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ENERGY STORAGE REQUIREMENTS FOR SPACECRAFT

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Summary

The demands for electrical power on a spacecraft are reviewed. These requirements affect the choice of prime power source and energy storage devices, which must be selected as a system. The design of the electrical system is addressed by describing current hardware characteristics and how these interact. Particular attention is given to batteries for energy storage.

Electrical power requirements

The demands for electrical power vary during different spacecraft operations, which may include standby, prelaunch, ascent, orbital, maneuvering and peak demands. These operational states are described below.

Standby or launch ready usually implies that the satellite is in a reserve or inactivated state, but can be activated on very short notice.

Prelaunch activated periods are usually brief and allow for a transition from ground power to spacecraft power.

Prelaunch, launch and *ascent* to orbit power can be provided by any power source which does not require deployment of hardware beyond the confines of the launch shroud. This excludes solar arrays which if body-mounted require a launch shroud or if stowed must be deployed on-orbit.

Orbital conditions define significant requirements for the power system, most important being those imposed by solar eclipse periods. In low earth orbit, near half of a 90 min period may be in eclipse, whilst at geosynchronous orbit the eclipse period varies from 0 to a maximum of 72 min per day, only 84 days a year. The eclipse demands that solar energy converters be augmented by energy storage devices.

Use of solar radiation imposes a pointing requirement. On a solar array panel input is a function of the cosine of the angle between the incident solar radiation and the panel; the cosine function allows a wide margin in pointing accuracy. Use of a solar concentrator demands higher pointing accuracy. Solar array sun tracking systems are used frequently on stabilized spacecraft;

however, spinning satellites require either body-mounted solar arrays or despinning of the solar arrays and their mounting structures.

Space trajectory or *probe* spacecraft may not have solar eclipse conditions, but since the solar incident energy is a function of distance from the sun, extreme temperatures and lower efficiency or lower output may occur. The fact that available solar energy decreases as the square of the distance from the sun favors the use of non-solar, such as isotope, fueled power sources for outer planetary missions.

High power surges or short-duration demands which may be 5 to 10 times average load can only be accommodated by batteries, fuel cells, or chemically fueled dynamic engines. Radioisotope thermoelectric or dynamic generators as well as solar arrays sized to meet the average demand cannot accommodate any significant increase in load. Therefore, these power sources when used in this kind of application must be supported by an energy storage device which can "load level" high peak demands.

Voltage transients caused by rapid switching of loads or electromagnetic interactions between equipment can be attenuated by low impedance power sources. Batteries are relatively low impedance devices; therefore they are valuable in this function.

Reserve power for emergency or back-up situations is best satisfied by a passive, non-degenerating, instantly available power source with a high energy density or by over-designing the basic power system.

Analysis of system requirements

The above power requirements are analyzed to define the worst case conditions which will size the power system. The analysis is complicated because all elements of the power system and its users dynamically interact. Figure 1 indicates the major functions generally required for spacecraft power systems. Working back from load to source, a first task is to characterize and quantify all loads; this will establish what, if any, power conditioning might be required and will give power as a function of time at point A of Fig. 1. If no energy storage is required, the three functions shown below points A and B are absent.

If power source operation is discontinuous, as in a solar eclipse with a solar array, or if periodic power demands exceed source output, energy storage will be required. In this case power at point B during charge of the energy storage subsystem must support power demand at point A in addition to charging power and inefficiencies of charge, regulation and control. The power regulation between points B and C may or may not be required, depending upon the electrical characteristics of the power source and the energy storage hardware.

The conditioning and distribution of power to spacecraft loads will not be addressed, since those elements do not usually impact the choice of the energy storage subsystem.

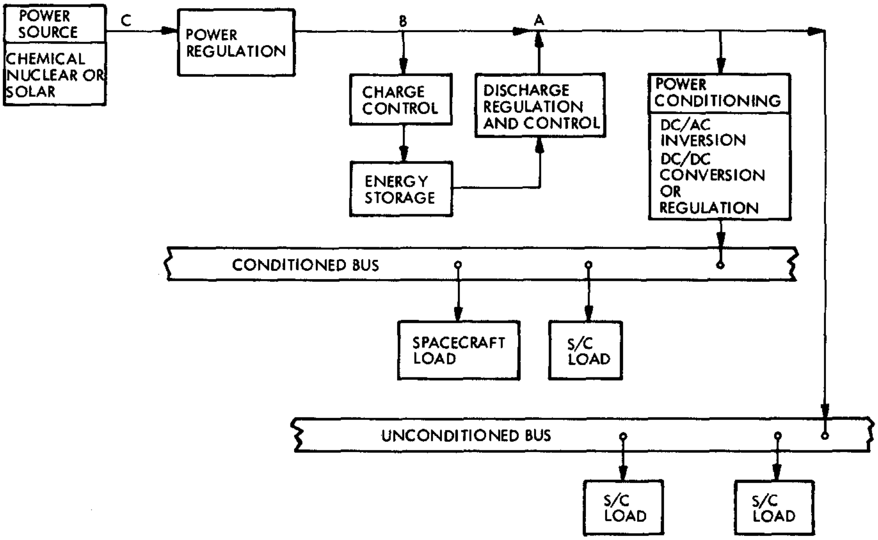

Fig. 1. Functional block diagram.

Power generation and energy storage

Designers must take all of the above requirements into consideration when evaluating power system alternatives. Life and power level would be first used to identify possible alternatives and secondly, weight, volume, availability and cost would be used to determine feasibility. The next selection set would deal with system effectiveness and include reliability, maintainability and safety. In synthesizing power systems one to three major components may be required; these are: power source, power converter, energy storage device. At this time four power source types are in use; they are solar, chemical, isotope and nuclear. The solar flux is converted to electrical power by photovoltaic cells. Figure 2 is a solar array panel which is typical of many flown. In this photograph one segment of the array is being illuminated to check electrical performance.

Chemical power sources may be divided into two groups. First, chemical combustion to drive heat engines or thermal converters and second, electrochemical which includes primary batteries and fuel cells.

Isotope fuel elements and nuclear reactors are heat sources for thermal/electrical converters such as thermoelectric generators and heat engines, the latter including Brayton, Rankine and Stirling cycles.

Isotope half-life determines the system life limit. The most commonly used isotope is ^{238}Pu which has a half-life of 89 years, which means a power loss of only 7.8% after 10 years.

The Garrett Corporation [1] is developing a Brayton cycle, isotope power system (BIPS). A model of a 1.3 kW BIPS is shown in Fig. 3; the photograph shows a turbine–alternator–compressor assembly (center), two

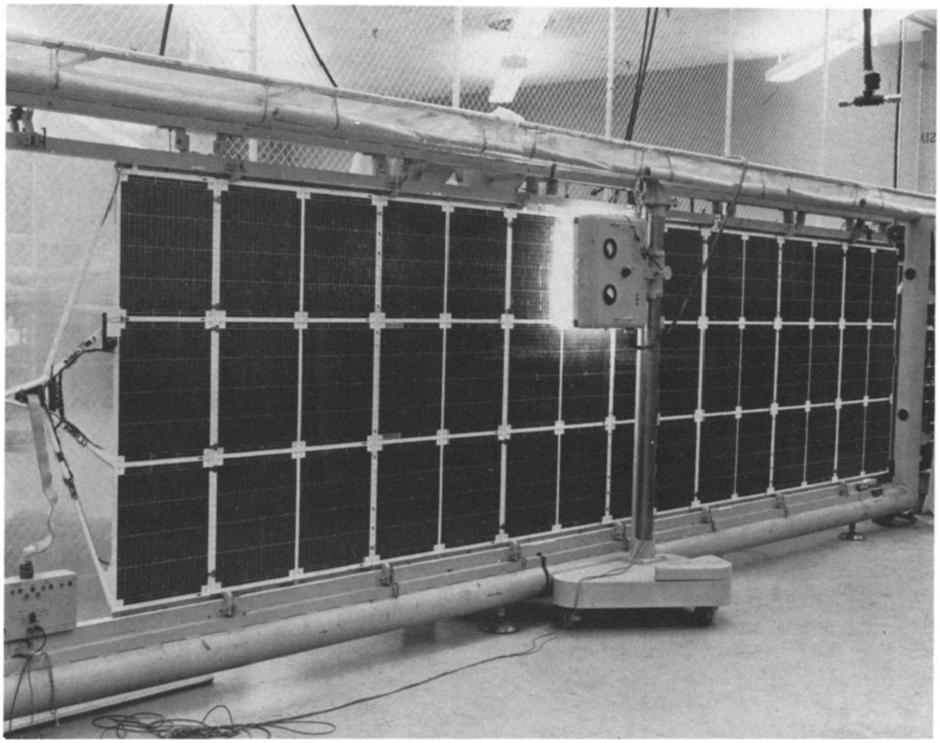

Fig. 2. Solar array wing in electrical checkout.

isotope heat source assemblies (left and right) and a heat exchanger (below). The 1.52 m diameter structure is designed to mount a radiator. The BIPS uses a helium–xenon gas mixture as the motive fluid.

The Sunstrand Corporation [2] is developing an organic Rankine cycle, 0.5 to 2.0 kW isotope power system (KIPS). A cutaway perspective of a 1.3 kW KIPS is shown in Fig. 4. Dowtherm A is the working fluid which is vaporized in the isotope-heated boiler and expanded through the turbine, which drives an alternator rotor and system pump on a common shaft. The vapor exchanges heat to the boiler feed and is condensed in a jet condenser fed by subcooled fluid returning from a radiator. The radiator shown is 1.22 m in diameter × 3.23 m long.

Radioisotope thermoelectric generator (RTG) systems have been employed in space since the launch in 1961 of the U.S. Navy Transit IVA navigation satellite, which carried a 2.7 W RTG. The Apollo 12 mission left an Apollo Lunar Surface Experiments Package with a 73 W Snap-27 RTG on the Moon. As in the above isotope power systems, the isotope provides a heat source. The electrical energy is derived from heat flowing from the heat source to a heat sink, usually a radiator, through a thermocouple pair. Power conversion efficiencies achieved have been in the 5 to 6% range. Teledyne Energy Systems [3] and the 3M Company [4], under Department of Energy

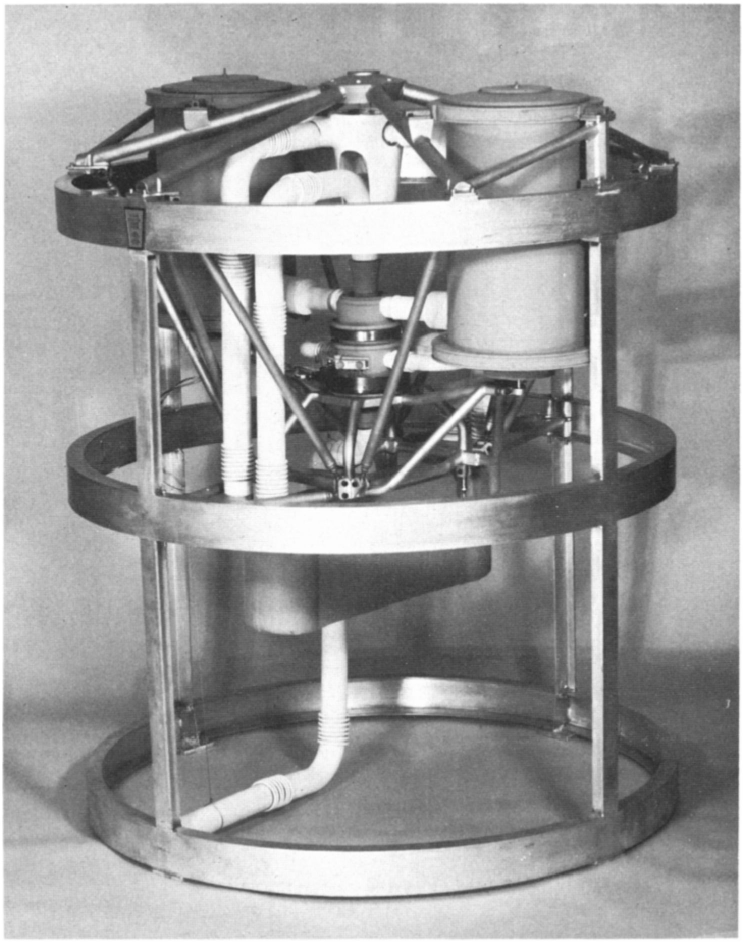

Fig. 3. Mechanical model of 1.3 kW (BIPS) (courtesy of Garrett Corp.).

support, describe, respectively, a 10% efficient RTG using copper–silver selenide and gadolinium selenide thermoelectric couples.

Energy storage can be chemical, electrochemical, electrical, mechanical or thermal; however, only chemical and electrochemical storage have the necessary energy density and storage properties for spacecraft power systems. Chemical storage is applied in the hydrogen–oxygen fuel cell system, where the reactants are stored as gases or cryogens in tanks and subsequently electrochemically converted to electrical power in fuel cells. Electrochemical storage of course is in batteries. Energy storage can be for either one-time, primary use or repeated, so-called secondary use.

Hydrogen–oxygen fuel cells of the primary type have flown on the Apollo moon missions. Secondary or regenerative fuel cell systems of two types have been under development; one type uses the H_2–O_2 fuel cell in reverse to electrolyze water, while the other type uses a separate water electrolysis unit.

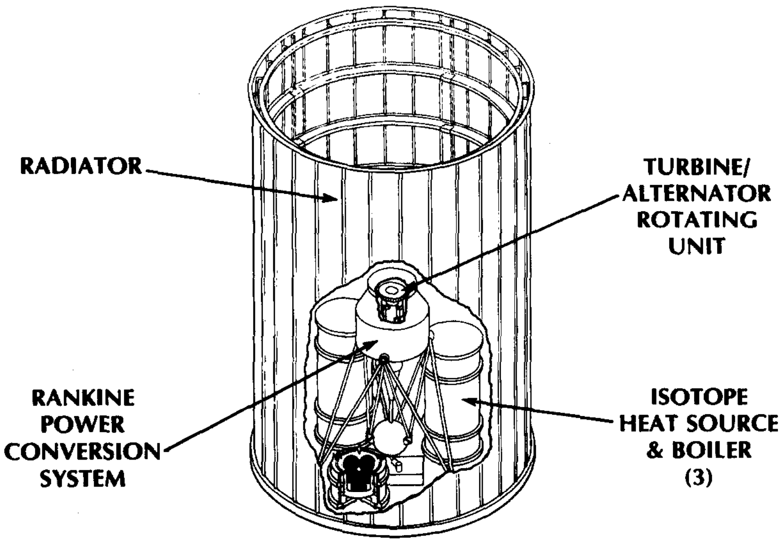

Fig. 4. Kilowatt Isotope Power System (KIPS) (courtesy of Sunstrand Corp.).

Silver–zinc batteries have been the mainstay of primary systems, because of their high energy density — up to 280 Wh/kg at low discharge rates. Ag–Zn secondary batteries may be applied where calendar and/or cycle life requirements are limited to hundreds of cycles.

Nickel–cadmium secondary batteries have been applied extensively wherever long calendar and/or cycle life is desired. The aerospace Ni–Cd cells are hermetically sealed and have the ability to accept overcharge — indefinitely at a low rate.

The most prominent competitor to Ni–Cd is the Ni–H_2 battery which at an operational energy density of 22 Wh/kg offers a potential 50% weight savings. The Ni–H_2 cell combines the nickel oxide positive electrode from the Ni–Cd cell with a fuel cell type hydrogen electrode. Elimination of the cadmium electrode promises longer life and the increase of hydrogen gas pressure with charged capacity is a good state-of-charge indicator. Each Ni–H_2 cell is designed as a pressure vessel since maximum operating pressure may reach 4×10^6 Pa (40 atm.).

Figure 5 is a photograph of a Ni–Cd battery assembly with control electronics. Figure 6 is a photograph of the Ni–H_2 battery type flown by the U.S. Air Force in a flight experiment in 1977.

System design

The above components are the basic building blocks for most systems. However, before selecting them, it is necessary to match requirements with element and system characteristics.

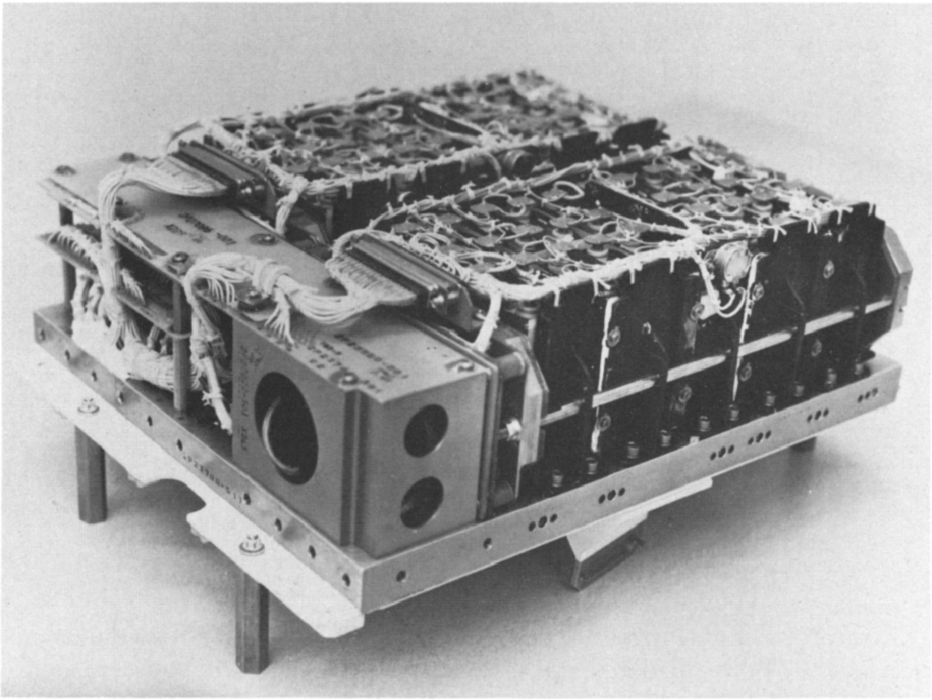

Fig. 5. Nickel–cadmium spacecraft battery assembly.

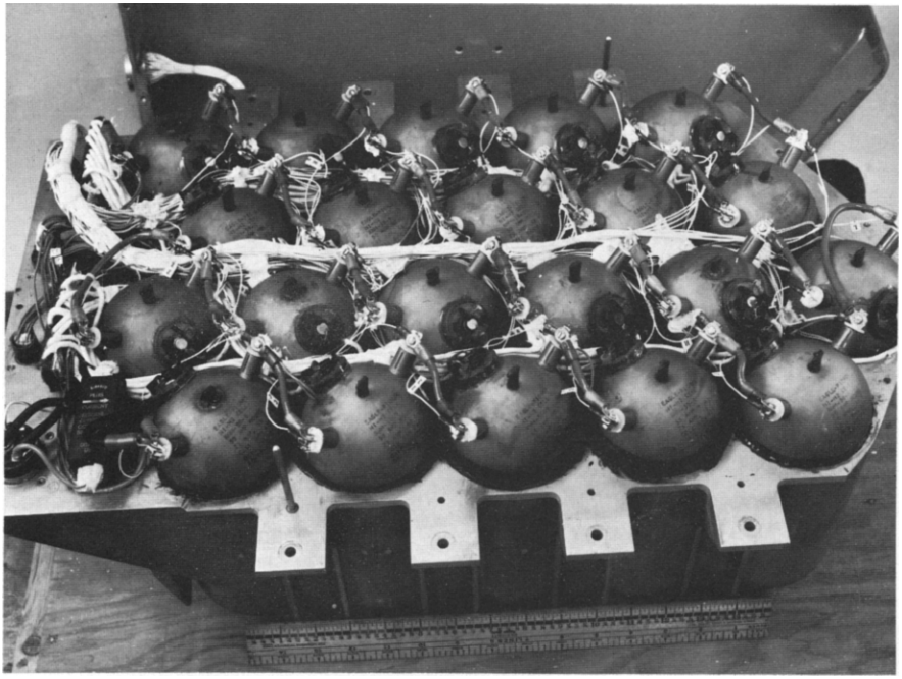

Fig. 6. U.S. Air Force nickel–hydrogen flight experiment battery.

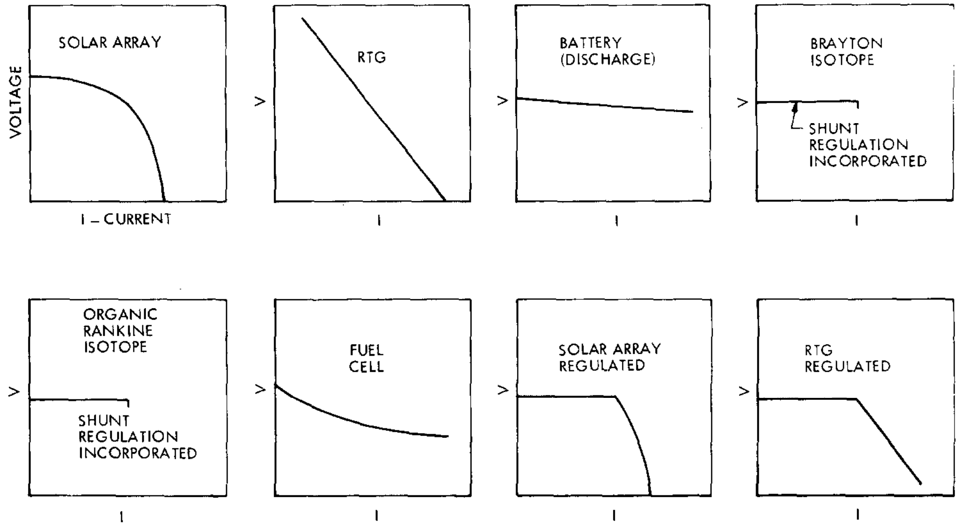

Fig. 7. Power source current–voltage characteristics.

Figure 7 presents current–voltage characteristics of the above-mentioned power sources, both regulated and unregulated. Those power sources, such as batteries and fuel cells, which are not limited by rate of availability of reactant, do not have a bounded I–V curve. Conversely, solar or isotope power conversion devices have their power output limited to power in, except for brief reliance on thermal capacitance. Figures 8 and 9 illustrate these functional differences between systems. These input-limited energy conversion devices also have poor voltage regulation characteristics; the RTG is seen to have a linear I–V curve. This demands the use of shunt regulators in parallel with system loads to maintain the necessary voltage regulation and further requires that the power source be sized to meet maximum demand.

If peak loads are much higher than average power demand or if the power source output is intermittent, as with a solar array during eclipse, energy storage is necessary. Secondary batteries have long been used successfully with solar arrays and would be used with other prime power sources. In Fig. 10 composite I–V curves are plotted to indicate how the addition of a secondary battery affects the system. These plots are only qualitatively indicative of spacecraft bus voltage.

Another way of showing how the energy storage device supports the prime power source for surge loads is to plot power as a function of current as in Fig. 11 for each device. The isotope heat source energy converters and the solar array are power limited/energy unlimited devices which we attempt to operate near their maximum power point. Conversely, batteries are energy-limited (without recharge) and usually voltage-limited before reaching peak power.

Figure 12 shows a composite power *vs.* current curve of prime power sources acting in parallel with secondary batteries. So long as the power

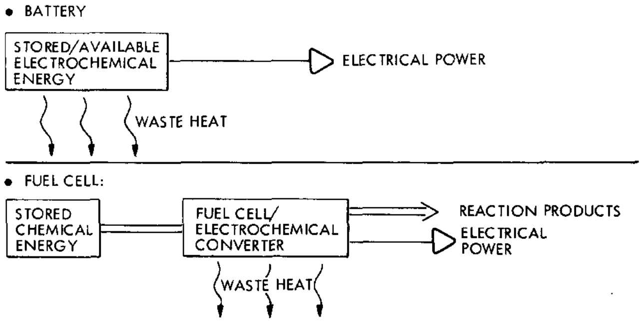

Fig. 8. Non-rate-limited energy converters.

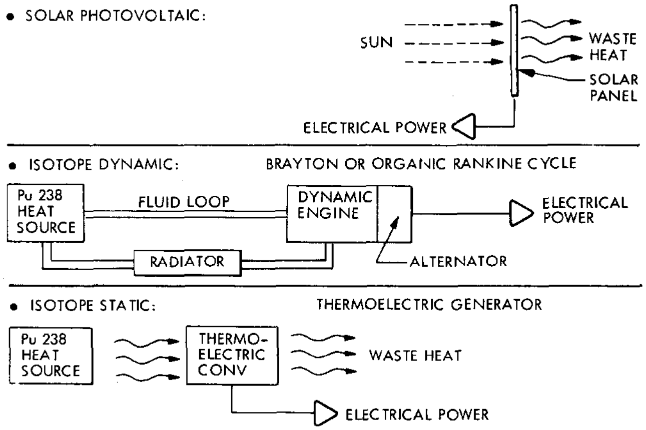

Fig. 9. Rate-limited power converters.

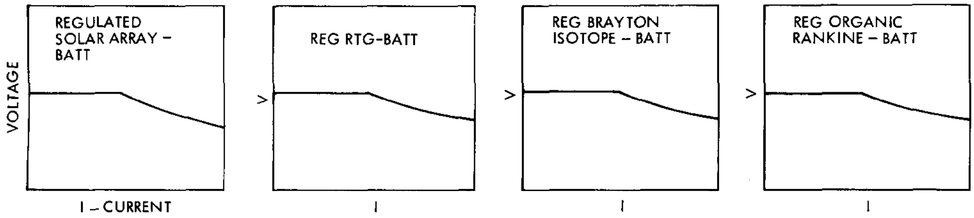

Fig. 10. Power source — secondary battery current — voltage characteristics.

demand is below the maximum output of the prime power source, batteries can be charged; when demand exceeds the prime source capability the batteries make up the deficit. Note that curves 1, 2 and 3 of Fig. 12 reflect the characteristic output of each power source so long as it supports the load; when the battery begins supporting the load it allows the prime power source to continue operating near its peak power point.

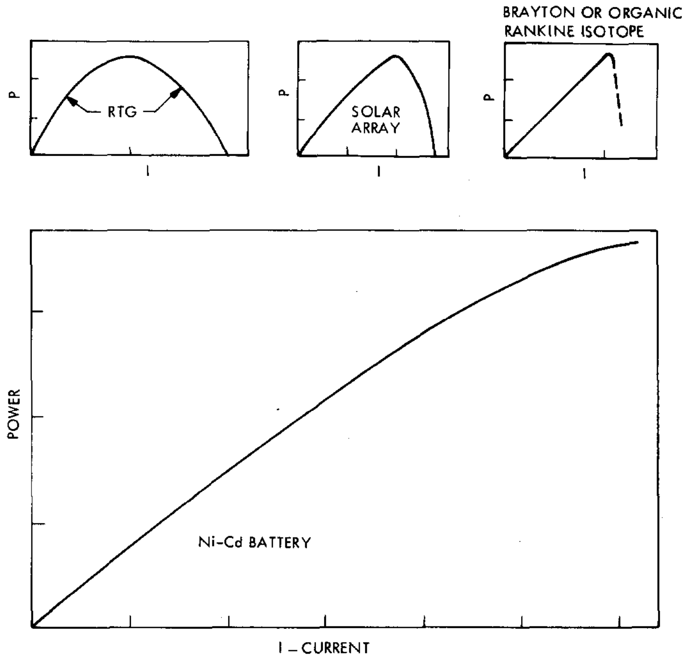

Fig. 11. Power *vs.* current (typical curves).

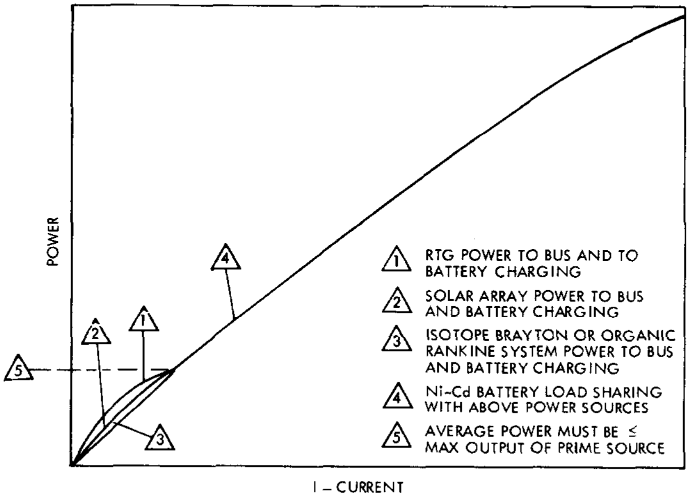

Fig. 12. Power source — secondary battery system power *vs.* current.

Battery design

Battery characteristics, such as ease of charge control, overcharge acceptance, and charge efficiency and life as a function of temperature, current and depth-of-discharge must enter into the development of battery requirements. The singular significant difference between a battery supporting an isotope power source and one supporting a solar array is the array's total loss of output during an eclipse, which places all loads on batteries and increases the capacity requirement.

A detailed battery specification will be prepared to satisfy the above requirements while at the same time not placing unrealistic demands on the manufacturer. There is a strong inclination to conservatism in selecting long-life batteries, because of the expense and time required for life testing and the demand for hardware with demonstrated reliability.

Conclusions

Energy storage requirements exist in most spacecraft electrical systems, even where continuous output power sources are used. Batteries will continue to be used for energy storage, because of their low impedance–high power characteristics and ease of integration with most power sources. An example of how an isotope-fueled power system with surge batteries might be sized to meet a given set of requirements is shown in the Appendix.

Appendix: Example of system sizing

Spacecraft requirements

Orbit period	90 min
Shade/sun times	40/50 min
Average load	1000 W
Peak load	5000 W
duration	10 min
frequency	once/orbit
Nominal voltage	28 V
Life	3 years (17,500 cycles)

Design philosophy and generalizations

(a) Cost, weight and reliability are the primary drivers in selecting a spacecraft system.

(b) Primary power sources are costlier and heavier than secondary batteries for meeting peak power demands. This suggests maximizing demand on batteries and minimizing the size of the primary power source.

Assumptions

(a) Energy storage will be in nickel–hydrogen batteries having an energy density of 44 Wh/kg. Allowing a 30% depth-of-discharge yields an effective 13.2 Wh/kg; depth-of-discharge is limited due to the high number of cycles the battery must withstand.

(b) A dynamic engine, isotope-fueled, power system can be provided which will deliver 6.4 W/kg.

(c) Overall battery charge power efficiency will be 70%.

Computations

(a) Peak demand

let $P(b)$ = peak power delivered by batteries
let $P(s)$ = power from prime power source
$5000\ \text{W} = P(b) + P(s)$ (1)

(b) Battery recharge rate

energy removed $= P(b)\ (0.167\ \text{h})$
energy replaced $= P(b)\ (0.167\ \text{h})/0.7$
recharge time available $= (1.5 - 0.167)\ \text{h}$
$= 1.333\ \text{h}$
recharge rate $= P(b)\ (0.167\ \text{h})/0.7\ (1.333\ \text{h})$
$= 0.179\ P(b)$ (2)

(c) Power source rate, P(s)

$P(s)$ = av. load + battery recharge rate
$P(s) = 1000\ \text{W} + 0.179\ P(b)$ (3)
Solving eqns. (1) and (3) simultaneously,
$P(s) = 1607\ \text{W}$
$P(b) = 3393\ \text{W}$

(d) Battery sizing

Energy removed $= P(b)\ (0.167\ \text{h})$
$= 567\ \text{Wh}$
Battery energy $= 567\ \text{Wh}/0.30$
$= 1889\ \text{Wh}$
Battery weight $= 1889\ \text{Wh}/44\ \text{Wh/kg} = 42.9\ \text{kg}$

(e) Battery discharge rate in terms of battery capacity

$3393\ \text{W}/1889\ \text{Wh} = 1.80 \times$ "C" rate

(f) System weight estimate

Isotope power system	250 kg
Battery	43
Charge control and regulation	20
Total	313 kg

Discussion

The battery capacity might be increased if better voltage regulation is desired, depth-of-discharge may be decreased to improve reliability, or the cell design may be tailored to best meet specific requirements.

It should be noted that sun/shade times have no impact (aside from thermal) for the isotope fueled system whereas a solar powered system would require battery power for the full shade period.

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