The MSX Spacecraft Power Subsystem

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The Midcourse Space Experiment (MSX) spacecraft is the fourth in a series of APL spacecraft sponsored by the Ballistic Missile Defense Organization. Though similar to its predecessors in mission objectives, the MSX differs notably in the performance and environmental requirements that drive the design of the electrical power subsystem (EPS). Diverse modes of operation and autonomous fault tolerance over a 5-year mission having a near-polar, low Earth orbit provide challenging engineering constraints to the EPS design. To meet these requirements, the MSX EPS consists of a 1200-W silicon solar array, a 50-ampere-hour nickel–hydrogen battery, and a microprocessor-based power management system. These components have been procured or fabricated by APL, integrated with the MSX spacecraft, and tested to assure successful mission performance. The MSX EPS design demonstrates a viable concept in power system engineering to meet the requirements of highly diverse, high-power, low Earth orbit applications.

INTRODUCTION

The MSX payload is similar to that of its three predecessors: Delta 180, Delta 181, and Delta Star (Delta 183), also sponsored by the Ballistic Missile Defense Organization (formerly the Strategic Defense Initiative Office).[1] These spacecraft all have a suite of scientific instruments or experiments that can be physically pointed at calibration stars or targets to obtain signatures against the various space backgrounds of a low Earth orbit.

Unique in the MSX payload is the large solid hydrogen dewar used to cool the sensing elements of the Space Infrared Imaging Telescope III (SPIRIT III). Thermal control of this dewar requires the instrument section to be kept very cold and to be thermally decoupled from the electronics section. The resulting thermal environment of the spacecraft requires the addition of several heaters on components that must be protected from the cold. The heaters create a major additional power load, with a possible peak power consumption of over 2100 W.

The predecessors of MSX relied on the final stage of the McDonnell Douglas Delta launch vehicle and its guidance computer for attitude control. While the missions of MSX's predecessors were all less than 1 year long, the MSX mission will last at least 4 years, with 5 years as a design goal. The ability of MSX to provide precise, fast-reacting, variable attitude control for instrument pointing over its long life comes from the use of reaction wheels, which the MSX predecessors did not have. The four wheels create another major additional power load, with each wheel requiring a peak power of up to 1000 W.

The Delta 180 and 181 missions were so short that their electrical power subsystems consisted of several nonrechargeable batteries. The Delta Star mission, which was originally to last 90 days but was eventually extended to 1 year, used one of the Air Force's available Global Positioning System (GPS) satellite power systems. The GPS electrical power subsystem (EPS), integrated by Rockwell International (RI), provided a regulated 28-V bus and consisted of a single-axis Sun-tracking solar array, three nickel–cadmium (NiCd) batteries, a battery charge regulation subsystem, and a DC/DC converter. Because of its applicability, availability, and proven flight heritage, a modified GPS Sun-tracking solar array was procured from RI for use on the MSX. The GPS DC/DC converter subsystem, however, could not support the MSX's highly variable loads, which required a low-impedance energy source on the bus. A direct energy transfer topology, which connects the solar array and battery directly to the loads instead of through a converter, efficiently supports the MSX loads.

The MSX spacecraft design, therefore, while similar to the previous Delta-series spacecraft in variable attitude pointing and scientific instrument payloads, represents a significant design departure. Its attitude control subsystem (ACS) and EPS were completely integrated by APL. Its projected mission life is longer, and it can handle much higher variable power loads.

MISSION AND ENVIRONMENT

Modes of Operation

The subsystem loads require both regulated and unregulated 28-V power as the spacecraft operates in each of three different modes:

1. *Parked Mode*. The purpose of this mode is to maintain the spacecraft in a condition of energy balance and thermal equilibrium. This mode generally has steady unswitched loads, automatic heater loads, regular housekeeping and maintenance loads (which include light ACS loads), and occasional tape recorder playback and downlink transmitter loads.
2. *Track Mode*. While in track mode, the spacecraft performs aided target acquisition; it then tracks and points at the target and collects signature data. Each use of this mode lasts about 35 min; this mode is used approximately 15 times, at widely spaced intervals throughout the mission. Virtually all the spacecraft loads are powered for this mode except the operational heaters, which are turned off before the event to conserve power.
3. *Background Mode*. When the spacecraft is in background mode the instruments observe target backgrounds and collect data while the spacecraft is maintained in a condition of energy balance and thermal equilibrium. This mode is like parked mode with a 10- to 20-min observation added once per orbit. The observation uses the experiment loads, experiment downlink loads, ACS loads, and data recording loads.

The spacecraft may undergo more than 20,000 orbits in the background mode. Parked mode is typically used for several orbits before and after a track mode. Parked mode also can be used to safely rest the spacecraft, possibly for extended periods.

The load profile of the parked mode approximates a steady value of 600 W. This value is the average spacecraft power required to achieve thermal equilibrium, and it is regulated by thermostatically controlled heaters. Since background mode must also achieve thermal equilibrium, the average spacecraft power dissipation is also 600 W, but the load profile can vary as approximated in Fig. 1. An approximate load profile of the track mode is shown in Fig. 2. In both the background and the track mode profiles, all instruments are powered to warm up the sensors, and the spacecraft is slewed quickly (100° in 2 min) into position. In the track mode, the initial observation is of a calibration star, after which the spacecraft is again slewed quickly to the expected target coordinates. After the track mode event, the spacecraft is slewed back to the calibration star to perform the postcalibration. When all observations and calibrations are complete, the spacecraft is slowly slewed back to the parked attitude, where it cools down and recharges its battery. An animated simulation of the MSX performing an attitude maneuver

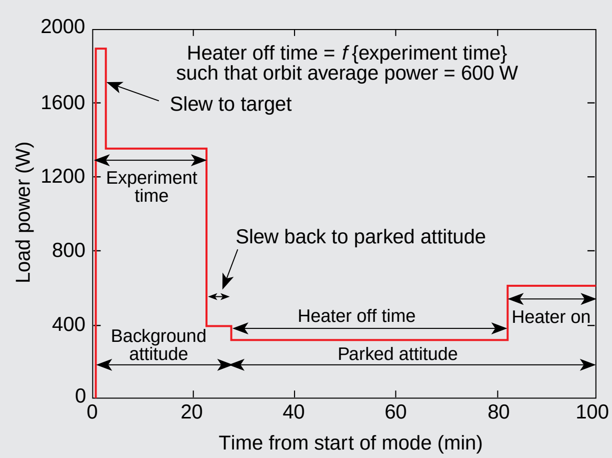

Figure 1. Load profile for MSX in a generic background mode. The orbital power requirement is driven mostly by the data collection event (DCE), which begins and ends with an attitude excursion. Heat generated during the DCE should decrease the need for heaters, keeping the overall average to about 600 W.

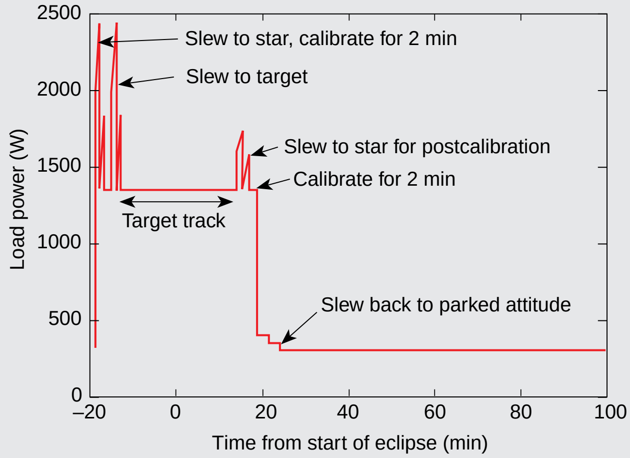

Figure 2. Load profile for MSX in a generic track mode. This extended data collection event includes pre- and postevent calibrations and can cause the battery to infrequently discharge to as much as 70% depth of discharge (DOD). Subsequent orbits are reserved for battery and thermal recovery.

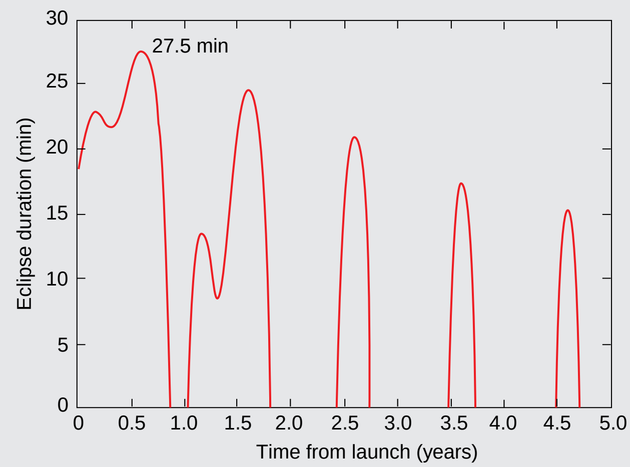

Figure 3. MSX eclipse time. The duration of the spacecraft's time in Earth shadow varies as the orbit precesses, causing longer periods of 100% sunlight as the mission proceeds. Orbital period is 103 min.

can be found on the World Wide Web at the location http://sd-www.jhuapl.edu/MSX/heyler.mov.

Mission Orbit and Attitude

The EPS must meet its operational requirements in the environment specified by the MSX orbit and attitude. The altitude and inclination were chosen to achieve a particular solar lighting geometry for target launches and also to create a repeating ground trace that is minimally dependent on the spacecraft launch date. The orbit that meets these criteria, however, is such that small deviations in the altitude (898 km nominal) and inclination (99.23° nominal) can cause large changes in the rate of precession of the ascending node. These changes, in turn, affect the eclipse profile of the orbit throughout the mission. The mission eclipse profile (calculated on the basis of the nominal injection parameters) is illustrated in Fig. 3. The maximum eclipse time is 27.5 min. The accumulated number of eclipse orbits over the mission life may exceed 20,000; this number defines the battery cycle life.

The MSX instrument suite and modes of operation impose constraints upon the spacecraft attitude. The SPIRIT III instrument must maintain attitude "keepout zones" to prevent damage to the infrared sensor or accelerated depletion of the hydrogen coolant from looking at hot objects (the Sun, Earth, or Moon). In addition, during tracking and background observations, the spacecraft must maintain precise attitude control in pointing at targets, which restricts activities such as tracking the Sun with the solar array.

The timing or phasing of background mode observations within an orbit relative to that orbit's eclipse affects EPS performance. Performing an observation during eclipse greatly diminishes stored battery energy, whereas performing the observation during sunlit periods hampers the battery recharge recovery from the discharge of the previous eclipse.

Radiation

The worst-case radiation environment, consisting of trapped charged particles and solar flares, could potentially cause a silicon solar array to degrade 21% by the spacecraft's end of life at 5 years. However, the launch date is not during the Sun's active cycle, so MSX may not experience any significant solar flare degradation until the third year of the mission.

Temperature

The MSX orbit induces thermal cycling of the spacecraft components. The solar array and other items that extend from the spacecraft body and have comparatively low heat capacity are subject to temperature extremes larger than those experienced by the rest of the spacecraft. When the spacecraft exits the Earth's shadow, the solar panel temperatures may be as low as –85°C. Upon reentry into sunlight they warm rapidly, peaking in the worst case at approximately 70°C (Fig. 4).

The MSX modes of operation induce thermal concerns from within the spacecraft. Periods of high activity during background and track modes result in elevated levels of heat generation, and the heat must be removed from the spacecraft. The internally produced heat affects component performance. The radiative surfaces required to remove spacecraft heat create the need for supplemental heaters to prevent the spacecraft from cooling too much when heat-generating loads are not active. These thermostatically controlled heaters contribute to the MSX's varying load profile.

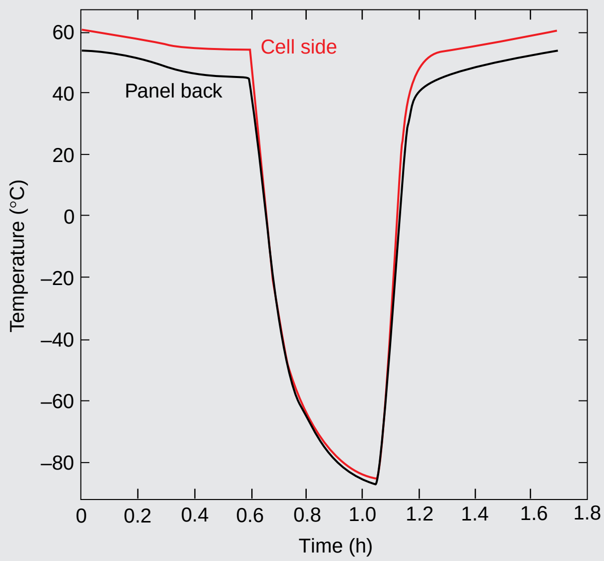

Figure 4. Typical solar cell temperature range. Temperatures will average about 60°C when the cells are illuminated; during eclipse, temperatures will plunge to –85°C.

OVERVIEW OF THE MSX ELECTRICAL POWER SUBSYSTEM

The MSX EPS uses an unregulated direct energy transfer topology. The EPS consists of the following four components, which are shown in the block diagram of Fig. 5 and discussed in more detail in subsequent sections of this article:

1. *Solar Array*. The solar array converts solar energy to electrical energy and delivers it to the loads and battery. The solar array consists of two wings, each comprising four solar panels supported by a boom and attached to a solar array drive (SAD). The SAD rotates the wings and electrically connects their solar cell circuitry, via a junction box, through slip rings to the main bus. The redundant control electronics unit (CEU) controls the drives so they can track the Sun. The solar panels are populated with silicon solar cells electrically connected into strings.
2. *Battery*. The battery accepts and stores electrical energy from the solar array. It provides a low-impedance source for powering transient loads, loads that exceed the capability of the solar array, and loads during eclipse. The battery consists of a series stack of 22 rechargeable nickel–hydrogen (NiH_2) cells, cell bypass circuitry, and monitoring electronics.
3. *Power Management System*. The power management system (PMS) controls the charging of the battery by shunting away excess solar energy. It consists of two digital shunt assemblies, which provide nondissipative shunting of the solar cell circuits; redundant analog shunt assemblies, which provide dissipative shunting at the main bus; redundant power management modules (PMMs), which monitor the battery and provide the main EPS command and data handling (C&DH) subsystem interfaces; and the dual shunt control electronics (DSCE), which control the analog and digital shunts and contain the low voltage sense system (LVSS) circuitry.
4. *Power Converters*. These components provide regulated DC power to the loads. There are 12 distinct converter designs and a total of 29 flight converters.

REQUIREMENT FLOWDOWN

The mission and environmental constraints translate into requirements for each of the EPS components.

Solar Array

The power requirements of the subsystem loads during the modes of operation and the sunlight availability as determined by the spacecraft orbit define the end-of-life power requirement of the MSX solar array. The solar array must be sized so that it generates enough energy to power the spacecraft loads and recharge the battery to a full state of charge (SOC) between eclipses. The sizing of the solar array must account for voltage and current degradation induced by the radiation environment. The sizing must also account for the possible occurrence of a single open-circuit failure of a solar cell

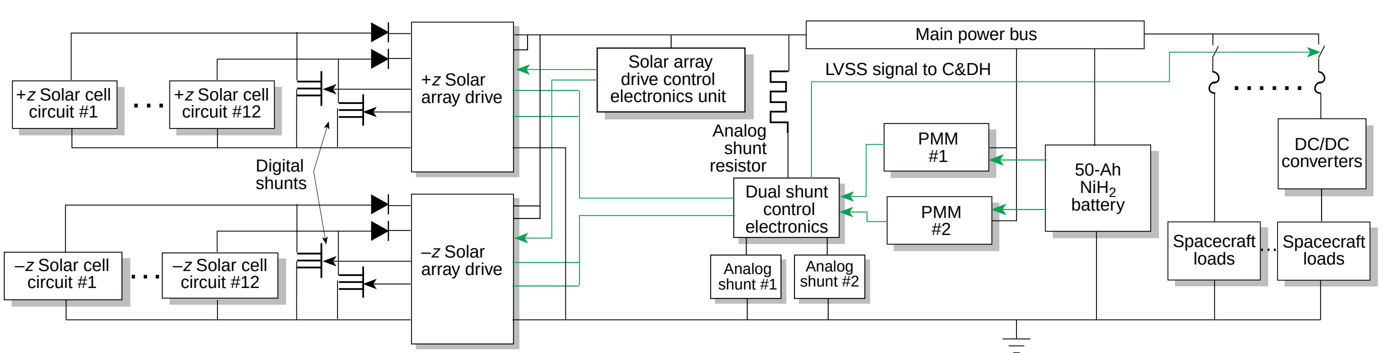

Figure 5. Block diagram of the MSX electrical power subsystem. (LVSS = low voltage sense system; C&DH = command and data handling; PMM = power management module.)

string. The solar array must not load the bus during eclipse, and it must not incur damage from partial shadowing during illumination. Array sizing must also account for the expected thermal variations that affect the array's current and voltage characteristics. Battery voltage variations also affect array sizing. Because the MSX EPS is an unregulated direct energy transfer architecture, with the battery directly connected to the solar array and bus, the operating voltage of the solar array is determined by the battery voltage, which varies as a function of temperature, current, and SOC. Since the EPS operates where the solar array current is fairly constant as a function of voltage, the depressed voltage of a warm, discharged battery lowers the solar array power output by lowering the array operating voltage.

Battery

The spacecraft loads and the mission eclipse profile define the requirements for the battery design. The battery must provide a low-impedance bus voltage in the specified range of 28 ± 6 V to power spacecraft loads during eclipse and to handle any loads that exceed solar array capability. The battery sizing is driven by resultant power demand, the duration of the power discharge, and the total number of discharge cycles required by the mission. The battery must be tolerant of the failure of a single cell in either an open or short-circuit condition. The battery design is also driven by restrictions of operating temperatures and temperature gradients.

Power Management System

The PMS must control the excess solar array power that is not used for the loads or for battery recharge. It must monitor and maintain the health of the battery by optimizing the battery charge rate during battery refill after all battery discharges. It must prevent damage to the battery and spacecraft loads by protecting against an overvoltage or undervoltage condition of the power bus. Finally, it must interface with the spacecraft C&DH subsystem, providing EPS telemetry and receiving ground commands.

DESIGN OF THE MSX ELECTRICAL POWER SUBSYSTEM

Solar Array

The GPS Block II solar array is a flight-qualified, production line system that closely matches the MSX power requirements. Although GPS was designed for a 12-h orbit, the array was used successfully on the MSX predecessor, Delta Star, which had a low Earth (1.5-h) orbit. Like MSX, Delta Star also had to slew a large sensor suite to track a moving target. The GPS array is stiff enough to avoid attitude system interactions and thermally induced oscillations when passing into and out of Earth shadow.

APL contracted with RI to provide a modified version of the GPS array. Figure 6 is a photograph of the solar array. Since GPS did not use the entire panel surface, the 172.7 × 78.4 cm panel layout was redesigned to use the entire front surface. The GPS string length was also increased from 100 to 104 solar cells in series because of the low-Earth-orbit radiation and increased temperature environments of MSX. The design placed a 61°C end-of-life maximum power point of 34 V at the main power bus. Spectrolab, a subcontractor of RI, installed 4992 model K6700 cells. These cells are nominally 3 × 6 cm silicon and have aluminum-doped back sides and 12-mil ceria-doped microsheet cover slides. Their initial efficiency at 28°C is 14.6%. Two series-connected 1N5809 bypass diodes are connected across each third of every string for shadowing protection, totaling six bypass diodes for each of 48 strings. The GPS thermistors on each panel were replaced with platinum wire temperature sensors to accommodate the MSX telemetry system. Two parallel 104-cell strings form a circuit. Each of the three circuits per panel has a digital shunt across it. The PMS uses the shunt to remove excess power from the spacecraft

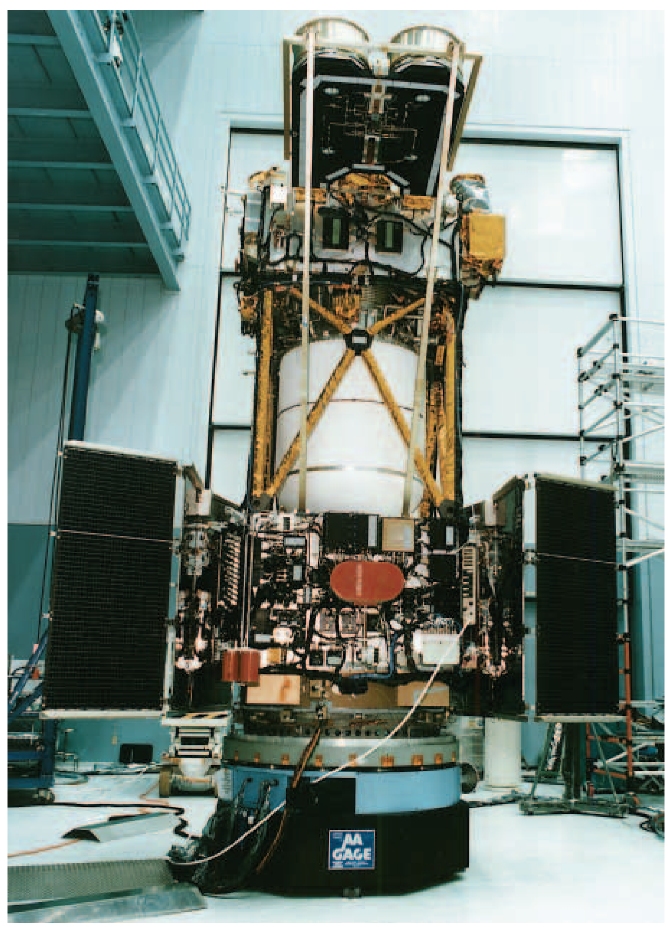

Figure 6. MSX solar array during deployment testing.

bus. The bypass diodes prevent destructive back-biasing of shadowed cells. The circuits are also isolated from the main power bus with blocking diodes to prevent the solar array from loading the spacecraft bus when it is not illuminated. The digital shunts, solar circuit blocking diodes, Sun pitch sensors, and digital shunt current monitoring resistor are housed in the boom that supports a solar array wing.

The boom supports four panels and, via the junction box, connects to the SAD, which contains slip rings and redundant motors that rotate the solar array wing through a full 360°. The SAD CEU can continuously and automatically point the two wings at the Sun, using the Sun sensors for feedback, or it can be commanded in a closed-loop fashion through the ACS. The CEU is commanded through the MSX ACS or by ground control. Whenever the spacecraft is not observing a target and SAD rotation jitter will not interfere with an observation, the CEU is commanded to automatically point the arrays at the Sun using the Sun sensors for feedback. The ACS meanwhile tries to orient the instrument suite away from keepout zones while maintaining the most favorable attitude for the solar array to stay normal to the Sun. When an observation event is about to begin, the ACS commands the CEU to drive the SAD and set the panels to the fixed position that provides the most power during the data collection event. The ACS uses SAD position signals from the CEU for feedback. The SAD and the ACS are redundant. If the SAD fails to correctly position the solar array, an autonomy algorithm within the C&DH subsystem can switch to the redundant SAD or to the ACS.

Battery

The selection of battery technology was reduced to two candidate chemistries. NiH_2 cell technology was selected over NiCd technology because of its lower weight and longer cycle life under deep discharge conditions. NiH_2 technology is also superior to NiCd in high-rate discharging, tolerance to voltage reversal in overdischarging, and acceptance of the high charge rates needed to replenish the battery before the next discharge. Another factor influencing the decision was a concern about the reliability of the available NiCd plate separator material.

Battery cell size selection was based upon the amount of charge the battery would need to supply during a discharge, the rate at which charge would be drawn from the battery, and the number of discharges required over the 5-year mission life. Cell capacity (C), or charge storage capability, was selected as 50 ampere-hours (Ah). This capacity was chosen largely because of the amount of low-Earth-orbit life cycle data available on the 50-Ah design and because Landsat 6 was slated to fly the same design. The other available capacities at that time included 65 and 90 Ah. However, these batteries did not have as much test time or projected low-Earth-orbit flight time, and they were too tall to clear the launch vehicle separation mechanisms.

The predicted loads for the MSX mission are capable of drawing sustained discharge rates on the order of 60 A (1.2C) with the possibility of 100-A (2C) peaks. Of the 38-A maximum solar array current, 30 A (0.6C) may be available to charge the battery. Track mode discharges a 50-Ah battery to 70% depth of discharge (DOD), which is acceptable for an infrequent, one-shot profile. Life projections proven by life cycle testing showed that the 50-Ah NiH_2 cell could deliver over 20,000 discharge cycles to 40% DOD at 10°C. Limiting the background mode to 40% DOD is acceptable. Two batteries connected in parallel would have added the extra margin of load sharing and redundancy, but there was room on the spacecraft to accommodate only one.

The decision to use only a single battery was significant because it introduced the possibility that a single cell failing in an open-circuit condition would end the mission. Theoretically, the NiH_2 pressurized seal can leak electrolyte, which can lead to the open-circuit failure condition. Each cell has a bypass circuit across it to minimize the otherwise catastrophic effect of one cell failing open. The simplest of bypass techniques is to use diodes to form the charge and discharge paths across the failed cell. The forward voltage characteristics of these diodes were selected to minimize power dissipation, limit the spacecraft bus voltage drop during discharge, reduce spacecraft bus voltage rise during charge, and avoid unnecessary discharge of a healthy cell. The circuit consists of a parallel matched pair of Schottky diodes for discharge bypass in parallel with three series-connected silicon diodes for charge bypass.

The 22-cell MSX battery was packaged by APL. Cells were procured from Yardney Technical Products, Inc. High-current battery cabling from cell to cell uses a flexible copper braid, and the power connectors use a bundle of twelve 12-gauge stranded copper wires. The battery is also instrumented to measure four cell pressures, four cell temperatures, full stack and half stack voltages, and stack current.

The battery assembly is mounted to the bottom of the spacecraft at the end opposite the optical instruments. Thermally nonconductive stand-offs maintain thermal isolation. The 31.5 × 62.2 × 96.5 cm, 89.8-kg package is topped with a large radiator covered with silver Teflon for passive thermal control. The cells are packaged in a "wine rack" configuration having a triangular pitch in four rows, alternating five and six cells per row. Figure 7 is a photograph of the battery.

Each 8.9-cm (3.5-in.) diameter, cylindrical Inconel cell case is partially enveloped by an electrically insulated thermal sleeve. The sleeve encloses three-fourths of the cell, extending slightly beyond one end. A flange about the sleeve near the center of the cell attaches to

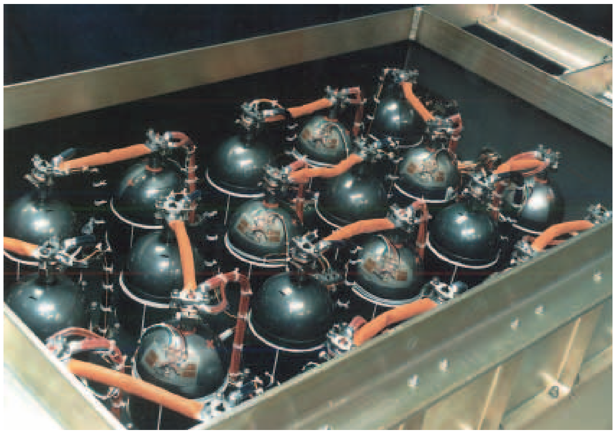

Figure 7. The MSX battery with the 22 NiH_2 cells assembled and mounted on the test fixture.

a structural plate that has been cut out to allow the cell to protrude. Another flange at the extended end of the sleeve is connected to the radiator plate. A bracket holding the five bypass diodes is thermally connected near the sleeve between the structural and radiator plates so as to conduct heat to the radiator.

The passive battery thermal control is designed for an operating range of 0 to 20°C and a survival range of –10 to 40°C. These thermal requirements were the most significant factor in determining the battery packaging design. Most difficult to overcome was the requirement that if a cell fails open circuit, bypass diodes must conduct large amounts of current and dissipate large amounts of power. The battery enclosure also houses the electronic circuit boards for monitoring and conditioning the AD590 temperature sensors, strain gauge pressures, and full and half stack voltages. The redundant 0–5 V signals are isolated and distributed to the MSX telemetry system for downlink and to the PMM for control of the battery charge rate. The PMM provides the battery with regulated voltages for powering the circuitry.

The battery electrically interfaces through five connectors: (1) positive battery terminal, (2) negative battery terminal, (3) umbilical, (4) test, and (5) PMM/telemetry. In the spacecraft harness, the power cable for the positive battery terminal extends out to the battery flight plug connectors before connecting to the solar power cabling at the main bus. Outside the battery housing, a Hall-effect current sensor monitors total load current in the main bus. The power cable for the negative battery terminal goes to a precision 625-$\mu\Omega$ current monitor, the other end of which connects to the main power ground. The main power ground and signal ground busses connect to the spacecraft chassis at a single-point ground.

Power Management System

If the power generated by the solar array exceeds the power requirement of the subsystem loads, that excess power is taken up by the battery. The purpose of the PMS is to manage the dissipation of excess solar array power not needed for battery charging. The charge control system consists of 24 digital shunts, redundant analog shunts, the DSCE, and redundant PMMs that can be selected by an autonomy algorithm within the C&DH subsystem.

The digital shunts can remove excess power from the spacecraft by nondissipatively shorting out the entire solar array in 1.5-A steps. Metal oxide semiconductor field effect transistors (MOSFETs) are used as digital switches to short out individual solar array circuits. Since each SAD contains only four slip rings capable of carrying high currents, the MOSFETs had to be placed on the boom. Each solar wing contains 24 individual cell strings, but only 12 slip rings are available to carry digital shunt control signals, and so the strings are paired. The negative terminals of the MOSFETs are bussed together at the digital shunt current telemetry resistor. The other end of the telemetry resistor is the common ground for the 12 solar circuits. It is connected to the spacecraft through two of the four power slip rings. The positive terminals of the circuits are bussed together through isolating diodes, and are connected through the other two power slip rings.

Redundant analog shunts act as a vernier in removing excess power by controlling battery charge current to within ±188 mA. An analog shunt consists of a resistor bank in series with a MOSFET. The resistor bank is implemented with Kapton-encapsulated thin-film strip heater elements that are distributed around the launch vehicle adapter ring.

The DSCE measures battery current through a precision current monitoring resistor, compares it to the battery charge current limit commanded by the PMM, and feeds back the difference to control the analog shunt. When the analog shunt reaches its defined operating extremes, the DSCE automatically switches the digital shunts on or off as appropriate. The DSCE also contains redundant LVSS circuits. If for any reason the battery voltage drops below an acceptable level, indicating a low SOC, the LVSS will signal the C&DH subsystem. The C&DH subsystem will respond by aborting any operation in progress, shedding noncritical loads, and returning the spacecraft to the parked attitude. This reduces the load on the battery and allows its voltage to recover. If the aborted operation was holding the solar array fixed and away from the Sun, then park mode allows the solar array to reorient toward the Sun and begin recharging the battery.

The PMM consists of a current integrator, a computer running a charge control algorithm, analog-to-digital and digital-to-analog circuits, and serial digital interfaces to the MSX C&DH systems. The PMM samples all battery data with the analog-to-digital circuit and uses the digital-to-analog circuit to command the DSCE. To best maintain battery health, the computer uses a two-step charge control algorithm based on the charge return to the battery. It also provides automatic charge cutback if it detects excessive battery voltage, pressure, or temperature.

The charge control algorithm (CCA) for the NiH_2 battery is based on the battery's charge return. The battery recharge controller allows all available solar current to charge the battery as long as the charge efficiency is high and the charge is not stressing the battery. As the battery approaches full SOC, the charging becomes less efficient and the process becomes increasingly exothermic. Before the heat dissipation becomes excessive, the controller must lower the charge rate from *full* to *reduced* charge; when the battery is full, it must reduce the rate further to *trickle* charge. The point where the controller should reduce the charge rate depends partly on the amount of exothermic stress that the battery can safely tolerate and partly on the heat rejection characteristics of the battery thermal design. These characteristics change over time, as does the battery response to exothermic charging, and the criteria are inexact. Therefore, the reduced charge and trickle charge rates and the criteria for their use can be programmed from the ground. Additionally, if a battery cell fails in an open-circuit condition, the excess heat dissipated in the bypass circuitry may cause the battery to overheat. Therefore, the full charge rate level also can be programmed from the ground during the mission.

The full charge rate can be set within the limits of 18.1 and 41.9 A; the reduced charge rate can be set within the limits of 9.4 and 21.1 A; and the trickle charge rate can be set within the limits of 0.0 and 5.7 A. Each limit can be set by ground command in 375-mA steps to optimize the charge regime for long battery life during either high performance or light duty operation. The CCA bases its charge rate selection on battery voltage, pressures, temperatures, and SOC. Assuming that during a nominal charge no excessive pressures, temperatures, or voltages show up, the SOC is the sole factor determining which charge limit to use.

The most important element in the algorithm is the charge counter, which calculates the battery SOC. The PMM implements the charge counter in software using inputs from a current integrator. The current integrator interrupts the processor whenever it measures 6 ampere- seconds of charge or discharge. If the interrupt is received on the discharge port, the PMM decrements the SOC accumulator, which is a 16-bit word in memory. If the interrupt is received on the charge port, then a programmable value of charge-to-discharge (C/D) ratio is added to a special temporary word. The *overflow* of that word, if any, increments the SOC accumulator. This procedure uses the C/D ratio to account for battery charging inefficiency. Therefore, a charge counter reading of full (SOC = 100%) means that the battery has accepted an amount of charge equal to the amount taken out multiplied by the C/D ratio. If the appropriate C/D ratio (which depends on the temperature and usage of the battery) has been set by ground command, then the charge counter should fairly accurately reflect the actual battery SOC.

On the basis of test data and mission telemetry, the point at which high-rate exothermic charging becomes damaging can be mapped to an SOC limit. If the SOC is less than this programmable limit, then the full charge limit governs. If the SOC is greater than the limit but the battery is not yet full, then the reduced charge limit is used. The battery charge rate is reduced to trickle charge when the SOC equals 100%. Whenever the SOC drops below 100%, the CCA determines the appropriate charge limit and charges the battery. The SOC limit that marks the beginning of reduced charge can be programmed from the ground to account for performance variations of the battery over its mission life.

If at any time the battery temperature or pressure exceeds its limits, then the charge rate is set to trickle. If at any time the battery voltage exceeds its limit, then the charge rate is reduced. If the reduced limit fails to rectify the overvoltage within 1 min, the battery is put on trickle charge. Whenever any limit is exceeded, a limit flag is latched and downlinked. The limit flag will unlatch when the battery goes into discharge or when it is reset by ground commands. All of the limits can be adjusted by ground command. If any sensor or conditioning circuitry is determined to be faulty, then the limit checking for that particular sensor can be masked. There is also a protected mode where the charge counter can be disabled to force charging if the battery is actually at a lower SOC than is calculated by the CCA. Figure 8 illustrates the CCA.

DC/DC Power Converters

MSX uses a low-impedance, unregulated battery bus with a single-point ground. Distributed DC/DC converters are local to the loads. Because the number of loads on MSX was large, APL subcontracted for the converters needed for in-house subsystem component designs or other procured payloads that did not come with compatible power converters. ST Keltec (formerly Quadri Electronics) provided MSX with 29 flight units and 12 engineering models, which represented 12 different converter designs. The designs use a 100-kHz switching forward converter to regulate line variations and input filters with active in-rush current limiting. Magnetic amplifier post regulators are used to control load variations. Converter packaging is designed to meet radiated emission and susceptibility requirements. The vendor performed flight qualification testing of the converters.

AUTONOMY

The EPS has some redundant components, including the SAD and PMS. In case of failures in these compo-

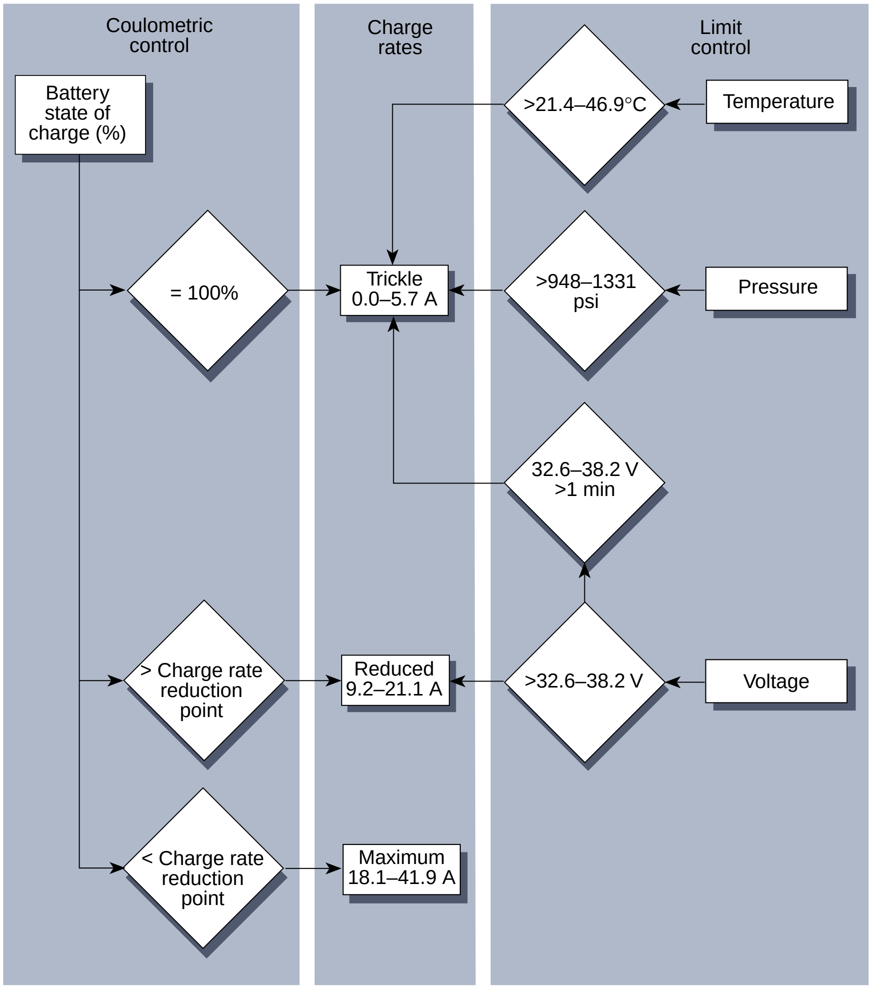

Figure 8. Flow diagram of the charge control algorithm. The algorithm uses a charge counter plus temperature, pressure, and voltage limits to determine battery charge rates.

nents that could affect the ability of the solar array to point or the battery to recharge, then complete functionality can be restored by switching to the redundant unit. The EPS, however, cannot perform this switching by itself.

The C&DH subsystem has a feature that allows autonomy rules to be programmed to routinely monitor measured values for out-of-tolerance conditions and to automatically take corrective action. Low battery pressure, for example, will provide advance warning of low SOC, triggering a switch to the redundant PMS before the battery voltage drops low enough to trip the LVSS circuit. Similarly, autonomy rules are devised to automatically switch to the redundant SAD, or also to the redundant ACS, which controls the SAD.

PRELAUNCH ANALYSIS AND TEST

The solar array vendor, RI, performed the appropriate structural, electrical, and thermal analyses required to prove the new solar array interfaces with the spacecraft and MSX environment. With its subcontractors Moog, Ball Aerospace, and Spectrolab, RI also performed acceptance testing on all solar array components. APL wired the junction box and, with RI's help, performed mechanical and electrical functional, acoustic, pyrotechnic shock, and deployment tests with the spacecraft. Using a large-area pulsed solar simulator to emulate solar spectra and intensity, APL also flash-tested the panels before final thermal vacuum bakeout. These simulator tests proved that the panels were operating with predicted efficiencies. On the basis of these measurements, the predicted current output at the beginning of solar array life is 37 A at summer solstice, 38.5 A during the spring and fall equinoxes, and 40 A at the winter solstice.

Flight qualification testing of the battery consisted of cell acceptance tests by the vendor and battery testing at APL. The battery bypass diodes were screened over temperature and current before matching, and the cells were checked for leaks before assembly. The finished battery was tested for capacity at 0, 10, and 20°C, cycled at 40% DOD (background mode) and 20% DOD (park mode), and tested for track mode performance. A 72-h open-circuit capacity test proved the battery could retain its charge during the trip to the launch pad and installation on the Delta II launch vehicle. Further tests simulating the prelaunch environment showed the battery could be safely maintained at or below 25°C before launch and still deliver 50 Ah. The launch loads are light, and the battery is not expected to discharge more than 10% before solar array deployment and illumination.

Battery performance in the various modes of operation was analyzed to characterize the bus voltage and to ascertain energy balance. A typical battery voltage profile through a background mode operation (Fig. 9) shows the voltage variations within the prescribed bounds of 28 ± 6 V of the unregulated topology. Since the solar array power output varies with the solar array temperature and the battery-driven operating voltage, analyses were performed to verify solar power output at end of life for various expected array temperatures as a function of the expected battery voltage. The results (Fig. 10) illustrate improved array performance at higher battery voltages. Performance diminishes at higher array temperatures.

Battery reconditioning was provided periodically throughout integration and test to maintain the performance of the cells and measure their capacity. For reconditioning, the battery was discharged at 25 A (C/2) until the lowest cell voltage fell below 1 V. The integration of the discharge current over the duration of the discharge provided a measure of the usable charge, or capacity. Then resistors were shunted across each cell to allow complete discharging to below 0.1 V per cell. The battery was then fully charged. During the discharge and charge activities, the battery was carefully monitored to avoid any overstressing. Thermal control was provided during reconditioning by air conditioners that maintained the battery temperature at about 15°C. The specified minimum capacity is 50 Ah at 20°C or 57 Ah at 10°C. Capacity measurements in the field have typically exceeded 58 Ah at average charge temperatures of about 15 to 18°C.

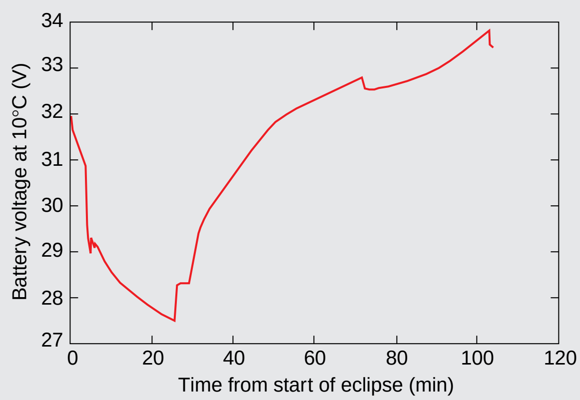

Figure 9. Profile of battery voltage at 10°C for a typical orbit with a 40% depth of discharge. Voltage ranges from 27 to 34 V.

Considering eclipse time, solar array power, a cold case average 672-W load, a degraded battery energy efficiency, and a 31-V average charge voltage, the design margin can be conservatively approximated as shown in Fig. 11. Assuming one solar cell string fails, a worst-case design margin minimum occurs about 7 months into the mission due to a 27.5-min eclipse. Note that design margin is more sensitive to eclipse time than it is to radiation damage, as is evidenced by the worst case being so soon after launch. Other brief minima can be seen at about 1.6, 2.6, 3.6, and 4.6 years. Background operations performed at these times may have to be restricted to protect the battery from excessive DOD.

Thermal vacuum testing at the spacecraft level explored power subsystem limitations. While simulating minimum power margin, situations that proved restrictive to the more demanding data collection events (DCEs) were found. DCEs beginning near the end of eclipse provide the worst-case orbit phasing for power: the battery is not allowed to charge because attitude maneuvers point the locked solar array away from the Sun. The thermal cold case also provides the worst-case spacecraft temperature for power availability because of the heavy heater load demand. Therefore, the loads for the test were reduced to a minimum to quicken the transition to cold. After just three of these cold-case, worst-phased consecutive DCEs, the battery had discharged to 70% DOD and the simulation had to be stopped. The largest contributor to the negative energy balance was the persistence of large heater loads. As indicated in Fig. 1, the heaters were supposed to turn off as the spacecraft warmed up during background mode DCEs, but when the spacecraft is thoroughly chilled, its large thermal mass tends to stay cold for several orbits. The lesson learned is that if the spacecraft is allowed to

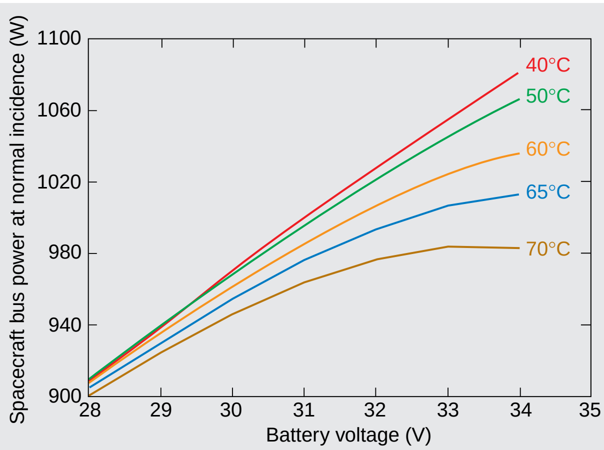

Figure 10. Solar power output at the spacecraft's end of life for various expected array temperatures as a function of the expected battery voltage. After 5 years of radiation exposure, solar power generation becomes sensitive to high array temperatures. At temperatures above 70°C, higher battery voltages cease to be advantageous.

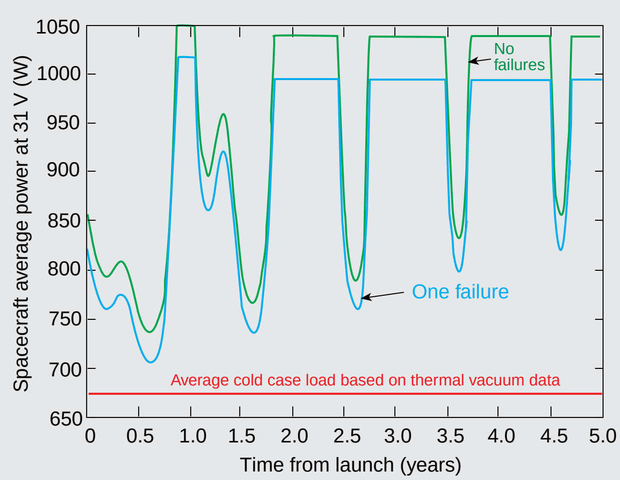

Figure 11. Average power MSX can support over mission life. Taking into account solar array radiation degradation, eclipse time, and battery inefficiency, the minimum orbital average power available is coincident with the maximum eclipse time, which occurs about 7 months after launch.

sit dormant for so long that many temperatures fall below thermostat set points, then strenuous DCEs cannot be supported right away. In this case, which is most critical during the minimum design margin times as illustrated in Fig. 11, DCEs should be gradually introduced to prevent the battery from overdischarging. The spacecraft should be warmed at the same time, thus allowing the heaters to eventually switch off.

CONCLUSION

The MSX EPS is characterized by utilization of state-of-the-art technology, tailoring of existing design elements, and simple, reliable approaches to fault protection. The use of NiH_2 technology for energy storage provides performance and reliability not otherwise attainable. The microprocessor-based PMS furnishes flexibility and functionality in maintaining optimum battery performance by employing charge control algorithms that account for battery aging and faults. The adaptation of the solar array from the GPS design effectively makes use of years of flight heritage reliability. Passive bypass circuitry in the battery design, PMS redundancy, and self-diagnostic autonomy algorithms provide effective and reliable protection from single-point failures.

This EPS design has demonstrated in tests and analyses the ability to meet or exceed the diverse requirements imposed by the mission environment and performance objectives. Operational constraints are few and minor. Its flexibility makes it a candidate for adaptation to other long-duration, high-power, low-Earth-orbit applications with high transient loads.

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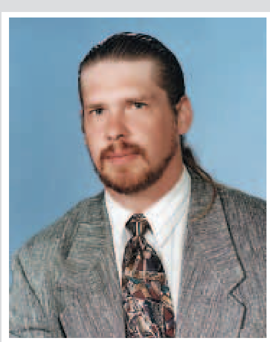

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