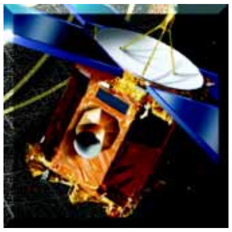

Power Subsystem Design and Early Mission Performance

Jason E. Jenkins, George Dakermanji, Michael H. Butler, and P. Uno Carlsson

The power subsystem for the Near Earth Asteroid Rendezvous (NEAR) spacecraft is a direct energy transfer system with an Eagle Picher Industries Super nickel–cadmium battery and fixed solar panels populated with solar cells of gallium arsenide on germanium. Power is controlled by digital and analog sequential shunts. During the mission the distance from the Sun varies significantly, causing large variations in solar array power and temperature. To minimize solar array size, the solar array and power system electronics were designed to allow operation at voltages both greater than and less than the array's maximum power voltage. The power system design emphasizes reliability with low weight and volume. Analyses verified subsystem capabilities over the varied mission environment, and early mission flight data indicate that performance is consistent with analysis models.
(Keywords: Gallium arsenide solar panel, Nickel–cadmium battery, Power.)

INTRODUCTION

The Near Earth Asteroid Rendezvous (NEAR) mission has in its charter a dedication to low cost, short schedule, low mass, and high performance. The power subsystem for this spacecraft must support this mission using low-risk, mass-effective technology operating in a variable interplanetary environment. This article describes the requirements imposed upon the power subsystem, characterizes the architecture and components, analyzes the projected performance, and reports flight performance from the early part of the mission.

OPERATIONAL MODES

The spacecraft has two primary modes of operation that dominate the mission timeline—cruise mode and asteroid mode. From launch until acquisition of the asteroid Eros, the spacecraft is in the cruise mode. Asteroid mode consists of all activities after rendezvous with Eros. Cruise mode is further divided into two operational states: outer cruise and inner cruise.

Outer Cruise Mode

During outer cruise, solar illumination (and, thus, available solar array power) is at a minimum for the mission, and spacecraft loads are at a survival minimum. During the deep-space burn and the flyby and observation of the asteroid Mathilde, a significant effort in load management was implemented to maintain a load power level below or near the nominal outer cruise load level, so that these events could be performed without reliance upon battery power.

Inner Cruise Mode

Inner cruise is defined as the periods prior to asteroid acquisition in which the spacecraft is close enough to the Sun to allow powering of additional loads while maintaining power margin. Inner cruise activities include maintaining thruster flange heaters and allowing instrument calibration before asteroid acquisition.

Asteroid Mode

The rendezvous burn denotes the beginning of the asteroid mode of operation. During this phase, instruments are activated, reaction wheel power is increased, and the spacecraft orbits and observes the asteroid. Average power level requirements during this mode of operation are the highest of the mission.

ENVIRONMENT

Illumination

Figure 1 illustrates the highly variable illumination environment. During cruise mode, spacecraft illumination ranges from 1.00 sun under air-mass 0 illumination to a low of 0.21 sun at a distance of 2.2 astronomical units (AU), increasing again to 1.03 sun during the Earth flyby. Solar flux input decreases to 0.33 sun at 1.7 AU at asteroid rendezvous. Once at the asteroid, the illumination increases to 0.79 sun at an asteroid perihelion distance of 1.1 AU, diminishing again to 0.58 sun at the conclusion of the mission at 1.3 AU. Variations in angle throughout the mission lead to additional variation in illumination. The nominal attitude for the spacecraft is Sun-pointing. However, the spacecraft must be capable of pointing directly at the Earth throughout the mission and must point up to 30° off Sun during observations at the asteroid. The spacecraft configuration was designed to minimize shadowing of the solar array. However, partial array shadowing, which could occur during integration and test, prior to array deployment, or during any loss of attitude in flight, had to be considered.

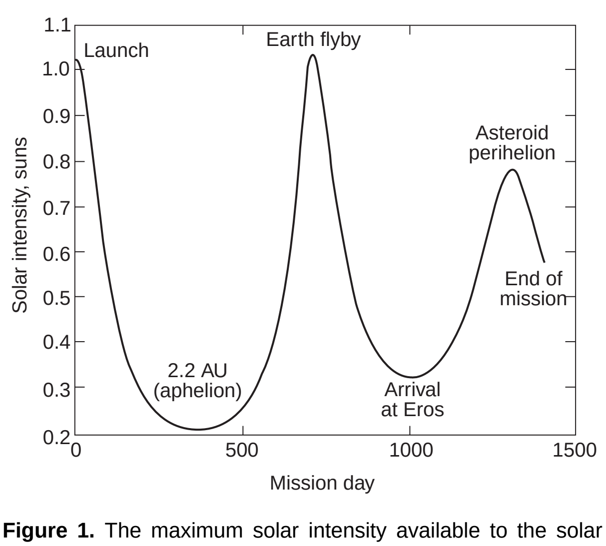

Figure 1. The maximum solar intensity available to the solar panels over the NEAR mission. Maximum intensity ranges from 1.0 to 0.2 sun over the mission timeline. Additional variations are introduced by spacecraft attitude requirements.

Thermal

Large variations in the thermal environment significantly affected the selection and sizing of the NEAR solar array. Although the spacecraft experiences a wide temperature range, it does not encounter rapid or numerous large thermal cycles, as in a geocentric orbit. Smaller thermal cycles are induced on the solar panel as the power system electronics (PSE) regulate the power delivered to the spacecraft bus. These cycles are caused by changes in absorptivity as array segments are shunted and by analog shunt elements located on the back of the panel.

Radiation

The radiation environment of deep space is dominated by unidirectional protons exhibiting an inverse-square dependence upon solar distance beyond 1 AU. The NEAR spacecraft was launched in February 1996 during a solar minimum period. During this time no significant solar activity was predicted, with a confidence of 95%. The start of predicted solar activity is coincident with the spacecraft's arrival at the asteroid, when the annual proton fluence is expected to increase to the levels shown by Table 1.

Contamination

Thrusters will be fired over several periods during the mission. Ultra-pure monomethylhydrazine is the propellant used by the attitude thrusters, and it is not

Table 1. Annual radiation fluence while the NEAR spacecraft is in asteroid mode.

Energy (MeV)	Annual fluence (cm^{-2})
>1	1.64×10^{11}
>4	5.49×10^{10}
>10	2.12×10^{10}
>30	5.81×10^{9}
>60	2.61×10^{9}

expected to contaminate the solar array or to adversely affect solar array performance. Bipropellant burns are all directed beneath the solar panel face.

Vibration and Static Loading

The vibrational loading the spacecraft had to withstand was determined by the Delta-II launch vehicle. The solar array, however, had an additional mechanical requirement: in the launch configuration, the panels were bound tightly to the spacecraft against a center standoff. This mechanical loading bent the panels, storing a small amount of mechanical energy to assist in panel deployment.

POWER SYSTEM DESIGN

Topology

The NEAR power subsystem is a direct energy transfer system (Fig. 2). The solar array is connected directly to the spacecraft bus without active series components. This bus voltage is maintained at 33.5 V ± 1% by a bus voltage regulator that uses digital shunts to coarsely shunt excess array current and by banks of sequential linear shunts for vernier bus voltage control. The regulator operation optimizes solar array sizing by operating on both sides of the solar array current–voltage (*I*–*V*) curve. The battery is isolated from the bus by a charger and discharge diodes, eliminating solar array power reduction from variations in the battery voltage. Battery charging and bus regulation are performed by hardware, with all autonomy algorithms being implemented in software by the command and data handling (C&DH) processor.

Subsystem Interfaces

Power Distribution

The power subsystem carries a power bus where voltage monitoring and regulation are performed. This bus is connected to a spacecraft main power bus on terminal boards, from which switched and unswitched load buses are created. The power system return bus is similarly connected to a common single-point ground on the terminal boards.

Telemetry

The interface to the C&DH subsystems provides lines to receive relay commands and to provide relay status, 0–5 V telemetry, and temperatures telemetry. Telemetered parameters include the solar array return current, aggregate digital shunt current, analog shunt currents, battery charge rates, bus and battery voltages, component temperatures, and subsystem relay states.

Attitude Control

A discrete line to the attitude interface unit of the guidance and control subsystem provides a measurement of the solar array current for use in backup Sun-locating algorithms.

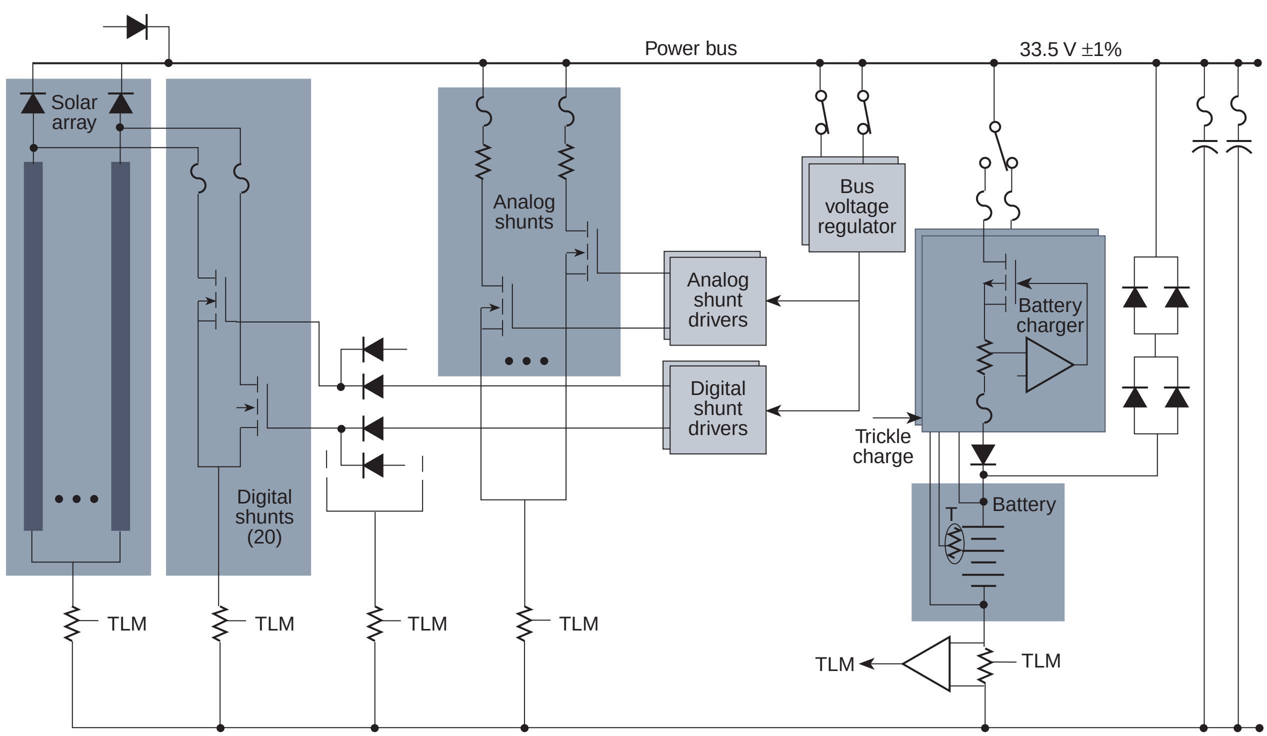

Figure 2. The NEAR power subsystem—a regulated, direct energy transfer design. The solar array provides power directly to the main power bus, which is regulated to 33.5 V by the power system electronics. Bus voltage is regulated using digital and analog shunts to dissipate excess array power. The battery is isolated from the main bus by the charger. (TLM = telemetry.)

External Interfaces

During integration and test, the spacecraft was powered by the solar array simulator or by a ground power unit via the spacecraft umbilical connector. The umbilical connection allowed the ground power unit to power the entire spacecraft via the main bus or to provide battery trickle charge and monitoring when the spacecraft was off. External connections to the main bus were protected by diodes housed within the PSE.

Solar Array

The solar array is the primary power source for the NEAR mission. It is sized to provide power throughout the mission (except during launch) without reliance upon the battery. Spectrolab, Inc., performed fabrication and laydown of the cells; British Petroleum fabricated the substrates.

Cell Interconnect Coverglass Assemblies

Each of the four solar panels is populated with 960 7.478 × 2.891 cm cell-interconnect-cover (CIC) assemblies. The CICs are fabricated, individually tested, statistically grouped, and assembled into circuits before laydown on the panel substrates. The solar cells are a gallium arsenide (GaAs) junction grown on 0.18-mm (7-mil) germanium substrates. The cell has an antireflective coating and is protected by a 0.15-mm (6-mil) ceria-doped microsheet (CMX) coverglass with an ultraviolet reflective coating. The minimum average efficiency of the CIC flight lot is greater than 19.1%. Compared with silicon solar cells, GaAs technology is less susceptible to the extremes of temperature throughout the mission, which allows a more efficient series/parallel arrangement of cells. The high power efficiency also provides better power margin throughout mission life, and even though the radiation environment of NEAR is not exceptionally severe, the radiation resistance of GaAs improves the power margin at the spacecraft end-of-life. The cells are interconnected with a silver mesh, which lies within the plane of the CICs when bonded to the substrates. The in-plane interconnect configuration was selected to reduce the risk of damage from the despin cables lying across the panel faces in the stowed launch configuration. Silver was selected to satisfy the magnetic cleanliness requirement.

Substrates

The solar array consists of four coplanar, single-sided panels. The substrates are a 2.5-cm-thick aluminum honeycomb core with 0.13-mm (5-mil) aluminum facesheets. The cell side is insulated by a 0.05-mm (2-mil) sheet of Kapton. Three Kapton-encapsulated Inconel strip heaters bonded to the back of each panel serve as the analog shunt resistive elements. The back of the panel is painted with A276 white paint, providing a high-emissivity surface for heat dissipation with a low absorptivity to mitigate the temperature increase from the Earth's albedo during the Earth flyby maneuver. All mechanical interfaces on the panel are located on the back or edges, allowing the entire front surface to be populated with cells.

Panel Assembly

The substrates were fabricated and delivered to the solar array vendor with the Kapton insulation and the analog shunt elements bonded and the back side painted. Before cell laydown, the CICs were electrically connected into series strings of 20 cells. Two strings are laid side by side and connected in series to form a single circuit of 40 series cells whose positive and negative contacts are located in close proximity, reducing magnetic field emissions. Each panel is populated with 24 40-cell circuits, which are bonded to the panel using CV-2568 room temperature vulcanizing silicone adhesive. The front of a NEAR solar panel is shown in Fig. 3. Circuits are arranged by flipping the positive and negative terminations down the length of the panel edges, alternating the direction of current loops to reduce magnetic field generation. The 24 circuits are electrically and physically grouped in parallel into five digital shunt groups of four or five circuits each. The total mass of the four panels is 46.3 kg.

Battery

The purpose of the NEAR battery is to provide power in the launch phase of the mission and a backup energy source in case of an excessive load excursion or temporary decrease in solar array power. The battery cells were fabricated by Eagle Picher Industries. Hughes Aircraft Company assembled and tested the battery. The battery cell technology selected for the NEAR program is Super nickel–cadmium (S-NiCd). Cells have a nameplate capacity of 9 A·h and have the standard NiCd prismatic construction. Physically, the S-NiCd technology differs from the standard NiCd technology in that the separator is a polymer-impregnated Zircar instead of nylon, the impregnation of the plates is electrochemical instead of aqueous, and the electrolyte contains proprietary Hughes additives. The S-NiCd battery has a preferred storage condition of 100% state-of-charge sustained by a trickle charge. It is thus well suited to the NEAR application, because it can be maintained during the mission in its optimum storage regime, with no need for reconditioning prior to use. The battery was selected to be rechargeable to provide a backup power supply for repetitive load demands throughout the mission. The NEAR battery

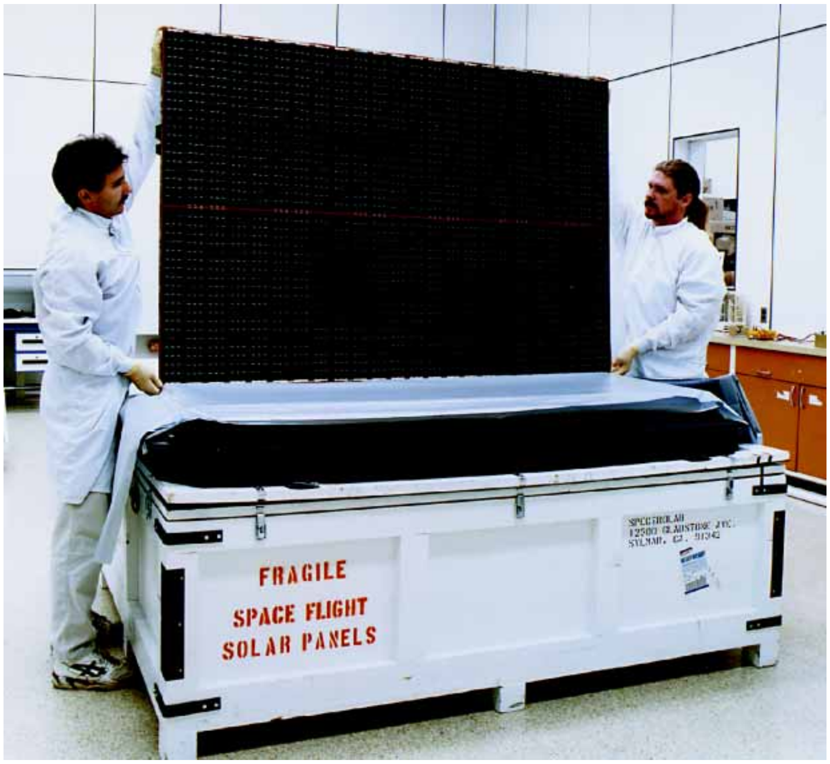

Figure 3. One of the four NEAR GaAs solar panels during incoming inspection at APL. Each panel is populated with 960 GaAs solar cells and produces 500 W of power under air-mass 0 illumination.

assembly consists of 22 cells electrically connected in series (Fig. 4). Each cell is wrapped in a 0.08-mm (3-mil) Mylar sheet to provide electrical isolation to the battery chassis. Wrapped cells are stacked into two rows, face to face, separated by thermal fins inserted between the cells. The cell stack is held in compression between two endplates by a pair of tie rods running laterally along each side of the battery. Electrical connections are made to the battery via a flight and a test connector located on either end of the assembly. The total mass of the NEAR battery (minus heaters and thermostats) is 12.3 kg.

To maintain battery performance, the battery operating temperature must remain between 0 and 15°C, although during rare excursions of discharge the battery temperature may get as high as 30°C. However, because discharge events are unlikely (and would be infrequent if they did occur), the temperature excursion was considered acceptable. The battery thermal design has a requirement that the maximum temperature gradient in a plane parallel to the baseplate be less than 3°C to facilitate temperature-compensated voltage (VT) limit charge. The maximum allowed gradient along the vertical axis of any cell is less than 5°C to avoid thermally induced electrolytic migration and dry spots in the separator material.

The baseline design of the battery assembly is a derivative of NASA's TOMS-EP battery. The TOMS-EP battery, however, is mounted to the spacecraft deck and uses that same interface to conduct heat from the battery. Since the thermal requirements of the neighboring instruments on NEAR are not compatible with those of the battery, the NEAR battery is thermally isolated from the deck. To create this thermal isolation, the endplates of the battery rise above the cell tops into a mounting bracket configuration, which is fastened to the spacecraft using fiberglass standoffs. The battery is radiatively decoupled from the spacecraft by a thermal blanket that surrounds the entire assembly except the radiator surface. The thermal fins between the cells conduct heat toward the radiative baseplate opposite the mounting surface. Redundant heater elements are located on this surface. Thermostats are located on an edge member.

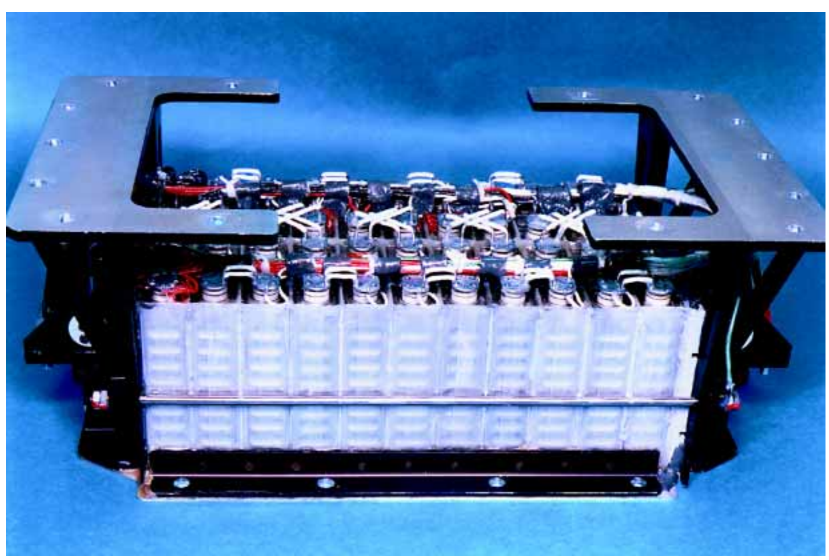

Figure 4. The NEAR battery, which consists of 22 Eagle Picher Super nickel–cadmium cells. The battery is mounted to the spacecraft by the large endplates. The bottom surface is the thermal interface from which the battery radiates excess heat to space.

Power System Electronics

Functional Description

The PSE houses the bus voltage regulator, battery charger, analog shunt controller, digital shunts, and main power bus. Power and telemetry from the battery and solar array are connected directly to the PSE. Except for battery and solar array temperature, all of the power system telemetry is available through the PSE interface.

Bus Voltage Regulator

The PSE houses two bus voltage regulators (BVRs). The primary BVR is set to regulate the bus at 33.33 V, and the redundant on-line unit regulates to 33.67 V, providing on-line protection from an overvoltage failure of the primary unit. A hard 40-V voltage clamp is also implemented to protect against voltage transients that are faster than the BVR control loop. Digital shunts are wired across each of 20 solar array circuit groups to shunt excess power from the main bus for coarse voltage control. The primary and redundant digital shunt drivers control the 20 digital shunts. Each digital shunt is a power metal oxide semiconductor field-effect transistor (MOSFET) sized to 3.5 A of solar array current under maximum illumination conditions. The BVRs use banks of sequential analog shunts that are connected directly to the main bus for fine bus voltage control. Each bank consists of six field-effect transistors (FETs), each directing up to 22 W of power through 50-W shunt elements located on the back sides of the solar panels. As each shunt element becomes saturated, the next element is turned on. As the last shunt element is saturated, a digital shunt is turned on, removing the power of a circuit group. Spreading the power removal capability of the analog shunts over six elements reduces the total dissipation in the FETs to facilitate thermal management of the PSE box.

In the varied mission environment, the I–V characteristics of the solar array vary with temperature and illumination. The series-parallel arrangement of solar cells was selected to optimize array output by maintaining the maximum power voltage of the array near the nominal bus voltage. As a result, the bus voltage at the solar array may lie on either the short-circuit current side of the I–V characteristic curve or on the open-circuit voltage side. The bus controller is designed to maintain stability when operating on either side of the I–V curve.

Battery Charger

The PSE houses two identical battery chargers, which provide an off-line redundancy. The battery charger isolates the battery from the spacecraft bus and charges the battery via a FET connected to the main bus. The primary charge control regulates the battery voltage to a VT limit. A current limiter control loop is run concurrently with the VT controller to limit the maximum charge rate to C/20 (0.45 A), where C is the rate in amperes required to discharge the battery of its nominal capacity in 1 h. This limiter recharges the battery after discharge or tops off the battery in preparation for discharge. A lower current limit, trickle charge, is selectable by relay command, which limits the current to C/75 (0.12 A).

REDUNDANCY AND RISK

Solar Array

The solar array is designed to augment the robustness and fault tolerance of the overall power subsystem. The array is sized to accommodate one group of five circuits more than is required by the loads, allowing the loss of a single digital shunt in a shorted state or multiple losses of strings. Individual strings are all isolated from the digital shunt busses and power busses with diodes preventing a shorted string from affecting power generation of the remaining strings. Wiring from the circuit busses to the connectors is redundant, as are power connections between two connectors. To mitigate the risk of cell cracking, the crystallographic axes of the cells are oriented such that a cell crack would propagate nearly parallel to the cell's contact grid, resulting in two smaller parallel cells rather than loss of a string.

Battery

The existence of a full healthy battery is not critical throughout the mission, but the battery must be available to clear fuses and provide power for attitude excursions. Separating the battery from the bus protects the main bus voltage from battery voltage variations resulting from thermal variations, changes in state-of-charge, or cell failure. Should a cell fail, the battery charger will maintain trickle charge, and ground command may select a lower VT limit to account for the missing cell. Because the battery is isolated from the bus by the charger, even a battery with several cell failures does not affect the main bus voltage, but should be able to clear fuse faults. Battery wiring is redundant, as are the VT temperature sensors and telemetry sensors. The battery chassis is grounded through 10 kW in the PSE instead of locally to the spacecraft deck, allowing operation in the event that cells develop a short to the chassis. An open-circuit failure mode of the battery is exceedingly unlikely and is not protected against. Such a failure would have been critical during launch; if it were to occur later in the mission, it would require active load management.

Power System Electronics

As shown in the PSE design description given earlier, elements of the PSE are redundant or fault tolerant. The bus voltage controllers are concurrently active, so an overvoltage fault is arrested by the redundant unit. The digital shunts may be controlled by either bus voltage controller and are sized in conjunction with the array to allow the shorted loss of any one and the open-circuit loss of at least three. Analog shunt banks are fully redundant and can tolerate the loss of at least one shunt element. The battery charge is fully redundant and may be switched by ground command or by autonomy algorithms.

Autonomy

The requirement that the power subsystem be tolerant of single-point failures is met by design and with the use of the onboard processor in the C&DH subsystem. Having no processor of its own, the power subsystem is protected by autonomy algorithms in the C&DH processor. Power telemetry is monitored to detect faults in the battery chargers and in the bus voltage controllers (including digital and analog shunt controllers). Should a fault occur that requires the removal of one of the controllers (such as digital or analog shunt controller failure), the C&DH detects the fault and powers off the failed side. If the battery charger fails, the C&DH subsystem disables the faulty charger and enables the redundant side. Additional autonomy algorithms are used to protect the battery from high temperature and low voltage. Low voltage sensing (LVS) protects the battery from excessive discharge. Two maskable levels of detection are used and are set for bus voltages of 26 and 25 V. During launch, the LVS levels were masked to avoid tripping during the launch discharge. Response to an LVS trip includes load shedding and solar attitude correction.

PERFORMANCE ANALYSIS

Analysis of mission performance accounted for the effects of the predicted mission environment. Analyses of the NEAR power subsystem performance consisted primarily of solar array analyses over nominal mission operation and case-by-case analyses of significant mission events.

Data on radiation spectra were converted to 1-MeV electron equivalence and derated for solar distance over the mission. Estimates of the bulk panel temperature over the mission were provided as an input to the array model from a more detailed spacecraft-level thermal analysis in terms of solar energy input and panel absorptivity. The power analysis iteratively calculated panel efficiency, which provided consecutively more refined absorptivity and temperature estimates until the solution converged. A significant and unique problem to interplanetary spacecraft design is the effect of low intensity low temperature (LILT) operation. The term "LILT" is used fairly loosely in the sense that low illumination effects tend to degrade voltage, regardless of temperature. The NEAR program study included a review of industry efforts to quantify the phenomenon, as well as LILT testing of NEAR GaAs cell assemblies. The NEAR results were conservative and consistent with those of industry, revealing a correlation between cell beginning-of-life electrical performance and the degree of voltage reduction. Evaluating only cells with efficiencies greater than 18% under mission aphelion conditions yielded a 3.5% degradation in open-circuit voltage and 4.8% degradation in the voltage at maximum power. However, considering the inclusion of lower-performance cells in the flight distribution, a 3.9% open-circuit voltage degradation and 7.6% voltage at maximum power degradation were evident.

Available Power Analysis Results

Figure 5 illustrates the available power over the mission for maximum required solar offset. The bottom curve illustrates the required load power. The load power minimum represents the outer cruise mode in which load management is applied to conserve power margin. The load maximum is the observation mode at the asteroid.

Voltage Profiles

Figure 6 shows the profiles of open-circuit voltage (V_{oc}, upper curve) and maximum power voltage (V_{mp},

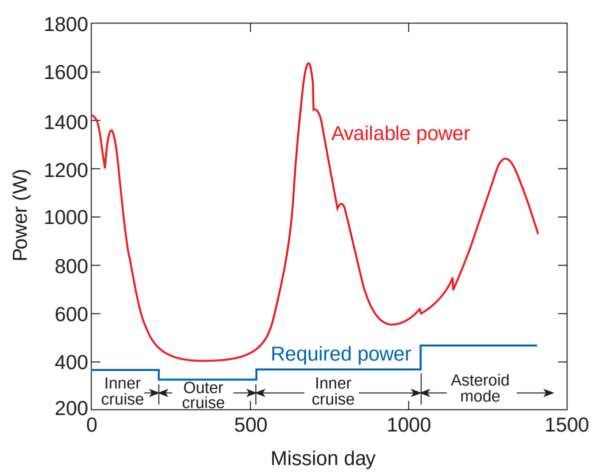

Figure 5. Available and required load power. Analysis results show sufficient available power to operate the spacecraft during each mission mode with minimum power generation during the 2.2 AU aphelion. Discontinuities in the available power curve are due to changes in spacecraft pointing.

lower curve) over the Sun-pointing mission. The straight horizontal line at 34.5 V shows that the array operating point is greater than the maximum power voltage several times over the mission. This reflects the series-parallel optimization to ensure sufficient parallel area to provide power margin at aphelion. The PSE is capable of stable operation in this range of the solar array *I–V* curve.

Burn Scenarios

Investigation of the burn scenarios at 2.0 AU and for Eros rendezvous show that load management can reduce the peak spacecraft loads to 432 W during the velocity adjustment. The deep-space burn has ample margin with 462 W of available power (including one assumed shunt failure). The rendezvous burn has a more favorable power margin with 575 W available before the burn (including shunt failure). However, the heating of the panel during the burn will cause a voltage depression on two panels. Conservative thermal analyses show that only circuits on the panel edge facing the thruster would be affected, and that 524 W of power is still available during the burn.

Mathilde Flyby Analysis Results

Analysis of power margin for the observation of the asteroid Mathilde showed that the spacecraft would have 286 W of available power during the 50° solar offset at 1.99 AU to power a peak load of 273 W. As with the burn scenarios, the power estimate assumed the failure of a single digital shunt element, although no such failure occurred before or during the Mathilde flyby. In addition, a conservative estimate of shadowing losses was applied to the estimate because of the large solar offset. Although the power analysis indicated that the battery would not be required even under worst-case situations of solar input and load requirement, analyses were done to predict battery performance in the unlikely event that a load transient initiated a discharge. The resulting decrease in bus voltage yielded a sustained discharge to a maximum depth of discharge of less than 12% for a 45-min observation.

FLIGHT DATA

Launch

Figure 7a shows the total solar array current during the first hour of the mission. When the fairing was

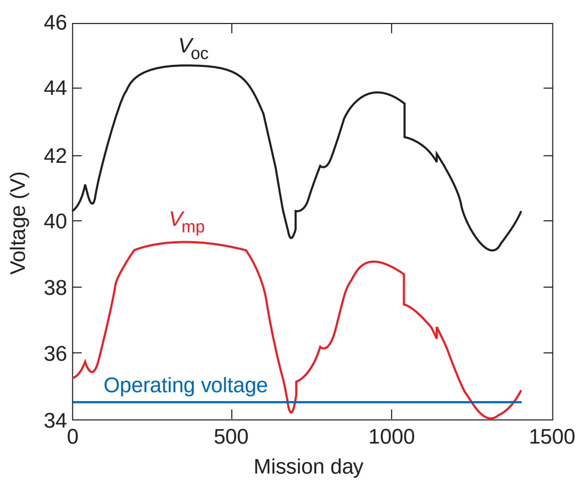

Figure 6. Solar array voltage over the mission. Voltage varies as a function of temperature, illumination, and radiation exposure. The series-parallel arrangement of the solar array, combined with the power system electronics design, optimizes power generation in this varied mission environment. (V_{oc} is open-circuit voltage; V_{mp} is maximum power voltage.)

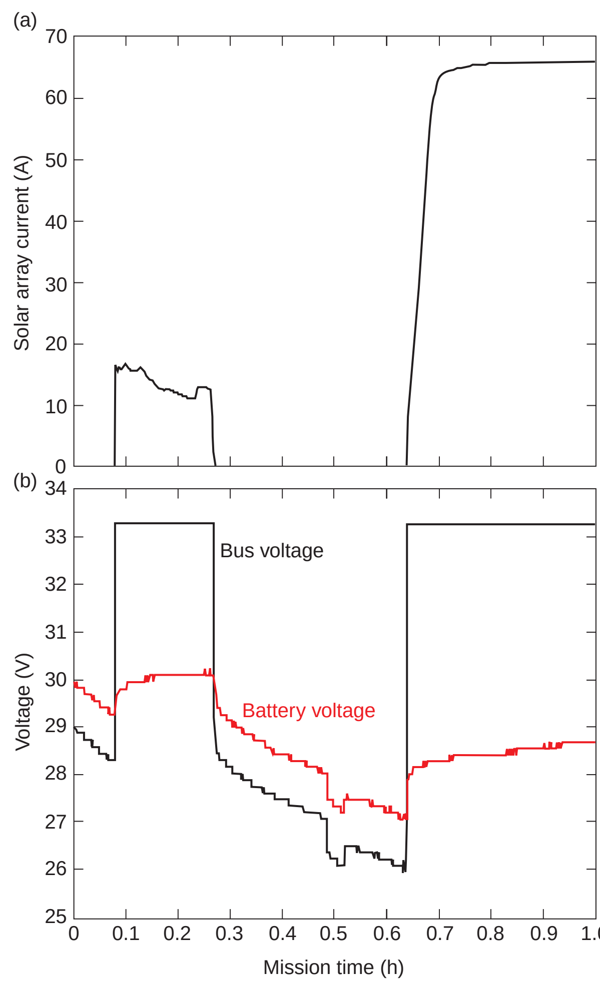

Figure 7. (a) Solar array current, and (b) bus and battery voltages, during the first hour of the mission. The spacecraft was released from the rocket fairing in sunlight, which generated enough solar array current to stop battery discharge before the panels were deployed. Discharge resumed when the spacecraft traversed into eclipse, during which time the panels were deployed. On exiting eclipse, a Sun-pointing attitude was achieved.

ejected 5 min after launch, power was being generated by the stowed panel until the spacecraft entered eclipse 11.5 min later. Panel temperature data (not shown) suggest that the majority of power was generated by the $+y$ panel, with possible minor contributions from $\pm x$ panels. Sufficient power was generated to stop battery discharge and bring the bus to regulation. Figure 7b shows the battery and bus voltages. Integration of the discharge current yields an approximate discharge depth of 15%.

Solar Array Performance

A plot of the predicted and actual total solar array current for the first year of the mission is shown in Fig. 8. Comparisons with results of array models were obtained using actual flight distances, Sun angles, temperatures, and digital shunt configurations. These parameters were input into the original array analysis software, which models the I–V response to radiation, temperature, illumination (including LILT), and voltage of the array circuits. To date, the flight array current has averaged 101 ± 1% of the modeled value.

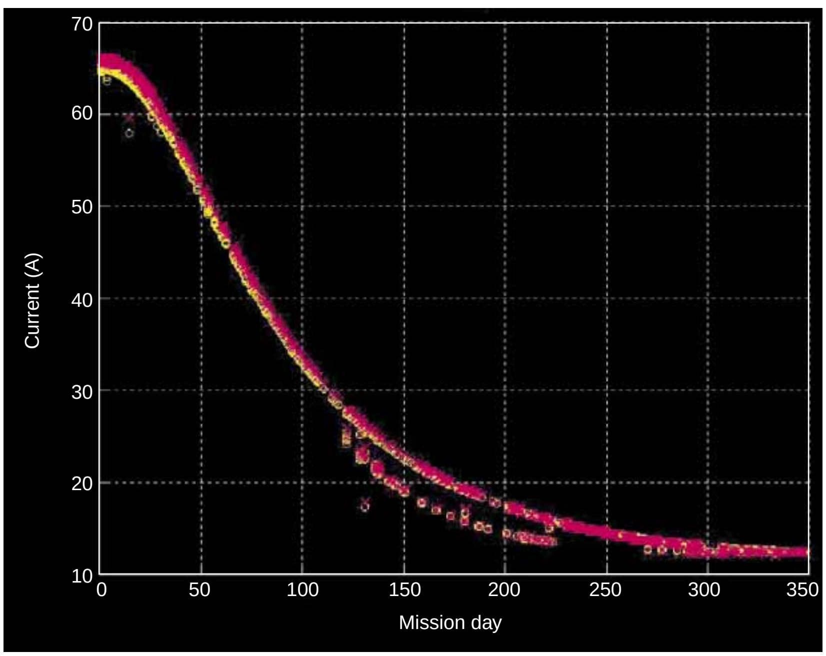

Figure 8. Predicted (yellow) and actual (magenta) total solar array current for the first year of the NEAR mission. Solar array output tracked modeled performance to within 1%. The total array current is a combination of current shunted by the digital shunts at 0 V and bus current (including loads and analog shunts) at 33.5 V.

On 22 July 1996, the NEAR solar array became the first photovoltaic primary power source beyond the orbit of Mars. In February 1997, the new photovoltaic solar distance record was set as NEAR flew through mission aphelion at 2.2 AU. Overall system performance has been nominal and consistent with predictions. The analog and digital shunts have been effectively maintaining a solid bus voltage of 33.28 V. Battery temperatures cycle between 0 and 5°C with about a 4.75-h period due to heater activity. The trickle charge has tapered from $C/75$ maximum current limit to a range of $C/80$ to $C/110$ because of the VT-5 charge limit. The mean voltage difference is 0.08 V between the top and bottom halves of the battery stack 180 days into the mission.

CONCLUSION

The power subsystem of the NEAR spacecraft is a simple, effective, robust design that provides power in an interplanetary environment. The direct energy transfer topology, GaAs solar array, and Super-NiCd battery were appropriately selected to provide a low-risk system over a highly variable range of operating conditions. Analysis of the overall mission, with emphasis on critical, atypical activities, shows sufficient power margin and fault tolerance to complete the NEAR mission. Flight data so far show satisfactory mission performance, which may be extrapolated to predict future mission success. The NEAR power subsystem represents a design that is a candidate for use as a baseline in future small-size, high-performance interplanetary missions.

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