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NTS-2 Nickel-Hydrogen Battery Performance

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The nickel-hydrogen battery was first used for satellite energy storage aboard the U.S. Navy's Navigation Technology Satellite-2 (NTS-2). A flight demonstration of this new battery was the prime object of the joint Navy/INTELSAT program. The battery's successful performance proves its usefulness for future satellite applications. The nickel-hydrogen battery supplied spacecraft power for one 30-day eclipse season during which two eclipse periods occurred daily, lasting no longer than one hour. Its unique cell reversal feature was successfully demonstrated several times when the battery was inadvertently discharged completely because the solar array lost track of the sun. The nickel-hydrogen battery also uses a novel concept in which temperature is used for charge control. Strain gauges measure cell pressure, thereby providing information which permits a direct appraisal of the battery's state of charge. The use of cell pressure to control the rate of charge also can be considered a suitable method. This paper describes the results of the environmental test program along with the prelaunch and orbital performance through one eclipse season.

Introduction

THE nickel-hydrogen battery concept originated six years ago with an exploratory R&D program, followed by development effort, which ultimately led to the successful flight demonstration of the nickel-hydrogen battery. This battery was used as the prime energy storage system aboard the Navigation Technology Satellite-2 (NTS-2),, which was successfully launched from Vandenburg Air Force Base on June 23, 1977. Since this was the first nickel-hydrogen battery to be flight tested, its success is a milestone in the development of this new battery technology.

Battery Design

A detailed description of the battery design and spacecraft configuration has been previously published.[1,2] Only the salient features will be described herein. The satellite, which is shaped like a right octagonal prism with a center tube, is gravity-gradient stabilized and uses reaction wheels for damping and yaw control. The Ni-H_2 battery is mounted on two of the satellite's three rectangular surfaces used to continuously view deep space (Fig. 1).

Two eclipse seasons, of approximately 30 days each, occur each year. The first eclipse season began July 10, 1977. The satellite is in a twelve-hour orbit; hence, two eclipse periods occur daily. The battery will be cycled 60 times during each eclipse season, twice per day. The longest shadow period, which is in the middle of the season as shown in Fig. 2, lasts 0.94 h.

Energy Requirements

The normal satellite design load is 325 W. During the longest shadow period of 0.94 h, the energy requirement for the battery is 351 Wh. This includes the loss due to the boost regulator which has an efficiency of 87%. To meet this requirement, the battery was designed to accommodate fourteen 35-Ah cells connected in series. The average battery discharge voltage expected is 17.5 V (1.25 V/cell), making the depth of discharge approximately 60%. To implement the battery into the spacecraft, two 7-cell modules were made.

Charge Control

Following an eclipse discharge, the battery is fully recharged at a constant current of 3.6 A (C/10 rate). Continuous overcharging at this rate will heat the battery, increasing its temperature at a rapid rate. When the battery temperature reaches 15°C, the battery charging current is automatically switched from the C/10 rate to a C/60 rate. At this low trickle charge rate, the battery temperature drops to about 10°C prior to the next eclipse period. The control temperature of 15°C was selected, based upon the thermal

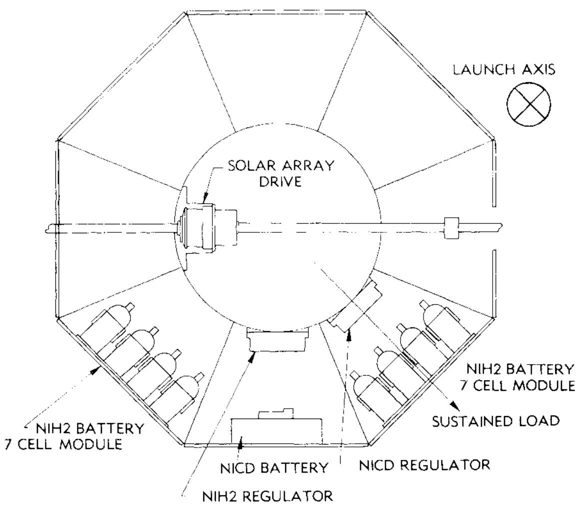

Fig. 1 Equipment location in spacecraft.

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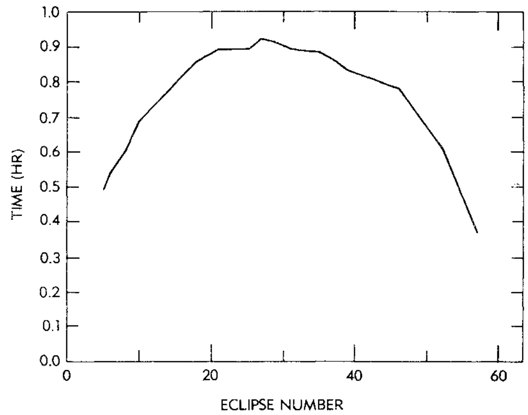

Fig. 2 Eclipse season one discharge times.

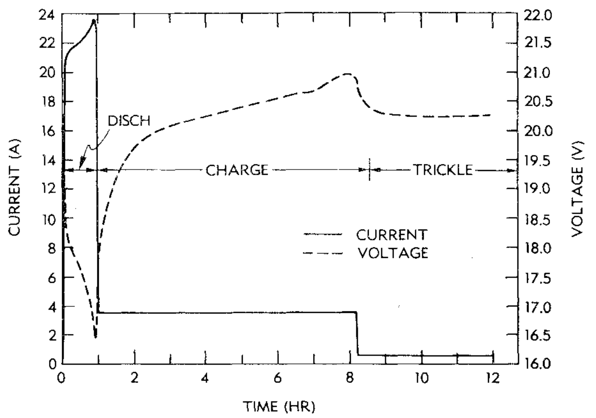

Fig. 3 Battery current and voltage.

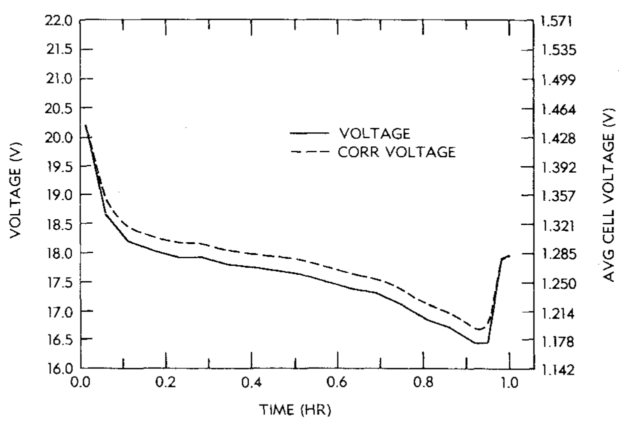

Fig. 4 Battery voltage.

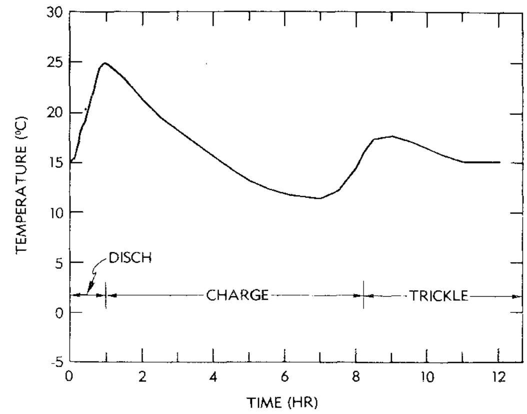

Fig. 5 Battery temperature.

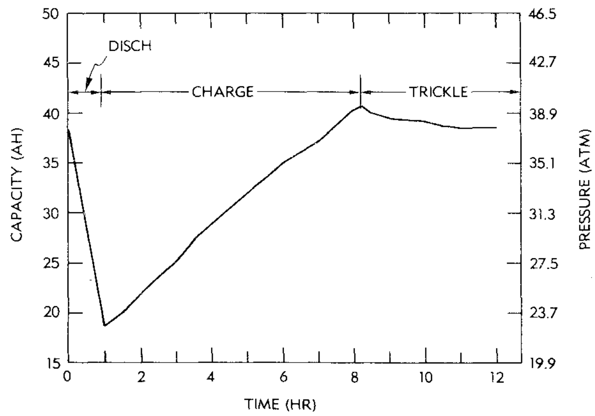

Fig. 6 Battery capacity and pressure.

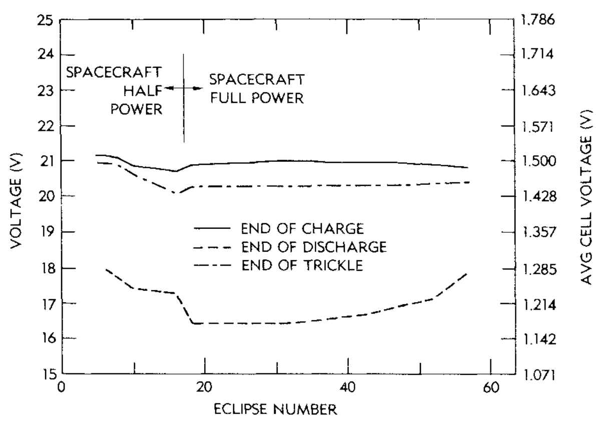

Fig. 7 Eclipse season one battery voltage.

model temperature predictions and thermal vacuum test data. To allow for temperature conditions higher than those predicted, alternate temperature set points are provided in the 18-24°C range.

Two additional operational provisions are incorporated in the system. An over-temperature limit of 35°C will switch the charge off, and an under-voltage condition of 14 V will shed noncritical loads. Both of these automatic controls can be disabled on command.

Temperature Control

Temperature control is essential to overall battery performance and life expectancy. A direct radiator battery design concept was employed. Design objectives were to maintain the temperature between 0 and 25°C during the longest eclipse day, and between 0 and 5°C between eclipse seasons.

Data Acquisition

Four of the cells have strain gage bridges mounted on the domes of their pressure shells. Pressure values deduced from these strain gage readings provide a direct measure of the cell capacity.

The following parameters are available through spacecraft telemetry: battery temperature and voltage, individual cell voltages, and cell internal pressure. The battery data are stored aboard the spacecraft every 102 s for retrieval by ground command.

Environmental Test Program

A prototype and a flight battery, each consisting of two 7-cell modules, were fabricated for this program.

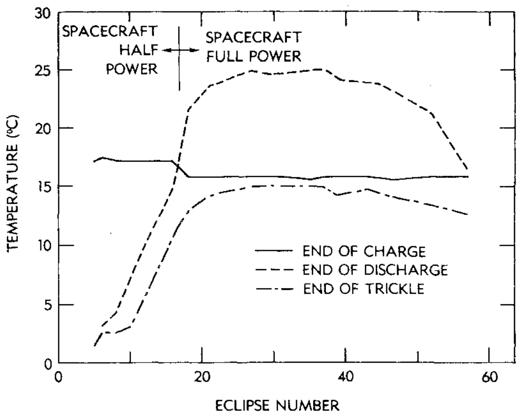

Fig. 8 Eclipse season one battery temperature.

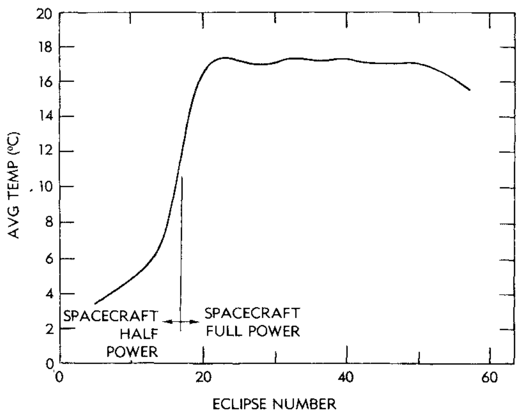

Fig. 9 Eclipse season one battery average charge temperature.

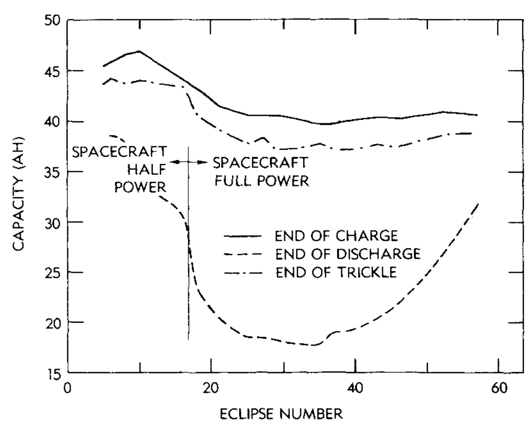

Fig. 10 Eclipse season one battery capacity.

Prototype Battery

The prototype battery was subjected to thermal vacuum, random vibration, and sustained load (acceleration) tests. The sustained load test was performed on one 7-cell module at a loading of 8 *g* for nine hours. The thermal vacuum test, conducted with a 3-bay spacecraft module, simulated three of the longest eclipses and a spacecraft hot and cold condition. The battery was operational during all the tests, and no malfunctions resulted from the thermal vacuum and sustained load tests.

Vibration testing of the prototype battery consisted of a low-level sinusoidal sweep in each of the three axes (0.5 *g*, 10-2000 Hz) and random vibration in each of the three axes at 18.1 *g* rms for 3 min per axis. A failure occurred at the seal weldment of one cell during random vibration; however, the weldment was modified, the failed cell was replaced, and the random vibration test was repeated with no further failures.

The prototype battery was integrated into the spacecraft, where it remained for about a year and performed flawlessly under the rigors of equipment integration and checkout. The battery was used during the electrical checkout and testing of the spacecraft.

Flight Battery

The flight battery was subjected to environmental testing at the component and spacecraft level. During the component vibration testing (13.1 *g* rms, 3 axes, 2 min/axis), the temperature measuring thermistors became unfastened from their mounts. The mounting technique was improved and both assemblies were again vibrated. No malfunctions were noted, and the battery was integrated into the spacecraft, as shown in Fig. 1.

The spacecraft was subjected to thermal vacuum, random vibration (8.2 *g* rms, 3 axes, 1 min/axis), and acoustic (144 dB, 40-10,000 Hz, 1 min) testing, respectively. The thermal vacuum test simulated continuous overcharge at the trickle rate, four longest eclipse cycles, a hot spacecraft conditon, and the transfer orbit. No battery anomalies occurred.

Prelaunch and Orbital Performance

Prelaunch and Transfer Orbit

The $Ni-H_2$ battery was exercised extensively during prelaunch operations at Vandenberg Air Force Base for electrical checkout of the spacecraft. State-of-charge indications from the strain gauges helped significantly during the spacecraft electrical checks. One hour prior to launch, the spacecraft began operating entirely from the battery and continued until the spacecraft acquired the sun approximately 30 min after launch. The satellite was launched into an elliptical transfer orbit and encountered three 30-min eclipses. For 33 h the satellite, which was spinning at 85 rpm, exerted a steady force of 6 *g* on the battery. The satellite was then boosted into a circular orbit and despun to 40 rpm, thereby reducing the force on the battery to approximately 1.4 *g* over the next 12 days. Finally, the satellite was despun and stabilized with its solar array deployed. Neither the dynamic forces during liftoff nor the steady load accompany spinning had any detrimental effects on the battery.

During the final spacecraft stabilization period, the battery was inadvertently discharged completely three times, causing cell reversals. Subsequent recharge revealed no detrimental effects, dramatically illustrating the battery's tolerance to overcharge. Since spacecraft stabilization, the battery has been operating in a 0-*g* environment with no adverse effects on performance.

Eclipse Operation

Battery performance during the first eclipse season is illustrated by telemetry data taken during the longest eclipse (eclipse No. 27). Figure 3 shows the battery voltage and current during the 0.94-h discharge and 11-h recharge periods. Figure 4 is an expanded version of the battery voltage during discharge. (Wire harness losses are not included in corrected voltage.) The end-of-discharge voltage for the battery is 16.4 V or 1.17 V per cell; the corresponding discharge current is 23.6 A, and the power output 387 W, which is slightly greater than the design load requirement (including regulator losses) of 374 W. Since the boost regulator maintains the bus voltage at 26.6 V, the actual voltage performance is as expected.

The battery temperature increases to 25°C at the end of discharge due to heat dissipation (Fig. 5). This is the maximum battery operating temperature observed and is in accordance with design objectives. The battery is recharged at a constant current of 3.6 A until the temperature reaches 15°C, thereby reducing the charging current to a trickle rate of 0.6 A, as shown in Figs. 3 and 5.

Capacity and corresponding cell pressure data over the 12-h charge-discharge cycle are shown in Fig. 6. Data from the

Table 1 14-Cell battery design

Parameter	Present battery (630 Wh)	Improved battery (630 Wh)	Improved[a] battery (900 Wh)
Top support	0.2	0.2	0.2
Sleeves	3.1	1.5	1.9
Cells	14.3	14.3	16.8
Connectors, wires, etc.	1.0	1.0	1.2
Total	18.6	17.0	20.1
Energy density at 100% DOD, Wh/kg	33.9	37.1	44.8
Usable energy density at 57% DOD, Wh/kg	19.3	21.1	25.5

[a] 50 Ah cells.

strain gauges indicate that 19.5 Ah were removed during the eclipse period. Calculations by integration of the current with time indicated that 20.0 Ah were removed, which verified the strain gauge as an accurate indicator of capacity. The battery state of charge at the beginning and the end of the cycle are almost identical, indicating proper recharging.

Figures 7-10 depict the battery voltage, temperature, average charging temperature, and capacity during the first eclipse season. Prior to eclipse no. 17 the satellite was operating at a reduced power load, approximately one-half the normal design load. After the 17th eclipse the satellite was operated at the normal design load for the remainder of the eclipse season. Figure 9 shows that the average charging temperature at the reduced power load was 3-8°C, and the average charging temperature at the normal spacecraft load was 17°C. Charging the nickel-hydrogen battery at the lower temperature results in a higher state of charge (ampere-hour capacity) as shown in Fig. 10. A somewhat flat trace of the end-of-charge capacity after eclipse no. 20 is also shown. If the switch to trickle charge had been controlled by this parameter, sufficient recharging of the battery would have resulted.

The temperature at which the high-rate (C/10) charging of the battery is terminated and the current is switched to the trickle rate is normally 15°C (T-set) (Fig. 8). T-set is increased by approximately 1°C if the battery temperature is below T-set at the end of discharge. This precludes a premature switch to trickle charge if the battery emerges from the shadow slightly below T-set.

Future Battery Design

The NTS-2 nickel-hydrogen battery has a respectable, usable energy density of 19.3 Wh/kg. However, this usable energy density may be improved by reducing the weight of certain components in the present design, and by increasing the capacity of the cells to 50 Ah. Table 1 is a tabulation of the component weight breakdown for the present and improved designs.

With the 50-Ah cell design, an improved NTS-2 battery (14 cells) operated at 57% depth of discharge (DOD) would have a usable energy density of 25.5 Wh/kg, which is about twice the usable energy density achieved with present Ni-Cd batteries in Intelsat IV and IV-A.

Conclusion

A nickel-hydrogen battery was built, tested, and orbited aboard the NTS-2 satellite. Ground testing demonstrated the battery's ability to endure the conditions encountered during spacecraft integration and environmental testing. No detrimental effects were observed from the dynamic forces during launch, the steady load accompanying spinning, or the 0-*g* environment. To date, one eclipse season has been completed with the battery supplying spacecraft power during the shadow periods. The automatic charge control system adequately recharges the battery while minimizing overcharge and internal heating. The strain gauges provide a reliable, operational, and useful state-of-charge indicator to aid in performance evaluation. The use of cell pressure or a combination of cell pressure and temperature should be considered as a means of charge control. The battery performance has currently met all of the design goals and objectives of this program.

The successful flight demonstration of the Ni-H_2 battery is a significant step toward the application of this technology in future communications satellites.

Acknowledgments

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