Journal of Spacecraft and Rockets

VOLUME 2 SEPTEMBER-OCTOBER NUMBER 5

Space Power Systems State of the Art

G. C. SZEGO
Institute for Defense Analyses, Arlington, Va.

THE technology of space power systems is playing an increasingly important role in determining our capabilities for space travel. A space vehicle has neither utility nor purpose without electric power. It can be said that power systems technology has replaced propulsion technology as the pacing item for our space programs. Great attention will be focused on the rate of progress in this important area. This paper surveys the state of the art of the space power systems and components and associated technologies in the United States. Wherever possible, details are given for systems under preliminary consideration which are not yet government funded. Some of the data shown are thus preliminary in nature and should be so interpreted. Reference 36 represents a selection of papers from the Second Biennial Space Power Systems Conference and would be an excellent bibliography for this review.

Solar Power Systems

Photovoltaic Conversion (Solar Cells)

Long-life satellites and probes have been almost exclusively dependent upon the solar cell for power. Even if no improvements are made in the presently available silicon solar cell, it probably will dominate the space power field for at least five more years. About one million silicon solar cells were delivered to users during the fiscal year July 1, 1962–June 30, 1963. Most of these were of the boron-diffused P/N variety, but large quantities of phosphorus-diffused N/P cells (superior in radiation resistance) are becoming available. Electrical performances for the two types of cells are comparable, with the useful yields from production runs falling between 10.5 and 11.5% for air mass zero or 12 and 13.5% for air mass unity. Price differential between the two types has been narrowed to about 10%, the N/P being more expensive. Satellite data have firmly established the superiority of N/P cells with respect to life in the radiation-belt environment. Bare cells cannot be flown on long-life missions; 6 mils of cover glass appears to be a practical minimum from the standpoint of protection from energetic particles and the handling of the cover slips during array construction.

Gallium arsenide, the nearest competitor to Si, has been developed in small-lot production, showing efficiencies around 8 to 9% for air mass unity. A few cells have exceeded 10 and 11% in ground sunlight. Although 6-mil shield

Dr. George C. Szego was graduated from City College, New York, and University of Denver, and received his M.S. and Ph.D. degrees in Chemical Engineering from the University of Washington. Currently, he is a Project Leader for areas of Space Technology at the Institute for Defense Analyses. Formerly, he was a Senior Member of the Technical Staff, Space Technology Laboratories, Inc. He has published many reports and articles in both open and classified literature and is co-author of a book on *Free Radicals as High Energy Propellants*, co-editor of a forthcoming book on space power systems, author of a forthcoming textbook on space power systems technology, and co-author of a forthcoming volume on propulsion. He has been an invited lecturer at the U. S. National Bureau of Standards, the U. S. Air Force Academy, the Faraday Society, the Czech Academy of Sciences, several international congresses, and many universities. He is a Member of the American Institute of Chemical Engineers and former Chairman of its National Program Committee, an Associate Fellow Member of the AIAA, a member of the AIAA Publications Committee, a member and past Chairman of the AIAA Committee on Electric Power Systems, and a Fellow of the British Interplanetary Society, Sigma Xi, and Tau Beta Pi.

Presented as Preprint 64-525 at the 1st AIAA Annual Meeting, Washington, D. C., June 29–July 2, 1964; revision received July 1, 1965. The author wishes to express his gratitude to the following individuals for their active assistance in the preparation of this study: E. Cohn, NASA; B. B. Rosenbaum, Bureau of Ships, Navy Department; N. Rasor, Thermo-Electron Engineering Corporation; K. P. Johnson, Aero-jet General Nucleonics; J. E. Taylor, Thompson Ramo Wooldridge; E. Zwick, Northridge, Calif.; D. Gignoux, Cosmic, Inc.; J. Neustein, Electro-Optical Systems, Inc.; D. Harvey, The Martin Co.; H. Dieckamp, Atomics International; J. A. Satkowski, Office of Naval Research, Navy Department; W. Cherry, Goddard Space Flight Center; P. Rappaport, Radio Corporation of America; M. Wolf, Heliotek, and the entire AIAA Electronic Power Systems Committee.

Table 1 Spacecraft voltaic power system characteristics

	NIMBUS "B"	EGO	OSO	OAO	UK-1	Expl. XII	Expl. XIV	RELAY	TIROS[b]	TIROS[c]	IMP	EARLY BIRD
Array oriented	Yes	Yes	Yes	No	No	No	No	No	No	No	No	No
Thousands of solar cells[a]	11.0	32.5	1.87	53.0	4.26	5.6	6.14	8.4	9.12	9.12	11.52	6.0 N/P
Cell mounting type	Panel	Panel	Panel	Paddle	Paddle	Paddle	Paddle	Body	Body	Body	Paddle	Body
Wt of array, lb	64	127	5.2	222	8.8	11.0	12.8	25.8	24.5	24.5	26.5	8.7
Total wt , system, lb	177	201	35.9	399	22.8	17.3	19.1	53.8	64.5	64.5	33.2	8.7
Spec. power, system, w/lb	2.3	2.8	0.86	1.94	0.52	1.2	1.8	0.65	0.79	0.39	2.2	5.2
Power/area, w/ft^2	9.5	7.2	7.7	4.14	1.1	1.3	2.3	2.0	2.9	1.4	2.6	3.1
Efficiency, %	7.3	5.5	5.9	3.2	0.85	1.0	1.8	1.5	2.2	1.1	2.0	2.3

[a] 2 cm x 2 cm cells.
[b] At 90° inclination to sun.
[c] At 45° inclination to sun.

GaAs cells are more radiation-resistant and have better temperature properties than Si cells, the current state of the art does not warrant their use on today's spacecraft mission, as might certain future missions involving high-temperature and high-radiation-field environments. Thin-film cells made by evaporation of CdS and CdTe have shown sunlight efficiencies of 4 to 5% for small areas (cm^2) but only 1 to 2% for large (36 in.^{2}) areas. All of the other materials, such as GaAs, InP, Se, GaP, are still in a primitive state of development. However, the way to extremely lightweight, low-cost solar cells appears to be in this direction.

Typical properties of spacecraft solar cell systems are shown in Table 1. The efficiencies are based upon solar array output under the optimum condition of the spacecraft aspect to sun angle; losses incurred by the power system's converters are not taken into account. Table 2 portrays the approximate dosages (ψ_c) of various monochromatic energetic particles necessary to degrade the sunlight maximum power to 75% of its initial value for N/P and P/N silicon cells, and for P/N GaAs cells. Solar cell power systems can be designed to provide electrical power for at least several years in the radiation belts by shielding and/or overdesigning. Radiation resistance varies appreciably among lots; hence, a better understanding of the effects of trace impurities, base resistivity, and impurity profile will lead to more radiation-resistant devices. The heavily protected Vanguard I has operated for over $5\frac{1}{2}$ years. Present cost is 5 to \$6/1 × 2 cm cell which corresponds to 250/w to \$300/w for unassembled Si cells. Laboratory-built GaAs solar cells cost around 150 to \$300/1 × 2 cm unit. Including the cost of substrates, cover glasses, and the cost of array assembly, Si cells range from \$1000/w for oriented arrays to as much as \$8000/w for spin-stabilized spacecraft. Automation techniques and cell standardization could reduce these costs substantially.

Solar Concentrators

For low- and moderate-power systems for long duration, solar concentrators should have weight and cost advantages, if they can be made practical and reliable. A sizable development effort is under way to demonstrate practical systems; this work ranges from studies of large and advanced mirrors through orbital flight test. At least one experimental concentrator has already been placed in an earth orbit. Typical characteristics of some of the mirrors that have been built and ground tested are given in Table 3 (from Ref. 2).

Solar power systems using concentrators can be divided conveniently into two categories: "low-temperature" systems using thermoelectric converters or dynamic engine-generator units, with absorber temperatures of 1400° to 2200°R; and "high-temperature" thermionic systems with absorber temperatures of 2900° to 3600°R. Only one-piece rigid mirrors have been made with accuracies suitable for use in thermionic systems; 5-ft-diam nickel electroformed mirrors have been made which can focus over 85% of the incident solar radiation through an absorber aperture of 0.5 in. in diameter. The accuracies obtainable with unfurlable petal or foam-rigidized mirrors, although not adequate for thermionic systems, appear suitable for thermoelectric or dynamic engine-generator systems.[3] The inflatable mirrors are of marginal interest because of low efficiencies and difficulty in maintaining inflation in space for extended periods. Three programs on considerably larger mirrors are under way (Table 4). Two of them involve automatically unfurlable mirrors. The third is for the development of a large, rigid, one-piece mirror for launching by a Saturn booster.

In an attempt to answer the question of how well these mirrors will withstand the rigors of the launch and space environment, three programs have been initiated by the Air Force and NASA (Table 5). The EROS program was canceled after the first launch, a Fresnel collector experiment. Telemetry data indicated only that the mirror had unfolded. The former reflector orbital experiment (ROE) program, now called ROE-1B, consists currently of a materials sampling program supporting the advanced solar turboelectric conversion (ASTEC) concept. Nevertheless, it is expected that concentrating space power systems will ultimately be successful and useful.

Solar Power System Status

Solar power plants (Table 6) utilize heat storage for periods when the vehicle is out of direct sun. The heat storage requirement currently limits solar power systems to a few peak cycle temperatures; e.g., the fusion temperatures of LiH and LiF are 1250° and 1550°F, respectively. These temperatures are consistent with hydride-moderated and uranium carbide reactors, so that conversion equipment senses about the same inlet conditions in both solar and nuclear systems. The LiF heat storage is much more attractive than LiH for Stirling and Brayton cycle use, whereas Rankine systems can use both, but not with the same working fluid; however, LiF has less than half of the heat fusion of LiH, so that weight advantages because of the higher tem-

Table 2 Approximate dosage that will reduce maximum power GaAs and Si solar cells to 75% of initial value

Bombarding particle and energy	Φ_c (particles/cm^2) × 10^{-11}		
	GaAs	Si N/P	Si P/N
0.8-Mev electrons	11,000	13,000	400
5.6-Mev electrons	2,700	300	20
0.1-Mev protons	~10	~100	~100
0.4-Mev protons	<1	~1	~1
1.8-Mev protons	24	1.3	0.4
17.6-Mev protons	57	4	0.7
95.5-Mev protons	>20	7	2

Table 3 Examples of fabricated solar collectors with Al reflective material[2]

Type	Construction	Rim angle, deg	Diam, ft	Measured reflectivity	Unit wt, lb/ft^2	Developer^b
Fresnel	Electroformed Ni	40.0	4.0	0.85	0.46	GM
Inflatable	Stretchformed Mylar	53.1	4.3	...	0.03^a	EOS
Inflatable	Mylar gores, foam-rigidized	60.0	10.0	...	3.82	GY
One-piece	Electroformed Ni	61.5	5.5	0.89 - 0.90	0.96	EOS
	Cast epoxy Fiberglas	60.5	5.0	0.755± 0.1	0.94	BC
Petal	Electroformed Ni	53.1	4.3	...	~0.3^a	EOS
	Stretchformed Al honeycomb	45.0	15.8	0.86	1.00	GE
	Welded lattice truss Al	45.0	10.0	...	0.29^a	R
	Stretchformed Al honeycomb	52.0	32.2	0.85	0.18	TRW
Umbrella	60 Al ribs and Mylar	90.0	10.0	0.83	0.11^a	LRC

^{a}Weight of primary construction material only.
^{b}GY = Goodyear; for other abbreviations, see Table 11.

perature are substantially offset by LiF weight. It has been demonstrated that H disassociation from the LiH because of temperature and cycling is less than 1 lb per year in Sunflower tests.[4] There has been no hydrogen embrittlement of materials at these high temperatures. The practicality of LiH has been demonstrated, whereas work with LiF is in process. The crossover point between thermal and electrical storage appears to be about 2.7% efficiency of the conversion device; below this figure, electrical storage is lighter, and vice versa.

The power conversion portion of a solar powerplant is similar to that of a nuclear plant. Much of the subsequent discussion on nuclear systems applies also to solar systems. Equipmentwise, this is an advantage, for the same conversion equipment may be used for either solar or nuclear applications. Another cycle of special interest to the solar plant is the Stirling cycle. This system utilizes an inert gas in a closed loop with a piston engine instead of a turbine and has high efficiency at low power levels. A conversion system efficiency of 30% and life >1000 hr at 4 kw have been demonstrated. Another advantage is that most moving parts operate at only 250°F. Principal problems are heat rejection at low temperatures (150°–200°F) and heat transfer in the regenerator, especially as complicated by lubrication and seal problems. High component efficiencies are needed to have a competitive system.

Systems analyses show that collector orientation within 0.5° to 0.7° is required to reach the system efficiencies shown in Table 6. These tolerances are within the state of the art of vehicle attitude-control systems. An important problem confronting the solar plant is integration of the collector and radiator with the vehicle. The collector design is greatly influenced by the vehicle and mission.

Chemically Fueled Power Systems

A number of chemical space power systems are under development (Table 6). The principal development problem is related to the prime mover. When turbines are used, they must operate at extremely low specific speeds. Reciprocating expanders pose problems in piston sealing and lubrication.[5]*

For re-entry applications and other short-time use, several chemical power systems are of interest. Important advantages are: no need for a radiator; proved technology; and, in some systems, integrated environmental and electrical capability. Fuel consumption has been reduced to the order of current fuel cell art with O_2 and H_2, and even better results are expected with reciprocating expanders. This is because of the use of waste heat from other systems and man as the ultimate energy source. As in the nuclear and solar plants, development will depend upon firming the missions and time requirements. Conventional chemical combustion engines have less than half of the fuel economy of fuel cells, and as soon as the power level exceeds that obtainable from waste

Table 4 Advanced mirror development programs

Type	Diam, ft	Agency	Contractor^a	Application	Status and comments
Mylar foam rigidized	44.5	ASD	GY	15-kw ASTEC solar dynamic	Hardware development. Form for front skin completed.
Electroformed petals	44.5	ASD	EOS	15-kw solar dynamic	Six petals completed. Optically and structurally tested.
Large high-performance one-piece	20 - 30	NASA-Lewis	EOS	Solar Brayton cycle	Study program in progress, electroforming.
			TRW	Solar Brayton	Study in progress, stretchforming of Al.
Inflatable rigidized	9.5	NASA-Langley	HA	Low temperature	Analytical and experimental research on predistributed rigidization materials.
High-precision master	9.5	NASA, JPL	GE	Solar thermionic	Hardware development. Fabrication complete; evaluation to be performed.
High-precision, high-temperature, rigid	5	NASA-Langley	EOS,TRW, BC	Solar thermionic	Hardware development and evaluation. Electroformed Ni, stretchformed Al, epoxy Fiberglas.

^{a}GY = Goodyear; for other abbreviations, see Table 11.

* Both Sundstrand and Kidde have developed single-disk pressure stages for re-entry turbines to achieve high efficiency in cryogenic power systems. Vickers and Marquardt are developing reciprocating expander technology for O_2-H_2 and $N_2H_4N_2O_4$ systems, respectively. These components are in a relatively advanced state of development at the present time.

Table 5 Summary of solar concentrator space testing programs

Program	Mirror type	Space test objectives	Agency	Contractor	Comments
EROS (Experimental reflector orbital shot)	Unfurlable Fresnel, 4-ft diam	Unfurl and determine reflectance and geometry degradation rates under non-oriented conditions	ASD	GM	First unit launched, but telemetry failed. Project cancelled.
ROE-1B (Reflector orbital experiment)	Unfurlable rigid-petal paraboloid, and unfurlable foam-rigidized Mylar, each 10-ft-diam on same vehicle	Same as EROS, except mirrors oriented	ASD	Prime: L = MSC; sub for petal mirror: EOS; sub for foam-rigidized mirror: GY	Preliminary design materials sampling program in support of ASTEC concept
Flight experiment on reflective surfaces	Under study	Coatings include Al and Ag; substrates include Ni, Al, epoxy, and Mylar	LRC	EOS	Study

heat or the duration exceeds a few days, expander engines, with or without oxidation, lose out to fuel cells.

Systems for Nuclear Auxiliary Power (SNAP)

It is generally agreed that nuclear power will be the only reasonable source of power in excess of a few tens of kilowatts for more than a few weeks duration. The crossover between solar, chemical, and nuclear power depends upon the specific mission; the nuclear system offers advantages of ruggedness, high power per unit area, no collector deployment, no orientation, continuous power, and minimum power storage requirements. In many cases, the added power availability of a nuclear system should offer significant operational flexibility and improved reliability through application of more conventional circuitry and instrumentation, and through redundancy. Despite the lack of specific plans for near-term application of nuclear power, the SNAP program has demonstrated a spectrum of units to fill future needs. Currently identified systems under development are described in Table 7.

Reactors

The SNAP hydride (U-ZrH_x) thermal reactor concept is employed in SNAP 10A, 2, and 8. Two full-power test reactors have accumulated a total ground operating time of >2 hr at temperature and power conditions up to the SNAP 10A and SNAP 2 requirements. The recent SNAP 10A flight, of course, conclusively demonstrated the operability in space of the basic reactor. A critical test, designed for the SNAP 8 higher power and temperature conditions, is in the early test phases.[6] A uranium nitride fast-reactor concept was considered the choice for SNAP 50. Considerable reactor experience at the SNAP 50 temperature levels had been accumulated. This project was canceled.

Shield weight is a stronger function of the mission and allowable integration configuration than it is of the power unit. For payloads comprised of semiconductor devices, dose levels below 10^{11} nvt and 10^6 rad probably require very minor restrictions on component selection. Payload hardening for 10^{19} nvt and 10^7 rad can be accommodated if properly considered from the outset of payload design. The shield weight for simple conical shadow-shield geometry can be in the region of 200–500 lb for SNAP 10A, 2, and 8. In the case of manned applications, the shield weight varies from 4000–7000 lb for a simple conical shadow-shield configuration of a 10-ft-diam space station to 15,000–20,000 lb for a 150-ft-diam toroidal station.

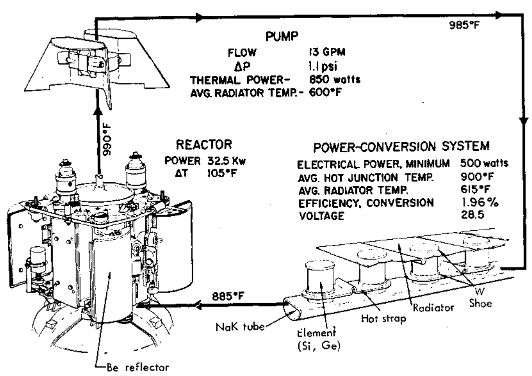


Fig. 1 SNAP 10A schematic and thermodynamic cycle.

The high energy-density advantage of nuclear heat sources directly implies that long life is necessary to achieve the full advantage of nuclear power systems. The only unique, self-imposed environment that could influence reliability is radiation. The more important influences of high temperature, corrosion, creep, high vacuum, micrometers, etc. are shared by other approaches to high-performance space power. The major problem becomes apparent when one considers the unreasonable time and cost associated with a statistical demonstration of reliability or with the corollary identification of failure modes with confidence. This basic dilemma is shared by many other aspects of the space program. The ultimate solution must rely upon simplicity, basic phenomenological understanding, and sound engineering. Basic system development must progress to a level which allows a valid judgment of inherent reliability, and considerable experience must come from interim usage.

Thus far, no insurmountable safety problems have been identified. The Atomic Energy Commission (AEC) has established an aerospace safety program for the specific purpose of developing technology to cope with any nuclear safety problems. In general, the reactor-powered SNAP unit can be transported, stored, installed, and checked out without nuclear hazard or personnel exposure, hence the use of a reactor-powered unit need not perturb the normal launch operations. During launch, the normal chemical exclusion radius is adequate to protect launch personnel from any unlikely nuclear hazard caused by a possible vehicle malfunction or abort. During the entire prelaunch and launch sequence, the reactor is basically inert and contains a negligible inventory of radioactivity; after start-up in orbit, the system can be shut down, and the accumulated radioactivity will decay to a safe level during the remaining time in orbit prior to re-entry. A recent suborbital flight test indicates that re-entry heating will assist in the dispersal of any remaining inventory; however, the exact influence has not been conclusively demonstrated as yet. Both the AEC and the Air Force (through its Nuclear Reactor Systems Safety Group activities) are seeking to establish that the use of nuclear power units in earth orbits will not constitute a radiological hazard to the general public.

Power Conversion and Nuclear System Status

The SNAP system (Fig. 1 and Table 8) employs Si-Ge alloys for thermoelectric direct conversion. A lower figure of merit $S^2/k\rho$ (where S is the Seebeck coefficient, v/°C; k is thermal conductivity, w/cm²-°C/cm; and ρ is the electrical resistivity, ohm-cm) was accepted for the Si-Ge alloys in order to achieve better fabricability and higher temperature capability than the more familiar Pb-Te. The Hg-Rankine cycle turbomachinery development for SNAP 2 (Fig. 2 and Table 7) has overcome the structural and thermal distortion problems that may have been limiting the reliability of the Hg-lubricated bearings. Rotating machinery endurance accomplishments on SNAP 2 and Sunflower (7000 hr on one bearing, including >2600 hr of continuous operation) have demonstrated the potential engineering feasibility of a hermetic machine with working-fluid lubrication. In a recent redirection of the SNAP 8 program, the lubrication scheme on the SNAP 2 concept has been abandoned. NASA feels that use of antifriction bearings, rotating shaft seals, and an organic lubricant (ET-378 polyphenyl) will allow a separation of development variables and will provide a closer relationship to existing technology for SNAP 8.[6] Whereas SNAP 2 employs direct condensation of the Hg in a combination condenser-radiator, SNAP 8 (Table 7) now includes a compact condenser with a liquid-metal heat-transfer loop coupling the condenser to the radiator and an organic loop and radiator to cool the lubricant. Turbomachinery development will reveal the practicability of applying conventional conversion machinery technology to space power,[6] as will be discussed later. The power conversion for SNAP 50 (Table 7) is in the research phase; several agencies are studying the problems relating to alkali-metal power conversion systems.

SNAP 10A has been orbited (Figs. 3 and 4). It is operated for 43 days, performing in excess of specifications. SNAP 2 is well into component development but is no longer a system program as such. SNAP 8 is in the component development and system definition phase; NASA has postponed specific flight-test plans until there is a better definition of the mission.[7] The basic cost of a SNAP 10A unit is estimated at <\$1 million, and the SNAP 2 unit should cost 1–\$1.5 million. An estimate of ≤\$5 million for a SNAP 8 unit seems reasonable.

Rankine cycle

Virtually all of the nuclear dynamic systems under development use the Rankine cycle because of its high actual efficiency between two given temperatures, tolerance to component efficiencies, and minimum heat exchanger and radiator areas. These traits permit compact systems with promise of operation for very long times. Areas of major concern have been a) high-temperature corrosion with a liquid-metal working fluid over long running time[8] and b) condensing under zero-*g*, which cannot be simulated on the ground. However, endurance tests of Sunflower and SNAP 2 power conversion systems, using Hg boiling at 1100°F, have shown that extended operation is possible with no deterioration in performance with proper materials and high cleanliness; up to 1200°F, metallurgical tests show comparable results.[9] Boilers can be made insensitive to zero-*g* conditions by designing artificial gravitational fields into them.[10] Experiment and analysis have also provided the data needed to design zero-*g* condensers. If the size is small, direct radiating condensers are straightforward.[11] For larger sizes, compact

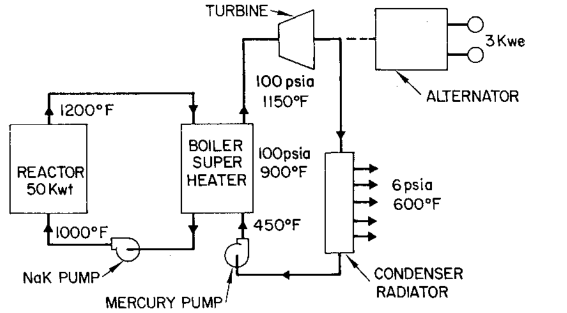


Fig. 2 SNAP 2 schematic and thermodynamic cycle.

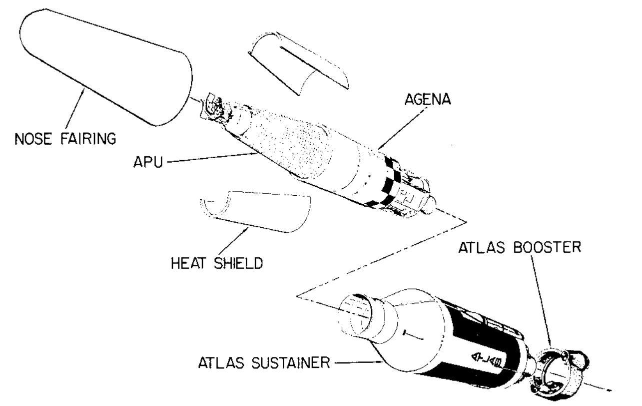


Fig. 3 Integration of SNAP 10A APU with Atlas-Agena.

condenser-radiator assemblies ease manifolding and flow-stability difficulties. A potassium heat-transfer loop is to be orbited in an Agena vehicle to study the effect of zero-*g* on boiling and condensing processes.[12] This program is sponsored by the Aeronautical Systems Division/U. S. Air Force (ASD/USAF).

The desire for low system weight and low radiator area is directly approached by increasing the peak cycle temperature and raising the radiator temperature.[13] A lithium-cooled reactor at 2000°F with an alkali-metal working fluid (*K*) is attractive for powers ≥300 kw.[14] The SNAP 50 program (sponsored by AEC/AF/NASA) is aimed at such a system. The increase in temperature, however, requires an intensive new research effort. Such a program will take considerable time and effort, and the systems cannot be available before 1972, but the work is so important that it must be pursued vigorously on a technology development basis and has indeed been underway since 1962. The bulk of experience with alkali metals is at temperatures below 1000°F. A considerable amount of high-temperature materials work in flowing loops is necessary. In addition, the alkali metals readily condense on reaching the saturation temperature and will cause turbine erosion if condensation occurs within the nozzle or blading. A third major area of study is *K*-lubricated bearings. Associated problem areas are 1400°F potassium vapor in the alternator cavity, control of the large powerplant, knowledge of boiling and condensing, flow stability of *k*, and long-time creep and fatigue of materials at high temperature. [This program was cancelled in July 1965.]

Brayton cycle[15]

During the long development period of the alkali-metal systems, plants up to 100 kw using the lower temperature Hg

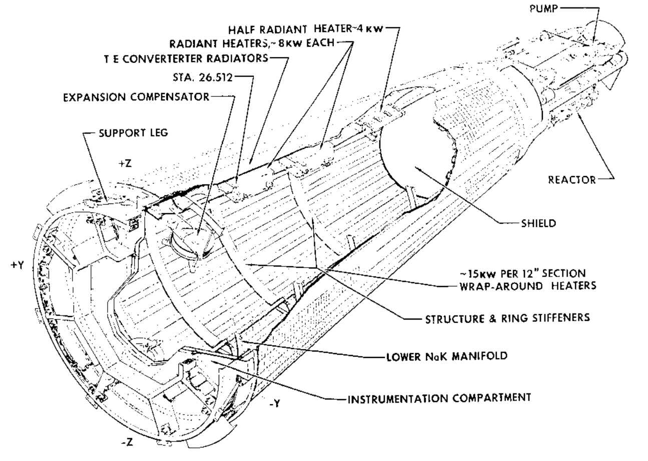


Fig. 4 SNAP 10A system.

Table 6 Chemical and solar dynamic systems

State of art

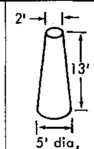
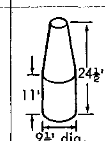
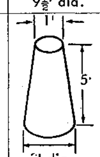
Name, power level, heat source	Organ.[c] (Sponsor)	Working fluid(s), Temperatures, °F[a]	Storage volume	Weight, lb	Demonstrated Endurance, hr	Efficiency[b] (Est. reliability)
CHEMICAL DYNAMIC						
Integrated accessory power and environmental control system, 50-40 HP, chemical combustion	WK (ASD)	H_2/O_2 (fuel rich), 0-400 precombustion 1400-1800 combustion	50 ft^3 for 8-hr mission 200 ft^3 for 36-hr mission at 25 HP	System 1000 tankage 360 env. control sys., 115 acc. pwr. sys. 125 total; 1600 for 8-hr mission, 4500 for 36-hr mission	Breadboard system has accumulated several hours	SPC = 1.1 at 25 HP (not applicable)
871A space power system, 10 KVA at 115/200 V, chemical decomposition	SA (ASD)	Anhyd. N_2H_4, 2100 = T_{TI}	12 x 18 x 23 in. + reactant supply	Prime mover 82 generator 65 hydraulic pump 8 total: 155 tankage depends on mission	4.5 (100 design)	SPC = 3.5 (MTBF = 1500 hr)
Cryhocycle R, 3 KVA at 115/200 V 400 cps or 28 VDC, thermal control system with supplementation from catalytic reaction chamber	SA (In house)	H_2/O_2 120 max	11 x 25 x 10 in. + reactant supply	Prime mover 31 generator 15 total: 46 tankage depends on mission	>200 at rated conditions	H_2 consumption 1.5 lb/kw-hr, (des. rel. = 0.9999, MTBF = 2000 hr)
876 B space power system, 8.8 GPM at 2700 psi 12 KVA at 118/205 V, catalytic reaction chamber	SA (ASD & BC)	H_2/O_2 (gases) 1500 = T_{TI}		Prime mover 125 generator and/or hydraulic pump 41.5 total: 166.5 tankage depends on mission	250 at rated conditions	43.6%, (MTBF, no redundancy, = 750 hr)
Model 152000 power system, 15 KVA at 400 cps, 115 VAC 12 GPM hydraulic flow, chemical decomposition	TRW, (ASD & BC)	Anhyd. N_2H_4 1600 = T_{TI}	15.8 ft^3 for 25-min mission	Weight for ~ 25-min mission, 243	50(air), 1/2 (N_2H_4) same unit reusable	SPC = 6.0 (des. rel. = 0.99)
SPU-2, 2.5 KWE, chemical combustion	MC (LRC/NASA)	N_2H_4 family and N_2O_4, 5000	0.65 ft^3 engine generator + attached accessories	Prime mover 26 generator 20 pumps and controls 11 total: 57		23% target SPC = 3 (des. rel. = 0.9999, MTBF = 1000 hr)
Hydrogen-oxygen internal combustion engine system, 3 kw	VI (LRC/NASA)	Steam and H_2 4960 combustion 1170 heat rej. 2710 cyl. head	2 ft^3	100		1.8 lb/kw-hr (because operation is H_2 rich a percentage figure is not meaningful)
SOLAR DYNAMIC						
Sunflower	TRW (NASA)	Mercury LiH heat storage	10-ft diam circular cyl. 8-ft high topped by 15° half-angle cone	Total, 700 lb (Based on 300-nmi orbit)	Turbo alternator 2300 hr and 2750 + hr at design	9.2 % over-all in full sun with zero misorientation
Binary, 5.50 KWE solar collector and heat storage	TRW (ASD)	Hg and hydrocarbon Rankine cycles with LiH heat storage, 1250 Hg superheat 1085 Hg boiling 650 Hg condenser 600 H_xC_y boiling 200 H_xC_y condenser	~1000 ft^3 for 15 kwe system	1023 at 15 kwe	None	12.3% overall in 300 nmi orbit 30% binary Rankine cycle conversion (Not available)
Stirling cycle, 3 KWE, solar	GM/A (ASD & GM/A)	He with NaK heat transfer, NaK inlet 1250, outlet 150	4.8 ft^3 + collector & radiator	555 at 3 KWE	1000	30.2% conversion efficiency demonstrated, 38% anticipated
Model 1400, solar dynamic space power system, 1.5 KWE, solar collector and heat storage	SA (Navy BUWEPS)	Diphenyl	36-in. diam x 48 in. long	287		14% design
Model 1350, solar dynamic space power system, 15 KWE solar collector and heat storage	SA (AFSC)	Rubidium, 1200 boiling 1800 superheat 1750 T_{TI} 680 T_{TE}	8 in. diam x 8 in. long	1155		18.2% design

[a] T_{TI}, T_{TE} = turbine inlet and exit temperatures, °F.
[b] Thermal efficiency given in %, SPC = specific fuel consumption, lb/hp-hr
[c] WK = Walter Kidde; SA - Sundstrand Aviation, Denver; for other abbreviations see Table 11.

Table 6 Continued

System status			Applications		
Availability	Development Status	Main Development Problem(s)	Use Factors	Unique Features	Probable applications
Not applicable; system designed for each application	Integration concepts have been demonstrated in breadboard testing	System control and performance matching	Requires gaseous start-up system for heat prior to use of cryogens	Combination of sub-systems offers potential of lighter overall vehicle launch weight	1kw and up for durations up to 1 week
Unit is ready for PERT tests	Initial units ready for prelim. flight rating tests	None	Immediate start and restart	Lightweight and storage	Emergency power reentry after long missions
Depends on funding	Feasibility demonstrated in both turbine and reciprocating version		Integration with thermal control system	Can provide cooling for crew or equipment	Manned vehicles for low-power-level missions
	First operating unit delivered to customer	Multi-stage, single-disc reentry type turbine to meet performance and life requirements	Depends on fuel supply available. Servicing of lubricants and control adjustment possible. Completely self-contained and designed for extended zero-g operation	Multi-stage, single-disc, turbine; low SPC; lightweight, high power output; series integration with environmental control system to provide heat sink	Manned vehicles where cryogenics are on board Spent boosters where boil-off can be utilized
Now	Operational unit used with USAF IM 99 B missile (BOMARC)	None Adaption to space vehicle only required meets USAF spec. MIL P-25413	Storable 2 yr in ready condition at S.L.T.P.	Proven operational unit; low-cost, storable fuel; 5-sec start, restartable	Emergency power reentry and landing
1965 dependent on funding	Development testing	Minimum heat rejection to cooling system	Propellant system temperature range 20°F - 150°F	Lightweight, small volume, unlimited start-stop capability, high power/weight and power/volume ratios, good efficiencies	Emergency power or primary power utilizing left-over propellants on vehicle
Technology problem only	Prototype being tested	O_2 injection mechanism; Improve SPC	Small radiator required; space start and restart capability		Short-time, low-level auxiliary electric power; prime mover for lunar roving vehicle
Dependent on funding	Active work has been terminated 4300 hr to accidental termination	Actual demonstration of start and 1-yr endurance demonstration is lacking	Orientation $\pm 3/4°$ collector opens after launch into space Designed for 10-g boost, ± 1 and zero-g orbital loads	Load variable between 0 and 3 kw Continuous operation for 1 yr	Long-term applications, manned or unmanned for power levels up to 15 kwe in earth orbits MOSS
1968 dependent on funding	Second phase study program	Low temperature working fluid stability	Orientation	Low specific weight	Orbiting applications up to 50 kwe MOSS
1966	Performance development completed 1000-hr endurance test completed	Shaft seals	Orientation $\pm \frac{1}{2}$ deg Low heat rejection temp. means large radiator	High conversion efficiency especially important in solar power system - no zero-g influence with He working fluid	Satellite and space power from 1 to 50 kwe MOSS
Mid-1966 man-rated 1968	R & D	Space inflatable concentrator heat storage	Orientation ± 6 min	Low launch volume	Long-duration space missions Orbiting space stations COMSAT
	R & D	Space inflatable concentrator Heat storage Rubidium bearings	Orientation ± 6 min	Relatively low launch volume	Satellite and space power, small MOSS

Table 7 Dynamic nuclear systems

		State of art					System status				Applications	
Power plant	Organ. (Sponsor)	Fluids[a] and T, F°	Storage volume	Weight, lb	Endurance, hr	[b]η_{th},% (R_{est})	Availability	Status	Main development problems	Usage factors	Unique features	Probable applications
SNAP 2 3 kwe, thermal reactor	AI/TRW (AEC)	RC = NaK 78 WF = Hg T_{RE} = 1200 T_{RA} = 620		750+550 shield	Obj: 10,000	68 overall	No longer a system development program	Conversion system Startup and restart demonstrated PCS endurance being demon.	Long time Hg-materials compatibility	Orbital startup Compatible boost vehicle Atlas-Agena, Saturn, etc. Requires nuclear shielding Radiator area, 120 ft^2.	Zero-g operation No orientation required Short-circuit stable, aircraft quality power, 1800 cps Low area requirements High power by paralleling conversion units (10 kw)	MOSS manned and unmanned military satellites Communications Navigation Applications Lunar base (Modules)
SNAP 8 (Ref. 6) 35 kwe, thermal reactor	AG/AI (NASA/AEC)	NaL 78 reactor RC=Hg WF = ET-378 bearing lubricant RHX = NaK T_{RE} = 1300 T_{RA} = 650	Depends on radiator configuration which is not now established	3160, excluding radiator & reactor shield	Obj: 10,000	8 overall (80)	1970	Components in design and development	Seal development Nuclear radiation of lubricant Bearings Materials (Corrosion, fabrication)	Radiator area: 1400-1800 ft^2 Radiation configuration compatible Boost vehicles Titan III Saturn	Higher power potential, 50 kwe Flexibility, 400 cps power	Space stations Lunar base Manned/unmanned interplanetary missions Communication satellites
SNAP 50/SPUR 300 kwe, UC fast reactor	AEC,AF/ASD (In house)	RC = Li WF = K RC = Li T_{RE} = 1800		3790 unshielded 4590 unmanned shielded	Obj: 10,000		Ground test capability by 1968 Flight test by 1972 (Note: this program was cancelled in July 1965.)	System analysis, Materials & component development	Turbine erosion Materials compatibility Refractory metal design & fabrication technology Bore seal	Compatible boost vehicles, Titan III Saturn Same nuclear freedoms & restrictions as above	Minimum shielding. Higher power by paralleling conversion units (1 Mw)	Propulsion Lunar base Lunar ferry Megawatt unit
Isotope 0.5-3 kwe, Radioisotope	TRW (In house)	WF = Hg T_{TI} = 1230 T_{RA} = 630		356 at 1 kw	Obj: 10,000 Dem: 2,500 (pcu)	11.1 (99)	2 yr ARO to develop isotope boiler	Conversion equipment endurance of 2500 hr demonstrated & flight tested in SPUD program	Requires isotope boiler development	Isotope hazards associated with launch & reentry Redundant weight penalty, 10 percent	Independent of environment No orientation required Low weight Conversion equipment already flight tested	Orbital mission Longer than weeks Planetary Landings Space probes . Lunar base Portable power
HIPO-1 15 kwe, thermal reactor	TRW (In house)	RC = NaK 78 WF = Hg T_{RE} = 1300 T_{RA} = 750	342 ft^3 (conical radiator)	1374 + shield wt.	Obj: 20,000 (5 yr w/main.)	15 (98)	Flight test, 1968 Operational, 1969 Based on 1965 go ahead	Preliminary system design Component evaluation	Very long life w/ minimum or no maintenance Radiator design min. meteorite protection wt.	400 cps, AC aircraft Quality power Reactor start in orbit ~2 hr start time Uses SNAP 8 reactor	Easily formed into parallel systems Uses much proven technology	Space station Lunar base On-board Power for satellites
HIPO-2 30 kwe, UC reactor	TRW (In house)	RC = NaK WF = Ne or A/He T_{RE} = 1600 T_{RA} = 225	900 ft^3 with deployable radiator leaves folded	2014 + shield wt.	Obj: 20,000 5000 hr demonstrated on turbo-compressor component w/ fluid bearings	25 (98)	Flight test, 1971 Operational, 1972 Based on 1965 go ahead & reactor availability	Preliminary system design Component evaluation	Gas bearings Radiator size & weight High component efficiency requirements	Orbital start Limited to about 40 kw due to radiator size and weight	Uses existing facilities Uses inert gas—no corrosion in working fluid loop Alternator easily cooled	Space stations Lunar base
35 kwe, UC reactor	AIRE-SEARCH (In house)	RC = NaK WF = Ne T_{RE} = 1600 T_{CI} = 80		1747 + reactor + shield	Obj: 27,000 Over 13,500 hr accumulated on bearing test rig in single unit	25 (High)	Avail. depends on funding and reactor availability	Preliminary system designs Component devel. & optimization	Radiator size & weight— config. optimized to applications	1630 ft^2 radiator Pressurized gas or electric power req'd for start Orbital start Can be paralled for higher power	Max. use of existing facilities Uses inert gas as working fluid Utilizes some proven state of art in open-cycle gas turbines	Space stations Lunar base

[a] RC = reactor coolant; WF = working fluid; BL = bearing lubricant; RHX = radiator heat-transfer fluid; T_{RE} = reactor temperature; T_{RA} = radiator temperature; T_{TI} = turbine inlet temperature; T_{CI} = compressor inlet temperature; [b] η_{th} = thermal efficiency, %; (R_{est}) = estimated reliability based on no meteorite puncture in 1 yr, %.

technology can be built. Studies show that systems utilizing the SNAP 8 reactor and present conversion technology with a higher temperature radiator could be available by 1968 or 1970 and could have weights of 100 lb/kw at the 30-kw level and 35 lb/kw at the 100-kw level. Notwithstanding these accomplishments, the concern over liquid-metal corrosion and the possibility of using the carbide reactor or solar heat plus LiF heat storage as a higher temperature heat source has created a renewed interest in the Brayton cycle. The closed Brayton cycle is an adaptation of open-cycle gas-turbine technology to space power. An inert gas (A or Ne) is used as the working fluid, thus eliminating any corrosion in the working-fluid loop. The single-phase fluid is independent of gravity, so that zero-g operation is easily simulated, and no artificial gravity need be used in heat exchanger design. However, the Brayton engine is less efficient and larger than a Rankine engine operating between the same temperatures. As shown in Table 7, HiPo-2 and a typical 35-kw AiResearch system have relatively high cycle efficiencies obtained by raising the reactor temperature and lowering the radiator temperature of the Brayton system which reduces gas flow and heat rejected. The compressor horsepower required is proportional to inlet temperature, so the inlet temperature must be kept low ($\sim$540°R). The headers and the radiator are large because of handling a gas operating at low temperature and being nonisothermal; this is a major problem area because of meteorite vulnerability[16] and the high surface area requirements,[15] as well as the associated weight.[17]

High component efficiency is required for the Brayton cycle. Typical values are regenerator effectiveness of 0.85 or more, compressor efficiency of 0.80 or more, and turbine efficiency of 0.85. Careful attention is being paid to flow channels to minimize pressure loss, for the Brayton cycle optimizes at low pressures (about 30 to 50 psia depending upon the power level). Component development and system design are being done in-house by several contractors and by NASA. Emphasis is on obtaining the high component efficiencies at low-power levels (6–15 kw), system dynamic analysis, and radiator size reduction. Studies show that the Brayton cycle is probably of little interest for power levels >40 kwe because of the size of the system. An important factor in possible system application is the availability of a carbide reactor in flight configuration. Work done in the power conversion system, however, shows that component efficiencies can be met. They may be marginal at low-power levels when actually used in a closed system.

Future direction

Performance advances will be paced by materials technology. In the reactor, temperature and fuel material selection will determine the useful power output before failure due to fission-product-induced fuel swelling. In the remainder of the system, temperature and material selection will limit life because of corrosion,[18] creep, sublimation, etc. The output of SNAP 10A and subsequent thermoelectric systems is proportional to the fifth power of the source temperature. The low efficiency disadvantage of thermoelectric systems will probably be offset by the inherent reliability of static power conversion up to multikilowatt levels. The performance of Hg-Rankine cycles is limited by the practical working pressure limit and the thermodynamic properties of Hg. The major improvement in Rankine cycles occurs with the change in working fluid that, unfortunately, involves a temperature jump of 600° to 800°F to the SNAP 50 conditions.[19] Future Hg systems probably will employ the currently available, lower-temperature technology and hence will be relatively large. The Brayton cycle, even though it requires a larger radiator area at a given heat source temperature limit, should be more versatile in accepting the increased heat source temperature capability that future technological improvements will yield; this potential for a more continuous performance growth

Table 8 SNAP 10A characteristics and program participants

Power, kwe	0.5
Reactor power, kwt	30
Efficiency, %	1.6
Reactor outlet temp., °F	1000 U-ZrH$_x$ thermal
Primary coolant	NaK-78
Power conversion	GeSi
Hot junction temp., °F	930
Cold junction temp., °F	615
Radiator temp., °F	615
Radiator area, ft^2	62.5
Spec. area, ft^2/kwe	125
System unshielded wt, lb	650
Spec. wt, lb/kwe	1300
Development agency	AEC
Flight-test agency	AF
System contractor	AI
Power conversion contractor	RCA
Reactor contractor	AI
Flight-test contractor	Lockheed

could have a significant influence on long-term reliability achievement. ASD is supporting work on Rankine systems with water due to our vast experience in handling this working fluid.

As a backup to the SNAP 50 program, the AEC has initiated a development program at Oak Ridge National Laboratory (ORNL) to evaluate the feasibility of a system employing a fast reactor with direct alkali-metal boiling in the reactor core. The maximum performance system of the future, 5 to 10 lb/kw, is the reactor thermionic system. This concept has been called SNAP 70, but no development program has been established. Two system concepts are being advanced: one employs diodes outside of the reactor (perhaps in the radiator), which forces the entire reactor structure, control, coolant, pumps, etc., to operate in the 2000° to 2500°F region; the other places the diodes directly in the core as an integral subassembly of the fuel element, so that the nuclear material operates at the cathode temperature of the space-charge-neutralized diode, $\sim$3000°F, whereas the remaining reactor structure, control, coolant, and pumps operate at the anode temperature, $\sim$1500°F. This concept is illustrated in Fig. 5. The basic phenomenon of thermionic conversion is reasonably well understood as evidenced by demonstrated conversion efficiency and power-density accomplishments. The problems of materials selection for the environment and for useful diode life are less well understood. The technology necessary for the selection of a reactor fuel material with appropriate

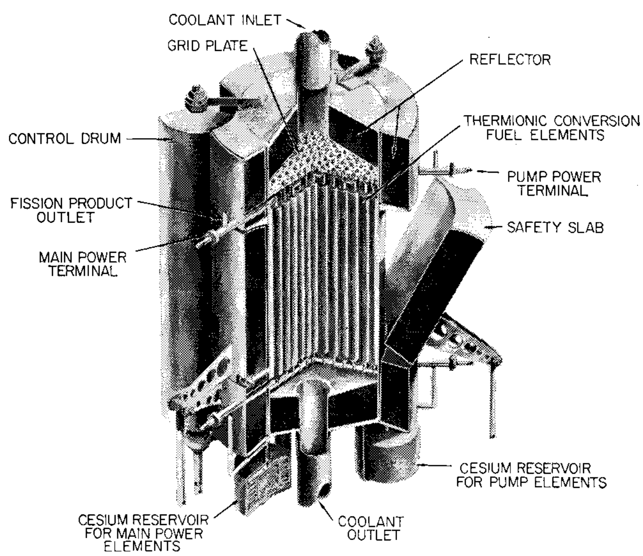


Fig. 5 Schematic of thermionic reactor.

physical properties and high-energy output capability is currently far from the status required to support serious system design and development.

Dynamic Machinery for Space Power

Turbines

Dynamic nuclear systems are summarized in Table 7. Liquid-metal Rankine systems utilize turbines as prime movers. Development problems with these turbines include 1) achieving efficiency at low percent admission for the low-power systems; 2) loss of performance because of moisture and wet vapor expansion; 3) erosion of the turbine disk, blades, and housing materials by moisture in the wet vapor expansion; and 4) materials compatibility.[20,21] All of these problems have their counterparts in conventional ground steam powerplants. Performance loss, erosion, and tip-speed limitations are well documented for steam turbines, but no correlations have been verified which would permit extrapolation of these results to other working fluids and materials of construction. Analytical studies indicate that the performance loss in liquid-metal expansions may be comparable to that expected in water vapor, i.e., 1% loss of efficiency for every percent of moisture.[22] Experimental data do not yet exist for the alkali metals. There is evidence that superheated Hg vapor tends to supersaturate without condensation during expansions far into the vapor dome at equilibrium moisture content much higher than 3%, which normally is considered the maximum supersaturation achievable in water.[23] This may help to account for the freedom from apparent performance deterioration of the Sunflower mercury turbine, which has operated continuously for over 4000 hr,[4] and the South Meadow turbine, which ran 116,000 hr. Analysis indicates that alkali metals will not supersaturate significantly. Their surface tension properties are more nearly comparable to those of water.[8] Wet expansion of alkali-metal vapors through a turbine is currently under investigation in connection with the now defunct SPUR/SNAP 50 program.

The design requirements of the bearing and shaft support are long life and ease of integration into the power system. Since properly designed journal bearings theoretically have almost infinite life, they are preferred to antifriction bearings, which have a finite fatigue life. Use of the system working fluid as the bearing lubricant eliminates seals and their development problems. Recent experiments have demonstrated that liquid-metal-lubricated bearings are feasible, but development is difficult because of the severe requirements imposed by the application, the nature of the lubricant (low kinematic viscosity), and the limited analytical techniques available. Bearing operation in the turbulent regime was previously experienced only in a few machines operating with water-lubricated bearings. Major efforts by the Air Force and NASA have been initiated to generate an understanding of the theory and practice of turbulent bearing lubrication. The transition from laminar to turbulent operation occurs when the Reynolds number exceeds 41.1 $(r/c)^{1/2}$, where r is the bearing radius, and c is the radial clearance. The bearing behavior in the turbulent regime, as compared to laminar, is characterized by 1) greatly increased power loss, 2) increased load capacity, and 3) greatly reduced flow rate for a given Δp.[24] Considerable effort is being expended to obtain analytical and experimental data on performance characteristics of turbulent-liquid-metal bearings. Journal and thrust bearings lubricated with Hg have been operated in a Sunflower system for over 7000 hr, and Rocketdyne (USAF sponsorship) completed a 700-hr run of an experimental potassium-lubricated bearing operating under conditions comparable to those for space power applications. Power loss data from such experiments verify Smith's and Fuller's empirical correlation for friction coefficient with water as a lubricant.[25] Load-carrying capacity is not as well established for alkali metals.[26] Acquisition of valid data in this area is greatly hampered by the lack of suitable instrumentation; USAF-sponsored research on instrumentation is in its early stages.

Proper design for adequate rotor critical speed is a difficult analytical problem at all times. The problem is complicated by the finite stiffness of the liquid-metal bearings and the difficulty in predicting fluid film stiffness and damping characteristics in the turbulent regime. Proper application of existing techniques with experimental values of bearing stiffness and damping, combined with good machine design, should permit control of rotor dynamic stability.

Generators

Two types of generators are presently under consideration. For more conventional applications, a brushless alternator would be used with characteristics similar to those used in conventional aircraft powerplants. Frequencies of 1000 to 3200 cps are preferred to minimize the weight of other electrical components in the system. For those applications where the powerplant is directed specifically toward an ion propulsion engine, an electrostatic generator would be desirable because of the high-voltage, low-current, d.c. output of such devices. Development of brushless alternators in conjunction with present Hg programs is proceeding relatively well with respect to the high-temperature materials and high-rotative speed problems. A more difficult problem is that of sealing the windings from the corrosive alkali-metal vapors.[27] Electrostatic generators are attractive for space power systems because they can operate at high temperatures (they do not contain magnetic materials, which are affected by high temperatures), and they have high efficiency and power/weight ratio.[28,29]

Thus far, feasibility tests have been accomplished only with some small-scale experimental models and with only two designs, one using capacitance variation in a single capacitor, and the other using the work of the forces caused by an oblique field in a gap between dielectric surfaces.[28] Two engineering problems remain to be solved: rotating seals and bearings.

Heat-Transfer Components

Development of boilers and condensers for Rankine cycle powerplants has been somewhat empirical because of the lack of adequate heat-transfer data with the materials in question.[30] Recent measurements show no significant discrepancies in the basic heat-transfer phenomena involved with liquid metals. Boiling and condensing coefficients are more or less in the range expected for low-Prandtl-number fluids. For the case of Hg in a direct radiator condenser, existing heat-transfer correlations are adequate, because the outside radiation resistance is controlling. The various regimes of Hg boiling have been identified and described mathematically.[4] Flow stability in condensers under zero-g conditions appears to be achievable, although further attention may be desirable for larger systems with extensive headers.[31]

High-temperature emissive coatings are under development,[11] but the effects of ultraviolet and proton bombardment in a space vacuum are difficult to assess on earth; hence the durabilities of these coatings are not known. Protection against meteoroid penetration looms as the largest single factor influencing the design of radiators for dynamic power systems.[16] Indirect radiators, which use a heat-transfer fluid between the working fluid and the radiator can only be justified on the basis of reduced area sensitive to penetration. An extreme example of design for minimum sensitive area is the belt radiator.[32] The use of this type of design can be shown to yield significant reductions in total powerplant weight, if the meteoroid problem is as severe as it presently appears. Because of its great importance to powerplant weight and reliability, it is vital that accurate data be acquired for meteoroid flux distribution in both near-earth and interplanetary space.

Radioisotope Space Power

SNAP isotopic power systems are currently under development for the AEC to fulfill several space mission power requirements for NASA, the Navy, and the Air Force. Advantages of isotopic power supplies lie principally in a) their ability to operate for long periods in an unoriented space environment, b) their attractive power/weight ratios via high-power-density fuel forms and compact high-temperature radiator designs, c) simplicity of systems operation (a SNAP generator and a solid-state converter regulator to fulfill all of the load distribution requirements), and d) their ability to operate in the high radiation fields of the Van Allen belts without degradation.

Current Systems

On June 29, 1961, a 2.7-w, plutonium-fueled, thermoelectric generator was placed into a 500-mile circular orbit aboard the Transit 4A navigation satellite. Telemetry data confirm the successful operation of the isotopic generator and d.c. step-up transformer in the space environment. The continued reception of precise radio signals from the 54- and 324-Mc satellite transmitters by Transit tracking stations throughout the world gives further assurance of successful generator operation. A similar device aboard Transit 4B (launched November 15, 1961), which maintained continuous operation for 28 weeks (before some malfunction shorted the generator output), confirmed the repeatability of performance parameters. Through August 1963, over 22,000 hr of successful operation had been logged in space with these two satellites.

In August 1961, the SNAP 9A development program was initiated to produce a plutonium-fueled, thermoelectric generator producing 25 w and a power/weight ratio of 1 w/lb. Electrically heated prototype units were operated at design power and subjected to design qualification and reliability tests. Three flight units were fueled with Pu-238 by October 1962. Systems integration testing and flight qualification have since been satisfactorily accomplished. Generator output at the beginning of life is in excess of 27 w(e). More than 20,000 hr have been accumulated under fueled conditions with three SNAP 9A generators, thus assuring long-term reliability of such devices.

The SNAP 11 program was established in February 1962 by the AEC at the request of NASA to develop a flight-acceptable radio-isotope-fueled thermoelectric generator for use on Surveyor A-25 spacecraft. Specifically, it is to provide 25 w(e) under lunar night conditions to permit the accumulation of surface data in the absence of solar energy and to supplement a solar cell power supply system during lunar day conditions. Operating life is 120 days minimum. Two prototype electrically heated generators are in design qualification test status, and two more will be used for spacecraft integration tests. The unit weighs 30.4 lb, including 5 lb of shielding to reduce radiation on sensitive instruments. This generator will be fueled with curium-242, a 163-day half-life isotope. As a result, the unit is fitted with a power-flattening thermal shutter system. This system is completely redundant to satisfy stringent reliability aspects of systems integration.

A program to develop a radioisotope thermoelectric generator (RTG) has been established by the AEC for NASA to develop a flight-acceptable system for prime electrical requirements of the Interplanetary Monitoring Probe-C (IMP-C) and subsequent scientific satellites. Because of prerequisite flight payload space limitations and the requirement for satillite spin orientation, two RTG units, each of approximately 20-w power output, will be required per satellite launching. Enough Pu-238 will be employed to provide 2-year operational life and 1-year shelf life. An appropriate regulated voltage converter will also be developed to provide about 32 w of usable power to the IMP payload. Current state-of-the-art developments resulting from the SNAP 9A and SNAP 11 programs will be employed wherever possible, and the RTG diameter will be kept within a 7-in. annular space envelope. Phase I, completed June 30, 1963, emphasized the establishment of system specifications by the contractor in joint cooperation with the AEC and NASA. Phase II, a 12-month effort, concentrated on power system fabrication, assembly, and test of a prototype and flight hardware. A detailed safety analysis was conducted.

SNAP 13 represents the first marriage of a radioisotope heat source to a thermionic conversion unit. A demonstration device is being developed under the current effort, which will employ Cm-242 with a Cs-filled thermionic diode designed to operate in the ionization mode and produce 12.5 w at a system weight of 4.5 lb. Two electrically heated units produced under this program have been tested for more than 4000 hr using electrical heaters to simulate isotope thermal conditions. Preparations are now under way to modify these prototypes so that they can withstand launch environments. Plans are to fuel and demonstrate advanced units. The high power/weight ratio of this type of system offers specific advantages for advanced space missions.

Future Systems

The following study programs are under way to advance the technology of radioisotope thermoelectric power: 1) an Sr-90 power system for communication satellites; 2) 200- to 500-w systems for such missions as Voyager, Nimbus, etc.; and 3) 3- to 5-kw systems for use in manned space stations. These studies have proved the feasibility of such concepts, and advanced thermionic systems are being examined to show the feasibility of power supplies up to 500 w. Development efforts have shown that reliable, compact, isotopic power supplies in the 1 to 4 w/lb range are feasible and can take the rigors of launch and continuing operation for extended periods in space environments. By more than 10^6 hr of SNAP generator operation using Po-210, Pu-238, and Sr-90 isotopic heat sources, it has been shown that such systems can provide reliable power for missions such as 1) earth-orbiting missions of ≥ 1 year; 2) lunar surface exploration probes for which dust and terrain uncertainties or lack of solar energy would dictate an unoriented, self-energized power supply; 3) deep space probes for which temperature levels or lack of solar energy favor use of isotopic power units; and 4) higher power levels (1–6 kw), where lower launching weights are achievable with isotopic power supplies.

Thermoelectric Systems

There was a heavy research effort in the period 1958–1962 to improve the conversion efficiency of thermoelectric materials. Figures 6a and 6b illustrate the state of the art for N-type and P-type materials, respectively, as submitted by the laboratories indicated. The efficiency ϵ is an "ideal" or theoretical maximum efficiency for the specific material tested at the temperature shown. The cumulative efficiency η values shown (21.7 and 20.5%, respectively) represent the best values theoretically obtainable if it were possible to multistage a very large number of thermoelements, each operating over a very narrow temperature range at the highest $\epsilon(T)$ values indicated. The Ge-Si solid-solution alloy has poorer thermoelectric properties below 500°C than the other materials shown, but it is much more stable physically and chemically at high temperature and low pressure, and it has high mechanical strength and low density. In actual systems, the materials efficiencies of Fig. 6 are degraded by junction and contact losses, thermal heat shunting around thermocouple legs, etc. Furthermore, the concept of multistage

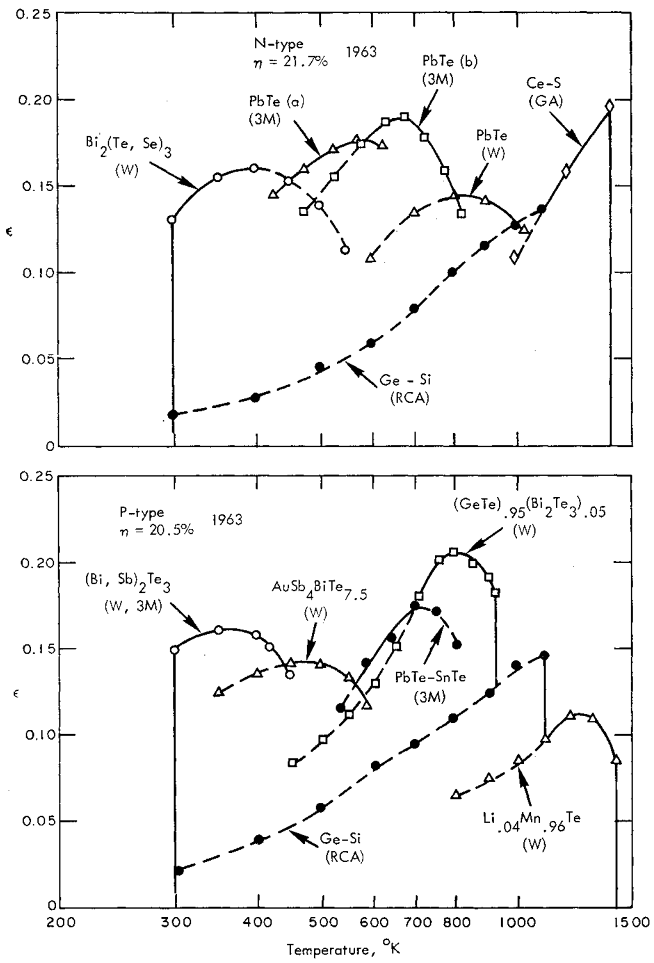


Fig. 6 Efficiencies of *N*-type (top) and *P*-type (bottom) thermoelectric materials.

devices utilizing different materials in their optimum temperature ranges has so far been proved impractical because of thermal and electrical impedance mismatches and physical and mechanical incompatibility between the different materials.

Radioisotopic, solar, and nuclear reactor thermoelectric systems have been built, or are being built, with varying degrees of success. As previously noted, two Pu-238-fueled SNAP 3 units were launched in the Transit 4A and 4B satellites. SNAP 9 is under development for Navy navigational satellite applications; the 25-w unit is to use Pu-238 and to weigh 27 lb for a planned life of 120 days. SNAP 11 is a 25-w generator for use on the Surveyor soft-lunar-landing spacecraft under development by NASA. Fuel is 25,000 curies of Cm-242 for a planned life of 120 days at a weight of 30 lb. Design of the generator has been completed. Flight-acceptable fueled generators are expected to be delivered during 1965. Because radioisotopic thermoelectric generators have higher power densities than solar cells, for some future space missions RCA is developing in-house a prototype radioisotope generator in the 50- to 75-w output range.

Solar Thermoelectric Generators

The flat panel concept, introduced by General Atomic Division in 1960, has been the most successful solar application to date. Ground acceptance tests on 4 × 4-in. panels (18 thermocouples each) gave 1.33 w/ft² panel area with the collector at 250°C and the radiator at 20°C. Weight/power ratio was 96 lb/kwe. With a collector temperature of 300°C, output was 2 w/ft². Panels were thermally cycled 690 times in vacuum without significant degradation and operated over 9000 hr under steady-state conditions with less than 10% degradation. Thermoelements are *P*-type ZnSb (0.055 × 0.055 × 0.106 in.) and *N*-type PbTe (0.04 × 0.04 × 1.106 in.) sandwiched between Al foil collector and radiator surfaces, properly coated, and reinforced by Al honeycomb for structural strength. In the first orbital flight of test panels, power output degraded during 15 orbits to about 25% of that obtained during acceptance tests. Temperature variations were not sufficient to explain the decrease. On the second flight, three 4- × 4-in. panels produced 2.5, 2.6, and 1.3 w/ft² at an average weight of 60 lb/kwe. These early experimental panels were greatly stiffened with auxiliary structure to survive launch vibration, thus considerably increasing the system weight. Similar systems are now being investigated by General Instrument Company for the Air Force and by Melpar for NASA.

A thermoelectric system using parabolic collectors was studied by Hamilton Standard Division of United Aircraft (UAC). A 5-w model was made up to 56 unit cells connected in series; each cell consisted of a 4-in.-diam parabolic concentrator with a single concentrator at the focal point. Degradation was excessive.

The Aerospace Electrical Division of Westinghouse built a thermoelectric generator with an integral cavity solar absorber surrounded by an LiH storage system. The model has been operated in a space chamber with electrical heaters and on the ground with natural sunlight. The LiH was used to maintain a continuous supply of energy for the thermopile during simulated 90-min orbits with 35 min of darkness. The model produced >17 w at 10 v for the 90-min period with variation of only 7%. Over-all efficiency of 3.78% was observed (electric power output to solar power intercepted by the collector concentrator). Based on these tests, an analysis was performed to determine the power/weight ratio, efficiency, and envelope dimensions of power modules with an output between 50 and 180 w. A power module with an output of 180 w coupled with an 8-ft-diam collector would have a specific power of 2.25 to 2.30 w/lb at an efficiency of approximately 4.5% and would be capable of continuous operation for the full 90-min orbit. The average hot and cold junction temperatures for the cycle are 520° and 198°C, respectively. The thermoelements are *P*-type Ge-Bi-Te and *N*-type PbTe. All of the preceding research in solar thermoelectric generators has been supported by ASD/USAF.

Nuclear Reactor Thermoelectric Generator[33]

The 500-w SNAP 10A thermoelectric generator (Table 8) is primarily intended as a forerunner of space power generators using a nuclear reactor as a heat source, a liquid metal (NaK) as a heat-transfer mechanism on the hot side of the generator, and radiation to space on the cold side. The thermoelectric generator assembly is made up of 1440 couples of *N*- and *P*-type Ge-Si alloy. A 300°F ΔT is maintained across the thermoelements. Each couple produces 0.37 w at 0.1 v to a matched load. The Ge-Si alloy exhibits excellent strength and stability and low vapor pressure at the elevated temperatures and space environmental conditions. Over 50,000 hr of test time have been accumulated on modules to be used in the generator. Individual element failure rates are now less than 1 per 1000 during the acceptance test, yielding an over-all component survival probability of better than 95%. The allowable degradation over a year's life is now less than the specified maximum of 10%. The generator has operated at temperatures approaching the operating temperature of the reactor, thus making it possible in later designs to increase the power with little additional development effort. Over-all conversion efficiency is about 1.7% at an average hot junction temperature of 930°F. The generator development work was conducted by the Electron Tube Division of RCA, Harrison, New Jersey. As mentioned, the flight performance of the SNAP 10A system has been excellent, operating for 43 days and meeting its predicted performance goals.

Thermionic Energy Conversion

The simple cesium vapor diode, with elemental metal electrodes, continues to be the most efficient and powerful form of converter at all practical temperatures. Cylindrical electrodes provide the most efficient configuration for internally heated converters in a nuclear reactor, but planar electrodes have been used predominantly in other applications with external heat sources. A typical configuration is shown schematically in Fig. 7. Thermionic converters can presently be purchased in quantity, meeting $\pm 10\%$ output specification, with powers, efficiencies, and lifetime suitable for many practical applications. Problems associated with integrating the converter with a hostile environment, as in contact with a nuclear fuel or chemical flame, are being given increased attention in research activities.

In Fig. 8, an attempt has been made to define the best reported performance consistent with a high degree of design confidence for present applications. The power density to be expected in Ce-vapor-diode converters depends upon emitter temperature and electrode spacing. Most advanced design points fall in the shaded region, depending upon system requirements. Reported values falling below this region usually have off-optimum spacings, compromised for mechanical and lifetime reasons, or do not use refractory metal emitters (e.g., the carbide emitters). The efficiencies are based on measured power and heat flow. Most design points fall within the shaded region, depending upon the degree to which radiant heat losses are prevented and the degree to which optimum converter parameters are consistent with system constraints. Because of the ambiguous nature and inadequate statistics associated with lifetime tests to date, the summary in Fig. 8 is useful only to illustrate trends. Some test data, which illustrate typical performance obtained during 1963 and 1964 in research and application-oriented converters, are given in Table 9. Table 10 summarizes the states of systems that have been developed to the point where significant test data have been obtained. A maximum performance chart for Re-Mo converters is shown in Fig. 9.

The total support for thermionic conversion R and D is presently about $12 million/year. Approximately half of this supports the activities of the first five organizations in Table 11. The AEC and DoD contribute at least $3 million apiece, and the rapidly increasing support by NASA presently totals about $2 million. The remaining support is by smaller government agencies and private industry.

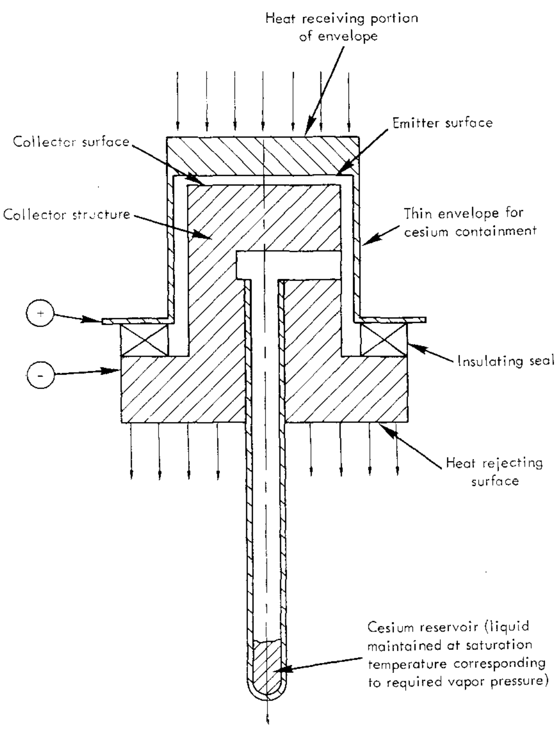


Fig. 7 Basic thermionic converter structure.

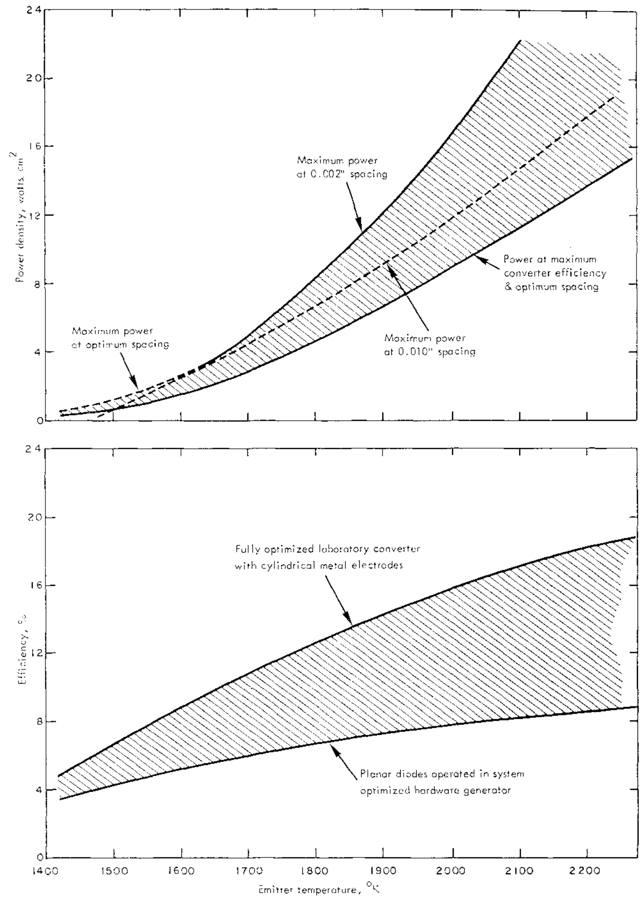


Fig. 8 Approximate summary of verified state-of-the-art power densities and efficiencies for thermionic converters.

Magnetohydrodynamic Power Generation

The critical problem in the use of a magnetohydrodynamics (MHD) device with a reactor heat source is the attainment of acceptable plasma conductivities at temperatures permitting long reactor life. Since at temperatures near 1700°K thermal ionization cannot produce acceptable plasma conductivities, a considerable amount of effort is being devoted to "nonequilibrium" plasmas in which the electron temperature is increased above the gas temperature. The only experimental verification of "nonequilibrium" ionization applicable to MHD power generation has been a favorable comparison of diode test results with theory and the experimental observation of magnetically induced effects in an MHD shock tube. The data shown in Fig. 10 were obtained from tests on a 0.23-m-long diode tube with hot electrodes and filled with cesium-seeded helium. These points were evaluated from voltage and current readings about 25–50 μsec after the initiation of the discharge, which was long with respect to the time required for collisional ionization equilibrium but short with respect to the time for heating the gas. Thus, all of the data can be considered to have been 523°K. The results show that conductivities on the order of 100 mho/m are obtained at current densities on the order of 10×10^4 amp/m² with an electric field on the order of 1200 v/m.

The results in Fig. 11 were obtained from an MHD shock tube having a generator section $2 \times 2 \times 10$ in. long. The test gas in this case was xenon; the temperature behind the shock was 5000°K. It can be seen that the conductivity increases roughly as the square of the magnetic field. The

Table 9 Recent performance achieved in some typical, advanced thermionic converters

Application	Organization*	Sponsor*	Emitter Material (Physical characteristics)	Collector Material	Emitter Area cm^2	Spacing, mils	Elec. Geom. †	Wt., g	Heat Source Used †	Emitter Temp. °C	Max. Obs. Power Density w/cm^2	Output Voltage at Max. Power v	Max. Eff. % †	Total Operating Time, hr[1]
Laboratory test vehicles	FIC	ONR	Impreg. W		8.5	195	C	--	E	1300	2.	--	6	--
	GE	-	Nb	S.S	10.7	11	C	--	E	1680	2.87	--	(9)	8538
	GE	ASD	Mo	Mo	5.0	5-10	P	--	E	1450	2.2	0.37	7	1400
	RCA	BSN	Mo	Ni	60	9	C	--	E	1350	2.66	--	10.5	1000*
	RCA	ASD	Mo	Ni	40	30	C	--	E	1200	1.2	--		800*
	TEEC	ARPA, ONR	Re,W,Ta,Mo, Nb,Ir	Mo	4.2	variable	P	--	E	Experimental results given in Fig. 13[2]				
Isotope	TEEC MM	DRD	See Table 11											
Nuclear	GA	DRD	W-clad UC	Nb	14	10	C	--	N	1750	5.0	--	(11.5)	21
	LASL	DRD	UC-ZrC	Ni	4	40	C	--	N	1880	9.0	--	(4)	284
	LASL	DRD	UC-ZrC	Ni	5	40	C	--	N	1990	15.0	--	(8)	112
	RCA	BSN	Mo-clad UO_2	Ni	60	9	C	--	N	1350	2.25	--	(8.5)	285
	TEEC	ASD	Re-clad UC	Mo	2.5	5	P	--	E	1600	8.0	0.5	--	200
Solar	AT	JPL	Mo	Mo	2.0	3	P	960	E	1700	10.25	0.6	6.5	50
	EOS	JPL	Ta	Mo	2.0	--	P	213	E	1700	6.9	0.6	--	--
	GE	ASD	Mo	Ta	3.0	2-5	P	--	E	1680	9.6-6	--	--	620
	RCA	JPL	Ta	Mo	2.0	--	P	550	E	1700	10.2	0.6	--	--
	TEEC	JPL	Ta	Mo	2.0	2	P	160	E	1700	18.6-10.8	0.8-1.0	9	1020*

Notes: 1. Life data should be obtained at specified maintenance-free conditions, and it should include the degree of performance degradation observed. A consistent set of such data is generally not available. Asterisk(*) indicates still operable.
2. This data is included here since it has been used extensively in many design calculations for thermionic systems. It generally represents the best state-of-the-art observed performance characteristics, and it has been reproduced in several types of research & hardware converters.
3. Results of tests conducted by JPL for identical test conditions (JPL Space Program Summary No. 37-21 or ASME Paper #63-MD-54).

† C = cylinder, P = planar, E = electrical, N = nuclear, M = measured, () = estimated.

* See Table 11 for organization and sponsor code letter identification.

emission spectra in the generator were also observed during these experiments. A series of runs at 8000°K showed a dramatic increase in the luminosity of both the continuum and line spectra as the magnetic field was increased from 0 to 20,000 gauss. This may be an independent verification of magnetically induced ionization.

Several MHD research loops designed for continuous operation with cesium-seeded helium at 1000°–2000°K are undergoing instrumentation calibration runs at this time. The major objective of these loops is to demonstrate "nonequilibrium" ionization under actual motion-induced electric field conditions on a continuous basis. A positive indication of the value of electron-heated plasmas for MHD power conversion should result within 1 year.

There has been substantial progress in the development of open-cycle MHD generators. A self-excited 20-Mw (net) generator will be completed shortly. Water-cooled peg-wall construction will be used for the channel. Running time will be limited to 3 min by the magnet coils (33 kgauss), which act as a heat sink. The cost of development and test of this unit is approximately $40/kw. It will operate on seeded products of combustion of ethyl alcohol and oxygen with stagnation conditions of 8 atm and 3100°K. The gross dimensions of the generator, including the magnet, are

Table 10 Thermionic space power systems development

Type of system	Organization	Sponsor	Design objectives	Status	Availability
ISOTOPE SNAP 13	TEEC MM	DRD	Meet Surveyor requirements: 25 w at 1 v for 130 day	Power objectives met. Generators are 3.3 w/lb incl. fuel. Those on electrically heated life test still operable.	Fueled within 1 yr Available 2 yr
NUCLEAR In-pile Concept (SNAP 70)	AI, B&W, GA, GE, LASL, MM, P&W, RCA, TEEC	ASD, DRD, LRC, ONR, BSN	0.3 to 10 Mw for 1 yr. Estimated unshielded system wt, 4 to 10 lb/kw, for T_{fuel} of 1500° to 1900° C	Basic feasibility still uncertain from fuel & materials standpoints. Various electrical & in-pile tests of fueled converters have short lifetimes (<300 hr).	Variously estimated at 5 to 10 yr
Pile-Surface Concept (STAR-R)	GE	--	10-40 kw. Estimated unshielded system wt., 20 to 65 lb/kw for fuel temperatures of 1900° to 2300° C	Feasibility still uncertain. No fueled converter tests under assumed conditions	Requirement uncertain. Duplicates SNAP 8 power range
Thermionic Radiator Concept	AGN RCA	--	0.3 to 10 Mw. Estimated unshielded spec. wt., 10 to 20 lb/kw for fuel and coolant temperatures of 1200° to 1500° C	Feasibility still uncertain for coolant, fuel, and materials. Li-heated converters demonstrated for short time, (AGN-RCA). P&W also planning converter tests	Unknown (3-7 yr)
SOLAR TEE-100 watt	TEEC TRW	ASD	100 w at 1700° C using 5-ft concentrator	5-diode generator. Electrically tested; 120 w at 1700° C. Solar tested; 19 w at 1380° C	Unknown
STEPS II	GE	ASD	Demonstrate system feasibility 5-ft concentrator	3-diode generator. Solar tested; 27.9 w at 1670° C	Unknown
SET I	TEEC EOS	JPL	135 w at 1700° C using 5-ft concentrator; 3000 hr	SET Ia (5-diode generator): solar tested: 41 w at ~ 1700° C. SET Ib: electrically tested: 82.5 w at 1700° C JPL will solar test Specific power: gen. = 24 w/lb; complete system = 7.8 w/lb	Flight test in 2 yr (est.) Available in 4 yr

Table 11 Organizations[a] and government agencies active in thermionic conversion R & D

Major contractors (> $10^6, alphabetically)
GA—General Atomic
GE—General Electric
LASL—Los Alamos Scientific Lab.
RCA—Radio Corp. of America
TEEC—Thermo Electron Engineering Corp.
Other organizations (alphabetically)
AGN—Aerojet General Nucleonics
AC—Aerospace Corp.
ANL—Argonne National Lab.
AERE—Atomic Energy Res. Est. (Earwell, England)
AI—Atomics International
BC—Boeing Co.
CC—Consolidated Controls
EOS—Electro-Optical Systems
FIC—Ford Instrument Co.
GEL—General Electric Co., Ltd. (England)
GM/A—General Motors, Allison Div.
HA—Hughes Aircraft
MC—Marquardt Corp.
MM—Martin Marietta
RA—Republic Aviation
SCNS—Saclay Center for Nuclear Studies (France)
TRW—Thompson-Ramo-Wooldridge
TI—Texas Instruments
UC—Union Carbide
UA—United Aircraft (P&W)
W—Westinghouse
3M—Minnesota Mining and Mfg. Co.
GA—Gen. Atomic Div., General Dynamics
Government agencies (alphabetically)
ARPA—Advance Res. Proj. Agency, DoD
ASD—Aeronautical Sys. Div., USAF
BSN—Bureau of Ships, Navy
CRL—Cambridge Res. Lab., USAF
DRD—Div. of Reactor Dev., AEC
GSC—Goddard Space Center, NASA
JPL—Jet Propulsion Lab., NASA
LRC—Lewis Res. Center, NASA
ONR—Office of Naval Res., Navy
RADC—Rome Air Dev. Center, USAF
SRDL—Signal R & D Lab., Army

[a] In addition, many universities and institutes, here and abroad, have research activities in thermionic conversion and related technologies.

approximately 15 × 9 × 11.3 ft; its weight, which is nearly all field coil, will be 200,000 lb. The gross input power from the combustion chamber is approximately 360 Mw.

The scaling laws for MHD generators favor large size. It is conceivable that 10^{10}-w units for limited duty cycle operation could be built for about $3/kw installed. Using a copper coil, the generator would weigh between 400 and 1500 tons, depending on the amount of field dissipation that would be tolerated. If field dissipations of up to 10% of the gross output of the generator could be used, it is conceivable that the weight could be reduced to the vicinity of 100 tons. These remarks apply to the case of MHD generators driven by typical rocket propellants.

In summary, very little progress has been made toward defining and developing space applications for MHD power conversion. Although commercial exploitation of open-cycle MHD for power generation seems within our grasp, it is obvious that a number of fundamental technical advances must take place before we can say the same for the nuclear space application.

Batteries and Fuel Cells for Space

The material in Tables 12–14 represents an attempt to condense into manageable proportions the vast and heterogeneous information available on electrochemical power generation and storage systems for space use. (In the section

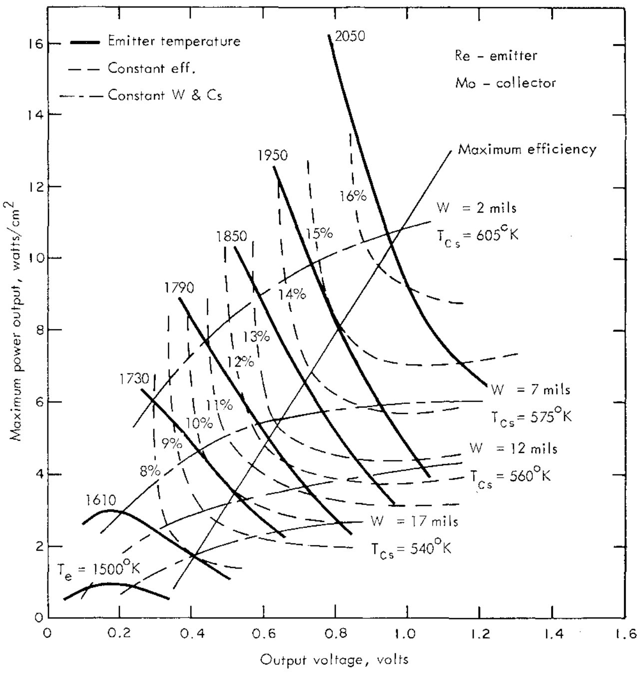


Fig. 9 Maximum performance chart for Re-Mo converters, showing values of optimum spacing W and cesium reservoir temperatures T_{Cs}.

on fuel cells there is no mention made of the Apollo program, because data on the Apollo program are unavailable for publication in the open literature.) Electrochemical devices are often employed in conjunction with other systems for storage purposes, particularly for solar systems where intermittent operation is due to the device being in the shadow of the earth or other geocenter. Electrical storage devices are also required for those missions in which the power-time profile contains brief, high-power peaks. Thus, few space power systems operate without some form of electrochemical energy storage.

A review of fuel cells for space application can be found in Ref. 34. In the Gemini and Apollo development programs, engineering difficulties are being experienced, but the scientific principles of the ion-exchange membrane and Bacon fuel cell systems are considered to be sound, and the problems will

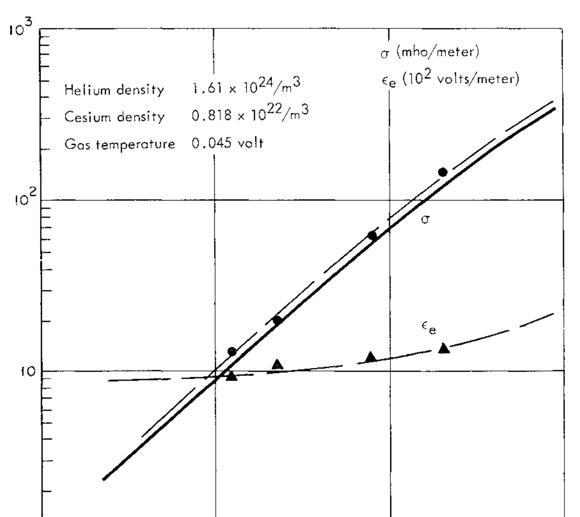


Fig. 10 Experimental data on breakdown potential ϵ_e and conductivity σ vs breakdown current density J_e for cesium-seeded helium in an MHD shock tube.

Table 12 Primary and secondary space-type fuel cells*

	Eagle Picher	GE Gemini (Advanced)	UC C elect. (C-metal)	TRW	GM – ALLISON	TRW	IGT	TI	W
Fuel/oxidant	Zn/O_2	H_2/O_2	H_2/O_2	H_2/O_2	K-Hg/K-air[e]	H_2/Li	H_2/CO_2-air	H_2/O_2	H_2, CO, CH_4, Coke/O_2, Air
Anode/catalyst	Pt	Ta & Pt	C+cat. (C+ metal)	Pt black	---/---	Li pool	Pt-Ag foil/Ni	---	Porous Pt
Cathode/cat.	Ag	Ta & Pt	C+cat. (C+ metal)	Pt black	---/Ag	No diaphragm	Ag paint	---	Porous Pt or Ni
Electrolyte	KOH	ion exchange membranes containing polystyrene H_2SO_4	KOH	KOH	KOH+KI+KBr[f]	LiH/LiF/LiCl	Molten carbonate	$LiNaCO_3$	Stabilized ZrO_2
Operating range, T, °C	-20-50	25-60 (20-70)	20-80	0-100	275-425/300 & 100[g]	550	450-700	500-600	800-1100
P, atm	---	0.5-2	1-2	1-2	1	1	1-1.1	1	1[i]
Light load, amp/ft^2	1	5.3 (7)	50	20	300/100	0-100	15	75	10-20
v/cell	1.45	0.9 (0.93)	0.9	0.95	0.44/2.4	0.46	0.7/0.92	0.92	1
life, hr	1000+	2000[a]	6000+[b] (2000+)	200	1 yr/---	---	---/1500+	1000+	2000+
efficiency, %	(200 w-hr/lb)	60-62	61	71	10-63	---	---/20	65	90 (free-energy eff.)
Overloading, amp/ft^2	---	40 (150)	200[d]	250	700-300	3500	---/60	200	750
v/cell	---	0.77 (0.73)	0.77	0.5	0.50/1.8	shorted	---/0.71	0.6	0.5
Normal load, amp/ft^2	5	11.5 (100)	100	100	500/200	1400	25	125	250
v/cell	1.40	0.88 (0.80)	0.85	0.8	0.48/2.1	0.30	0.5-0.6/0.85	0.8	0.8
life, hr	400+	a	4000[b] (on test)	200	1 yr/---	---	5000/1000+	1000+	---
efficiency, %	(160 w-hr/lb)	59 (54)	57	57	7/54	---	10/20	50	75
w/ft^3	---	590 (1400)	400 (740)	4000	---/2000[h]	400 w/ft^2	200-400/800-1200	300	3000-4000
w/lb	---	14.6 (35)	8.6 (16.2)	40	10,000/11,000[h] 10-30/56	---	1.5-2.5/4-5	5	20-25
Cost, \$/kw	---	47,500 (---)	---	---	---	---	1500-2000/200-300	---	---
Useful power range, kw/battery	---	1-1.5 (1-5)	0.005-1000	0.05-10	0.5-60/100-1000	---	1-100	2+	0.5+
Estd. yr available	1965	Now (1965)	Now	1966	1967	1968	1965	1965	Now

*All data presented as received from manufacturer. May apply only to individual cells, especially with respect to hfc.

[a]With Gemini load profile, life is now 1000 hr.

[b]Life is now limited by auxiliaries to 2000-3000 hr.

[c]Including cell and accessories except reactants and tankage for primary system, or complete secondary system, but exclusive of radiator and power conditioning equipment cases.

[d]Maximum power at 350 amp/ft^2; usable for up to 0.5 hr.

[e]The K-air system does not include recharging accessories.

[f]The K-air system requires a separate aqueous KOH electrolyte, also.

[g]300°C for K and 100°C for air.

[h]Specific power of K-air system is based on 150-kw system exclusive of tankage and radiator.

[i]Or any other pressure as long as $\Delta P \approx 1$ atm on both side of electrolyte.

undoubtedly be overcome. Table 12 presents fuel cell information obtained from various manufacturers. Unfortunately, no data were received from some of the major firms in this field, e.g., Pratt and Whitney Aircraft, Allis-Chalmers, and Electro-Optical Systems. Allis-Chalmers plans to orbit a hydrogen-oxygen fuel cell on their Air Force program, and on a NASA program they are developing a simple and reliable method for removing heat and water with the least number of mechanical moving parts and minimum need for parasitic power. This goal should be attained by evaporating water through a capillary membrane adjacent to the electrodes, the cavity behind the membrane being evacuated to the vapor pressure of the KOH electrolyte at its operating temperature of about 200°F.

Electro-Optical Systems has completed work on one contract and is beginning follow-up studies on electrolytically regenerative hydrogen-oxygen cells, using a 50-50 mixture of

Table 13 Primary space batteries

Anode/cathode	Zn/AgO	Zn/Ag_2O	Cd/AgO	Cd/Ag_2	Mg/AgCl	Mg or Ca/$CaCrO_4$
Electrolyte	KOH	KOH	KOH	KOH	KSCN-NH_3	Fused salt (thermal cell)
Separator(s)	Semi-permeable, nylon + cellophane cynpor + cellophane, or cellulosic		Semi-permeable		Special	Special
Seal or vent	Automatically activated, with pressure vents, or sealed		Automatically activated and with pressure vents		Automatically activated	Automatically activated
Case	Plastic or nylon, rect. or cyl.		Plastic or nylon, rect. or cyl.		Special	Drawn stainless-steel cyl.
Perf. at 25°C:[a]						
v/cell	1.5 - 1.7(1.82)	1.2-1.6(1.59)	1.1(1.38)	1.05(1.16)	2.15(2.35)	
w-hr/lb[b]	50-110(230)	30-90(124)	25-50(143)	20-35(79)	8 (185)	
w-hr/in.^{3}[b]	5-10(61)	2.5-8(31)	2.8-6(40)	2.3-4.2(22)	0.9 (39)	
Wet-stand loss, %,[c]					Automatically activated	Automatically activated
at 25°C, 1 mo	10	< 10-15	20	5		
6 mo	40	< 40	50	20		
Cost, \$/w-hr at 25°C:[d]						
vented	0.35 - 1.0		0.60 - 2.0			
sealed	0.35 - 1.5		0.75 - 2.5			5.00 - 1000
Avail. sizes, amp-hr	1 - 400		0.5 - 300		Exp. models	0.001 - 0.5

[a]Numbers in parentheses are theoretical values. Actual values for Mg/AgCl obtained at -65°F.

[b]Dependent upon cell size, discharge rate, and number of cycles required.

[c]Percent loss of full (not rated) charge during stand. Losses are 1/3 to 1/2 as great at 0°C, much higher at 50°C.

[d]Inversely related to cell size and quantity.

Table 14 Sealed secondary space batteries (KOH electrolyte) available in cylindrical or rectangular stainless, plastic, or nylon cases[a]

Anode/cathode	Cd/Ni	Cd/Ag	Zn/Ag
Separator(s)	Pellon, polypropylene, or "semi-permeable"	Ion-exchange, Nylon + cellophane, or cellulosic	Same as Ag/Cd; also Visking
Seal alternates[a]	M,G,C	M,P,C	M,P,C
Theo. perf. at 25°C:			
v/cell	1.30	1.16 - 1.38[c]	1.59 - 1.82[c]
w-hr/lb[b]	148	79 - 143[c]	124 - 230[c]
w-hr/in.^{3}	27	22 - 40[c]	31 - 61[c]
Actual perf. at 25°C and 50 % disch.[d]			
v/cell	1.24 - 1.25	1 - 1.1	1.4 - 1.55
w-hr/lb	6 - 9	7 - 15	4 - 8[e]
w-hr/in.^{3}	0.6 - 0.9	0.9 - 1.6	1.2 - 3
Life, thousands cycles[f]			
T, °C - % Disch.			
0 - 25	10	7+	0.5 - 0.6
0 - 50	5	4+	0.3+
0 - 65	4	1.5+	0.15+
25 - 25	10	3 - 12+	0.3+
25 - 50	4	1.5 - 7+	0.15+
25 - 65	2.5	1 - 1.8	0.05+
50 - 50	2.5	0.5 - 0.6	0.05+
Shelf life, yr			
at 0°C, disch.	8	3 - 4	1 ± 3
charged	8	2 - 3	1
at 25°C, disch.	8	2 - 4	1 - 3
charged	8	1 - 2	0.5 - 0.7
at 50°C, disch.	4	½ - 3	0.4 - 0.5
charged	4	¼ - 1	0.2 - 0.3
Cost, $/w-hr (at 25°C and and 50 % disch.)	1 - 3; 20 - 50[g]	1.20 - 4	0.75 - 3.50
Available amp-hr sizes	0.5 - 50	0.1 - 300	0.1 - 300

[a]M = mechanical, G = glass, C = ceramic, P = plastic.
[b]Based on active materials only.
[c]Lower voltage for argentous, higher for argentic oxide.
[d]For Ag, only argentous data are given.
[e]Lower range for 100-min cycle, higher for 24-hr cycle.
[f]Thousands of 100-min orbits; numbers of cycles for 24-hr orbit are 0.1 to 0.5 as large for Cd/Ag, almost unchanged for Zn/Ag; Cd/Ni not recommended for Cd/Ni.
[g]Estimated ranges by two different sources.

platinum and palladium as the anodic catalyst and pure platinum as the cathodic catalyst. Optimum operating pressure appears to be in the range of 50–400 psig; the optimum operating temperature has not yet been established. Present power densities are up to 90 Mw/cm^2 at room temperature and 150 Mw/cm^2 at 70°C.

Silver-cadmium secondary batteries have now been flown on several satellites and have given quite satisfactory service. Testing has advanced sufficiently to indicate that these batteries, with their considerably higher energy density, have cycle lives at least equal to the nickel-cadmium system. The most significant recent development in the field of secondary batteries is the introduction of a third electrode (usually connected to the cadmium electrode), which permits a much higher rate of charging than has been possible heretofore.[35]

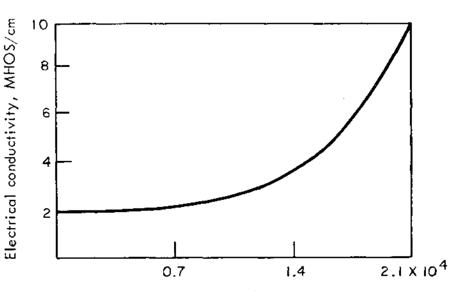


Fig. 11 Experimental effect of magnetic field on conductivity for xenon in an MHD shock tube.

Table 15 Future power conditioning (d.c. to d.c.) equipment estimates for 28-v output (Mil Spec 704) (data from the General Electric Company)[a]

Power, kw	Input, v	Voltage reg., %	Spec wt, lb/kw	Spec vol., in./kw	Full-load eff., %
0.025	1	100	28	360	85
	9	100	18	240	90
	20	100	8[b]	100	90
0.150	6	100	18	240	90
	12	100	12	160	92
	20	40	6	85	94
1.0	12	100	11	140	91
	20	40	5.5	75	93
10	20	40	5	70	92

[a] For d.c. to a.c.: weights and volumes > twice those for d.c. to d.c.; efficiency about the same, i.e., 90 + %.
[b] Assumes eventual availability of high-speed, switching transistors no now in hand.

Table 16 Direct current-alternating current inverters 3 ph, 115 v, 400 cps, fully regulated output (data from Westinghouse Electric Corporation)

Application	Power out, KVA	$V_{in.}$, v. d.c.	lb/kw	Full-load eff., %	Spec vol., in.³/kw
Apollo	1.25	25–31	30	80	1000
Centaur	0.6	25–31	42	77	1000
Lunar base	10.0	50–60	16	90	420
MOSS	1.0	50–60	25	81	1000
d.c.-d.c. converter (42 v. dc., current limited, output)					
Saturn	1.0	50–60	11	90	180

The auxiliary electrode, essentially a fuel cell oxygen cathode, can be used either for sharply controlling the cutoff for charging or simply as a gas recombination device that avoids the danger of excess pressures being built up. Third electrodes will probably become available soon in both Ni-Cd and Ag-Cd systems.

A novel concept, on which development was started in 1963 under a NASA contract, is the dry tape fuel cell or battery. In principle, this consists of a porous tape that is coated with anodic material on one side and cathodic material on the other. Electrolyte is added just before the tape is pulled through two current collectors with which it is in contact just long enough to discharge the electrochemical couple. Crude Ag-Zn dry tapes have already given performance equal to currently available, conventional Ag-Zn batteries. Optimization, as well as the use of other couples, is expected to yield a device with very high energy and power-density capability. This concept may yield lightweight reserve-type batteries and may make possible the use of normally incompatible chemicals. Tables 13 and 14 contain data on the current state of space batteries as provided by a number of manufacturers and laboratories.

Power Conditioning

Some of the key power-conditioning functions are as follows: a) accept prime power and transform, convert, invert, rectify, regulate, filter, or limit it in order to satisfy all of the user subsystem requirements; b) respond to external commands by applying power, removing power, and changing characteristics; c) operate in an open- or closed-loop control system in response to external control functions; and d) take corrective or protective action in the event of malfunctions. The power-conditioning system thus has interface requirements with almost every other subsystem and figures strongly in system evaluations from a standpoint of reliability, efficiency, weight, and general probability of mission success. The problem is to interject power-conditioning technology into the system design early enough to allow optimization of the power-conditioning system.

Although an a.c. prime power source is versatile and simplifies the power-conditioning system, the majority of applications today are based on the use of low-voltage d.c. originating from solar cells, thermoelectric converters, and batteries. The Ranger spacecraft power system is one example of a design that utilizes solar panels, batteries, distribution in the d.c. phase, and individual conversion at the location of the using device. The data in Tables 15 and 16 were provided by General Electric and Westinghouse to indicate some of the probable characteristics of initial power-conditioning systems for space vehicles.

References

[1] Brown, D., NASA Lewis Research Center, private communication (1964).

[2] Heath, A. R., Jr., "Status of solar energy collector technology," ARS Paper 2535-62 (1962).

[3] Egli, P. H. and Sherman, G. W., "Energy sources for the future," Society of Automotive Engineers Paper 645B (January 1963).

[4] Technical staff, "Sunflower status and application considerations," Thompson Ramo Wooldridge Inc., Report (May 1963).

[5] Morgan, N. E. and Dittman, B. F., "Hydrox engine development demonstrates advantages for space power applications," ARS Paper 2519-62 (1962).

[6] "Redesigned Snap 8," Nucleonics **21**, 79 (July 1963).

[7] Scott, W. C. and Schulman, F., "Space electric power," Astronautics Aerospace Eng. (May 1963).

[8] Benjamin, W. D. and Vargo, E. J., "The current status of materials compatibility with two-phase alkali metals," Thompson Ramo Woolridge Inc., TM 3697-67 (May 1963).

[9] Diedrich, J. H. and Lieblein, S., "Materials problems associated with the design of radiators for space power plants," ARS Paper 2535-62 (1962).

[10] Coombs, M. G. and Stone, R. A., "SNAP-2 radiative-condenser design," *ARS Progress in Astronautics and Rocketry: Space Power Systems*, edited by N. W. Snyder (Academic Press, New York, 1961), Vol. 4, pp. 301–324.

[11] Askwith, W. H., Hayes, R. J., and Mikk, G., "The emittance of materials suitable for use as spacecraft radiator coatings," ARS Paper 2538-62 (1962).

[12] Brooks, R. D. and Sowochka, S. G., "Alkali metal two-phase heat transfer for space power—present status," ARS Paper 2547-62 (1962).

[13] Edwards, E. K. and Rodrick, R. D., "Spectral and directional thermal radiation characteristics of surfaces for heat rejection by radiation," ARS Paper 2546-62 (1962).

[14] Daughman, C. L. and Stump, F. C., "The effect of system parameters on space electric power system rates," AIAA Paper 63-243 (1963).

[15] Norman, L. W., "The application of the recuperated Brayton cycle to space power conversion systems," AIAA Paper 63-220 (1963).

[16] Bjork, R. L., "Meteroroids versus space vehicles," ARS Paper 1200-60 (1960).

[17] Loeffler, I. H., Lieblein, S., and Clough, N., "Meteoroid protection for space radiators," AIS Paper 2543-62 (1962).

[18] Shure, L. I., "Heat transfer test capsule design report," Thompson Ramo Wooldrige Inc., Rept. ER-4559 (July 1962).

[19] "Spur power system," Quarterly Progress Rept. SY 5396-R4, AiResearch Manufacturing Co. (April). 1963.

[20] Balje, O. E., "A study on design criteria and matching of turbo machines; Part A Similar to relations and design criteria of turbines," American Society of Mechanical Engineers Paper 60-WA-230 (1960).

[21] Silvern, D. H. and Balje, O. E., "A study of high energy level, low power output turbines," Sundstrand Turb. Rept. AMF/TD 1196 (April 1958).

[22] Kearton, W. J., "Some experiments on the flow of mercury vapor thru nozzles," Proc. Inst. Mech. Engrs. (London) (December 1929).

[23] Hill, P. G., Whitting, H., and Demetri, E. P., "Condensation of metal vapors during rapid expansion," American Society of Mechanical Engineers Paper 62-WA-123 (1962).

[24] Smith, M. I. and Fuller, D. D., "Journal bearings operations at super laminar speeds," Trans. Am. Soc. Mech. Engrs. **78,** 469–474 (1956).

[25] Constantinescu, V. M., "Analysis of bearings operating in turbulent regime," J. Basic Eng. 139 (1962).

[26] Hall, J. and Spies, R., "Determination of working fluid lubrication capability in journal bearings," U. S. Air Force Rept. ASE-TDR-62-630, Pt. II (May 1963).

[27] Briggs, R. W. and Dobler, F., "Development of potassium cooled AC generators for space applications," AIAA Summer Meeting, Los Angeles, Calif. (June 1963).

[28] Gignoux, D., "Constant oblique field electrostatic generator," ARS Paper 2556-62 (1962).

[29] Coenraads, C., Denholm, A. S., Lavelle, J., and McCoy, F., "Electrostatic power generators for space," ARS Paper 2555-62 (1962).

[30] Krebbs, R. P., Wench, B. M., and Lieblein, S., "Analysis of a megawatt level direct condensor radiator," AIS Paper 2545-62 (1962).

[31] Papell, S. S., "An instability effect 2-phase heat transfer for sub-cooled water flowing under conditions of zero gravity," AIS Paper 2548-62 (1962).

[32] Burge, H. L., "Revolving belt space radiator," ARS J. **32,** 1243–1248 (1962).

[33] Anderson, G. M., "Nuclear reactor systems," Astronautics Aerospace Eng. (May 1963).

[34] Szego, G. C. and Cohn, E. M., "Fuel cells for aerospace application," Astronautics Aerospace Eng. **1,** 107–111 (May 1963).

[35] Shair, R. C., "Sealed secondary cells for space power systems," AIAA Paper 64-455 (1964); also J. Spacecraft Rockets (to be published).

[36] Zipkin, M. A. and Edwards, R. N. (eds.), *Progress in Astronautics and Aeronautics: Power Systems for Space Flight* (Academic Press, New York, 1963), Vol. 11.