HETE SATELLITE POWER SUBSYSTEM

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Abstract

The HETE (High-Energy Transient Experiment) satellite, a joint project between MIT's Center for Space Research and AeroAstro, is a high-energy gamma-ray burst/X-Ray/UV observatory platform. HETE will be launched into a 550 km circular orbit with an inclination of 37.7°, and has a design lifetime of 18 months. This paper presents a description of the spacecraft's power subsystem, which collects, regulates, and distributes power to the experiment payload modules and to the various spacecraft subsystems (radios, electronics, sensors, and actuators). The spacecraft uses four solar power arrays for primary power during orbit-day, and six rechargeable nickel-cadmium batteries during orbit-night. Two Power Point Trackers (PPT) are used to regulate the operating points of the solar arrays. The outputs from the PPTs provide the spacecraft's main power bus, which is used to recharge the batteries through three Battery Charge Regulators (BCR), and to supply power to distributed DC-DC switching converters for individual subsystems. The benefits gained by using a distributed-power scheme are improved regulation, electrical isolation of payload subsystems, simplified wiring, and fault-tolerance. Each experiment subsystem can be powered on and off independently, with minimum disturbance to other subsystems.

Design Strategy

The strategy adopted for this design is to build a low-cost, simple, distributed, and fault-tolerant power system capable of providing power to diverse payload instruments and spacecraft subsystems during both orbit-day and orbit-night periods. This requires the use of low-cost, commercially available electrical components wherever possible, robust and efficient switching converters, small and lightweight parts, and a combination of solar panels and rechargeable batteries.

A distributed power system provides separate power regulators for each subsystem that uses electrical power. There are several advantages to be gained by this:

- Electrical isolation of loads (experiments, SC systems)
- Improved regulation of output power.
- Improved fault tolerance.
- Simpler and more efficient routing of power lines.

These advantages also come with several penalties, however, when compared to a local power system design. These disadvantages are:

- Decreased electrical efficiency.
- Increased number of EMI/RFI noise sources.
- Power is nearer to sensitive systems.
- Thermal considerations must be carefully addressed.

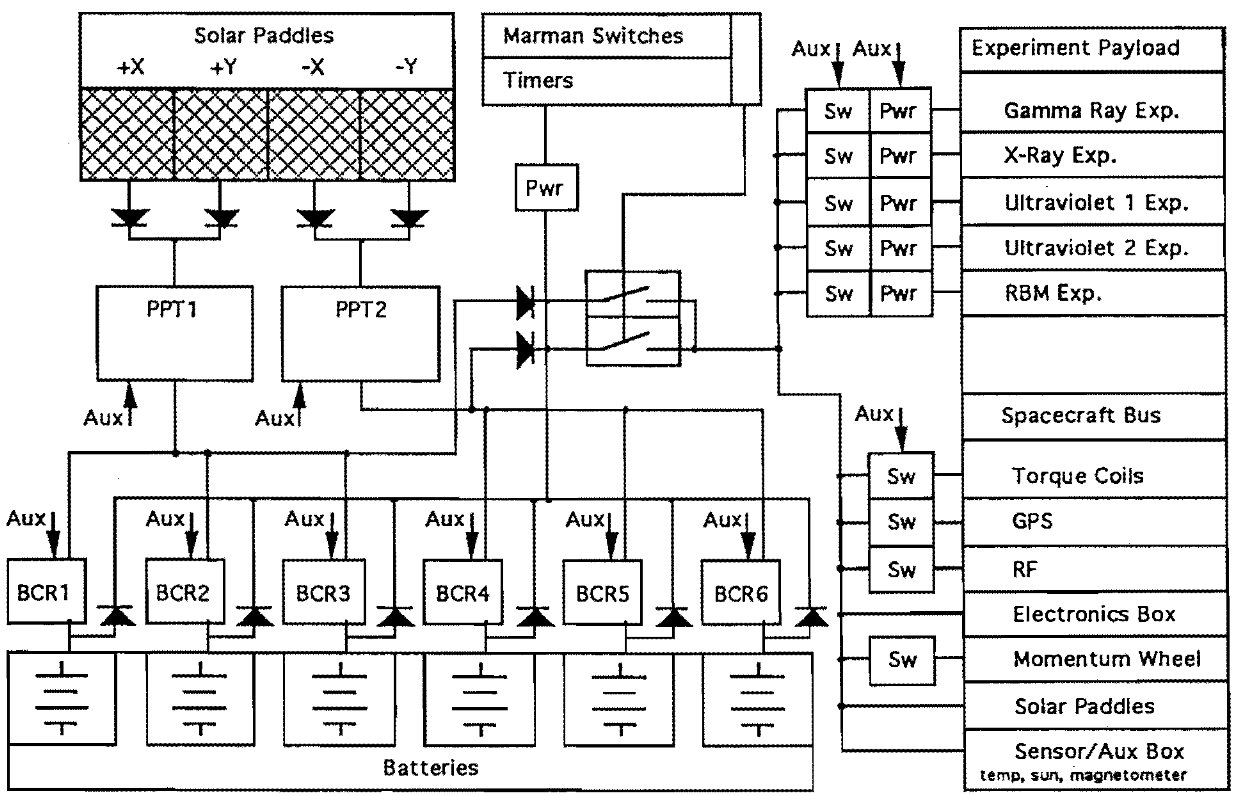
The design of a local power system would use a single, centralized power converter to provide power to all subsystems. The single converter would provide all the voltages required by the instruments and other systems, and each instrument would be share its power lines with the others.

AeroAstro has determined from direct experience that the advantages of the distributed system outweigh the disadvantages, and that a distributed design will provide better performance and a simpler instrument interface. This is especially true for a satellite with multiple sensitive instruments that need to be independently powered on and off.

The decision to use a distributed power system means that individual spacecraft instruments have their own isolated and fully regulated power supplies. A set of DC-DC power switching converters is provided for each payload instrument and for each subsystem on the spacecraft bus (here "payload" is used to refer to the science package flown on the satellite, and "bus" is used to refer to the satellite flight, control, and data processing systems). The switching converters are isolated regulators that are powered by the solar arrays or batteries, and which provide power at the specific voltage and current levels required by each instrument. For example, one instrument needs +5.2 volts (V) at 140 milliamperes (mA), -5.2 V at 50 mA, +15 V at 220 mA, and -15 V at 130 mA. Another instrument needs +5.1 V and -5.1 V each at 300 mA, +15 V at 180 mA, -15 V at 60 mA, and 28 V at 20 mA.

In order to maximize the performance of the satellite on orbit, the power system has been designed to withstand the failure of some power components while still providing enough power to permit nominal or near-nominal satellite operations. One of the ways in which this has been accomplished is by providing two separate solar panel regulators, which each regulate the power derived from two of the four solar panels. These regulators are a special design which will force the panels to their most power-efficient operating points when the satellite power demand is sufficiently high to require maximum power from the solar arrays. The output from each PPT provides power to three of the six battery charge regulators (BCR). Each BCR controls the charging and discharging of its associated battery pack. The block diagram below illustrates the power system structure.

Power System Block Diagram



Power Budgets

There are two power budgets used to indicate the amount of power available to the satellite versus the amount of power consumed by the electrical systems. The first budget applies to orbit-day operations, when the satellite is powered by the illuminated solar arrays and the batteries are being recharged. The second budget applies to orbit-night, when the batteries are discharged and the solar arrays are dark. A key figure of merit here is the level of discharge required from the batteries to sustain the satellite during the duration of its orbit-night period. A nickel-cadmium rechargeable cell will exhibit its longest lifetime when it is operated at only a fraction of its capacity, typically 20%. HETE's orbit, a low-inclination circular LEO, causes the satellite to experience regular solar eclipses throughout its lifetime, which means the batteries will be subjected to a high number (approximately 4500 per year) of charge/discharge cycles. Each orbit-night period lasts on average about 35 minutes, compared to the typical orbit-day period of about 65 minutes. The sizing of HETE's batteries has been selected to provide full spacecraft power for discharge depths of up to 20% at the beginning of the satellite mission. The declining capacity of the batteries as they age will result in a depth of discharge of about 30% near the end of the useful lifetime of the satellite.

During orbit-night operations, the science instruments operate at their peak, while during the day, the instruments are partially inactive. Also, during the day, a large portion (nearly half) of the power demand is used for recharging the depleted batteries. The paddles were redesigned to increase their maximum output power capabilities by using slightly larger silicon cells. This was done to improve the orbit-day power margins, especially during operations late in the satellite's lifetime. Increased margins will also permit more attitude maneuvers to be executed and provide greater tolerance to sun-pointing errors.

Orbit-Night Power Budget:

(Units are in Watts)		Typ	Max				
Reg Power Demand		57.1	73.8				
Converter Efficiency	80%						
Unreg Power Demand		71.4	92.3				
		Typ	Max	Typ	Max	Typ	Max
		DOD =	20%	DOD =	25%	DOD =	30%
Battery Power (6 sets)		Margins		Margins		Margins	
BOL, nom, 35 min.	370.0	2.7	-18.3	21.2	0.2	39.7	18.7
EOL, nom, 35 min.	300.0	-11.3	-32.3	3.7	-17.3	18.7	-2.3

Orbit-Day Power Budget:

(Units are in Watts)		Typ	Max			
Reg Power Demand		92.7	104.9			
Converter Efficiency	80%					
Unreg Power Demand		115.9	131.1			
		Typ	Max		Typ	Max
Solar Pwr	1 *	Margins		2 †	Margins	
BOL, nom	145.0	29.1	13.8	165.0	49.1	33.8
EOL, nom	134.0	18.1	2.8	153.0	37.1	21.8

* Original solar paddles
† New boosted paddles

Commercial Batteries

The nickel-cadmium cells which have been chosen for the HETE mission are commercial sub-C size 1.2 Ampere-hour (A-h) cells. The prohibitive cost of aerospace cells necessitated this choice. The cells are a custom sintered-sintered design optimized for high discharge currents and long charge/discharge cycle lifetimes. There are six batteries, each consisting of 23 series cells, on the satellite. The batteries are constructed in pairs, with three battery boxes each containing two sets of cells. Some important specifications are:

- 23 cells per battery, 7.2 A-h per cell => 4.8AH @ 30V max available battery capacity.
- Batteries can provide 216 Watts for up to an hour, or up to 371 W for 35 minutes.
- At 19% DOD, batteries can provide approximately 72 W for 35 minutes.

The battery charge regulators (BCR) control the charge and discharge rates of the batteries. There are three distinct programmed charge rates, and one programmed discharge rate.

A slow charge rate (approximately C/15, where C represents a charge current of 1.2 Amps) is used as the default. This slow rate is appropriate for charging batteries which are already in or very near a fully-charged state. A nominal charge rate (approximately C/4) is selected for recharging the batteries after a typical orbit-night, when the batteries need to be brought back up from a 20% discharged state. A fast recharge rate (C/2) is selected to recharge batteries that have been deeply discharged. This could occur if a fault or unbalanced condition exists which causes one battery to be loaded more heavily than other batteries, or it could occur when a battery has been deliberately discharged as part of a reconditioning cycle.

The programmed discharge rate (roughly C/20) is used to slowly discharge a battery to an almost fully-depleted state (corresponding to a battery voltage of about 23 volts, or 1 V per cell), after which the battery is recharged. This reconditions the battery, and can extend a battery's useful life by restoring some of the lost capacity of a battery that has been through many charging cycles. During normal orbit-night operations, the batteries will discharge at a rate (estimated to be about C/3) that meets the power needs of the spacecraft electrical systems.

The BCRs operate by monitoring the terminal voltages, battery currents, and battery temperatures. A programmed voltage cut-off level determines when a charging battery is switched from a nominal recharge rate to a slow recharge rate (trickle charge). In addition, the temperature profile of the battery is measured to look for the sudden rise in temperature gradient that typically indicates a fully-charged battery, which begins to convert incoming current to heat most effectively when charged to near full capacity.

Solar Panels

HETE is equipped with four silicon-cell, deployable solar panels. The satellite is launched with its panels stowed, and does not deploy them until the attitude control system has established a spin-stabilized, sun-pointing orientation. During this acquisition period, the spacecraft systems are powered by the spacecraft batteries. Once stabilized, the panels are permanently deployed by a command from the on-board processing system.

Some relevant specifications of the solar panels are as follows:

- 109 silicon cells per panel
- Cells are 1.24" x 2.36", 0.01" thick
- Total area of cells for four panels is approximately 1273 in^2.
- 38.3 Watts, 47.2 Volts per panel at end-of-life (EOL)
- Total solar power available is roughly 153 W EOL
- 85 V maximum open-circuit voltage at beginning-of-life (BOL), -15°C
- 38 V minimum open-ciruit voltage at EOL, +70°C

Solar power from the panels is regulated by the power point trackers (PPT), which regulate the power generated by two of the panels. The panels act as paralleled power sources, and have an output characteristic that degrades over time (due to exposure). Peak performance is obtained when the cells are cold and new, and as the cells are heated or exposed to the elements, the cell putput voltage deteriorates. Some of the factors that cause the cells to age are thermal cycling, exposure to ultraviolet and charged-particle radiation, and contamination. Each panel has a built-in blocking diode which allows current to flow out of the panel, but not in the opposite direction. This has the detrimental effect of a small power loss, but protects other panels from being affected if one panel should fail and become a virtual short-circuit.

Additional fault-tolerance is obtained by using two PPTs. If one should fail, then the other will supply power to the spacecraft and attempt to make up for the loss. Each PPT supplies power to three of the six batteries, but a provision exists to allow all six batteries to be charged from a single PPT in the event that one of the PPT circuits fails or is performing poorly.

Power Controls

The power subsystem is controlled by the spacecraft processor. This processor, an Inmos Transputer, sends messages to the power system using a proprietary redundant serial communications link. This serial link (called the Aux bus) is supported by a set of custom FPGA designs which decode and respond to the serial transmissions. Each subsystem on the spacecraft bus has its own Aux bus chip or chips to link it to the spacecraft processor. For the power subsystem, the Aux link is used for commanding the different states of the power components (on/off switches, charge rates, etc.), and for determining the operating state of the system. The power system is equipped with several analog-to-digital converters which monitor analog voltages and currents at points in the system. These data can be interrogated using the Aux bus, which keeps the spacecraft processor informed of all pertinent operating points in the power subsystem.

The Aux bus allows full control over the PPT and BCR circuits, so that optimum performance can be obtained from the power system as the power load, sun visibility, and circuit performance changes throughout the satellite mission.

The PPT is a buck converter which is controlled by a pulse-width modulated waveform generated by a control chip. This control chip monitors the output voltage and current of the regulator, and adjusts the duty-cycle of the buck converter's pass transistor to maximize the output voltage. With a current-limited power-source (the solar panels), this circuit will maximize the power obtained fron the input source when the output load requires maximum power. The PPT has a programmed maximum output voltage to protect the circuit elements downstream from it.

Summary

A low-cost, compact, and reliable satellite power system has been designed to provide the power needs of a low Earth orbit (LEO) science satellite with a diverse array of payload instruments. The HETE satellite will use a combination of solar-generated power and rechargeable nickel-cadmium batteries to provide power to the satellite systems, and can be monitored and controlled by a sophisticated on-board microcontroller, the Inmos Transputer. The power design implements a distributed power scheme, with power conversion performed at each subsystem by high-efficiency DC-DC switching converters. The power system is capable of providing up to 60 Watts of regulated power from its on-board batteries during orbit-night periods (with a battery depth-of-discharge of 20%), and up to 120 Watts of continuous regulated power from its solar arrays during orbit-day periods. This power is used to energize the instrument payload, consisting of gamma-ray, X-ray, and ultraviolet detectors, and the spacecraft bus, which consists of a powerful processing system, telemetry radios, attitude-control sensors and actuators, and on-board navigational electronics.

Acknowledgements

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