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Designs and Technologies for Future Planetary Power Systems

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Abstract

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 paper discusses **new** electric power system design and component technologies that provide the basis for planetary exploration in the 1990's and far beyond, especially for small spacecraft. 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 Watts 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 tong **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 paper describes emerging technologies that will 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 dry mass is presented in Figure 1. It is evident that power electronics mass percentage increased toward 10O/. as spacecraft dry mass approached 200 kg with conventional packaging and older power system topologies.

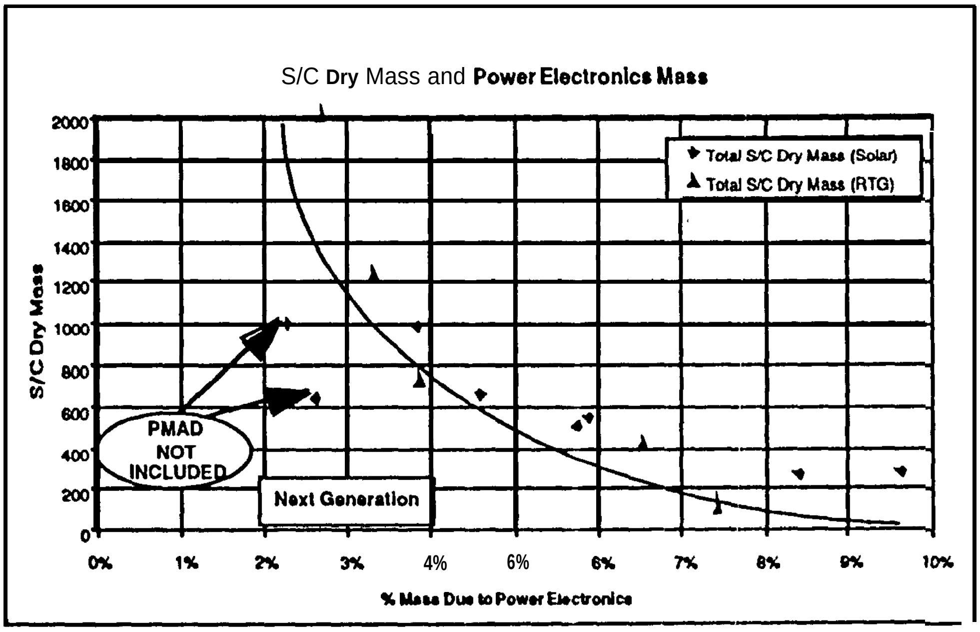

Figure 1. Historical Power Electronics Mass Performance

Mass and volume allocations for the next generation of planetary spacecraft will demand power electronics having high density packaging and high efficiency power conversion for krw load powers. A goal for future small missions is the achievement of a mass that is 2 to 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 Figure **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 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 interlace 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 indude a power density of 0.02 **W/cm^3** (0.34 **W/in^3**) and a power to weight ratio on the order of 0.05 Wig.

A modular power system design based on the **components** of Figure 2 is translated into the functional building blocks shown in Figure 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 Pyro Electronics functions are designed to provide incremental growth with end user complexity and size.

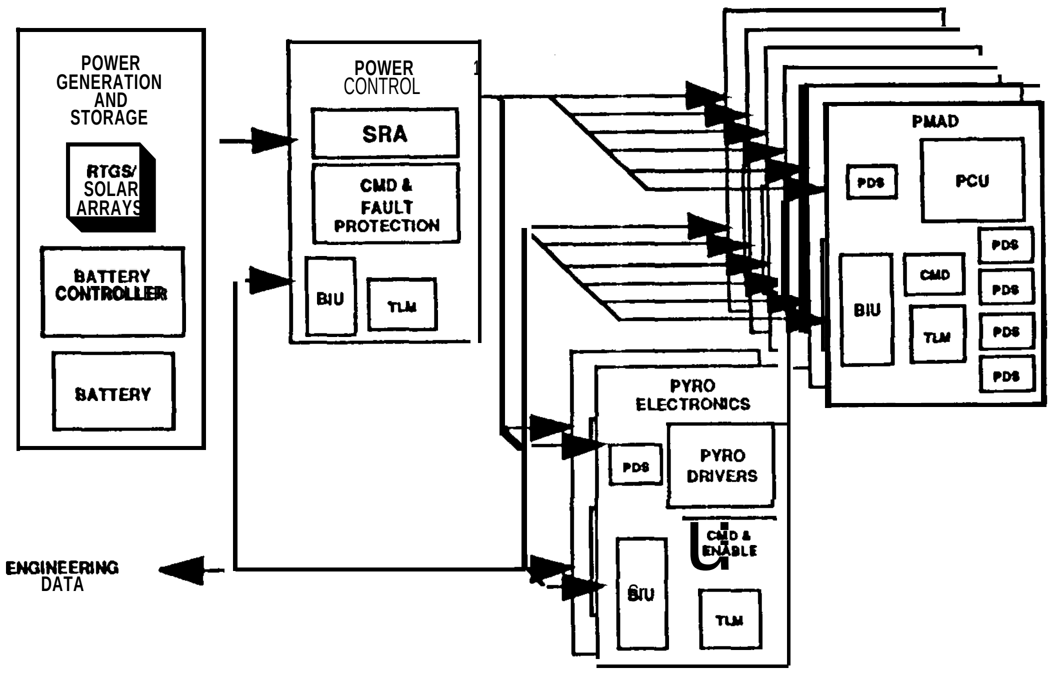

Figure 2. Conventional Power System circuit Blocks

Hybridization Technology

Miniaturization of the power subsystem is planned through the extensive use of hybridization technology. A power hybrid was introduced 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 flight stores. 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 Figure 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 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

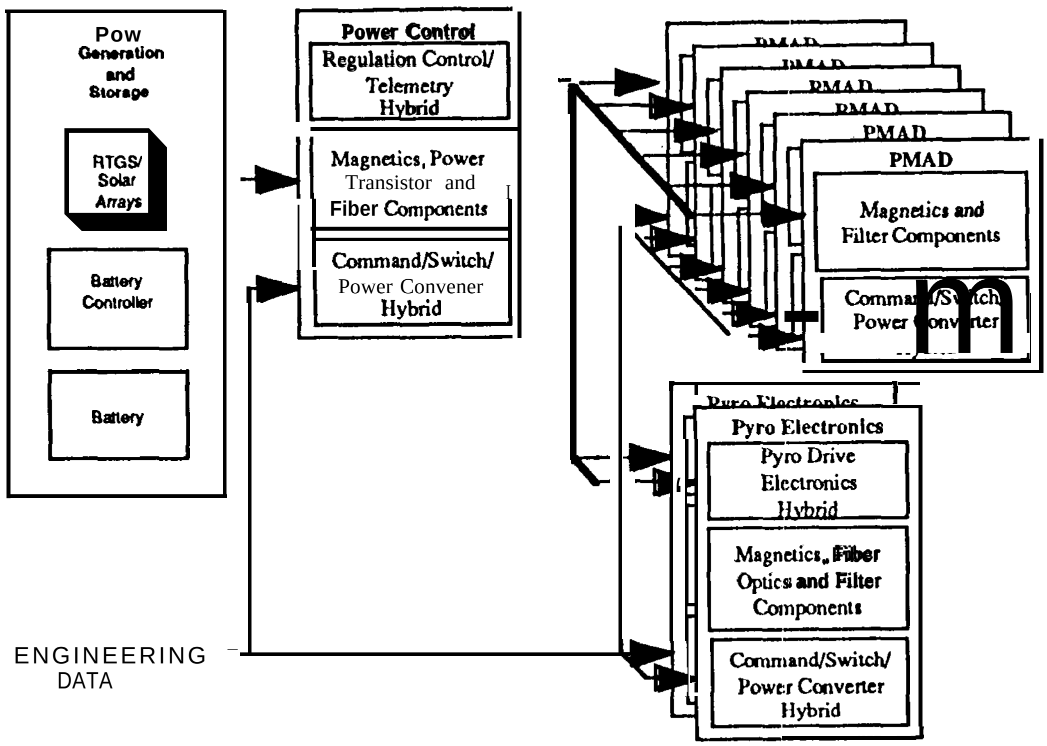

Figure 3. Advanced Hybrid Functional Building Blocks

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 over-current trip and load current telemetry. A load fault current limit will ensure a 4 Ampere 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 Vott dc switching and O to 2 Ampere continuous load current. High density design using mixed signal ASIC and hybridization will enable 4 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. 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 some 7 to 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 Amperes for load powers up to 50 Watts. Design of a complete de-to-de 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 above. 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.

Pyro electronics 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, the ill-fated mission to the “Red Planet’. The recent emphasis on smaller, lower cost missions is expected to dramatically increase the number of 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 LILT (low intensity - low temperature) 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 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 missions to Mars (Pathfinder, Global Surveyor, and Mars ‘98), solar orbiting (SIRTF), near sun (solar probe), and asteroid and cometary encounters (Near Earth Asteroid Rendezvous, NEAR), a mix providing for a wide range of environmental conditions,

The critical element in the solar array design is the solar cell. A summary of cell performance and relative costs are 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 higher cost GsAs cell. Due to the latter’s higher radiation resistance, higher efficiency, and continued cost reduction, a number of new missions have selected this cell. Although array low cost is not always achieved with GsAs, 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 GsAs compared to 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. At present, 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 US[3] and in Europe[4] has identified approaches that will prevent or mitigate this LILT degradation with the result that for cells used at 3 AU or greater, silicon efficiencies are capable of exceeding those of GaAs. These cells are not in production at this time so at present the more power at those solar distances. expensive GaAs cell is the only device suitable for PV power at those solar distances.

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 multi-function solar cells based on the GaAs/Ge technology.[5] These later devices may achieve efficiencies of up to 22% at standard test conditions (air mass O and 28°C), nearly 20% higher than GaAs/Ge. However, the behavior of these cells in the interplanetary conditions are yet to be established. Initial characterizations will most likely address Earth orbiting applications. The multi-junction cell will represent a significant characterization challenge since the typical solar simulators used for laboratory measurements may not be capable of achieving measurement accuracies 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 IR region.

Table 1. Photovoltaic Cell Performance/Cost Summary

	1 AU Eff.	4 AU Eff	Status	Rel. Cell cost	Rel. Array cost	Array Pwr. (W) @ 1AU-1 M²	Array Pwr. (w) @ 4AU-1 U²
Silicon	1 4-15%	16-22%	Production	1	1	130-140	7-12
GaAs	1 8-19%	18%	Production	5	1.6	160-170	10
GaAs/Ge Dual	22% Est.	219'0 Est.	Lab. Near Pilot	7	1.9	190-200	12 (?)
Advanced Silicon	20% Est.	20-24% (?)	Lab. Near Pilot	3	1.5	190-200	12-13 (?)
GaAs/GaSb	23% Est.	20-24% (?)	Lab.	30	7	195-210	12-13 (?)
InP	17-1 8% Est.	17-1 8% (?)	Lab.	40	9	150-160	10 (?)

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 500W 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 APSA design[6] used on EOS to concepts still in the breadboard phase. In general, these designs offer options to 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 two compared to technology used in past interplanetary missions. Acceptance of these technologies will require demonstration of reliable space flight capabilities, Concentrating systems most likely will have" an additional hurdle to cross before being fully accepted. Benefits of concentrators include improved radiation protection and potentially low cost, but the former is not always significant and the latter has not been demonstrated on an actual spacecraft. 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 a design "safety" measures, however, it remains to be seen how convincing these can be.

Regardless of the cell or array technology that will be used on future interplanetary missions, the unique requirements of these missions can only be effectively met 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 where cell *performance* data is incomplete and additional margins must be used to ensure success.

Radioisotope Power Sources

Missions needing nuclear power sources represent a range of environments ranging from hot, gaseous planetary atmospheres, e.g., Venus, to the vacuum of space at great distances from the sun. In fact, eight of the past eleven 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 $100M, 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 6.4%. They utilize PuO_2-based heat sources operating at over 1200K, producing power via thermoelectric uncouples. The half-life of plutonium is 87 years. However, the costs of acquiring and processing the fuel, and assembling the PuO_2-based GPHS modules may become prohibitively expensive as the ability of missions to bear the costs diminishes. Also, availability of the fuel modules may become problematic as production at nuclear facilities declines,

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: advanced thermoelectric materials, alkali metal thermal-to-electric conversion (AMTEC), thermophotovoltaics (TPV), and 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 Fast Flyby 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.

Table 2. Advanced Radioisotope Power Source Performance Comparison

Converter	# GPHSs	Mass (kg)	Power (W) EOM, 10 y	Efficiency EOM, 10 y
Baseline RTG	6	17.8	74	6%
AMTEC	2	6.1	85	22%
TPV	2	7.2	94	24%
Stirling	2	11.3	85	22%

Assumptions:

1. All designs are conceptual and specifically aimed at the Pluto spacecraft concept.
2. For AMTEC and TPV, power decays with the decay of the thermal source.
3. For TPV, power decays with the decay of the thermal source plus 1 “/i/year for radiation damage to the PV cells.

Of the four technologies, the free-piston Stirling engine is the most mature in terms of lifetime and efficiency **demonstrated.**[9] It's deficiencies relate to the possible need to compensate for engine vibrations, concerns about the reliability of devices with moving mechanical parts, and level of redundancy achievable with multiple engines. In addition to the Pluto Fast Flyby design, a duel 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 there are feasibility demonstrations 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. 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.

TPV is based on the response of **photovoltaic** cells to infrared 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. 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 band-pass filter between heat source and TPV cell will likely require development to achieve the highest projected efficiencies. The purpose of the filter is to tune the incident radiation to the band gap 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 Fast Flyby design exceeds the **AMTEC** and Stirling radiator areas by more than 5X. 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 radioisotope heater unit **(RHU)** that has been extensively used for electronics thermal control in planetary spacecraft. In the Powerstick concept, one thermal watt of **PuO_2fuel** would heat a **bismuth-telluride thermopile** to produce approximately 40 **mW** of continuous power, The power produced would charge **lithium** battery cells for subsequent discharge in a burst mode (28 **Wh/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 pre-prototype 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 is unpredictable. The development of **alternative** conversion technologies that conserve fuel or much smaller sources such as the **Powerstick** maybe 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 light weight 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 **($Ni\text{-}H_2$)** batteries are too heavy and bulky for many of the future planetary missions. In some cases they do not meet lie 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 3-4 times savings in weight and volume 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-H2,** single pressure vessel $Ni\text{-}H_2$ battery, **Ni-MH)** and lithium **($Li\text{-}TiS_2$,** Li-ion, lithium polymer) batteries, A comparison of the specific energy and energy density of SOA and advanced secondary batteries is given in Figures 4 and 5, respectively. 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 below.

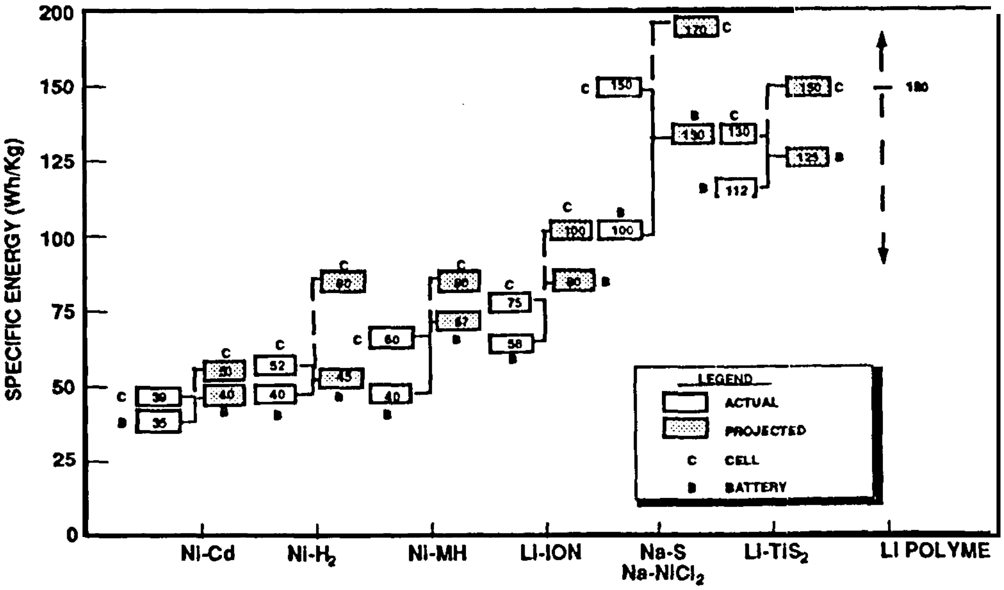

Figure 4. Specific Energy of Rechargeable Cells and Batteries

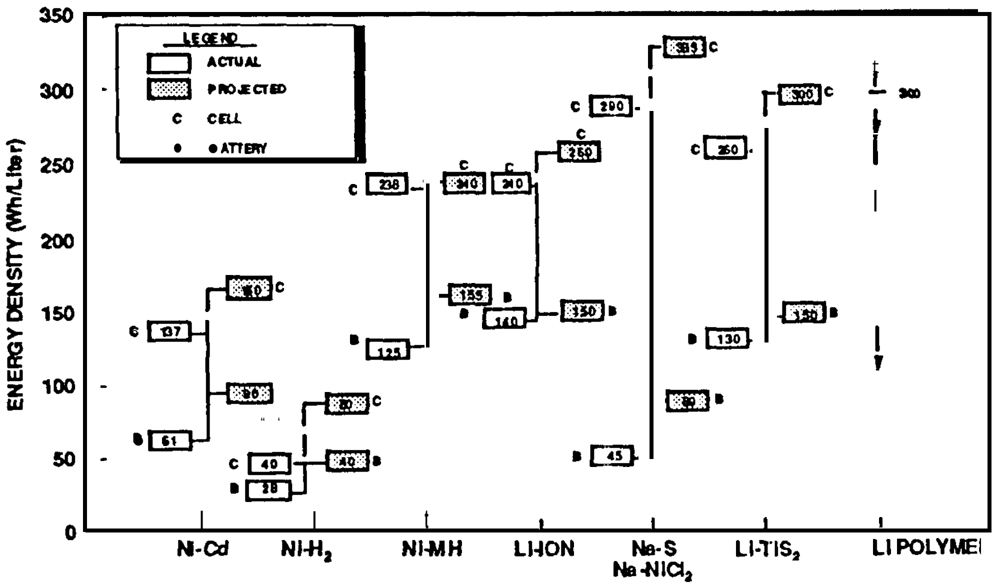

Figure 5. Energy Density of Rechargeable Cells and Batteries

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* **8 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, launch vehicles, etc. 'A number **of lithium** primary batteries such as Li-CF_x, Li-MnO_2, Li-I_2, Li-SO_2, Li-$SOCl_2$ Li-BCX, etc., 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 Wh/kg) and 6-8 years of active storage life. $Li\text{-}SOCl_2$ batteries have 20-30% higher specific energy compared to $Li\text{-}SO_2$ batteries. In view of this, these batteries were selected for use in JPL's Mars Pathfinder Rover. The main function of the $Li\text{-}SOCl_2$ battery in this mission is to meet peak power loads.

Advanced Nickel Battery Systems

Advanced Ni batteries that are presently under development for small spacecraft applications are: 1) two cell common pressure vessel (CPV) $Ni\text{-}H_2$, 2 single pressure vessel (SPV) 22 cell $Ni\text{-}H_2$ battery and 3) Ni-MH. Some of the important performance characteristics of small $Ni\text{-}H_2$ batteries are given in Table 3. These advanced $Ni\text{-}H_2$ batteries provide 10-30% higher specific energy and *energy* density compared to the state of the art individual pressure vessel $Ni\text{-}H_2$ 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-H2 was used on MISTI-2, TUBSAT and APEX.[14] Martin Marietta has selected two cell CPV for the Mars Global Surveyor mission based on preliminary ground test data. The Naval Research Laboratory has recently flown a single pressure vessel 22 cell $Ni\text{-}H_2$ battery on the Clementine mission that included LEO cycles and mapping of the moon. NASA is also considering this type of battery for several small spacecraft (Discovery Missions). Development of NI-MH batteries for space applications is presently in progress. Small capacity prismatic cells were constructed and testing of the cells is in progress.

Table 3. Small (<30Ah) NI H2 Battery Characteristics

Property	IPV NI H_2 (2.5")	2 Cell CPV (3.5")	22 Cell SPV
Voltage (Vdc)	20	20	28
Capacity (Ah)	20-30	10-25	10.5
Cycle Life	-65,000 (LEO) 30% DOD	>15,000 (LEO) 15% DOD	9,000 (LEO)
Specific Energy (Wh/Kg)	30-40	30.5	47
Energy Density (Wh/L)	16	18	50

Secondary Lithium Batteries

Ambient temperature secondary lithium batteries[16, 17, 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-H2 batteries. Successful development of these batteries will yield large pay-off such as 2-3 fold increase in energy storage capability and a longer active shelf life of 2 to 4 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 considering these batteries for future small planetary spacecraft, Such spacecraft may require batteries that can provide a specific energy of 100 Wh/kg and an energy density of 250 Wh/l and 500-1000 cycles. These batteries may also be attractive for rovers, astronaut equipment, and GEO spacecraft.

Three types of rechargeable lithium batteries: $Li\text{-}TiS_2$, Li-ion and Li-Polymer are presently under development at JPL (Table 4). The $Li\text{-}TiS_2$ system is considered suitable for planetary missions that require high specific energy (> 130 Wh/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 Wh/kg and can be used in a variety of configurations. JPL has developed small capacity $Li\text{-}TiS_2$ cells capable of providing a specific energy >130 Wh/kg and 1000 cycles at 50% depth of discharge (DOD), JPL is developing Li-ion cells that employ graphite as the anode and $LiCoO_2$ as the cathode. Experimental cells have completed 400 cycles at 100% DOD. These cells are projected to have a specific energy of 85-100 Wh/kg. JPL, in cooperation with Yardney Inc., is planning to scale up lithium ion cell technology to 10-2O Ah cell level for future spacecraft applications. Lithium polymer cells are in early stages of development at JPL and elsewhere. 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.

Table 4. Status of Rechargeable LI Technology at JPL

	LI-TIS_2	U ION	U POLYMER
ANODE	u	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 (Ah)	1-3	1-3	<0.2
CYCLE UFE	1OOO(5O%DOD)	1000 (100% DOD) *@	>1oo(1oO%DOD)
OPERATING TEMPERATURE ("c)	-20 TO 60	-20 TO 60	RT - 60
SPECIFIC ENERGY (Wh/Kg)	132	65	150'
ENERGY DENSITY (Wh/l)	260	240	350'

"PROJECTED
@ 200 CYCLES DEMO TODATE WITH INHOUSE CELLS

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, the advantages of a combined power and telecommunications system using a deployable concentrator/antenna is being studied 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.

Acknowledgments

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