ON-ORBIT NiH₂ BATTERY PERFORMANCE AND PROBLEM SOLVING ON THE APEX SPACECRAFT

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Abstract

The Advanced Photovoltaic and Electronics eXperiments (APEX) spacecraft, launched in August 1994, contained an Electrical Power Subsystem which included two 10 Common Pressure Vessel nickel hydrogen batteries manufactured by Eagle-Picher Industries, Inc. The spacecraft bus has fully supported over 18 months of on-orbit payload operations, well exceeding the mission requirement. Over the duration of the mission, three fundamentally different battery charging algorithms were used, as necessitated by two hardware failures. After each failure, engineers evaluated the available options for extending the mission and implemented the most desirable option. All three charging methods -- pressure-based, V/T-based, and recharge-ratio-based (with significant overcharging) -- have demonstrated acceptable battery performance for periods of at least several months each. The on-orbit phase of the APEX program demonstrated the flexibility of the APEX Electrical Power Subsystem and the robustness of the APEX batteries. This paper gives an overview of the APEX Common Pressure Vessel and battery designs, summarizes the different charging methods that have been used, and presents on-orbit performance data for each method. Data for an incident where the batteries got very cold and may have begun to freeze are also presented.

Electrical Power Subsystem (EPS) Design

The APEX EPS autonomously collects solar power and provides power to the spacecraft bus and payloads at all times. The EPS consists of three solar array panels, two nickel-hydrogen battery assemblies, and a suite of electronics units for power conversion and distribution and battery charge monitoring and control (see Figure 1). All EPS functions are controlled by a dedicated MicroController Unit (MCU) and software. Dual switching regulators convert solar array power to charge the batteries and power the spacecraft loads during sunlight. Both the MCU and switching regulators are fully redundant. The EPS software allows most important operating parameters to be changed via ground command, providing on-orbit operational flexibility.

The three solar panels are deployed into a single plane and contain four strings of 99 silicon cells each. Two 6 amp-hour, 10 Common Pressure Vessel (CPV) (20 cell) battery assemblies provide power to the spacecraft bus and payloads during eclipse, with a worst-case peak Depth of Discharge (DoD) of about 35%. During sunlight, Peak Power Tracking (PPT) is autonomously performed on the arrays to support the bus and payloads and maximize battery charge currents. When the batteries approach full charge, the MCU limits the charge currents to reduced levels and then to trickle charge.
Common Pressure Vessel Design

The APEX battery assemblies are made up of RNHC-6-1 nickel hydrogen CPVs (Figure 2), which are part of an Eagle-Picher line of spaceflight qualified 2.5 inch diameter CPVs of various capacities and physical configurations. The pressure vessels are cylindrical, with one domed end and one flat end. Each CPV contains two 6 amp-hour capacity (nameplate) electrode stacks electrically connected in series within a single pressure vessel. Actual capacity is approximately 7.5 amp-hours at 100°C, and working voltage is 2.5 volts. Table 1 lists the important RNHC-6-1 design parameters.

Battery Assembly Design

Due to spacecraft volume constraints and thermal design requirements, the APEX battery assemblies have the CPVs arranged in a flat-pack configuration, with two rows of five CPVs (see Figure 3). Each CPV is clamped within a sleeve that mounts to the battery baseplate. CPV-to-sleeve interfaces are electrically isolating and thermally conductive. CPV terminals face the battery centerline to minimize wiring losses. All ten CPVs in a battery are wired in series to provide a nominal 28 volt output.
Two CPVs in each battery are instrumented with strain gauges to measure internal CPV pressure. These same CPVs are also instrumented with thermistors, and an additional thermistor is mounted to each battery baseplate. Battery thermal control is provided by a radiator with visibility to space and heater elements (20 watts total dissipation per battery) bonded to the battery baseplates. Each battery is thermally isolated from the rest of the spacecraft through the use of insulating standoffs and thermal blankets. Battery assemblies (minus heaters, thermistors, and radiators) were supplied to OSC fully tested by Eagle-Picher.

![Figure 3. APEX Battery Assembly with Spare CPV.](image)

**Pressure-Based Battery Charging Phase**

**Background**

The APEX EPS was designed to perform PPT on the solar array I-V curve, while limiting the charge current to each battery to some time-varying maximum which is based on that battery's Depth of Discharge (DoD). When the batteries are significantly discharged, their charge current limits are well above what can be provided by the solar arrays, so true PPT is performed. When the batteries near full charge, their charge current limits drop below what can be provided by the arrays. When this happens, the EPS MCU reduces the battery charge currents by purposely operating the solar arrays at a voltage that is higher than their peak power voltage, reducing the amount of power coming into the system.

Each battery's DoD is calculated as a linear function of the average of its two pressure sensor readings. The terms used in these calculations were derived empirically from characterization testing of the flight batteries performed early in the APEX test program. Once DoD is calculated for each battery, that battery's charge current limit is calculated as a multi-step function of DoD. The limit is 1.5C (9 amps) for DoD greater than 8%, C/2 (3 amps) for DoD between 0% and 8%, and C/23 (0.26 amps -- trickle charge) for DoD less than 0%. This high trickle charge rate was necessary due to the coarse resolution of the battery charge current sensor readings. Since DoD is defined with respect to an empirical 0% DoD reference point, it is possible to have negative DoDs in normal operation (i.e. overcharged batteries). Battery voltages are telemetered to the ground for informational purposes but are not used for on-board charge control under normal conditions.

The MCU controls the heaters on each battery based on the average of the two CPV temperatures for that battery. The heater is turned on when the average temperature drops below -5°C, and turned off when the average temperature rises above 0°C. The design operating temperature range for the batteries is -5°C to 10°C at the CPVs.

**Performance**

Figure 4 shows the battery pressures versus time for a three-hour period about 45 days after launch. Eclipse period at that time was 34 minutes. The CPV pressures can be seen to decrease linearly during discharge, then increase rapidly during charge until the charge rates drop to C/2. Once the batteries reached full-charge and were trickle-charged, the pressures were flat or nearly flat. Figure 5 shows the pressures converted into DoDs. It is apparent that battery 2 overcharged significantly more than battery 1 (reaching -9% DoD, versus -2% for battery 1). This was probably due to slight offsets in the sensors used to control charge currents, resulting in a higher actual trickle-charge rate for battery 2.

![Figure 4. Pressure-Based Charging, Battery Pressures versus Time.](image)
Figures 6 and 7 show the battery currents and voltages for the same three-hour period. Positive current represents discharge in all battery current graphs. The batteries discharged at 1.5 to 2.5 amps each during eclipse, varying as a function of the load on the system. In PPT mode, for the first few minutes after the spacecraft entered sunlight, enough power was available at the arrays to produce charge currents of 5 to 6 amps per battery. The charge currents decreased gradually while in PPT mode due to the decreasing output power from the arrays as their temperatures rose. Once the battery DoDs dropped below 8%, the charge currents were decreased to a steady 3 amps. Trickle-charging began when the batteries reached 0% DoD and continued until entering the next eclipse.

Figures 8 and 9 show the battery temperatures for this same three-hour period. At the beginning of the period, both battery heaters were on, resulting in rising temperatures and the baseplate temperatures running very close to the CPV temperatures. When the heaters turned off, the baseplate temperatures dropped to about 6°C lower than the CPV temperatures due to the thermal resistances between the CPVs (heat source) and the radiators/baseplates (heat elimination elements).

Whenever the spacecraft was in an eclipse season, both battery heaters cycled on and off and the CPV temperatures cycled between -5°C and 0°C. The fairly short heater cycling period was due to the relatively small thermal masses of the batteries and the fact that they were thermally isolated from the rest of the spacecraft. Since the thermal behavior of the batteries was dominated by the heaters and radiators rather than dissipations in the CPVs themselves, there was no apparent correlation between the cycling of one battery heater and the cycling of the other battery heater, or between the cycling of a battery heater and the charge cycling of that battery. In multiple-orbit full sunlight periods, with continuous trickle charging of both batteries, the CPV temperatures stabilized at between 0 and 5°C with the heaters off.
Overall, battery performance during this phase of the mission was very solid. No unusual behavior of any battery parameters was observed and the EPS supported all payload operations.

**Cold Battery Incident**

In early November 1994, three months after launch, a hardware failure occurred which corrupted all CPV pressure and temperature sensor readings from both batteries. A fault tree analysis showed that the most likely cause was the short-circuit of a strain gauge bridge ladder resistor to the bridge circuit mounting surface, and of that surface to the CPV itself. This resulted in the potential of the CPV pressure vessel being injected into the middle of the strain gauge bridge circuitry, causing the strain gauge output to reach a voltage much higher than would normally be expected. This caused the output of the signal amplifier for that strain gauge to overdrive one input of a CMOS multiplexer which routed CPV temperature and pressure signals from both batteries, corrupting all of those readings.

The primary effect of the failure was that the battery pressures read very high, causing the EPS to believe that both batteries were always fully-charged no matter what their true charge states were. The MCU therefore limited both battery charge currents to trickle charge at all times. Also, the MCU disabled the battery heaters because the battery temperatures readings were invalid. Since the spacecraft was in the middle of an eclipse season when this occurred, the batteries began alternating between discharging during eclipse and trickle charging during sunlight, with no higher-rate charging. The energy balance was slightly negative, so that the batteries gradually discharged from orbit to orbit. More importantly, because the heaters were disabled and there was no overcharging to generate heat, the battery temperatures dropped very rapidly.

Figures 10, 11, and 12 show the battery temperatures, currents, and voltages for a period beginning several orbits after the failure. At the beginning of these charts the batteries were already behaving adversely due to the very cold temperatures (baseplates near -25°C). Although the batteries are believed to have been moderately charged at the time, their voltages hovered around 16 to 17 volts (0.80 to 0.85 volts per cell) over the first few minutes of the period, indicating that the electrolyte may have been freezing.

One thing to note in Figure 12 is that for several orbits the voltages rose to 32 volts and stayed there for the remainder of the sunlight period. This 32 volt plateau was actually caused by logic in the EPS software. To ensure that the spacecraft bus voltage would not exceed the 32 volt specification, the software was designed to automatically reduce the charge current to a battery with a voltage greater than 32 volts. These reductions in charge current can be seen in Figure 11, at around 400 minutes for example. The fact that the battery
voltages rose to 32 volts (1.6 volts per cell) so rapidly with very low charge rates shows that the cold temperatures were causing very high battery impedances.

In response to the failure, two options were considered for generating heat to warm the batteries: using the battery heaters and using moderate overcharging. Although it is generally considered harmful, overcharging was chosen over using the heaters because it would cause a more gradual warming of the batteries. Turning on the battery heaters at this point would have created a very rapid temperature increase due to the oversizing of the heaters, with the associated risk of thermally shocking the batteries. A command was sent to the spacecraft to set the battery charge rates to a constant C/10 rate, giving a recharge ratio of roughly 250% (representing about 0.5 amp-hours of overcharge per battery per orbit). All payloads, which had already been autonomously switched off, were left off to maximize battery overcharging. As the graphs show, the battery temperatures gradually rose to acceptable operating levels and eventually their voltages fell back within a nominal range.

The batteries were operated this way for about six weeks while a new charging algorithm was developed and tested. Other than some adjustments made to the charge rate as the eclipse period varied, the performance of the batteries in this mode was stable.

V/T-Based Battery Charging Phase

Background

After the battery sensor failure occurred and the spacecraft was stabilized as described above, Orbital evaluated options for continuing the mission. After the failure, valid telemetry from each battery was limited to only charge/discharge currents, voltages, and baseplate temperatures. Thus, pressure-based charge control was no longer possible. Options considered for implementation at this point were a modified V/T algorithm and a recharge-ratio-based algorithm. The latter was discounted due to inaccuracies in the charge current sensor readings, and work began to define a V/T-type algorithm based on previous on-orbit battery data. The algorithm consisted of initially charging the batteries at a C/2 rate upon entering sunlight, then switching each battery to a C/17 trickle-charge when its voltage exceeded the following threshold:

\[ 29.5 \ (V) - T * 0.125 \ (V/°C) \]

T represents the battery's baseplate temperature.

Since a completely new charging algorithm with minimal testing (as compared to the original pressure-based algorithm) was being put into place, the higher trickle charge rate was chosen to ensure that the batteries would be fully charged every orbit even if minor adjustments were needed to the algorithm. The disadvantage of this trickle charge rate was greater overcharging of the batteries, and therefore a potential for reduced battery lifetime, but this risk was considered to be lower than the risk of undercharging the batteries if a lower trickle charge rate were used. Due to the corrupted pressure readings, there was no longer any direct indication of battery DoDs.

A new algorithm was also developed to control the battery heaters based on the battery baseplate temperatures, rather than the corrupted CPV temperatures. The new trip points were: heaters on at -17°C and off at -12°C. These trip points were selected to prevent the batteries from freezing while minimizing heater operation and its destabilizing effect on battery charging. Based on thermal characterization of the batteries using previous on-orbit data, these trip points would keep the CPV temperatures at -11°C or higher.

Several patches to the flight computer software were created to implement these algorithms and tested using the APEX development system at Orbital. The new algorithm was put into effect on the spacecraft and payload operations resumed in late January 1995.

Performance

Figure 13 shows the battery currents versus time under the V/T-based charging algorithm for a three-hour period about 8 months after launch. The eclipse period for these orbits was 36 minutes. The profiles are simpler than before, since there was no longer any operation in PPT mode. The MCU limited charge rates to 3 amps upon entry into sunlight, and held the battery currents at that level until each battery reached its voltage threshold and was switched to trickle charge.
Battery voltages for V/T-based charging are shown in Figure 14. The voltages dropped smoothly during discharge, increased smoothly during C/2 charging, and then dropped slightly more than a volt when the charge rates are dropped to trickle-charge. The two battery voltages tracked each other well, although this tended to vary from orbit to orbit depending on their relative temperatures. The voltages during charge were significantly higher than those shown above for the pressure-based charging algorithm due to the lower battery temperatures in this new mode.

The V/T-based charging algorithm performed very well. Although spacecraft telemetry had to be monitored more closely than early in the mission, the EPS once again operated autonomously and allowed unrestricted payload operations. The only action required by the ground was to reduce the trickle charge rate from C/17 to C/23 upon entering a full-sunlight period to keep the batteries from overheating, and to increase it back to C/17 when returning to an eclipse season. No unusual behavior of the batteries was observed while using the V/T-based charging algorithm.

Manual Battery Charging Phase

Background

The EPS continued to operate effectively using the V/T-based charging algorithm until 10 months after launch, when the failure of a unit external to the EPS made the battery baseplate temperatures unavailable. This left charge/discharge currents and voltages as the only remaining valid battery telemetry. The EPS was again put into a C/10 constant current charge state with all payloads powered off while options were considered.

Although most payload mission objectives had been met at this point, Orbital investigated possibilities for extending the mission. Payload scientists emphasized that collecting data for even a few more weeks would be valuable, and that consequences to the mission beyond a month or two into the future were not as important.

Based on the desire to support payload operations for a few more weeks regardless of a possible reduction in battery lifetime, and the minimal valid battery telemetry available, an open-loop recharge-ratio-based algorithm was developed. This algorithm consisted of charging the batteries at a C/3 rate for some period of time after entering sunlight each orbit, then switching to a C/17 trickle-charge. The time to charge at the higher rate was pre-calculated for each orbit based on knowledge of the load configuration for that orbit. Total battery discharge energy was calculated using the expected current draw for each load and the eclipse period. High-rate charge time was then calculated to give a 100% recharge ratio plus 0.5 amp-hours of overcharge per battery (including energy put into the batteries during trickle-charge). Commands to switch back and forth between C/3 and C/17 charge rates were uploaded by ground
command to the spacecraft computer and then automatically forwarded to the EPS MCU at the pre-programmed times.

Prior on-orbit battery data was examined to derive the value of the overcharge constant above. Since at this point no battery temperature information was available, the battery heaters could not be used and were locked off. The 0.5 amp-hours of overcharge energy was very high, resulting in recharge ratios under some conditions of 130% or higher, ensuring that the batteries would be fully-charged every orbit and that they would not freeze. The analysis of previous on-orbit battery data indicated that overheating the batteries was unlikely.

Performance

Figures 16 and 17 show the battery currents and voltages for a representative 200 minute period about 1 year after launch. Eclipse period was 36 minutes. Both the current and voltage profiles show that the behavior of the two batteries was not as well matched under the manual charging algorithm as it had been under the previous charging schemes. This was probably due to a larger temperature differential between the two batteries. Some unusual discharge current imbalances were observed between the two batteries, the characteristics of which varied greatly from orbit to orbit.

![Figure 16. Manual Charging, Battery Currents versus Time.](image1)

![Figure 17. Manual Charging, Battery Voltages versus Time.](image2)

Because of time constraints and the desire to quickly resume payload operations, the manual charging algorithm was derived over the span of less than a week and was not fully tested on the ground prior to implementation on-orbit. However, it performed very well on-orbit and the EPS was once again able to fully support payload operations. The only major drawback of the manual charging mode was the added resources required to pre-plan charge mode transitions and generate spacecraft upload files with the time-tagged commands. Even though the algorithm was created with an emphasis on safe short-term operations, the EPS continued to operate in this mode for 11 more months.

Conclusions

Due to the three different schemes used to charge the batteries over the mission, and the varying eclipse duration and bus load, no useful long-term trending could be done on the APEX battery performance. However, a number of lessons were learned and have been incorporated into the software and hardware design of other OSC spacecraft, including ORBCOMM, MicroLab-1, and SeaStar.

The principal lesson is that well-designed nickel hydrogen batteries are extremely robust, improving mission reliability and providing great flexibility to deal with and work around unforeseen situations on-orbit. The APEX batteries continued to operate well enough to support mission operations for 19 months after the "cold battery incident" where they began to freeze. They also tolerated about a year total time of being significantly overcharged on every charge cycle.
The other major lesson learned involves building operational flexibility into a spacecraft EPS design. There are two schools of thought on this. One says that a system should be made as flexible as possible to give the most freedom during operations. The other says that a system should be made as foolproof as possible, which inherently limits flexibility. Despite the fact that the APEX EPS incorporated many ground-commandable operating parameters to be very flexible, the engineering team's responses to on-orbit hardware failures were limited to some degree by constraints built into the software. Valid parameter value ranges that make sense when considered during spacecraft design and testing may turn out to be too restrictive when dealing with an unforeseen failure on-orbit. Defining these acceptable value ranges in the ground command system, rather than hard-coding them on the spacecraft, allows them to be altered after launch if necessary.

Finally, clamping circuitry was added to multiplexer inputs for future spacecraft to ensure that a single failed sensor could not corrupt readings from other valid sensors.

References
