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Designs and Technologies for Future Planetary Power Systems

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Planetary missions place unique demands on spacecraft systems and operations in terms of lifetime and autonomous operation. At the same time, the new faster, better, cheaper environment requires more technological innovation than ever before to enable us to continue to explore the planets with the same successes that we have enjoyed in the past. This article discusses new electric power system design and component technologies that provide the basis for planetary exploration in the late 1990s and far beyond, especially for small spacecraft. Power technologies for the New Millennium spacecraft series are presented. We discuss new concepts in power management and distribution technology, followed by an assessment of the status of photovoltaic and nuclear power source technologies, and we conclude with a discussion of advanced battery technologies for small spacecraft.

Introduction

FUTURE planetary mission planning is focused on spacecraft implementations having a dry mass in the range from a few kilograms to 200 kg. Power capability for this class of spacecraft will be about 100 W or less. Demand for increased payload on these new planetary explorers drives the allowance for the power system to a smaller fraction of the total spacecraft dry mass than has been achieved with previous technologies. This must be done while meeting the wide range of sometimes extreme environments encountered in planetary exploration. Requirements frequently include long lifetimes (up to 10 years or more) and the ability to operate under very cold or very hot conditions. Reactive planetary atmospheres may also be a factor. This article describes emerging technologies that enable miniaturization of future planetary spacecraft while maintaining a high level of science return under wide-ranging conditions and lifetimes.

Power Electronics

Historical mass performance of power electronics given as a percentage of total spacecraft (S/C) dry mass is presented in Fig. 1. It is evident that power electronics mass percentage increased toward 10% as S/C dry mass approached 200 kg with conventional packaging and older power system topologies. The various missions referred to in Fig. 1 are Cassini (CASS), Mars Observer (MO), Galileo (GLL), Viking (VO), Mars Global Surveyor (MGS), Mariner Mars (MM), Mariner Venus–Mercury (MVM), Mariner Venus (MV), Pluto Fast Flyby (PFF), Mars Pathfinder (MPF), and New Millennium.

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Mass and volume allocations for the next generation of planetary spacecraft demand power electronics having high-density packaging and high-efficiency power conversion for low load powers. A goal for future small missions is the achievement of a mass that is 2–4% of the total spacecraft dry mass. This goal drives electronics designs and technologies to mass and volume reductions of 65 and 80%, respectively.

Modular Power System Design

Power system components for a conventional planetary spacecraft power system are shown in Fig. 2.[1] In this block diagram, the grouping arrangement for discrete components has been made representative of an advanced hybridized topology. These component blocks include power generation and storage, power control, power management and distribution (PMAD), and pyrotechnic electronics.

Discrete circuit groupings are shown within each of these blocks. The power generation and storage grouping contains radioisotope thermoelectric generators (RTG) as one of the power sources for planetary travel beyond the Jupiter environment. Housed within the power control block is the bus regulator, shown as a shunt regulator assembly (SRA). Command (CMD) interface is performed by a bus interface unit (BIU) that is a discrete MIL-1553 bus input/output interface to the spacecraft command and data system. The PMAD block contains power distribution switches (PDS) and power converter units (PCU), as well as the other functional components.

Current packaging technology is based on discrete electronic parts mounted on planar printed circuit boards. A gate array is used for the command interface and the only hybridized circuit is the power switch.[2] Packaging with available surface mount technology will provide a small net mass and volume gain over conventional printed circuit board designs. Metrics for this form of power packaging include a power density of 0.02 W/

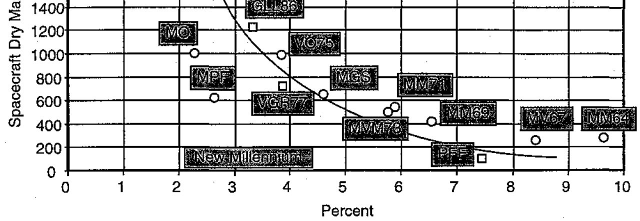

Fig. 1 Historical power electronics mass performance.

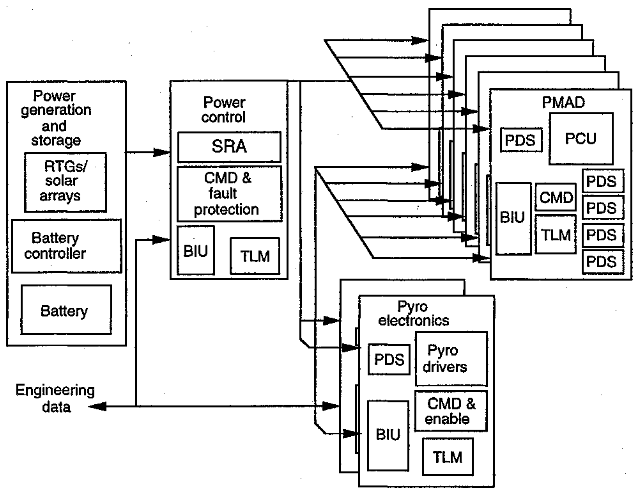

Fig. 2 Conventional power system circuit blocks.

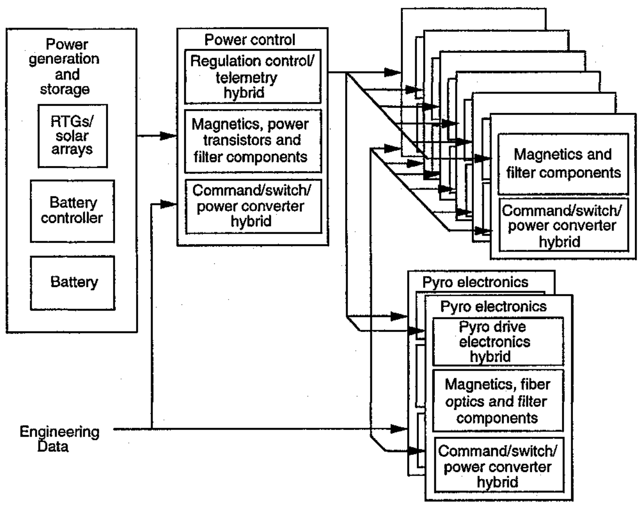

Fig. 3 Advanced hybrid functional building blocks.

cm^3 (0.34 W/in.^{3}) and a power-to-weight ratio on the order of 0.05 W/g.

A modular power system design based on the components of Fig. 2 is translated into the functional building blocks shown in Fig. 3. Each of the high-level blocks contains one or more of the hybrid function modules to be delivered by a technology development program. A modular approach permits a phased development program that incrementally builds capability while power system operational integrity is maintained. The core of this design is the PMAD block. Its basic elements, command and telemetry, power switching and isolated power conversion, are repeated in a number of the other blocks. The PMAD and pyroelectronics functions are designed to provide incremental growth with end-user complexity and size.

Hybridization Technology

Miniaturization of the power subsystem is planned through the extensive use of hybridization technology. A power hybrid has been implemented on the Cassini spacecraft with the development of a solid-state power switch (SSPS).[2] This hybrid switch was built to hybrid class K standards and 450 of these devices have been delivered to the spacecraft. Hybridization of a high-performance power conversion module is in process. A second-generation prototype hybrid converter is being fabricated now. Building upon this experience base, hybridization will be expanded to include a second-generation SSPS, a high-performance power converter module, and a field programmable gate array (FPGA) for command and telemetry interface.

The plan is to hybridize the most commonly used power functions first. Command interface, power switching, and power conversion are key functions that comprise most of the blocks shown in Fig. 3. After the key components are implemented, attention will be turned to power regulation and control circuitry, pyro-drive electronics, and battery control. These functions are more dependent on the particular spacecraft configuration and are therefore less likely to be a standardized set of hybrid modules.

Hybridization of the power electronics by function will allow for reconfiguration of the basic building blocks to accommodate different mission requirements. A functional approach permits a phased technology development that incrementally builds the technology base and time-phases delivery of increasingly complex products to a technology demonstration platform.

Core Hybrid PMAD Components

A command and telemetry (TLM) interface is based on the flexibility of a FPGA. Application of this concept allows the power system interface to the host spacecraft to be hardware independent. The plan is to have an adaptable software capability to reprogram the power command interface to accept the command protocol of the user spacecraft. PMAD telemetry interfaces to the spacecraft command and data subsystem are also controlled by the FPGA.

Power switching is accomplished by a second-generation SSPS. Lessons learned from the Cassini SSPS implementation and hybrid device build will be incorporated into the development process. The SSPS is used to connect and disconnect electrical loads while providing spacecraft power bus fault protection. Salient features of the SSPS include elimination of traditional relay/fuse spacecraft architecture, inrush current circuitry in user loads, and assurance of a hard power bus by rapid removal of disruptive loads.

Functional performance of the next-generation SSPS includes on/off command capability with programmable overcurrent trip and load current telemetry. A load fault current limit will ensure a 4-A limit for complete shorts on the output. Isolated switching elements will enable series/parallel connection of the SSPS. Load capability ratings for this switch are 5- to 30-Vdc switching and 0- to 2-A continuous load current. High-density design using mixed-signal application-specific integrated circuits (ASICs) and hybridization will enable four switches to be housed in one package.

High-performance power conversion is performed using a hybrid synchronous rectification power module. This module provides isolated low-voltage outputs for the spacecraft user loads from 5–50 W at an input voltage of 10–50 V. Development of this module has progressed to a second-generation hybrid prototype that is being fabricated under a contract to a hybrid foundry. Packaged within the hybrid are the entire converter power and signal switching circuits. Efficiency performance of the first prototype units peaks at 92% for a 5-Vdc output and 88% for a 3.3-Vdc output. These results are 7–10% better than typical custom power converters.

Flexibility in application and design are key attributes of this package. It can be used to derive isolated outputs from 2.5 to 12 Vdc with load currents to 10 A for load powers up to 50 W. Design of a complete dc-to-dc converter requires input and output filters, the isolation transformer, and compensation network. Only a knowledge of linear circuits and systems is required by the circuit designer to ensure stability of this power converter.

Power Control and Pyro-Hybrid Components

Power control development will utilize the PMAD elements described earlier. A command interface and solid-state power switching hybrids will be configured to provide control for the unique spacecraft power elements that are mission specific. Power for this function will be provided by a hybrid power converter module. Power bus regulation electronics will also be a hybridized module that will have digital signal processor control to adapt the regulation scheme to the particular spacecraft power bus configuration.

Pyroelectronics will also adapt the command and switching components from the PMAD building blocks to be used for switching the pyrotechnic devices. Again, this block will be powered by the basic hybrid converter module. Pyro-drive electronics will be another unique hybrid module.

Power Sources

Photovoltaic Power Sources

In the past, interplanetary missions accounted for a small fraction of space solar array usage. During the past 15 years, NASA has launched only two interplanetary missions powered by photovoltaics. These were Magellan, a mission to Venus, and Mars Observer (MO), the ill-fated mission to the "red planet." The recent emphasis on smaller, lower-cost missions is expected to dramatically increase the number of photovoltaic-powered (PV-powered) interplanetary spacecraft. However, even this increase will represent a relatively small quantity of annual space power, possibly on the order of 1–10 kW. Consequently, interplanetary photovoltaic power systems have been, and are expected to be, based on technology developed for Earth orbiting systems. Any special requirements are expected to be satisfied within the scope of modifications to existing systems. One possible exception is the development of solar cells for low intensity–low temperature (LILT) applications, which will be discussed later.

Although Earth-orbiting-based technology must be utilized, the requirements for interplanetary solar arrays can differ significantly from that of Earth orbiting systems. Temperatures may be much higher, such as those for near-sun missions, or much lower, such as missions to Mars and beyond. In addition, the distance to interplanetary targets often means a long cruise period of uninterrupted illumination, followed by encounters leading to frequent eclipses. A sampling of near-term missions and mission studies shows a mission to Mars (Pathfinder, Global Surveyor, and Mars '98), solar-orbiting Shuttle Infrared Telescope Facility (SIRTF) and near-sun (solar probe) missions, and asteroid and cometary encounters (near-earth asteroid rendezvous, or NEAR), a mix providing for a wide range of environmental conditions.

The solar array critical design element is the solar cell. A summary of cell performance and relative costs is presented in Table 1. For many years the only choice was the silicon cell. Showing small, but steadily increasing performance over the years, this device has powered spacecraft from the first satellites to the recent Magellan mission and will provide a portion of the Mars Global Surveyor power. Rugged, inexpensive, and well characterized, silicon has only recently received serious competition from the higher efficiency, and more costly GaAs cell. Because of the latter's higher radiation resistance, higher efficiency, and continued cost reduction, a number of new missions have selected this cell. Although low array cost is not always achieved with GaAs, the higher efficiency allows for smaller arrays. For relatively massive, rigid honeycomb structures, this can lead to lower overall array mass along with smaller stowage volumes and reduced deployment complexity (fewer panels). The higher voltage of GaAs compared to that of silicon also brings advantages for operating in high-temperature environments (near sun). At the same time, the GaAs efficiency advantage is reduced for outbound missions as the silicon cell efficiency increases rapidly. This efficiency increase is because of the greater voltage temperature coefficient for silicon (about twice that of GaAs). At present, commercial silicon solar cells suffer from LILT degradation that can significantly reduce cell performance at 2.5-AU solar distance and beyond. However, recent work in the U.S.[3] and in Europe[4] have identified approaches that will prevent or mitigate this LILT degradation with the result that for cells used at 3 AU or greater, laboratory measurements demonstrate that silicon efficiencies are capable of exceeding those of GaAs. These cells are not in production at this time, and so power at those solar distances will require the more expensive GaAs cell.

New cells presently under development and appearing capable of near-term implementation include advanced silicon (up to 20% higher efficiency than the 15% efficient conventional silicon cell) and multijunction solar cells based on the $GaInP_2/GaAs/Ge$ technology.[5] These latter devices may achieve efficiencies of up to 22% at standard test conditions [air mass 0 (AM0) and 28°C], nearly 20% higher than GaAs/Ge. However, the behavior of these cells in the interplanetary conditions is yet to be established. Initial characterizations will most likely address Earth orbiting applications. The multijunction cell will represent a significant characterization challenge since the typical solar simulators used for laboratory measurements may not be capable of achieving measurement accuracy equal to those obtained on single-junction cells. For these cells it will be necessary to accurately match current generation in both top and bottom cells, necessitating a much more accurate solar simulation, especially in the infrared (IR) region.

At the array level, which involves materials and structures, recent years have seen a number of new developments applicable to a wide range of power levels (from approximately 500 W to 10 kW at Earth). These advanced designs include both flexible and rigid substrates and planar and concentrator configurations. The maturities vary considerably, from the space-qualified advanced photovoltaic solar array (APSA) design[6] planned to be used on the Earth Observing System (EOS), to concepts still in the breadboard phase. In general, these designs offer options for lowering array mass, increasing strength, and/or reducing cost. The selection of any particular design will depend on a number of spacecraft constraints, such as power level, stowage volume, and environmental conditions. In practice, no single design will apply equally well to all missions. In addition, the selected array is often procured as part of an overall power system, further reducing options. In general, however, it is clear that existing advanced array concepts have the potential to reduce array mass by a factor of 2 compared to technology used in past interplanetary missions. Acceptance of these technologies will require the demonstration of reliable spaceflight capabilities. Concentrating systems most likely will have an additional hurdle to cross before being fully accepted. These designs introduce a number

Table 1 Photovoltaic cell performance/cost summary

Cell type	1 AU efficiency, %	4 AU efficiency,[a] %	Status	Relative cell cost	Relative array cost	Array power, W @ 1 AU-1M^2	Array power, W @ 4 AU-1M^2
Silicon	14, 15	16–22	Production	1	1	130–140	7–12
GaAs	18–19	19	Production	5	1.6	160–170	10
GaAs/Ge dual	22[b]	22	Pilot	7	1.9	190–210	12[b]
Advanced silicon	20[b]	20–24[b]	Laboratory near pilot	3	1.5	190–200	12, 13[b]
GaAs/GaSb	23[b]	20–24[b]	Laboratory	30	7	195–210	12, 13[b]
InP	17, 18[b]	17, 18[b]	Laboratory	40	9	150–160	10[b]

[a]For temperatures in the range from −100 to −80°C. [b]Estimated.

of new concerns. For example, array ground testing will be more complex with concentrator arrays because of the lack of large-area collimated AM0 light simulators. The need for low array-stowage volume may require additional deployment for the concentrator elements. Benefits of concentrators include improved radiation protection and potentially low cost, but the former is not always a critical mission requirement and the latter has not been demonstrated on an actual spacecraft.[7] That leaves a very real concern for the potentially catastrophic loss of power that can occur for off-sun pointing. This is being addressed at present through a number of design safety measures; however, it remains to be seen how convincing these can be. Many of these questions will be answered with a first concentrator array flight (the solar concentrator array with refractive linear element, or SCARLET) scheduled for New Millennium DS-1 mission.

Regardless of the cell or array technology that will be used on future interplanetary missions, the unique requirements of these missions can be effectively met only by obtaining knowledge of cell and array behavior under the appropriate operating and environmental conditions. At present, the new Mars missions have identified a number of areas (e.g., solar spectrum on the surface, insolation levels, dust environment, etc.) where cell performance data are incomplete and additional margins must be used to ensure success.

Radioisotope Power Sources

Missions needing nuclear power sources represent environments ranging from hot, gaseous planetary atmospheres, e.g., Venus, to the vacuum of space at great distances from the sun. In fact, 8 of the past 11 U.S. interplanetary launches have been powered by radioisotope-based power sources that have an extraordinary life and reliability record. However, since future thrusts now point to a preponderance of small or moderate missions, the use of nuclear power sources will be very limited. With budgets being capped at as little as $100,000,000, the future use of radioisotope-based power sources may depend upon 1) successful development of advanced converters that operate much more efficiently or 2) the use of radioisotope heater units (RHU) to power milliwatt-sized power sources. We will discuss these possibilities here. Finally, as budgets and launch vehicle options continue to be limiting, reactor-based power sources will remain unaffordable for today's planetary mission planners.

The general purpose heat source radioisotope thermoelectric generators (GPHS–RTG)[8] now flying on the Galileo mission to Jupiter and the Ulysses mission to the solar poles, and planned for the Cassini mission to Saturn, convert thermal energy to electricity at about 6.7% efficiency. They utilize 238 PuO_2-based heat sources, producing power via thermoelectric elements called unicouples operating at 1273 K. The half-life of plutonium is 87 years.

In recent years, the focus on radioisotope power technologies has been on increasing the efficiency of the converter, thereby reducing the fuel requirements for the same power level, and possibly reducing the total mass of the power source.[9,10] Four converter technologies have received considerable attention: 1) advanced thermoelectric materials, 2) alkali metal thermal-to-electric conversion (AMTEC), 3) thermophotovoltaics (TPV), and 4) Stirling cycle dynamic conversion. Work on advanced thermoelectric materials, in particular, improved silicon–germanium, have recently led to increases in performance by up to 20%. Other new thermoelectric materials may offer even greater advances. However, no new thermoelectric couple has yet demonstrated the potential for improvements that appear feasible with the other three technologies, i.e., 20% efficiency or greater, although research on new thermoelectric materials continues.

Table 2 shows a comparison of the AMTEC, TPV, and Stirling-based radioisotope power source concepts that were developed for the Pluto Express mission study along with a GPHS–RTG redesigned to meet Pluto mission requirements. The new technologies provide substantial mass and fuel savings in comparison with the conventional GPHS–RTG concept.

Of the three new technologies in Table 2, the free-piston Stirling engine is the most mature in terms of demonstrated lifetime and efficiency.[9] Its deficiencies relate to the possible need to compensate for engine vibrations, concerns about the reliability of devices with moving mechanical parts, and the level of redundancy achievable with multiple engines. In addition to the Pluto Express design, a dual-cycle Stirling engine concept design has been proposed for use on the surface of Venus. This novel concept would provide both power and electronics cooling for such a probe, utilizing the Stirling engine simultaneously in the power and cooling modes.

AMTEC, a thermally regenerative electrochemical cell, is the next most mature technology, yet feasibility demonstrations are still required. While it offers the advantage of no moving mechanical parts and life tests of several thousand hours have been successfully carried out, additional data are needed to provide complete confidence that 10-year (or greater) missions can be achieved.[11] Also, AMTEC system designs must provide two-phase fluid management and protect a ceramic solid electrolyte from the vibrations and shocks that the spacecraft experiences during launch and mission operations. A technology flight experiment is now being planned to resolve such issues by operating different AMTEC cell designs in space.

TPV is based on the response of photovoltaic cells to IR radiation from a high-temperature heat source; in this case, the radioisotope heat source. Current concept designs usually assume gallium–antimonide cells for TPV systems.[12] TPV also offers the advantage of a fully static system with the potential for very long life, especially since the power producing cells are at the heat rejection temperature (near room temperature). Its development would take advantage of the extensive photovoltaic systems capabilities in industry and government. However, it, too, faces major feasibility demonstrations before mission planners can confidently baseline a radioisotope-based TPV power source. For example, prototypical optical cavities (converter modules) have yet to be built and tested to verify the high efficiencies envisioned for TPV systems. Also, a narrow bandpass filter between heat source and TPV cell will likely require development to achieve the highest projected

Table 2 Advanced radioisotope power source performance comparison[a]

Converter	No. of GPHSs	Mass, kg	Power, W, EOM, 10 yr	Efficiency, %, EOM, 10 yr
Baseline RTG	6	17.8	74	6
AMTEC	2	6.1	85	22
TPV	2	7.2	94	24
Stirling engines	2	11.3	85	22

[a]Assumptions:
1. All designs are conceptual and specifically aimed at the Pluto Express concept.
2. For AMTEC and Stirling Engines, power decays with the decay of the thermal source.
3. For TPV, power decays with the decay of the thermal source plus 1%/yr for radiation damage to the PV cells.
4. Power source mass only; PMAD not included.

efficiencies. The purpose of the filter is to tune the incident radiation to the bandgap of the TPV cell. An alternative approach would be to coat the heat source with a selective wavelength emitter material (rare Earth oxides are under development). Finally, the need to cool the TPV cells to near room temperature results in the need for a large heat-rejection radiator. While advanced-material and heat-pipe technologies can be used to minimize the mass of the radiator, the total area of the radiator for the Pluto Express design exceeds the AMTEC and Stirling radiator areas by more than five times. This may eventually lead to system integration and packaging difficulties.

A substantially different approach to radioisotope power source development is represented by the Powerstick.[13] Powerstick is a concept that utilizes the RHU that has been extensively used for thermal control of electronics in planetary spacecraft. In the Powerstick concept, one thermal watt of PuO_2 fuel would heat a bismuth–telluride thermopile to produce approximately 25 mW of continuous power. The power produced would charge lithium battery cells for subsequent discharge in a burst mode (28 W-h/month). Applications for the Powerstick generally involve small planetary probes that require extremely low power levels for long periods of time (sleep mode), followed by more active periods when data are gathered and transmitted. However, the Powerstick is at the earliest state of development of all the radioisotope sources discussed here. A concept design has been formulated and a preprototype device is being fabricated for initial testing.

In summary, the availability of radioisotope power sources for future missions will depend upon many complex factors. While the conventional GPHS–RTG has proven to be an extraordinarily reliable power source, its future cost and availability are unpredictable. The development of alternative conversion technologies that conserve fuel or much smaller sources such as the Powerstick may be essential to ensure the future ability to conduct extended missions to the outer planets or in harsh planetary environments.

Advanced Batteries

Future planetary missions require lightweight and compact batteries with long cycle life capability. Some of the missions require operation of the batteries at extremely low temperatures. State-of-the-art (SOA) silver–zinc (Ag–Zn), nickel–cadmium (Ni–Cd), and nickel–hydrogen (Ni–H_2) batteries are too heavy and bulky for many of the future planetary missions. In some cases they do not meet life and environmental requirements. A number of advanced primary and secondary battery systems are presently under development at various organizations for commercial and space applications. Among the primary batteries, lithium batteries are most attractive for planetary and space applications as they can provide three to four times the savings in weight and volume as compared to the other primary batteries. Primary batteries are used mainly in probes, penetrators, etc., as primary power sources or for meeting peak load demands. Secondary batteries that are of interest for planetary missions include advanced nickel (two cell common pressure vessel Ni–H_2, single-pressure vessel Ni–H_2 battery, Ni–MH) and lithium (Li–TiS_2, Li–ion, lithium polymer) batteries. A comparison of the specific energy and energy density of SOA and advanced secondary batteries is given in Fig. 4. Significant mass and cost advantages are projected with the use of these advanced batteries. A brief discussion of the advanced batteries and their planetary applications is given next.

Primary Lithium Batteries

Lithium primary batteries have higher specific energy and energy density than any currently available primary cells. Other desirable features of these cells are higher operating voltage, excellent voltage stability over 95% of the discharge, operating capability over a wide operating temperature range, and exceptionally long active storage life. In view of these features, NASA is considering these cells/batteries for several space missions, such as planetary probes, penetrators, astronaut equipment, and launch vehicles. A number of lithium primary batteries, such as Li–CF_x, Li–MnO_2, Li–I_2, Li–SO_2, and Li–$SOCl_2$, were developed for commercial and aerospace applications. Among these, Li–SO_2 and Li–$SOCl_2$ batteries are most attractive for planetary applications in view of their higher rate capability and improved low-temperature performance capability. The Li–SO_2 system was used in the Galileo probe and was selected for use in the Cassini probe. The key requirements for these missions are high specific energy (>250 W-h/kg) and six to eight years of active storage life. Li–$SOCl_2$ batteries have 20–30% higher specific energy compared to Li–SO_2 batteries. In view of this, these batteries were selected for use in Jet Propulsion Laboratory's (JPL) Mars Pathfinder Rover. The main function of the Li–$SOCl_2$ battery in this mission is to meet peak power loads.

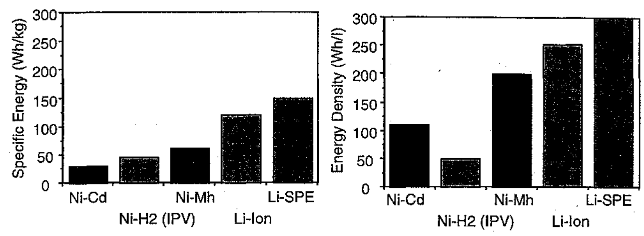

Fig. 4 Performance characteristics of various cells/batteries. (Li-SPE values are projected.)

Advanced Nickel Battery Systems

Advanced Ni batteries that are presently under development for small spacecraft applications are 1) two-cell common-pressure vessel (CPV) Ni–H_2, 2) single-pressure vessel (SPV) 22-cell Ni–H_2 battery, and 3) Ni–MH. Some of the important performance characteristics of small Ni–H_2 batteries are given in Table 3. These advanced Ni–H_2 batteries provide 10–30% higher specific energy and energy density compared to the state-of-the-art individual pressure vessel Ni–H_2 batteries. These advanced Ni batteries are presently being considered by several aerospace organizations for near-term space and planetary missions in view of their relative maturity compared to the rechargeable lithium batteries. Two-cell CPV Ni–H_2 was used on miniature seeker technology integration-2 (MSTI-2), Technical University of Berlin Satellite (TUBSAT), and advanced planetary explorer (APEX).[14] Lockheed Martin Corporation has selected the two-cell CPV for the Mars Global Surveyor mission based on preliminary ground test data. The Naval Research Laboratory (NRL) has recently flown a single-pressure vessel (called a "common pressure vessel" by NRL) 22-cell Ni–H_2 battery on the Clementine mission that included (low Earth orbit) (LEO) cycles and mapping of the moon.[15] NASA is also considering this type of battery for several small spacecraft (Discovery Missions). Motorola is also considering this system for the Iridium program. Development of nickel–metal–hydride batteries for space applications is presently in progress. Small-capacity prismatic cells were constructed, and testing of the cells is in progress.

Secondary Lithium Batteries

Ambient temperature secondary lithium batteries[16–18] have several intrinsic and potential advantages, including higher energy density, longer active shelf life, and lower self-discharge over conventional Ni–Cd and Ni–H_2 batteries. Successful development of these batteries will yield large payoffs such as a two- to three-fold increase in energy storage capability and a longer active shelf life of two to four years over Ni–Cd. These batteries are very attractive for missions that are very critical in weight and volume, and they are likely to be useful at very low temperatures (such as on the surface of Mars). JPL is

Table 3 Small (<30 A-h) $Ni-H_2$ battery characteristics

Property	IPV[a] $Ni-H_2$, 2.5 in.	2-cell CPV, 3.5 in.	22-cell SPV
Voltage, Vdc	20	20	28
Capacity, A-h	20–30	10–25	10.5
Cycle life	~65,000 (LEO), 30% DOD	>15,000 (LEO), 15% DOD	9,000 (LEO)
Specific energy, W-h/kg	30–40	30.5	47
Energy density, W-h/L	16	18	50

[a]Individual pressure vessel.

Table 4 Status of rechargeable Li technology at JPL

Component/property	$Li-TiS_2$	Li–Ion	Li–polymer
Anode	Li	Li_xC	Li_xC
Cathode	TiS_2	$LiCoO_2$	$LiCoO_2$
Electrolyte	$LiAsF_6$ EC+2–MeTHF	$LiPF_6$ EC+DMC+DEC	$LiAsF_6$ PAN+EC+3–MeS
Voltage, V	2.1	3.8	3.8
Capacity, A-h	1–3	1–3	<0.2
Cycle life	1000 (50% DOD)	1000 (100% DOD)[a]	>100 (100% DOD)
Operating temperature, °C	−20 to 60	−20 to 60	RT–60
Specific energy, W-h/kg	132	125	150[a]
Energy density, W-h/l[b]	260	240	350[a]

[a]Projected; 200 cycles demonstrated to date with in-house cells. [b]Data are representative of small cells and large batteries.

considering these batteries for future small planetary spacecraft. Such spacecraft may require batteries that can provide a specific energy of 100 W-h/kg, an energy density of 250 W-h/l, and 500–1000 cycles. These batteries may also be attractive for rovers, astronaut equipment, and GEO spacecraft.

Three types of rechargeable lithium batteries 1) $Li-TiS_2$, 2) Li–ion, and 3) Li–polymer, are presently under development (Table 4). The $Li-TiS_2$ system is considered suitable for planetary missions that require high specific energy (>130 W-h/kg), limited cycle life, and small-capacity cells. The Li–ion system is suitable for missions requiring long cycle life and large-capacity cells. Li–polymer batteries are projected to provide a specific energy >150 W-h/kg and can be used in a variety of configurations. Small capacity $Li-TiS_2$ cells capable of providing a specific energy >130 W-h/kg and 1000 cycles at 50% depth of discharge (DOD) have been developed. Lithium–ion cells that employ graphite as the anode and $LiCoO_2$ as the cathode are under development. Experimental cells have completed 500 cycles at 100% DOD. These cells are projected to have a specific energy of 85–100 W-h/kg. There are plans to scale up lithium–ion cell technology to the 10- to 20-A-h cell level for future spacecraft applications. Lithium polymer cells are in early stages of development. Small-capacity lithium polymer cells were fabricated and tested for polymeric electrolyte assessment. State-of-the-art cells were found to provide >100 cycles at 100% DOD.

Concluding Remarks

The rapid transition to much smaller, less expensive, and yet capable planetary science spacecraft is a challenge to the power-technology community. However, recent and ongoing progress provides optimism that electric-power technologies will meet the new requirements. In addition to the component technologies discussed here, other strategies are being investigated to meet future needs. For example, there are studies of the advantages of a combined power and telecommunications system using a deployable concentrator/antenna that would enable a low-power photovoltaic power source to be used at greater distances from the sun while meeting high science telemetry data rates. Such a concept is a long way from being proved feasible, however, it, as well as other innovative concepts, deserve attention as the science and programmatic requirements continue to evolve.

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