Power Supplies for Primary Electric Propulsion Missions

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This paper reviews the status of and requirements on solar electric and nuclear electric power supplies for primary electric propulsion missions. The power supply requirements of power level, specific mass (kg/kWe), and lifetime are defined as a function of the mission and electric propulsion system characteristics for planetary missions. The technological status of silicon and gallium-arsenide solar arrays is discussed. Nuclear reactors and thermoelectric, thermionic, Brayton, and Rankine conversion technologies are reviewed, as well as recent nuclear power system design concepts and program activity. Technology projections for power supplies applicable to primary electric propulsion missions are included.

I. Introduction

ALTHOUGH it is impossible to say that any one subsystem in a primary electric propulsion system is more important than another, the source of system power (i.e., the power supply) is generally the largest subsystem in terms of physical dimensions, mass, and cost. Seventeen years ago,[1] primary electric propulsion was greatly hampered by the lack of a suitable power supply, although the thrusters were well along in their development. A similar situation exists in the United States today, in the sense that nuclear electric power supplies are far behind the highly developed 30 cm mercury electron bombardment ion thruster and power processor. Fortunately, thanks to the solar electric propulsion (SEP) array program and its successors, the technology for high-power (>20 kWe), low-mass solar arrays is very nearly equal (in terms of mission readiness) to the propulsion technology. With the exception of the SEP solar array, power supply development has not and generally will not take place exclusively for electric propulsion. However, recent developments in other areas that require high power in space (i.e., a space station and military applications) allow one to be optimistic that high-power space power supply development may soon move rapidly forward.

The objectives of this paper are to: 1) define requirements on the power supplies for interplanetary electric propulsion missions, 2) discuss the status of both solar electric and nuclear electric power supplies, and 3) present technology projections for power supplies applicable to interplanetary electric propulsion missions. In those instances where reference to a specific thruster is helpful, this paper refers to ion thrusters, although in general the results are independent of specific electric propulsion technologies. The power supply requirements of power, lifetime, and specific performance are summarized for both nuclear electric propulsion (NEP) and solar electric propulsion (SEP) missions. These requirements have been distilled from numerous studies of SEP and NEP mission performance performed at the Jet Propulsion Laboratory. An example of the influence of power supply performance on both a NEP and SEP mission is presented in Sec. II.

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The review of the status of nuclear electric and solar electric power supplies includes descriptions of past system concepts, a discussion of the technical maturity of systems and components, and recent program development directions. Nuclear isotope and solar thermal power supplies are not included. NASA's near-term planning does not include missions requiring electric propulsion. However when the missions now planned are completed, it is clear that the performance offered by electric propulsion systems will enable the exploration of the solar system to continue. References 2 and 3 present potential SEP and NEP interplanetary missions for the future.

II. Power System Requirements

Besides the requirements of power, lifetime, and specific mass, the power supplies for interplanetary electric propulsion missions must be compatible with the launch vehicle (Space Shuttle and Centaur upper stage) and the mission specific interplanetary meteroid and radiation environments. Most NEP missions require a coast period for optimal preformance. This coast period can last for 1-5 years, during which only a small fraction (~1%) of the power is required. Power supply integration with the spacecraft must also be carefully considered. The radiation environment produced by the reactor must be compatible with dose limits of the spacecraft. The radiation dose limits for the state-of-the-art interplanetary spacecraft, i.e., the Jupiter orbiter Galileo, are 7.5×10^4 rad and 2.5×10^{10} neutron/cm^2. The power supplies must also be compatible with the plasma environment produced by the electric thrusters.

The power supply performance requirements of specific mass, power, and lifetime can be determined only for specific missions. Traditionally, this is done by letting a power supply characteristic be a parameter on a mission performance figure.[4,5] It is much more convenient for the designer to have a power supply characteristic as the dependent variable and a mission variable as a parameter on the mission performance figure, as was done in Ref. 6. The method used in Ref. 6 produces two sets of results, both of which are for a specific mission (i.e., target body) and are best presented as a function of the specific impulse of the electric propulsion system. Figure 1 is an example of these results for a NEP Neptune orbiter mission. The assumptions for this mission are: 1) a constant power source (nuclear), 2) a constant specific impulse, 3) a low-thrust escape spiral trajectory from Earth starting from a

700 km nuclear-safe orbit, 4) an optimal interplanetary trajectory containing one or more coast phases, 5) a low-thrust spiral capture trajectory into an orbit at 15.8 Neptune radii, and 6) the power supply must support (at low power) the science experiments for about 1 year after reaching the final orbit around Neptune. Figure 1 presents the required specific mass of the power and electric propulsion system divided by the electric propulsion system total efficiency (α_{pp}/η) as a function of the mission specific impulse. The values of α_{pp}/η shown in Fig. 1 are those that a power and propulsion system must exhibit in order that the mission be performed. Mission duration and total propulsion system burn time are parameters in Fig. 1. The mission duration does not include the 1 year period in the final orbit. The difference between the mission duration and propulsion system burn time is the time spent coasting during the interplanetary trajectory. The final parameter for these requirement charts is the ratio M_x/M_0 where M_0 is the initial spacecraft mass in the 700 km circular orbit at the Earth. The initial mass includes the payload, power and propulsion system mass, propellant, and propellant tanks. M_x includes only the mass of the payload and propellant tanks. A value of M_x/M_0 equal to 0.1 (used in Fig.1) is a reasonable estimate of this "payload ratio" for interplanetary missions.

Figure 1 also presents the ratio of jet power ($P\eta$) to initial mass as a function of specific impulse for the Neptune orbiter mission. Again, the value of $P\eta/M_0$ is that which is required by the mission at that value of specific impulse. In Fig. 1, the power supply full-power lifetime must be equal to or greater than the propulsion system burn time. The power supply must also be able to be throttled down (or to dissipate the generated power) during the coast phase and brought back to full power for the capture and science investigation phases of the mission. In Fig. 1, the requirements on the key power supply performance parameters (i.e., power level, specific mass, full power life, and total life) are available for the assumed mission description. Note that a range of initial mass, payload mass, specific impulse, and mission duration (flight times) is accommodated by this procedure. It is not possible to separate the power and propulsion characteristics in Fig. 1, unless some knowledge of electric propulsion technology is available. The total system efficiency for mercury ion thrusters over the specific impulse range of 2000-6000 s is approximately 55-80%. Based upon previous conceptual designs, the ratio of power supply specific mass to propulsion subsystem specific mass for NEP and SEP systems averages roughly 2 and 1, respectively.

Figure 2 presents the required value of α_{pp}/η for a representative comet rendezvous mission using SEP. The assumptions for this mission include: 1) constant specific impulse; 2) a minimum power requirement set by propulsion system hardware constraints, below which the thrusters are not operating; and 3) launch of the SEP system by the Space Shuttle/Centaur G combination. Since the mission assumes an optimal combination of launch stage and SEP system energies, the quantity $P\eta/M_0$ does not have the same significance as it does for the NEP system where the initial mass is essentially unconstrained. As a consequence, the absolute value of initial mass for the SEP examples is constrained by the launch vehicle injection capability. It is also necessary to specify an absolute value of payload mass in order to calculate α_{pp}/η for these missions. The data presented in Fig. 2 assume payloads of 600 and 1000 kg.

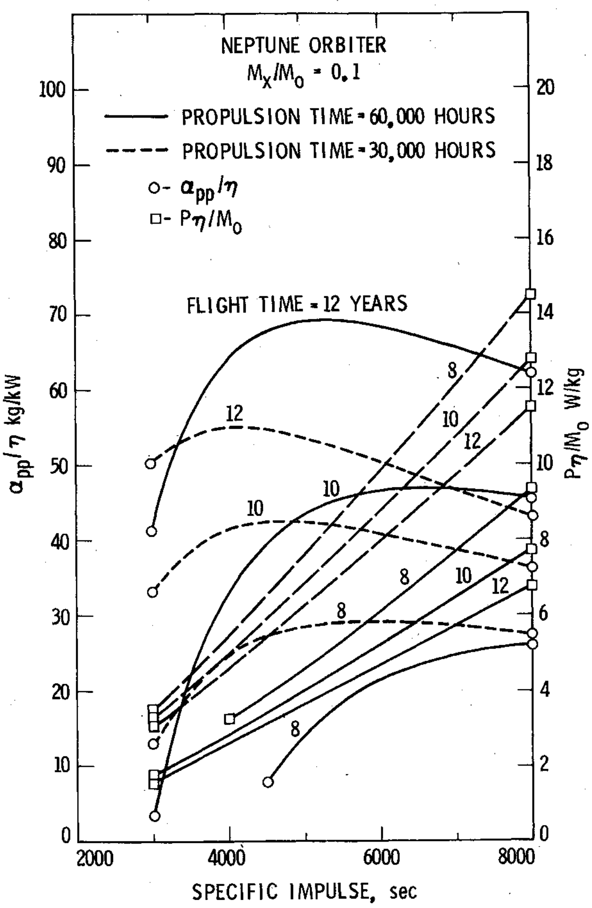

Fig. 1 Power supply requirements for a NEP Neptune orbiter mission.

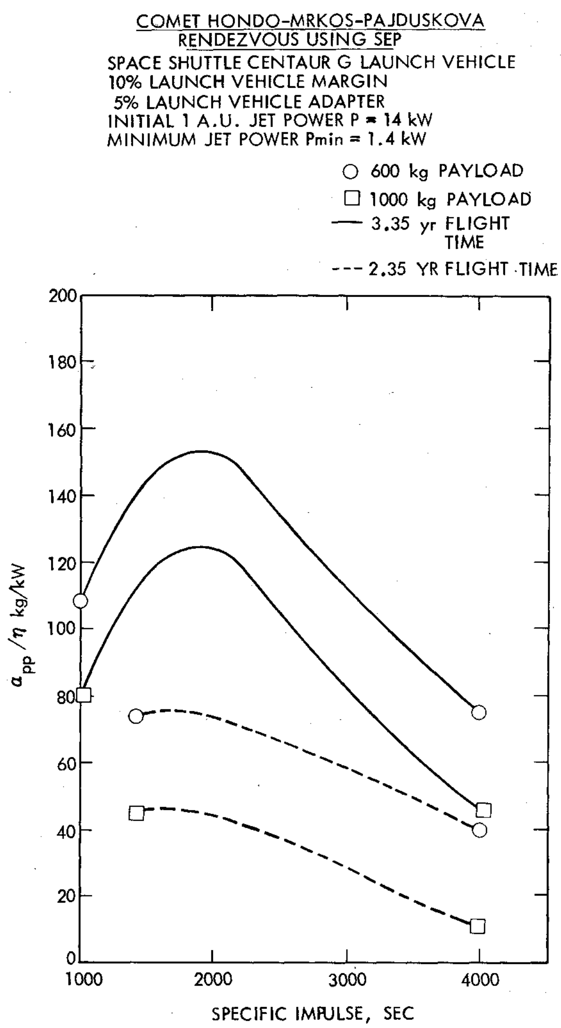

Fig. 2 Power supply requirements for a SEP rendezvous mission with HMP comet.

The initial power (P_0) at 1 a.u. and the minimum power (P_{min}) are defined as a function of the propulsion system efficiency as follows: $\eta P_0 = 14$ kW and $\eta P_{min} = 1.4$ kW. The maximum distance from the sun at which the thrust system is able to operate with the above assumptions is around 3.3 a.u. Beyond this distance, the thrust system is shut down and the spacecraft follows a ballistic trajectory.

Figures 1 and 2 are examples of NEP and SEP missions that have been studied. The Neptune orbiter is the most demanding NEP mission studied, while the Hondo-Mrkos-Pajduskova rendezvous is one of the least demanding SEP missions. Based upon the numerous past mission performance studies and the perceived requirements of future missions, the requirements on a nuclear electric power supply include: power level of 75-175 kWe, specific mass of 20-40 kg/kWe, full-power life of 3-6 years, and total system life of 3-12 years. The requirements on solar arrays for SEP missions include: power level of 25-50 kWe, specific power of 100-200 We/kg, and the cost of the solar array to be reduced by at least a factor of five relative to the SEP array.

III. Solar Electric Power Supply Status

NASA sponsored the technology for solar-powered electric propulsion for nearly a decade. This program was centered around the SEP (solar electric propulsion) array, an innovative concept developed by the Lockheed Missiles and Space Co., which was developed into prototype hardware aimed at demonstrating the salient features of the design. The mechanical portion of what began as the SEP array was successfully tested in space (September 1984) during the STS-14 mission. It is not the purpose of this paper to review that major effort; the main features of the array design (66 We/kg, 100 We/m^2) have been well documented in the literature.[7,8]

Performance (weight and size) and cost are two distinctly different, but equally important, aspects of solar power for electric propulsion. Due to the levels of power needed, it is necessary to develop an array that has very high [>50 We/kg at beginning of life (BOL)] specific power and the potential to be manufactured at significantly lower cost than conventional arrays ($1000/We at BOL). For these reasons, the SEP array was designed to minimize weight (flexible substrate, 200 μm cells, welding, Astromast deployer, use of composite structural materials) and be compatible with automated assembly processes (welding, wraparound contact cells, printed circuit bonding) that are inherently low-cost operations.

Since the initiation of the SEP program, a number of significant developments have occurred, including: 1) the implementation of advanced processes to improve greatly the efficiency, radiation resistance, and specific power of space-qualified silicon solar cells; 2) the utilization of techniques for producing GaAs solar cells that are superior to the most advanced silicon cells with respect to efficiency, radiation resistance, and high-temperature performance; and 3) perhaps most importantly, the major national commitment to terrestrial photovoltaics. From this last development has come a number of approaches to reducing the cost of space solar arrays without compromising performance and reliability.

Projections of solar array specific power based on the utilization of high-efficiency silicon solar cells in conjunction with lightweight array structures specifically designed to be dynamically compatible with ultrathin solar cell flexible blankets indicate that beginning of life performance characteristics exceeding 120 We/kg are highly likely.[9] Figure 3 shows a more ambitious strategy that, if successfully implemented, could achieve 300 We/kg at the array level. A NASA program aimed at demonstrating this goal before the end of this decade has been established.[10]

It has been demonstrated that high-efficiency (>13%) ultrathin (50 μm) silicon solar cells can be mass produced.[11] This is in large part due to the adoption of automated processing, an outgrowth of the U.S. terrestrial photovoltaics program. A new method has been developed to produce cover glasses less than 50 μm thick.[12] Preliminary evaluation of these "frosted" covers has been most encouraging. A polyimide-based encapsulant with acceptable transmission characteristics has been developed and tested for space use.[13] This material can be applied to cells using an extremely low-cost process. Screening tests (radiation, thermal vacuum, humidity, and vacuum ultraviolet) indicate that the material has potential for space applications, providing modifications in the synthesizing process can be made.

Parallel-gap resistance[14] and ultrasonic welding techniques have been used to interconnect ultrathin silicon solar cells without degrading the electrical output. New flexible substrated materials possessing near-zero coefficients of thermal expansion and better handling properties than Kapton have been produced for ultrathin cell blankets. Welded thin-cell/thin-cover modules bonded to various flexible substrate candidate materials showed no significant electrical (~1%) or mechanical degradation[15] after 2000 cycles between −175 and +65°C.

The success in demonstrating those elements necessary to achieve very high (>300 We/kg) solar array blanket specific power calls attention to the need to develop an array support and deployment structure that is much lighter than that developed for the SEP array. Figure 4 shows the impact of structure weight on the specific power of an array for various blanket designs. It is apparent that the baseline SEP structure weight limits the array performance in the case where silicon

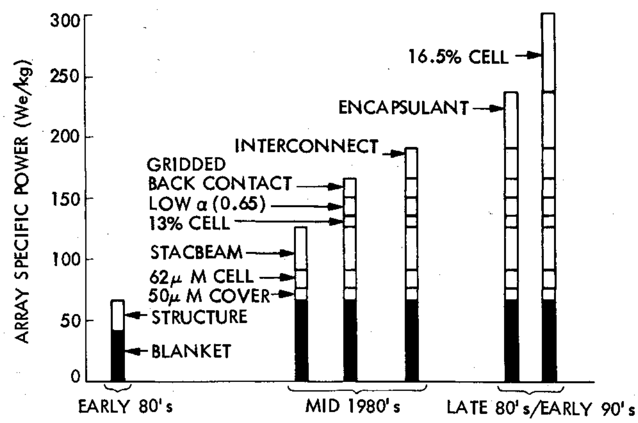

Fig. 3 Technology development strategy for a 300 We/kg beginning of life planar solar array.

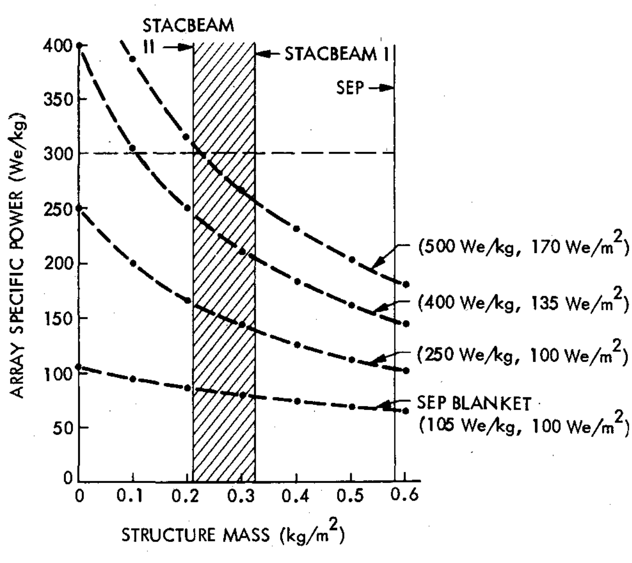

Fig. 4 Planar solar array specific power vs structure mass.

solar cells are used, because the output power per unit area of array blanket is determined by the realistic upper limit of silicon solar cell efficiency (14-15%).

A new lightweight array structure, the stacking triangular articulated compact beam (STACBEAM), is now under development. It is specifically designed to be dynamically compatible with the very lightweight blankets being developed for NASA. The details of the STACBEAM design are available in the literature.[16,17] Once the STACBEAM has been demonstrated by producing operating hardware and interfacing with the ultrathin cell blankets, which are in the final stages of development, it will be possible to demonstrate a prototype solar array capable of greater than 200 We/kg at the beginning of life.

The concept of designing an array blanket composed of materials that are highly transmissive in the infrared portion of the solar spectrum is now under evaluation. The concept has been enhanced by the development of gridded back-contact solar cells, which also have optical coatings that increase the transmission of unwanted radiation through the cell/adhesive interface.[18] If the proper combination of space-suitable materials can be identified and employed, a drop in array operating temperature resulting in a 10-15% increase in output power is anticipated without incurring a weight penalty.

Further major improvements in array specific power will probably depend on the efforts now underway to develop GaAs solar cells. Greater than 18% conversion efficiency has been reported for this device.[19] Since GaAs is a direct bandgap semiconductor material, a 5-10 μm thick solar cell is quite sufficient to obtain full conversion efficiency. Currently, the major emphasis is being placed on optimizing efficiency and radiation performance using very thick (350-400 μm) material. However, it has been experimentally demonstrated that ultrathin (<10 μm) GaAs cells with high conversion efficiencies are feasible.[20] Thus, it is not unreasonable to presume that 18% efficient GaAs solar cells with weight equal to, or less than, 50 μm silicon devices can be achieved in a relatively short time (5 years). An additional advantage offered by GaAs is its reduced sensitivity to high temperature. This is an important consideration for those applications of electric propulsion operation at less than 1 a.u.

When ultrathin GaAs solar cells become readily available, methods for assembling them into blanket modules will be available as a result of the development of manufacturing methods for producing ultrathin silicon solar cell blankets. Initial efforts to fabricate 50 μm cell-cover modules quickly showed that automated assembling approaches were necessary. A variety of automation options now exist, including at least one that is presently being used to manufacture space flight array hardware. One of the main reasons for developing this facility was the anticipation of an increased demand for solar arrays to support the needs generated by electric propulsion.

As stated previously, cost is a major consideration in developing the solar array technology necessary to satisfy the needs of electric propulsion. The original approach to SEP paid close attention to cost and was an excellent response to this challenge in view of the existing situation. The SEP program was quick to capitalize on the opportunities offered by the terrestrial photovoltaic program to enhance the baseline design. The major manifestation of this was the development of large-area wraparound contact cells that, in principle, provide cost advantages at the cell and module level. The STS-14 array experiment included a half-panel composed of 5.9×5.9 cm wraparound cells, devices greater than four times the size of the original SEP cell.

Solar concentration has been studied for use on missions beyond 1 a.u. Reference 21 has shown that concentration can potentially improve the specific power of a solar array operating between 3 and 6 a.u. It has also been suggested that the cost of a concentrator solar array can be reduced relative to an equal power planar array. The cost and performance of a concentrator solar array will have to be better than a planar array in order to balance the increased risk of employing concentration.

There is, however, one factor that might mandate the use of solar concentration for outbound missions using solar electric propulsion. It has been observed that silicon solar cells exposed to the low-intensity, low-temperature (LILT) conditions that would be experienced beyond 3 a.u. display a statistically significant and quite serious reduction in power output.[22] The cause of this phenomenon has been attributed to the random formation of a resistive layer at the metal contact/silicon interface during cell processing.[23] A solution to this occurrence has not been demonstrated. It has not yet been determined that GaAs solar cells exhibit this type of behavior under LILT conditions.

Paradoxically, the emergence of GaAs solar cells as a viable substitute for conventional silicon devices could either enhance or eliminate the concentrator option for solar-powered electric propulsion. If the GaAs cell fulfills its expectations with respect to cost and weight and the device does not suffer from the LILT effect, there will be little justification for solar concentration based on either cost or weight savings at the array level. Conversely, failure to meet anticipated objectives or evidence of the LILT phenomenon will likely force the consideration of solar concentration for some outbound electric propulsion missions.

Technology projections are difficult to make under any circumstances and are currently even more difficult for solar arrays. Although NASA has a number of impressive goals with respect to array power, performance, and cost, progress to meet these aims has been hindered by a reduction in the resources necessary to bring the many innovative conceptual approaches now available from the laboratory to the flight demonstration phase. As described previously, the technology elements to produce a 150 We/kg array at BOL are in existence, waiting to be integrated and tested in the space environment. Progress in GaAs solar cells indicates that arrays can be built that will produce 200 We/m^2 at BOL or even more if the solar concentrator array designs now being investigated prove feasible. A combination of ultrathin GaAs cells and the advanced array blanket and structure already shown to be capable of at least 150 We/kg could allow 250 We/kg arrays at the 10 kWe or greater power level. Conceptual approaches to package and deploy hundreds of kilowatts of solar array are waiting to be developed. Those factors necessary to bring about a tenfold reduction in the cost of space solar power are known. What prevents them from being executed is the lack of a sufficient demand for a standardized array module. The terrestrial photovoltaic program begun in the mid-1970s by the U.S. has already provided a convincing argument for the impact of market demand on the cost of solar power. There is little doubt that space power costs would repeat this trend if a similar situation were to develop.

The decision made by the United States to build a space station in low Earth orbit will have a tremendous impact on all areas of space technology and could provide the market demand that would stimulate some of the technology advances projected in the preceding paragraph. The need for extremely high power for the space station will drive solar array technology in the direction of large area and low cost. Unfortunately, the mass (critical for electric propulsion applications) of these high-power, low-cost arrays is not considered an important factor.

IV. Nuclear Electric Power Supply Status

This paper deals only with nuclear reactor power systems, not radioisotope power systems. This section will first briefly review the acceptability of space nuclear reactors. References 24-26 discuss safety and acceptability in more detail.

After the re-entry of the USSR nuclear-reactor-powered satellite, Comos 954, the United Nations established a working group on the use of nuclear power sources in outer space. This working group operates under the Scientific and Technical Subcommittee of the United Nations Committee on the Peaceful Use of Outer Space and has representatives from 32 nations. The working group has met four times and has received papers from several nations including the U.S., USSR, and U.K. The working group's most important conclusion was reaffirmed in their final report,[27] i.e., that nuclear power sources, including both radioisotope generators and uranium-fueled nuclear reactors, can be used safely in outer space, provided that all necessary safety requirements are met.

The U.S. has safely launched and operated 19 missions that received all or part of their power from radioisotope thermoelectric generators (RTG). A single nuclear reactor (SNAP-10A) has been orbited. All space nuclear power systems are reviewed by the Interagency Nuclear Safety Review Panel (INSRP). INSRP independently reviews the safety of space nuclear power systems and submits a report to the agency directors for review and approval. The approval sequence for space nuclear power systems ends in the office of the President.

The U.S. conducted an extensive space reactor development program in 1955-1973. All space reactor work was stopped in January 1973 due to the lack of clearly defined missions. In 1973-1979, only a small amount of nuclear power system work was conducted compared to the pre-1973 period. Table 1 (from Ref. 28) summarizes the principal space reactor programs. Also included in Table 1 are the reactor programs that were meant for nuclear rocket propulsion. These programs were very successful with the Rover program, demonstrating a cumulative test time of 879 min above 1 MWt and 255 min at much higher power levels.[29]

The major effort in reactor systems for space power was the Systems for Nuclear Auxiliary Power (SNAP) program.[30] The SNAP 10A and 8 programs were the most productive, the technology of which will be reviewed here briefly. Along with the SNAP 10A and 8 programs, the potassium Rankine, the in-core thermionic reactor program, and the 10 kWe/1150 K Brayton technology are the most applicable for the purposes of this paper and will also be reviewed briefly.

References 29-35 contain summary information of the SNAP 10A and 8 programs. SNAP 10A was designed to produce 0.5 kWe nominal power for 1 year and had a specific mass of approximately 870 kg/kWe. A SNAP 10A system was the first and only reactor power system launched by the U.S. It was launched on April 3, 1965, and successfully operated for 43 days before a voltage regulator failure caused shutdown of the experiment.[32] The reactor remains in a 1300 km circular orbit, which has a lifetime of about 3500 years. A second SNAP 10A system was operated on the ground for 10,000 h. The shield, control drum actuators, and electromagnetic pumps performed well.

SNAP 8 was designed to produce a minimum of 35 kWe for at least 10,000 h. All of the SNAP 8 Rankine cycle power conversion system components except the radiators were built and tested. The major components were tested[34] for at least 10,000 h and the complete power conversion system was tested for 7320 h. During the system test, 135 thermal cycles were obtained and the longest continuous test was 2000 h. Based upon such testing, the SNAP 8 mercury power conversion system was viewed as being technology-ready for 5 year missions.[35]

Two SNAP 8 reactors (S8ER and S8DR) were built and tested.[36] The S8ER produced between 0.4 and 0.6 MWt for 1 year with a coolant exit temperature of 980 K. The S8DR produced between 0.3 and 1.0 MWt for 7000 h at 980 K. Both reactors experienced cracks in the fuel cladding that required a change in the cladding material and a slight reduction of the reactor operating temperatures. As with the SNAP 10A program, the beryllium reflectors, control actuators, and bearings operated successfully.

Several different system configurations containing redundant power conversion systems were developed for the SNAP 8 with a power range of 35-54 kWe. For configurations applicable to unmanned electric propulsion missions, estimates of the SNAP 8 power and specific mass are 35 and 54 kWe and 250 and 185 kg/kWe, respectively.

The SNAP 10A and 8 programs demonstrated the feasibility of the uranium-zirconium-hydride reactor technology, but a higher temperature, more efficient thermal-to-electric conversion system was desired. A program was started to take advantage of the technology base of high-power, airbreathing Brayton cycle powerplants. The program goals were 10 kWe power, 1150 K turbine inlet temperature, and a 5 year design life.[31,33] Only the Brayton thermal-to-electric converter technology was developed. This Brayton system was tested in a vacuum chamber for about 2500 h and experienced 13 start/stop cycles. The measured efficiency of the system was 29%. Improvements demonstrated at the component level could have raised this efficiency to 32%. Two Brayton rotating units (BRU) were tested for 38,000 h each.[31] The BRU, supported on its gas bearings, performed successfully throughout the test period and no failure modes were identified that would prevent the attainment of the 5 year design life. This 10 kWe, 1150 K Brayton program successfully demonstrated its technology and gave confidence that similar Brayton systems could be built and used at higher power levels where the Brayton technology would be more broadbased.

Reference 37 describes several Brayton nuclear power systems for electric propulsion based upon an extrapolation of this 10 kWe, 1150 K technology. The concepts were designed for a life of 120,000 h and included two independent Brayton systems for redundancy. The reactor was a UO_2, heat-pipe-cooled design similar to the SPAR reactor (Table 1). The specific mass for systems producing 100, 400, and 1000 kWe were 29, 28, and 26 kg/kWe, respectively.

As can be seen in Table 1, the potassium Rankine cycle technology program spanned several reactor programs, the most notable of which was the SNAP-50/SPUR program. At the Lawrence Radiation Laboratory (LRL) uranium-nitride fuels for SNAP 50/SPUR were developed and tested for 6000 h at 1350 K. The SNAP 50/SPUR reactor was to have been a fast-spectrum, lithium-cooled, 2MWt reactor using refractory metals. The reactor was to operate for 2 years and provide lithium at 1480 K to boil potassium for the Rankine conversion cycle. No complete reactor and shield test took place; however, significant technology was developed and tested under the potassium Rankine technology program.[31,38,39] A potassium Rankine conversion system that was to be powered by a SNAP 50/SPUR reactor was designed.[40] The system was to have a net output of 375 kWe (at 27 kg/kWe) with a total efficiency of 18.7%. Reference 31 states that an improved design would have produced 404 kWe at 14 kg/kWe.

Detailed designs of the boiler, turboalternator, condenser, electromagnetic pumps, and radiators were completed for the 375 kWe system using refractory metals where necessary. The tantalum alloy T-111 was evaluated in two coupled potassium and lithium loops meant to simulate the system described in the preceding paragraph for 10,000 h and revealed no material problems.[31] Two turbines were built and tested[31] with saturated potassium at about 1100 K. Including the testing described above, approximately 800,000 h of potassium testing has been accumulated by the space power and terrestrial power technology programs.[38]

The final technology demonstrated in the years before the start of the SP-100 program of interest for this paper was the in-core thermionic reactor program.[31,41] Many thermionic fuel elements (TFE) have been successfully tested. For example, an electrically heated converter operated for 64 months at 1970 K with 8.0 W/cm^2 of emitter area. A TFE with two cells in series operated successfully in a reactor core for over 11,000 h at 5.6 We/cm^2 and 1820 K with uranium dioxide fuel. Another TFE with six cells in series operated in core successfully for over

Table 1 Principal U.S. space nuclear reactor programs (pre-SP-100)

Power plant	Purpose	Power level	Operating temp, K	Period	Reactor type	Fuel	Converter	Development level
Rover	Propulsion	365-5000 MWt	2,450	1955-1973	Epithermal	UC		Twenty reactors tested. Demonstrated all components of flight engine 2 h. Ready for flight engine development
Fluidized bed	Propulsion	1000 MWt	3,000	1958-1973	Thermal	UC-ZrC		Cold flow, bed dynamics experiments successful
Gaseous core reactors	Propulsion & electricity	4600 MWt	10,000 1,500	1959-1978	Fast	Uranium plasma, UF_6	Brayton	Successful critical assembly of UF6
SNAP-2	Electricity	3 kWe	920	1957-1963	Thermal	Uranium zirconium hydride	Mercury Rankine	Development level. Tested two reactors with longest test reactor operated 10,500 h. Percursor for SNAP-8 and 10A
SNAP-10A	Electricity	0.5 kWe	810	1960-1966	Thermal	Uranium zirconium hydride	Thermoelectric	Flight tested reactor 43 days. Tested reactor with thermoelectrics in 417-day ground test
SNAP-8	Electricity	30-60 kWe	975	1960-1970	Thermal	Uranium zirconium hydride	Mercury Rankine	Tested two reactors. Demonstrated 1 year operation. Non-nuclear components operated 10,000 h and breadboard 8700 h
Advanced hydride reactors	Electricity	5 kWe	920	1970-1973	Thermal	Uranium zirconium	Thermoelectric Brayton	PbTe thermoelectrics tested to 42,000 h
Medium power reactor experiment	Electricity	140 kWe	1,365	1961-1966	Fast	UO_2	Potassium Rankine	Stainless steel loop tested up to 2500 h. Fuel tested 14,000 h
SNAP-50/SPUR	Electricity	300-1200 kWe	1,365	1962-1965	Fast	UN, UC	Potassium Rankine	Fuels test to 6000 h
Advanced metal-cooled reactor	Electricity	300 kWe	1,480	1965-1973	Fast	Uranium nitride	Brayton and Potassium Rankine	Non-nuclear potassium Rankine cycle components demonstrated to 10,000 h. Ready for breadboard loop
710 gas reactor	Electricity & propulsion	200 kWe	1,445	1962-1968	Fast	UO_2	Brayton	Fuel element tested to 7000 h
In-core thermionic reactor	Electricity	5-20 kWe	2,000	1959-1973	Fast or thermal driver	$U0_2$ Uc-ZrC	In-core thermionics	Integral fuel element, thermionic diode demonstrated 1 year operation
Nuclear electric propulsion	Electricity	400 kWe	1,675	1974-1981	Fast	UO_2	Out-of-core thermionics	Limited testing of thermionic elements
SPAR	Electricity	100 kWe	1,500	1979-1982	Fast	UO_2	Thermoelectric	Limited testing of core heat pipes and advanced thermoelectric materials

8000 h at 3.0 We/cm^2 between 1740 and 1820 K, again with uranium dioxide fuel. Several other TFEs have operated at similar temperatures and power densities for thousands of hours ($\sim$8000) and have failed due to fuel swelling and other reasons. Reference 42 describes several in-core thermionic reactor power systems for electric propulsion. The power levels and specific masses were 70-270 kWe and 20-13.7 kg/kWe.

The preceding paragraphs have summarized five of the most significant technology development programs for nuclear power systems prior to the SP-100 program. The triagency (U.S. Department of Energy, U.S. Department of Defense, and NASA) SP-100 program[43] started in early 1983. The major program goal is to develop technology and system concepts for a nominal 100 kWe space nuclear power system. Communications, radar, and a growth space station are viewed as the most likely first users. Nuclear electric propulsion is also viewed as a potential user.

The SP-100 program is currently in the critical technology assessment and development phase. This phase is due to end in 1985. The next phase, called the ground test phase, is scheduled to start in October 1985 and last 5 years. It is anticipated that the first flights will take place in the mid-1990s. Within the SP-100 program, no item has higher priority than safety. Program policy includes the guidelines that the SP-100 power system will be launched cold (subcritical) and be operated only in nuclear safe orbits. Nuclear safe orbits are those where the natural decay time of the orbit due to atmospheric drag is long enough to allow the reactor radioactivity to diminish to safe levels before re-entry.

The SP-100 program has two major activities: system design and technology development. In order to compare different system designs, the following primary system requirements were established: 1) power level, 100 kWe; 2) mass, 3000 kg; 3) stowed size, one-third of the Shuttle payload bay length; 4) full power life, 7 years; 5) total system life, 10 years; and 6) full-power, 7 year radiation dose at 25 m from the reactor, 5×10^5 rad (Si) and 1×10^{13} neutrons/cm^2. Three system design approaches are being pursued, all of which use a fast spectrum reactor. The first system design employs a lithium-cooled reactor and thermoelectric conversion. The lithium is circulated by electromagnetic pumps and the thermoelectric elements are radiatively coupled to a secondary cooling loop. Waste heat is removed by a deployable heat pipe radiator that is conductively coupled to the thermoelectric elements. The second system concept uses in-core thermionic conversion and a mixture of sodium and potassium (NaK) as the coolant in a pumped loop. Waste heat is removed again by a heat pipe radiator. The final system concept uses a low-temperature reactor (1000 K compared to 1500-1700 K for the static systems) and a Stirling engine conversion system. Again, the reactor is cooled by a pumped NaK loop and a heat pipe radiator.

Technology development in the SP-100 program is focused on the conversion systems and reactors. To meet the system design requirements, the thermoelectric figure of merit must be raised from the state-of-the-art value of about $0.7\times10^{-3}\text{K}^{-1}$ to at least $1.0\times{}^{-3}\text{K}^{-1}$. The operating temperature of the thermoelectric converter must also be raised several hundred degrees kelvin from the current limit of 1280 K. Boron carbon (p type) and lanthanum sulfur (n type) are the thermoelectric materials that are being developed to meet these technology goals.

The two areas in thermionic conversion technology being addressed are: 1) emitter swelling at high temperature (1700-1800 K) that causes electrical shorting and 2) neutron irradiation degradation of the electric insulators that causes electric shorting. One potential solution to the emitter swelling problem is to increase the gap between the emitter and collector from 0.25 to 0.50 mm. To maintain the converter performance, the collector may be oxygenated.

Free-piston Stirling engine development has recently begun. The goals for this development within the SP-100 program include: 1) engine scalability from the current level of 3 to 25 kWe, 2) a demonstration of an efficiency 70% of Carnot at hot-to-cold temperature ratios of 1.5/2.0, and 3) establishing the feasibility of producing engines with a specific mass of 6 kg/kWe.

One of the major areas of research in the area of nuclear technology is the area of coolant/cladding/fuel compatibility. The system concepts use either UO_2 or UN fuel and lithium, sodium, or NaK coolants. Potential cladding materials include molybdenum/rhenium, tungsten/rhenium, and tantalum alloys. The rates at which oxygen and nitrogen diffuses through the cladding materials and into the coolant is being determined. An understanding of the physical and mechanical properties of refractory alloys before and after irradiation will be obtained.

Besides the 100 kWe systems, there is current interest in multimegawatt (MMWe) nuclear electric power systems. In the MMWe regime the fixed-bed reactor (FBR),[44] which uses a gas-cooled uranium dioxide particle fuel, is perhaps the best known concept. A very promising concept for a low-specific-mass MMWe power system combines the FBR and a state-of-the-art (temperature) Brayton thermal-to-electric conversion subsystem (based upon the Brayton technology discussed previously in this section) and an innovative radiator concept. The radiator concept called a liquid droplet radiator (LDR)[45] eliminates the need for radiator micrometeroid protection (a major contributor to radiator mass) by using a stream of small mass-to-area ratio liquid droplets.

The LDR uses a low-vapor-pressure liquid (tin, tin-lead-bismuth eutectics, vacuum oils) in a recirculating system in which the droplets are projected through space over a distance of tens to hundreds of meters from a droplet generator to a droplet collector. Due to the potential very low mass per unit area of the LDR, the power system can be operated at low-heat-rejection temperatures (high areas) that can increase the thermodynamic cycle efficiency or enable the use of low-temperature ($\sim$1000 K) reactor and conversion technologies. Reference 46 describes a FBR/Brayton/LDR concept that could produce 8.7 MWe with a specific mass of 1.9 kg/kWe.

Figure 5 presents the specific mass and power of over 20 conceptual nuclear power systems. These concepts were designed specifically for an electric propulsion application or are compatible with an electric propulsion mission, i.e., long life and non-man-rated shadow shields. These system concepts span over 20 years and some are clearly suspect in the light of current technology. In any event, certain trends are clear, i.e., 100 kWe, 400 kWe, and 10 MWe were the favorite power levels and the system-specific mass scales favorably with the power. Based upon the information of Fig. 5 and the past and current state of nuclear power system development in the U.S., a nuclear power system producing between 50 and 150 kWe with a specific mass of between 50 and 30 kg/kWe could

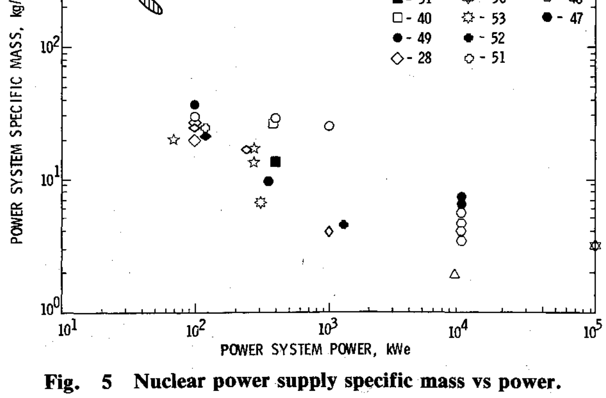

Fig. 5 Nuclear power supply specific mass vs power.

be ready for use in the early 1990s. This first system would probably not have the long life required for the more ambitious outer planet explorations. Starting at the same point and noting the interest in MMWe systems and the good technology base in gas-cooled reactors and Brayton conversion systems, a MMWe system in the 1-10 MWe range could be ready in the same time frame at a specific mass below 10 kg/kWe.

V. Summary and Conclusions

This paper has presented power supply requirements for interplanetary electric propulsion missions and has reviewed the technical status of both nuclear electric and solar electric power supplies. The requirements on the power supply specific mass were presented in the form of the total power and propulsion specific mass divided by the total electric propulsion system efficiency. From this parameter (α_{pp}/η), the required power supply specific mass can be determined using information about electric propulsion system technology. The power level was presented in the form of jet power divided by the vehicle initial mass. The power supply requirements of specific mass, power, and lifetime are very mission dependent. For any given mission, there is an optimum value of the electric propulsion system specific impulse that minimizes the requirements on the power supply specific mass.

With the inception of the SP-100 and space station programs and the successful Space Shuttle test of some features of the SEP solar array, the outlook for high-power space power supplies in the U.S. is brighter than at any time in the past. A solar array at 25 kWe and 66 We/kg and a nuclear power system at 100 kWe and less than 30 kg/kWe with at least a 40,000 h full-power lifetime will be the critical item to go along with the well-developed 30 cm mercury ion thruster technology to complete a viable primary electric propulsion system. Assuming vigorous and successful development programs, power supplies meeting or surpassing the requirements stated above are possible. By the early 1990s, nuclear power supplies characterized by and 50-150 kWe, and 50-30 kg/kWe, and megawatt power supplies in the 1-10 MWe range with a specific mass less than 10 kg/kWE are possible. Solar photovoltaic arrays characterized by 200 We/m^2, 150-250 We/kg, and 10-100 kWe are possible before 1990.

Finally, although there has been tremendous progress in power supply development over the past 20 years, technology development directly oriented to electric propulsion applications has been slow. Some programs do exist that can indirectly support this technology, such as the NASA high-performance solar array development effort for geosynchronous applications and the SP-100 program. However, power supply development for electric propulsion applications will benefit greatly when the electric propulsion community identifies specific mission opportunities, implying a set of requirements that will focus space power technology development, and develops a constituency for their objectives outside of the electric propulsion community.

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References

[1]Mickelsen, W. R., "Auxiliary and Primary Electric Propulsion, Present and Future," *Journal of Spacecraft and Rockets,* Vol. 4, Nov. 1967, pp. 1409-1423.

[2]Jones, R. M. and Sauer, C. G., "Advanced Solar Electric Propulsion (ASEP) for Planetary Missions," Paper 84-47 presented at 17th International Electric Propulsion Conference, Tokyo, 1984.

[3]Jones, R. M. and Sauer, C. G., "Nuclear Electric Propulsion Missions," *Journal of the British Interplanetary Society,* Vol. 37, August 1984, pp. 395-400.

[4]Stearns, J. W., "Electric Propulsion Requirements for Planetary and Interplanetary Spacecraft," Jet Propulsion Laboratory, California Institute of Technology, Pasadena, Rept. TR-32-403, March 1964.

[5]Mickelsen, W. R., "Power Needs for Electric Propulsion," *Proceedings of the 3rd IECEC,* IEEE, New York, Aug. 1968, p. 1033.

[6]Nagorski, R., "Advanced Power and Propulsion Study, Phase I: Nuclear Electric Applications," Jet Propulsion Laboratory, California Institute of Technology, Pasadena, Pub. 715-99, Dec. 1980.

[7]Elms, R. V. Jr. and Young, L. E., "SEP Full-Scale Wing Technology Development," *Proceedings of the 12th IECEC,* American Nuclear Society, La Grange Park, Ill., Aug. 1977.

[8]Elms, R. V. Jr. and Young, L. E., "SEP Solar Array Development Testing," *Proceedings of the 14th IECEC,* American Chemical Society, Washington, D.C., Aug. 1979, p. 1279.

[9]Scott-Monck, J. A., "Prospects for Enhancing SEP Array Performance," *Space Photovoltaic Research and Technology Conference,* NASA CP 2169, Oct. 1980, p. 351.

[10]Mullin, J. P. et al., "Advances in Space Power Research and Technology at NASA," *Proceedings of the 16th IECEC,* ASME, New York, Aug. 1981, p. 355.

[11]Lindmayer, J. and Wrigley, C. Y., "Ultrathin Silicon Solar Cells," *Proceedings of the 13th IEEE Photovoltaic Specialists Conference,* IEEE, New York, June 1978, p. 450.

[12]Westney, R. P., Bessey, J. S., and Sachs, I. M., "Textured Solar Cell Covers for Light Weight and High Performance," *3rd European Symposium on Photovoltaic Generators in Space,* European Space Agency, Paris, ESA SP-173, May 1982, p. 223.

[13]Naselow, A. B., Dupont, P. S., and Scott-Monck, J. A., "Development of a New Integral Solar Cell Protective Cover," AIAA Paper 83-0076, Jan. 1983.

[14]Mesch, H. G. and Rockey, D. E., "High Temperature, Low Mass Solar Blanket Development," *Proceedings of the 14th IECEC,* American Chemical Society, Washington, D.C., Aug. 1979, p. 1264.

[15]Patterson, R. E., Mesch, H. G., and Scott-Monck, J. A., "Progress in Developing Ultrathin Solar Cell Blanket Technology," *Proceedings of the 17th IEEE Photovoltaic Specialists Conference,* May 1984, p. 320.

[16]Rockey, D. E., Hedgepeth, J. M., and Adams, L., "High Performance Silicon Solar Arrays Employing Advanced Structures," *Proceedings of the 16th IECEC,* ASME, New York, Aug. 1981, p. 374.

[17]Adams, L., "STACBEAM: An Efficient, Low Mass, Sequentially Deployable Structure," *Proceedings of the 17th IECEC,* IEEE, New York, Aug. 1982, p. 1578.

[18]Ho, F. et al., "Recent Advances in Thin Silicon Solar Cells," *Proceedings of the 16th IEEE Photovoltaic Specialists Conference,* IEEE, New York, Sept. 1982, p. 156.

[19]Kamath, G. S., "GaAs Solar Cells for Space Application," *Proceedings of the 16th IECEC,* ASME, New York, Aug. 1981, p. 416.

[20]Fan, J. C. C., Bozler, C. O., and McClelland, R. W., "Current Status of Thin-Film Cleft GaAs Solar Cells," *Space Photovoltaic Research and Technology,* NASA CP 2256, April 1982, p. 91.

[21]Baker, I., "Enhancement of Solar Arrays in Space Through the Use of Concentrators," *Proceedings of the 14th IECEC,* American Chemical Society, Washington, D.C., Aug. 1979, p. 1278.

[22]Liebert, C. H., "Solar Cell Performance at Jupiter Temperature and Solar Intensity," *Proceedings of the 7th IEEE Photovoltaic Specialists Conference,* IEEE, New York, Nov. 1968, p. 92.

[23]Weizer, V. G. et al., "On the Cause of the Flat-Spot Phenomenon Observed in Silicon Solar Cells at Low Temperatures and Low Intensities," *3rd European Symposium on Photovoltaic Generators in Space,* European Space Agency, Paris, ESA SP-173, May 1982, p. 65.

[24]Buden, D. and Bennett, G. L., "On the Use of Nuclear Reactors in Space," *Physics Bulletin,* Vol. 33, No. 12, 1982, pp. 432-434.

[25]Bennett, G. L., "Overview of the U.S. Flight Safety Process for Space Nuclear Power," *Nuclear Safety,* Vol. 22, No. 4, July-August 1981, pp. 423-434.

[26]Buden, D., *The Acceptability of Reactors in Space,* Los Alamos Scientific Laboratory, N. Mex., Rept. LA-8724-MS, April 1981.

[27]United Nations, "Report of the Working Group on the Use of Nuclear Power in Outer Space on the Work of its Third Session," United Nations, New York, UN Doc. A/AC.105/287, Feb. 1981.

[28]Boudreau, J. E. and Buden, D., "A New Generation of Reactors for Space Power," *Proceedings of a Symposium on Advanced Compact Reactor Systems,* National Research Council, Contract N0014-82-C-0373, Nov. 1982.

[29]Layton, J. P. et al., "A Survey of High Temperature, Compact Nuclear Fission Reactors," Techno-Systems Analysis Corp. Rept.

82-001 prepared for the Committee on Advanced Nuclear Systems Energy Engineering Board, Commission on Engineering and Technical Systems, National Research Council, Sept. 1982.

[30]Staub, D. W. et al., "SNAP Programs Summary Report," Atomic Energy Commission, Rept. AI-AEC-13067, July 1973.

[31]English, R. E., "Power Generation from Nuclear Reactors in Aerospace Applications," NASA TM 83342, Nov. 1982.

[32]Wilson, R. F. and Dieckamp, H. M., "What Happened to SNAP 10A?," *Astronautics and Aeronautics,* Vol. 3, Oct. 1965, p. 60.

[33]Lubarsky, B., "Nuclear Power Systems for Space Applications," *Advances in Nuclear Science and Technology,* Vol. 5, 1971, pp. 224-319.

[34]English, R. E., "Technology for Nuclear Dynamic Space Power Systems," *Aerospace Nuclear Applications,* American Nuclear Society, La Grange Park, Ill., April 1970, p. 283.

[35]Derow, H., "SNAP-8 Power Conversion System Breadboard Assembly—Materials Evaluation after 8700 hours Operation," *Proceedings of the 5th IECEC,* American Nuclear Society, La Grange Park, Ill., Sept. 1970, pp. 11-24.

[36]Wilson, R. F., "The ZrH Reactor and Thermoelectric Conversion Systems," *Aerospace Nuclear Applications,* American Nuclear Society, La Grange Park, Ill., April 1970, p. 303.

[37]Harper, A., "Study of Reactor Brayton Power Systems for Nuclear Electric Spacecraft," NASA CR-168942, Sept. 1979.

[38]Zimmerman, W. F., "Alkali Metal Space Power Technology Applicable to National Energy Research and Development," *Journal of Energy,* Vol. 1, May-June 1977, p. 159.

[39]Manson, S. V., "A Review of the Alkali Metal Rankine Technology Program," *Journal of Spacecraft and Rockets,* Vol. 5, Nov. 1968, p. 1244.

[40]Heller, J. A. et al., "Study of a 300 KWe Rankine-Cycle Advanced Nuclear Electric Space Power System," NASA TMX 1919, Nov. 1969.

[41]Synder, H. J., "Some High Temperature Reactor Technologies," *Proceedings of a Symposium on Advanced Compact Reactor Systems,* National Research Council, Contract N0014-82-C-0373, Nov. 1982.

[42]Mondt, J. F., "Thermionic Reactor Power Systems for Electric Propulsion," *Proceedings of the 5th IECEC,* American Nuclear Society, La Grange Park, Ill., Sept. 1970, p. 14-11.

[43] Davis, H. S., "Progress in Space Nuclear Reactor Power Systems Technology Development: The SP-100 Program," AIAA Paper 84-1132, 1984.

[44]Powell, J. R. and Botts, T. E., "Particle Bed Reactors and Related Concepts," *Proceedings of a Symposium on Advanced Compact Reactor Systems,* National Research Council, Contract N0014-82-C-0373, Nov. 1983.

[45]Mattick, A. T. and Hertzberg, A., "The Liquid Droplet Radiator — An Ultralightweight Heat Rejection System for Efficient Energy Conversion in Space," *Acta Astronautica,* Vol. 9, No. 3, 1982, pp. 165-172.

[46]Beals, G. et al., "Light-Weight Nuclear-Powered OTV Utilizing a Liquid Droplet Radiator," AIAA Paper 83-1346, June 1983.

[47]Pitts, J. H. and Jester, M. H. L., "Rankine Cycle Systems Studies for Nuclear Space Power," *Proceedings of the 3rd IECEC,* IEEE, New York, Aug. 1968, p. 290.

[48]Seikel, G. R. and Nichols, L. D., "Potential of Nuclear MHD Electric Power Systems," *Journal of Spacecraft and Rockets,* Vol. 9, May 1972, pp. 322-326.

[49]Morgan, R. E., Chi, J. W. H., and Pierce, B. L., "Comparison of High Power Nuclear Systems for Space Applications," *Proceedings of the 18th IECEC,* American Institute of Chemical Engineers, New York, Aug. 1983, p. 977.

[50]Nock, K. T. and Garrison, P. W., "Final Report of the Advanced Power and Propulsion Study Phase II—Nuclear Electric Propulsion Implication," Jet Propulsion Laboratory, California Institute of Technology, Pasadena, JPL Doc. 725-71, Nov. 1981.

[51]*Nuclear Electric Propulsion Mission Engineering Study—Final Report* Vol. I, "Executive Summary," NASA CR-131599, March 1973.

[52]Beale, R. J., "The Conceptual Design of a Nuclear Electric Power Spacecraft for the Exploration of Jupiter," Jet Propulsion Laboratory, California Institute of Technology, Pasadena, Rept. JPL TR-32-231, May 1962.

[53]Beale, R. J. and Speiser, E. W., "A Nuclear Electric Spacebus for Planetary Landing Missions, Jet Propulsion Laboratory, California Institute of Technology, Pasadena, Rept. JPL TR-32-231, May 1962.

[54]Peelgren, M. L. and Mondt, J. F., "Advanced Design Concepts in Nuclear Electric Propulsion," Jet Propulsion Laboratory, California Institute of Technology, Pasadena, Rept. JPL TM-33-664, Aug. 1974.

[55]Buden, D. and Angelo, J. A., "Space Reactors—Past, Present and Future," *Proceedings of the 18th IECEC,* American Institute of Chemical Engineers, New York, Aug. 1983, p. 61.

[56]Loeb, H. W., "Electric Propulsion for Manned Spacecraft of the Future," AIAA Paper 81-0707, April 1981.