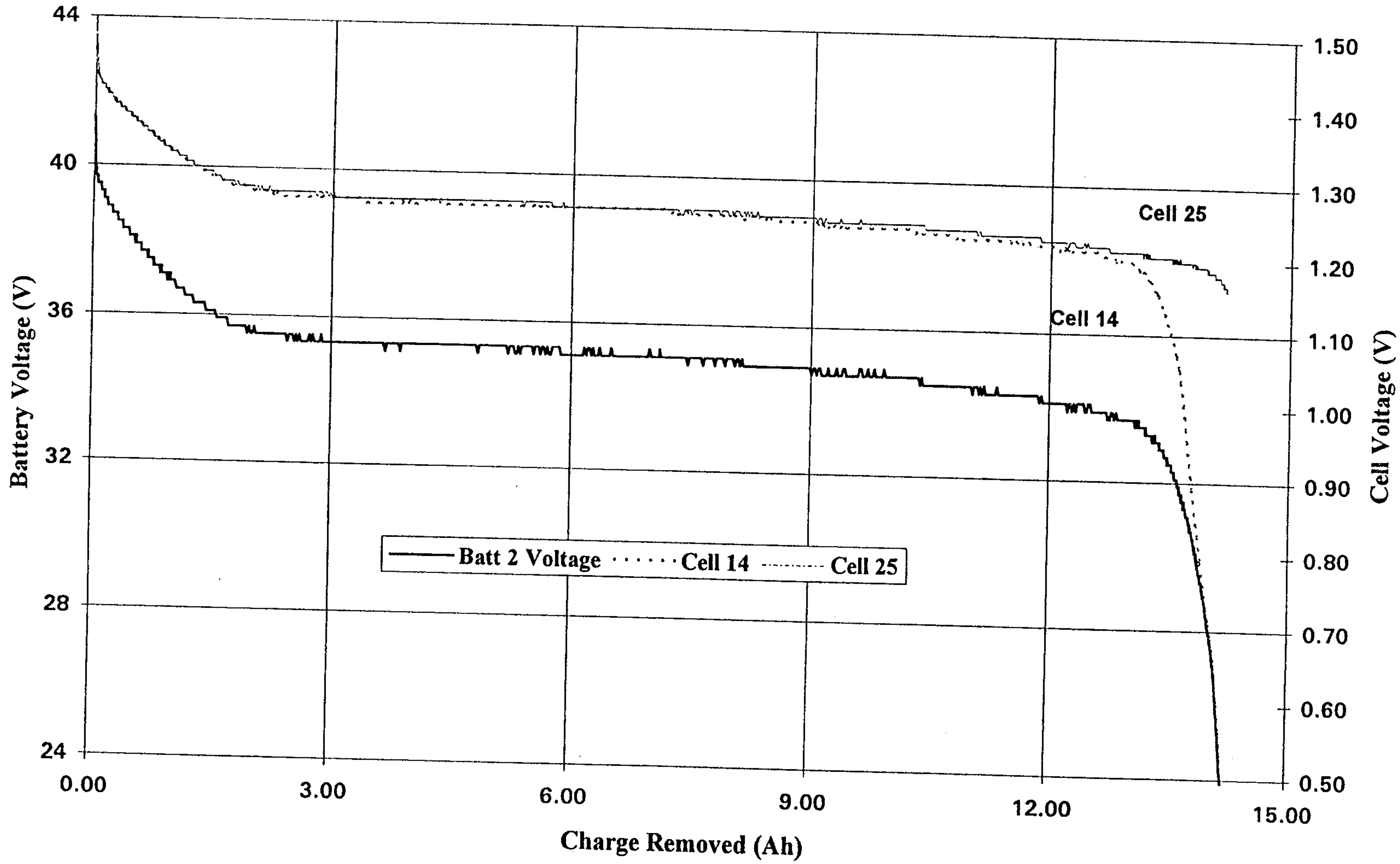


Figure 8. GOES-8 Fall 1994 Battery 2 Reconditioning: Battery and Cell Voltages for the Weakest (Cell 14) and Strongest (Cell 25) Cells as a Function of Charge Removed from the Battery.

Table 4. Battery Reconditioning Results for GOES-8 and -9.

		GOES-8		GOES-9	
		Ah Removed	EOD Voltage	Ah Removed	EOD Voltage
Fall 1994	Battery 1	14.14	25.1		
	Battery 2	14.21	23.9		
Spring 1995	Battery 1	15.25	19.7		
	Battery 2	15.41	19.1		
Fall 1995	Battery 1	15.32	20.3	14.06	22.9
	Battery 2	15.60	19.7	13.82	27.5
Spring 1996	Battery 1	15.89	18.1	15.23	18.9
	Battery 2	15.92	18.3	15.14	19.7
Fall 1996	Battery 1	15.57	19.3	15.30	19.3
	Battery 2	15.73	18.9	15.31	20.1
Spring 1997	Battery 1	16.00	16.3	15.77	17.7
	Battery 2	16.09	17.1	15.60	19.5
Fall 1997	Battery 1	15.68	18.9	15.40	18.7
	Battery 2	15.69	19.5	15.26	19.7

Battery Performance During The Eclipse Seasons

The GOES spacecraft experience the loss of solar power during the semi-annual eclipse seasons that last approximately 45 days each centered on the Vernal and Autumnal equinox. The duration of the eclipses vary from a few minutes to a maximum of 72 minutes on or near the equinox. We have analyzed the eclipse data for all 13 eclipse seasons (7 for GOES-8, 5 for GOES-9, and one for GOES-10) and found it to be consistent and predictable. We provide details of the latest (Fall 1997) eclipse season below along with some statistical data from the previous cases.

Figure 9 shows the daily battery DOD during the Fall 1997 eclipse season for GOES-10. The figure shows the DOD for batteries 1 and 2 and for the battery system as a whole. As the eclipse duration gets longer, the batteries are discharged for a longer period resuting in almost a linear increase in the battery DOD. The DOD curve, as shown, is not smooth because of the active load management that had to be performed to stay within the 60 % DOD limit. Operationally, we have the so-called "20-minute" rule that was imposed after a power amplifier failure on GOES-8. The uplink carriers from the ground station to the spacecraft provide extra heating to the power amplifiers during the eclipse but result in a higher DOD that would exceed the 60% limit during the longer eclipses. A balance is struck between the two requirements by keeping the carriers up during the eclipse except during the central portion of the season when the carriers are brought down at the start of the eclipse and then brought up 20 minutes before the end of the eclipse. During the central seven days of the eclipse season, this 20-minute rule was further modified to a 10-minute rule. In addition, the Attitude and Orbit Control Subsystem (AOCS) team kept both Earth Sensors operating during the eclipse season except for the days when the power engineer required one of them to be turned off to avoid excessive drain on the batteries.

Figure 10 shows the total DOD for the battery system for all three spacecraft for the Fall 1997 season. The data for GOES-8 and -9 also show the effects of the power management by modifying the times for the uplink carriers. However, other aspects of active power management are evident only for the GOES-10 spacecraft since this was the first eclipse season for it.

Figure 11 shows the battery discharge currents for Day of Year (DOY) 266 (September 23, 1997) for the three spacecraft. The eclipse times are separated by 2-hour intervals reflecting the fact that the three spacecraft are geostationary at 75 degrees W (GOES-8), 105 deg W (GOES-10), and 135 deg W (GOES-9) longitude. The figures show a sharp rise in the discharge currents near the end of the eclipse due to the uplink carriers (approximately 90 Watt increase in power consumption).

Figures 12 through 14 show the battery minimum voltage versus DOD for batteries 1 and 2 for GOES-8, -9, and -10 respectively. GOES-10 data shows larger spread between the battery voltages at the same DOD (encountered before and after the maximum duration eclipse), probably reflecting the fact that these batteries had not gone through reconditioning, as mentioned earlier. Figure 15 shows the battery minimum versus DOD data for all 6 batteries together. The results clearly indicate that the batteries are behaving as a family.

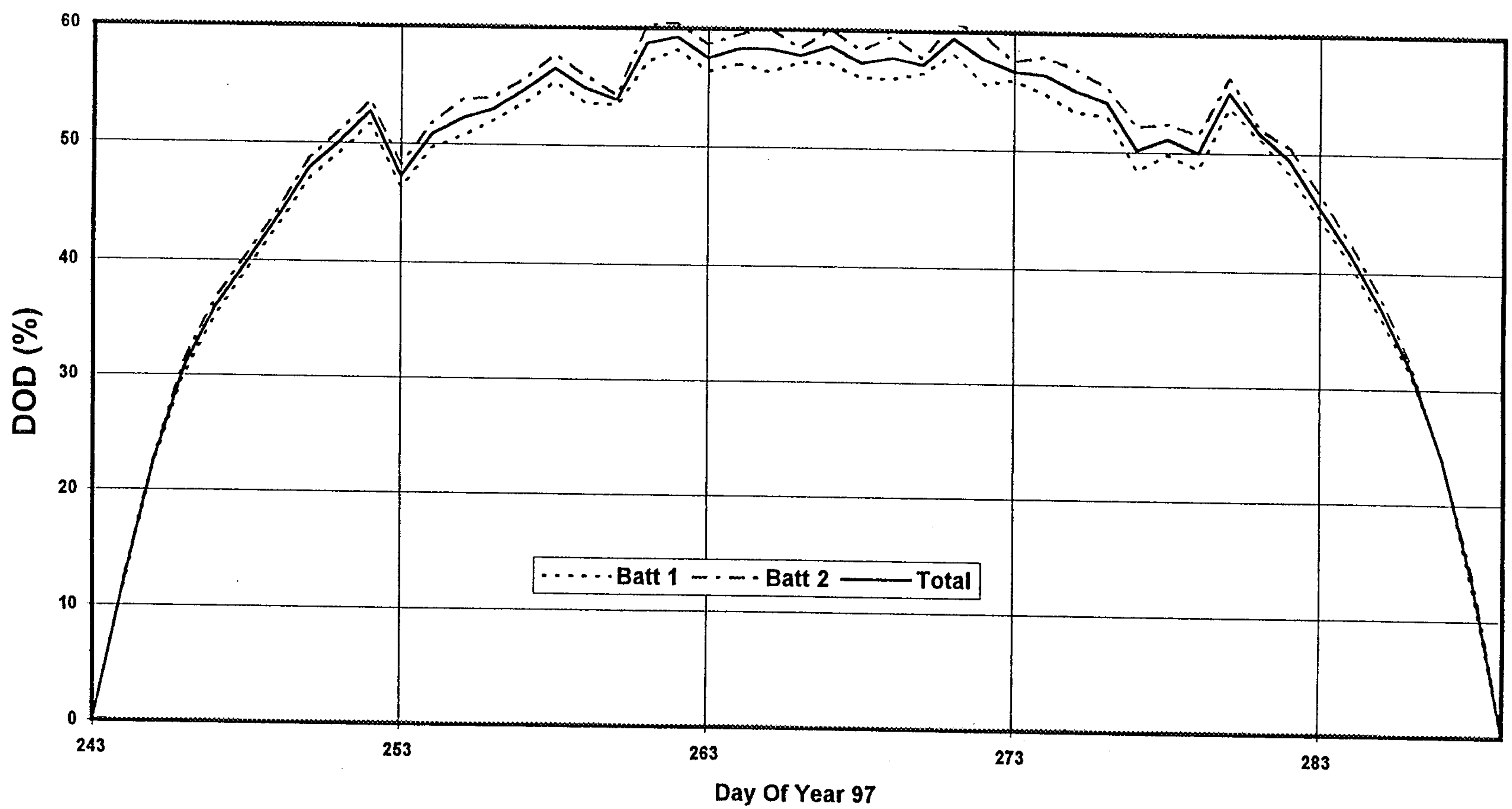


Figure 9. GOES-10 Battery Depth of Discharge (DOD) During the Fall 1997 Eclipse Season.

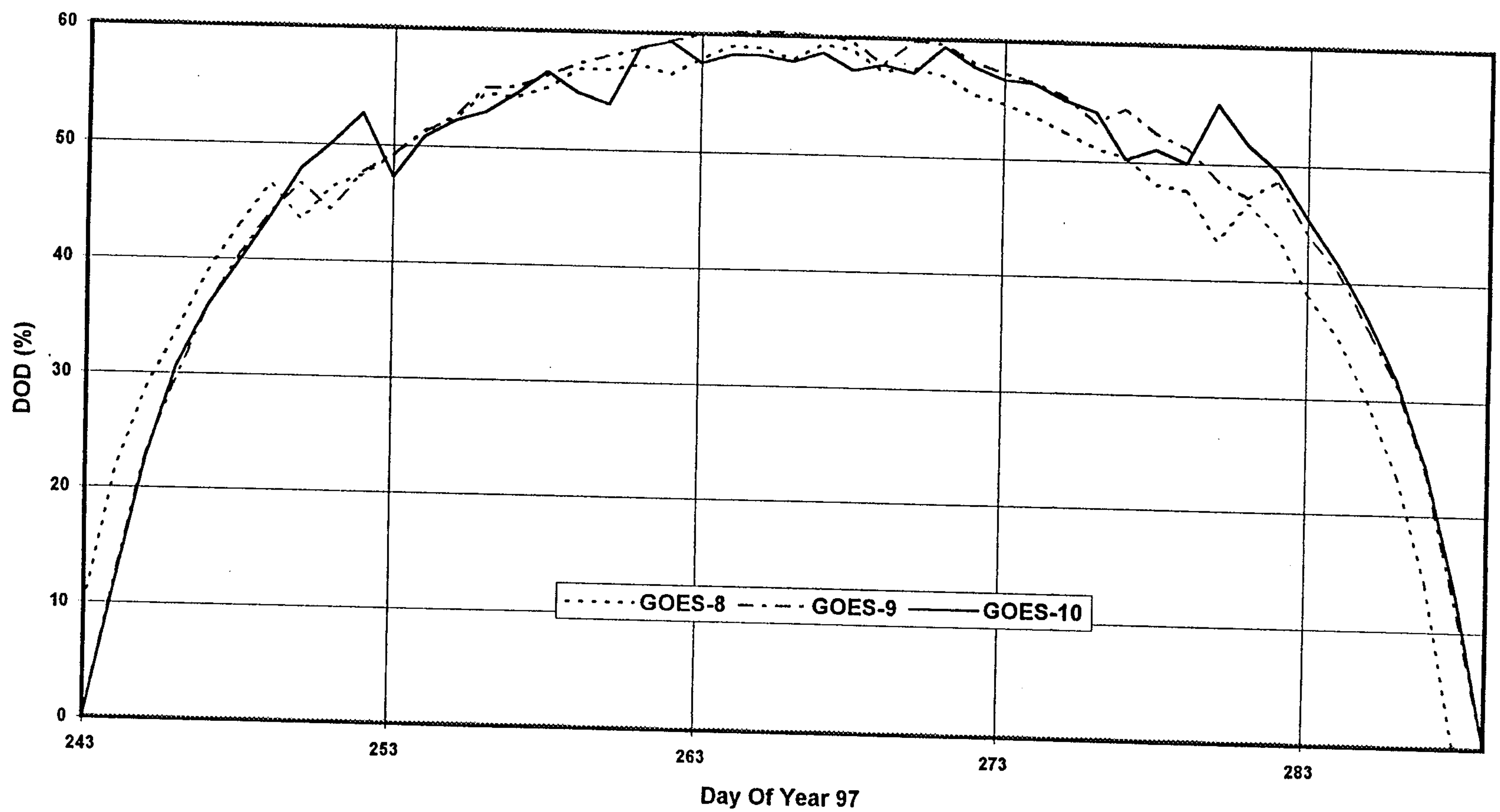
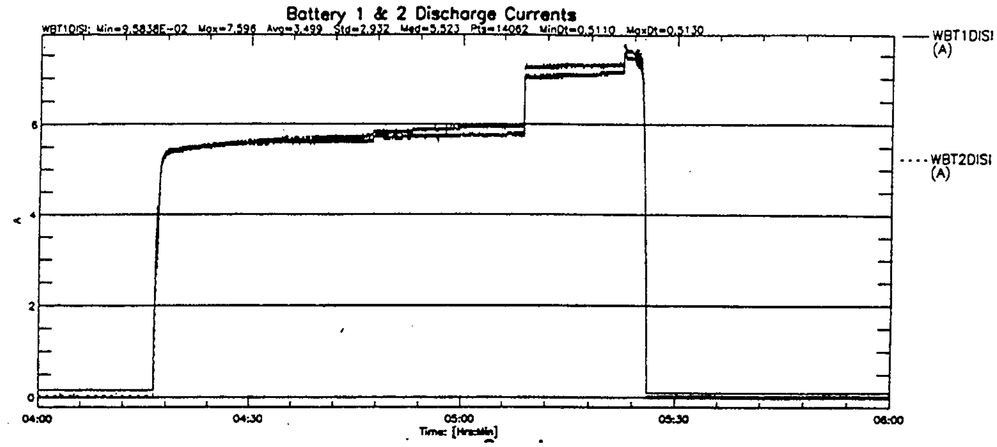


Figure 10. Battery Depth of Discharge (DOD) During the Fall 1997 Eclipse Season for GOES-8, GOES-9, and GOES-10.

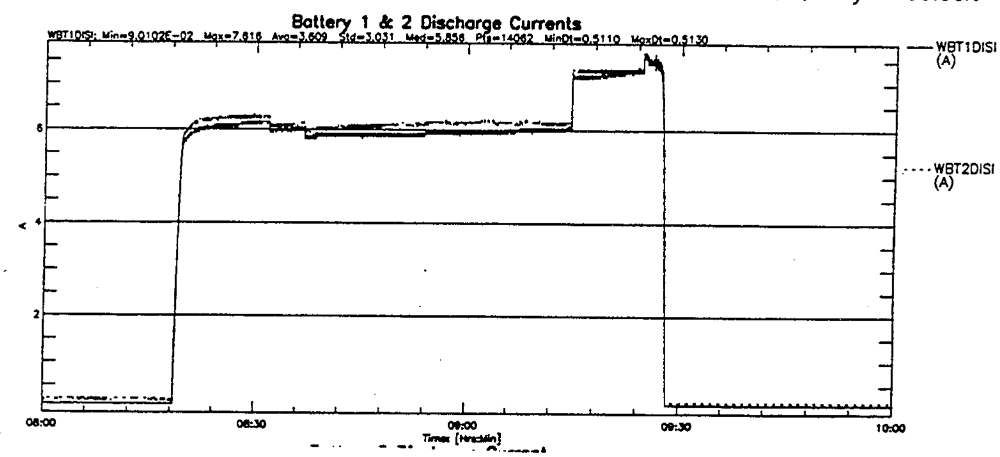
1997/266/04:00:00–1997/266/06:00:00
ALL POINTS RETRIEVED
SatPal Singhal (EPS)

Page: wbatdis_J SC08 (TEST1)
Tue, 4–Nov–1997 10:17:13
Minor Frame Quality: 100.00%



1997/266/08:00:00–1997/266/10:00:00
ALL POINTS RETRIEVED
SatPal Singhal (EPS)

Page: wbatdis_J SC09 (TEST1)
Tue, 4–Nov–1997 10:17:51
Minor Frame Quality: 100.00%



1997/266/06:00:00–1997/266/08:00:00
ALL POINTS RETRIEVED
SatPal Singhal (EPS)

Page: wbatdis_J SC10 (TEST1)
Tue, 4–Nov–1997 10:18:08
Minor Frame Quality: 100.00%

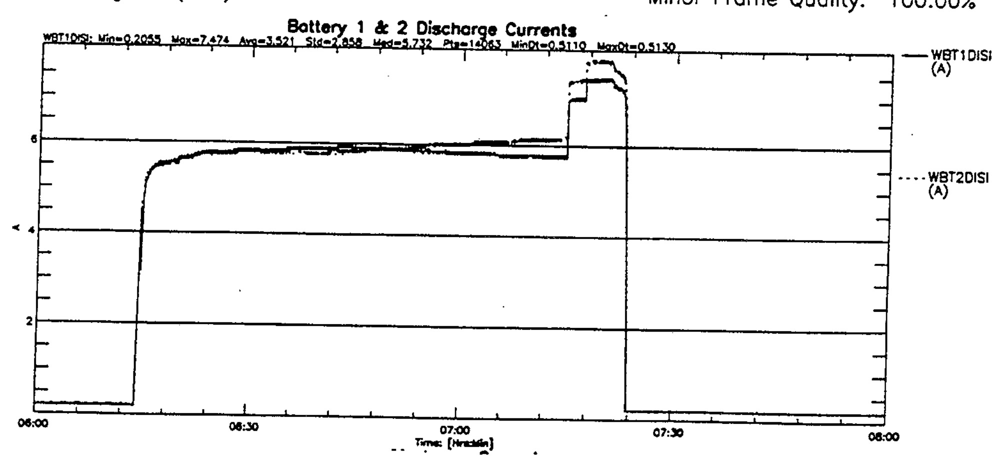


Figure 11. Battery Discharge Currents During the eclipse on Day 266 (September 23) of the Fall 1997 Eclipse Season for (a) GOES-8, (b) GOES-9, and (c) GOES-10.

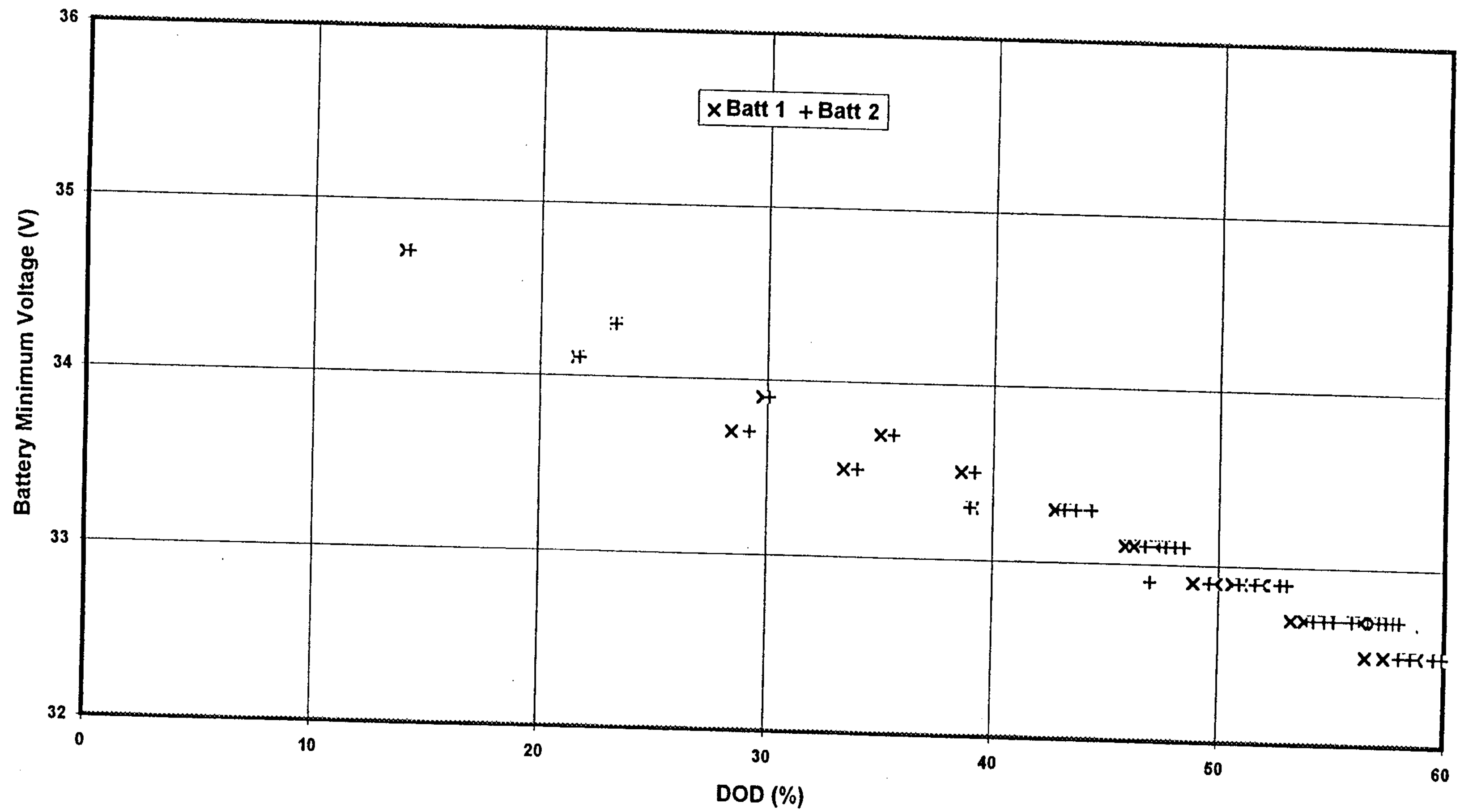


Figure 12. Battery Minimum Voltage as a Function of Battery Depth of Discharge (DOD) During the Fall 1997 Eclipse Season for GOES-8.

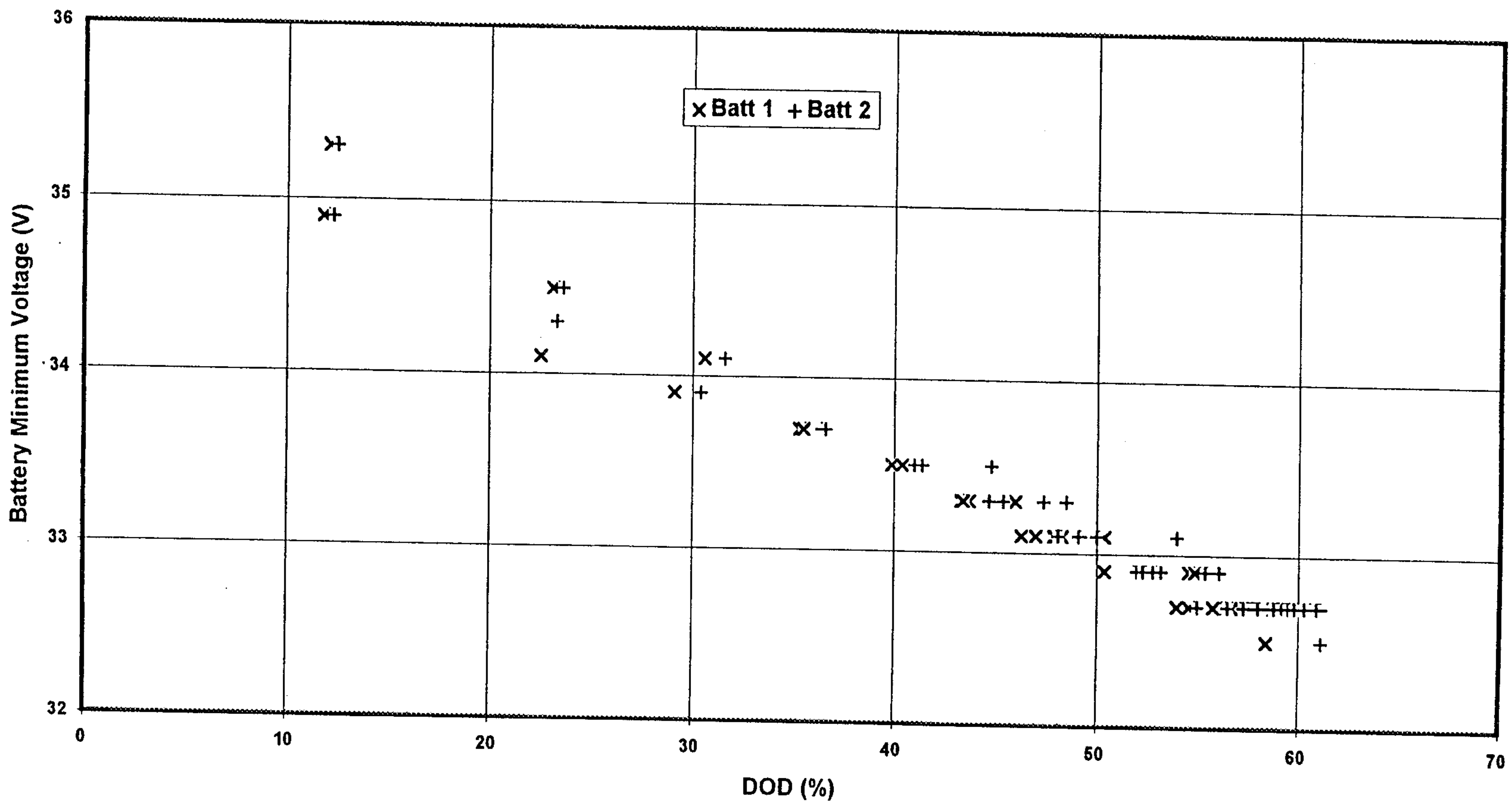


Figure 13. Battery Minimum Voltage as a Function of Battery Depth of Discharge (DOD) During the Fall 1997 Eclipse Season for GOES-9.

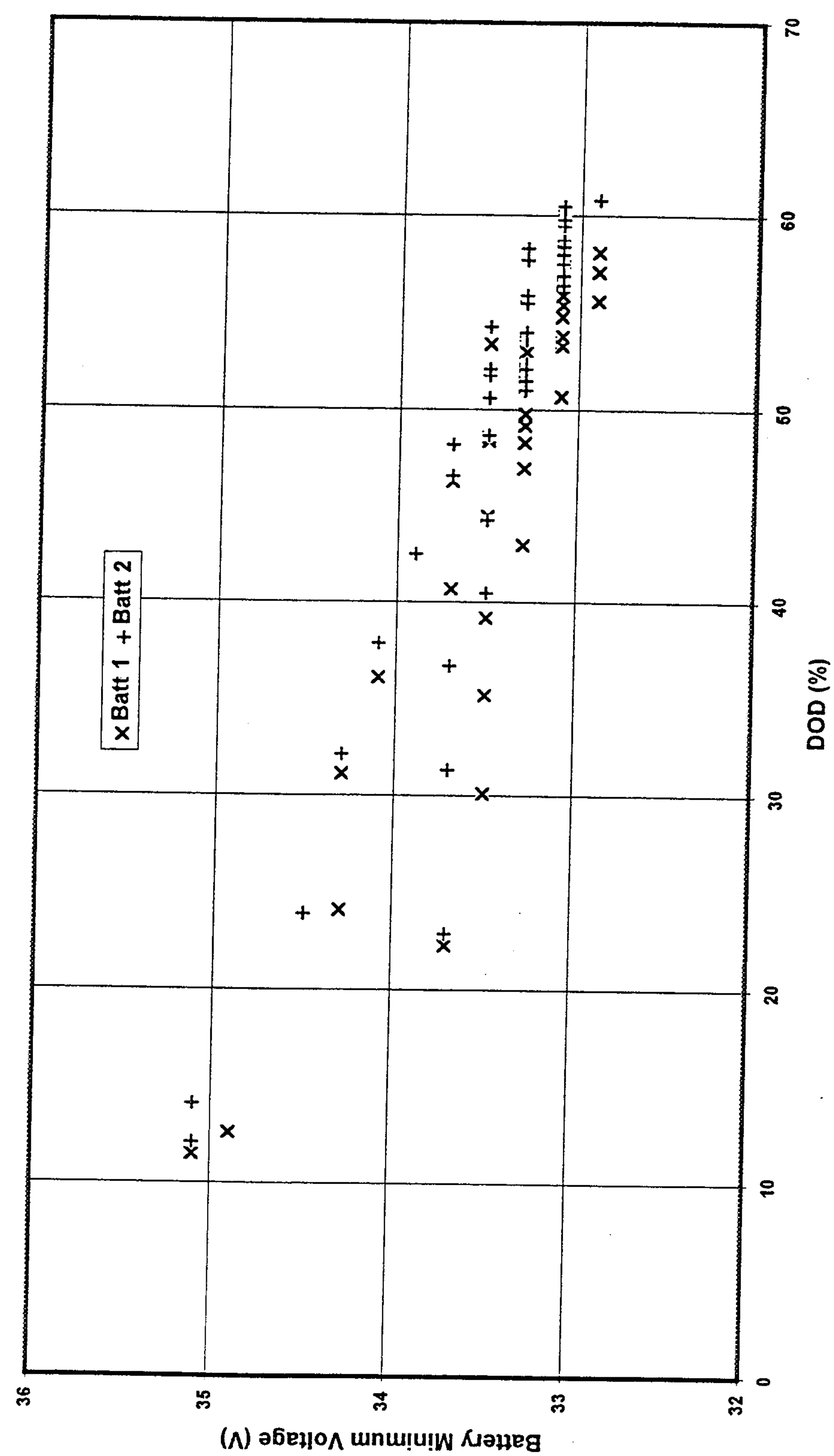


Figure 14. Battery Minimum Voltage as a Function of Battery Depth of Discharge (DOD) During the Fall 1997 Eclipse Season for GOES-10.

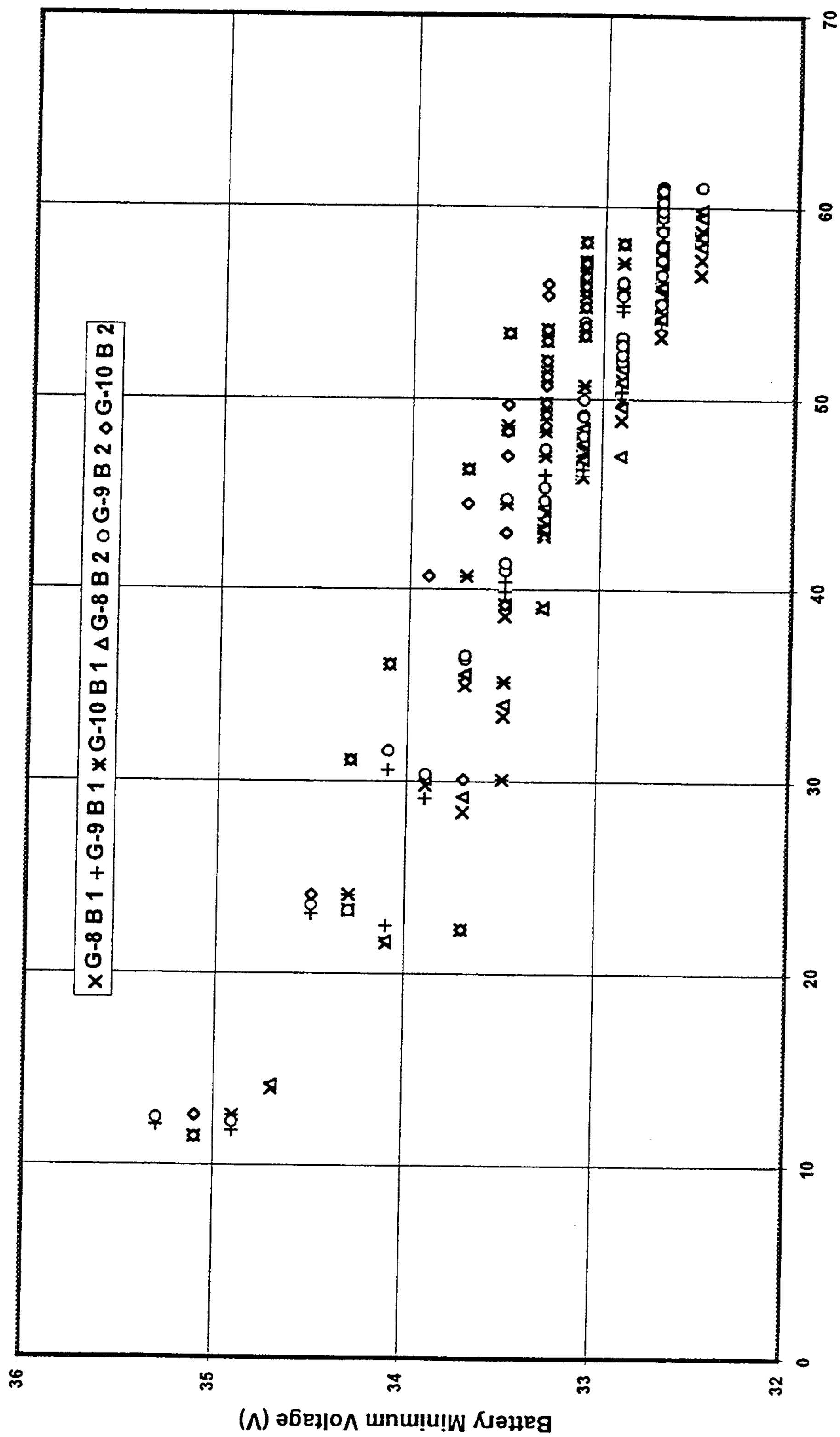


Figure 15. Battery Minimum Voltage as a Function of Battery Depth of Discharge (DOD) During the Fall 1997 Eclipse Season for all 6 Batteries on the GOES-8, -9, -10 Spacecraft.

During the Fall 1997 eclipse season, the minimum battery voltages recorded were 32.5 V, 32.5 V, and 32.9 V (both batteries) for GOES 8, 9, and 10 respectively. The minimum cell voltages recorded for the three spacecraft were: 1.160 V (Battery 1) and 1.161 V (Battery 2) for GOES-8; 1.161 V (both batteries) for GOES-9; and 1.174 V (Battery 1) and 1.186 V (Battery 2) for GOES-10.

The battery temperatures ranged from approximately 0 °C to 9 °C for GOES-8, 1 °C to 12 °C for GOES-9, and 2 °C to 13 °C for GOES-10. Daily peaks in the battery temperatures were observed near the end of the discharge cycle while the minimums were recorded a few hours before the end of charge cycle, as expected.

Table 5 lists the minimum battery voltages during all of the eclipse seasons encountered by the three spacecraft. As expected, the minimum battery voltages get lower with the aging of the batteries but they seem to have leveled off at 32.5 Volts. The battery temperatures have stayed in the same range for all eclipses (values as indicated above for the latest eclipse season).

Table 5. Minimum Battery Voltages During the Eclipse Seasons

	GOES-8		GOES-9		GOES-10	
	Battery 1	**Battery 2**	**Battery 1**	**Battery 2**	**Battery 1**	**Battery 2**
Fall 1994	**33.3 V**	**33.3 V**				
Spring 1995	**33.1 V**	**33.1 V**				
Fall 1995	**32.7 V**	**32.7 V**	**33.1 V**	**33.1 V**		
Spring 1996	**32.5 V**	**32.5 V**	**32.9 V**	**32.7 V**		
Fall 1996	**32.7 V**	**32.5 V**	**32.7 V**	**32.7 V**		
Spring 1997	**32.5 V**	**32.5 V**	**32.7 V**	**32.7 V**		
Fall 1997	**32.5 V**	**32.5 V**	**32.5 V**	**32.5 V**	**32.9 V**	**32.9 V**

SUMMARY AND CONCLUSIONS

On-orbit performance of the batteries on GOES 8, 9, and 10 spacecraft indicates that the batteries are performing within specifications and results behave as a family. In addition, the battery capacities to date have shown improvement and indicate a leveling off at approximately 16 Ah under the C/48 discharge conditions. Ground test data indicate that the batteries will meet power requirements for the spacecraft mission life of 5-7 years.

ACKNOWLEDGMENTS

The authors wish to thank NASA for support of this work under contracts NAS 5-32590 and NAS 5-32600. They also wish to thank the following individuals for their help during preparation of this paper: Mr. Steve Hall (Naval Surface Warfare Center, Crane) for life cycle test data, Mr. Ron Hudak (SS/L) for activation and acceptance test data, Dr. Milt Phenneger (CSC), and Mrs. Cheryl Singhal for review of the paper.

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ACRONYMS

Ah	Ampere hour
AMF	Apogee Maneuver Firing
AOCS	Attitude and Orbit Control Subsystem
BOL	Beginning of Life
DOD	Depth of Discharge
DOY	Day of Year
EED	Electro-Explosive Device
EOD	End of Discharge
EOL	End of Life
GOES	Geostationary Operational Environmental Satellite
NASA	National Aeronautics and Space Administration
Ni-Cd	Nickel-Cadmium
NOAA	National Oceanic and Atmospheric Administration
NWS	National Weather Service
PCE	Power Control Electronics
PCU	Power Control Unit
SA	Solar Array
SS/L	Space Systems Loral