ARTICLE NO. 79-1337R

A80-060

Nuclear Electric Power System for Solar System Exploration

,0004
10002

W.M. Phillips*
Jet Propulsion Laboratory, Pasadena, Calif.

This paper discusses the design of a nuclear electric propulsion system for outer-planet exploration. A nuclear electric power system with a power level of 200-250 kWe is recommended. Current technology appears capable of accomplishing the early missions, and growth potential exists for accomplishing more difficult later missions without significant changes in the basic system.

Introduction

THE national program on unmanned exploration of the solar system was initiated in the early 1960's with reconnaissance of the moon, Mars, and Venus. In the same time frame, reconnaissance of the outer planets was initiated. These programs were accomplished through a combination of chemical propulsion, radioisotope thermoelectric power conversion, and photovoltaics.

The mass of the spacecraft used for each of these solar system missions is shown in Fig. 1 both as a function of solar system target and as a function of the year when data returned to Earth from the spacecraft. Planned missions are shown as dashed lines in this figure. Superimposed on this data is the capability of chemical propulsion and solar electric propulsion (SEP) for direct flight missions. Gravity assist has been and will be used to increase the payload capability of both chemical and SEP-powered spacecraft, particularly for reconnaissance or fly-by missions. This approach is indicated in Fig. 1 via payloads in excess of direct flight capability. This technique is limited by the increase in mission time required and by the need for retropropulsion to place the spacecraft in orbit about the target body. Optimization of orbital missions using gravity assist require extensive detailed calculation beyond the scope of this paper. Variations of SEP capability also occur because of the target planets relation to the sun which varies with launch date and affects power available to the propulsion subsystem. Thus, from Fig. 1 it can be concluded that continuation of a balanced program of planetary exploration and Earth orbital missions can be maintained through the 1980's with improved versions of the same power and propulsion systems as have been used in the past. Beyond this time frame, sun-independent spacecraft for exploration and intensive study of the outer planets will be required. Design efforts have evolved a design concept for a nuclear electric propulsion (NEP) spacecraft over a range of power levels.

Selection of the power level for NEP systems will depend not only upon science payload requirements, but also upon Shuttle capability and orbital altitude before system startup. Other possible applications for a space nuclear power system must be considered to amortize the cost of applying nuclear space power to these missions. Evaluation of some of these factors are presented in this paper on the selection of an NEP system.

Presented as Paper 79-1337 at the AIAA/SAE/ASME 15th Joint Propulsion Conference, Las Vegas, Nev., June 18-20, 1979; submitted July 10, 1979; revision received Jan. 18, 1980. This paper is declared a work of the U.S. Government and therefore is in the public domain.

Index categories: Electric and Advanced Space Propulsion; Spacecraft Configurational and Structural Design (including Loads).

*Engineering Group Supervisor. Member AIAA.

NEP Mission Analysis

The objective for NEP in planetary exploration is the delivery of payloads of several metric tons to the outer planets in reasonable missions times (10 yr or less) for intensive study of these planets. Advantages of NEP as compared to SEP arise from the fact that NEP uses electric thrusters for both acceleration and deceleration to place the spacecraft into a circular orbit about the target planet or moon whereas chemical retropropulsion is required for the SEP system. The SEP requires chemical retropropulsion because of the low power output of the solar cells at the outer planets. Hence, the solar cells and the electric thrusters of an SEP spacecraft are separated from the payload and only the payload is put into planetary orbit. To maintain an adequate payload capability the orbit is ellipsoidal rather than circular and a second power source such as radioisotopic thermoelectrics must be a portion of the payload. NEP has onboard power and telemetry eliminating this weight requirement as compared to SEP.

Mission analysis of low-thrust spacecraft was done using a computer program developed especially for this method of interplanetary transfer. In all cases, the spacecraft was assumed to be Shuttle-launched to low Earth orbit of 6800 km. Launch capability of the Shuttle was assumed to be 29,000 kg.[1] Departure from low Earth orbit was assumed to be spiral. A spiral capture into a circular orbit at the target planets was also assumed. Radius of the orbit at the target planet was taken identical to that of its major moon, i.e., Titan at Saturn, Titania at Uranus and Triton at Neptune. Mission time includes Earth escape, transit, and planet capture times.

Mission analyses were done at three power levels—200, 300, and 400 kWe—input to the power conditioning system which

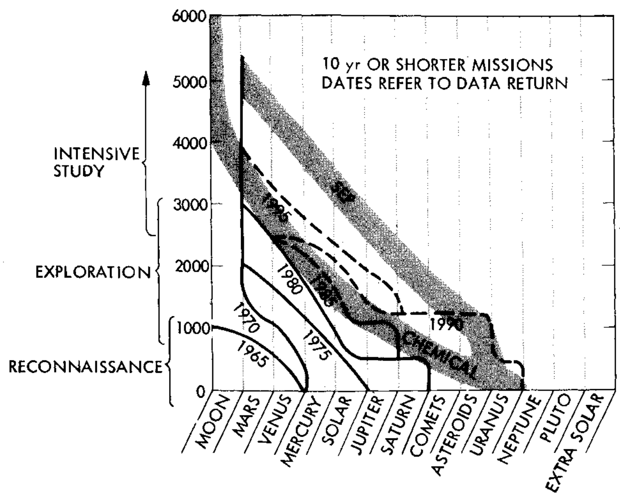

Fig. 1 Summary of previous and planned solar system missions.

was assumed to be 90% efficient. Thruster efficiency was assumed to equal

$$\frac{1}{1+\left(\dfrac{25.19}{\text{ion escape velocity, km/s}}\right)^2}$$

Launch dates were close to optimal in 1993, however, some data scatter is present as a result of failure to use the best date in every case.

Specific impulse of the thrusters was limited to either 6400 s or 11,000 s. The significance of these limits will be discussed later. Mission analysis results of these calculations in terms of total mass in orbit as a function of mission time are shown in Figs. 2-4.

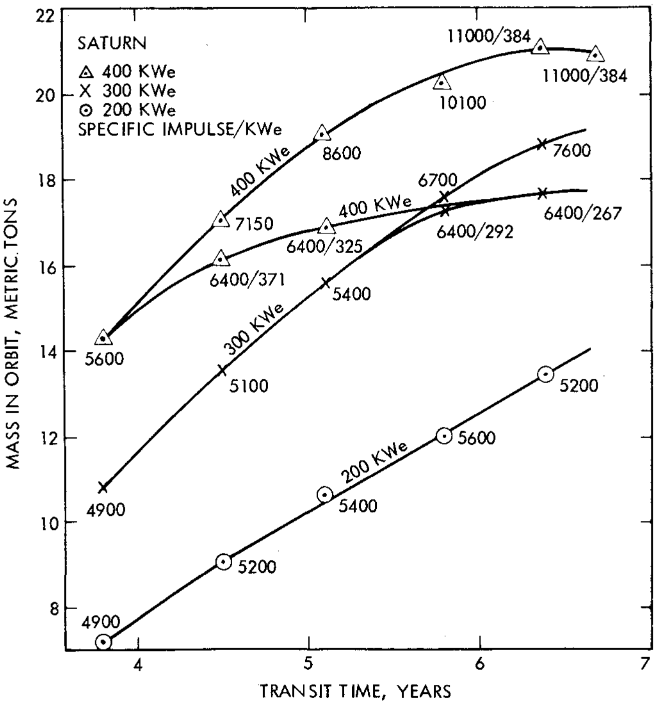

Fig. 2 Total spacecraft mass in orbit about Saturn as a function of mission time, system power level, and thruster specific impulse.

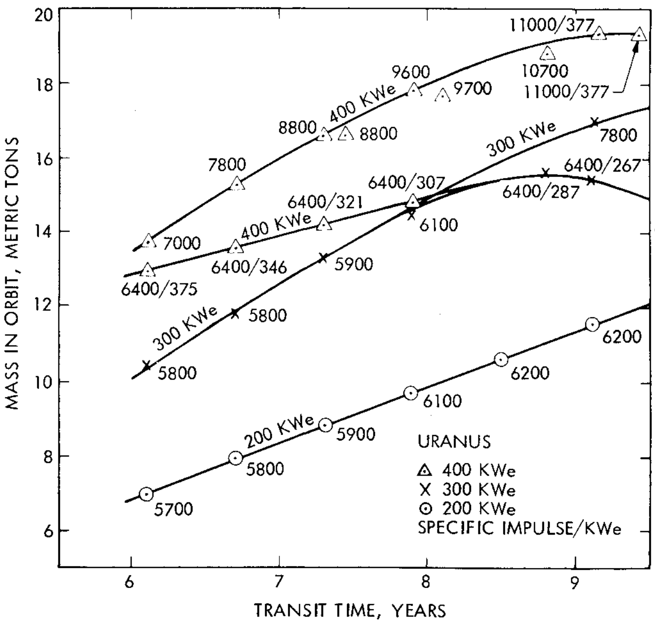

Fig. 3 Total spacecraft mass in orbit about Uranus as a function of mission time, system power level, and thruster specific impulse.

In all three figures, when an assumed thruster specific impulse limit is reached optimum power decreases. This is illustrated in Fig. 4 for a 400 kWe system. Optimum specific impulse increases from 8000 to 11,000 s as mission time increases from 7¾ to 10+yr and optimum power is 400 kWe. At 10½ yr with specific impulse limited to 11,000 s optimum power is 392 kWe. At 11½ yr this power decreases further to 367 kWe. If specific impulse is limited to 6400 s, 400 kWe is required only for a 7¾ yr mission. At 9 yr optimum power is 313 kWe and 271 kWe at 10½ yr. Similar effects occur in all missions. Thus, it can be concluded that for all three planetary missions, Saturn, Uranus and Neptune, the optimum power level for thrusters limited to a specific impulse of 6400 s and a launch mass of 29,000 kg is 200 to 300 kWe. For a specific impulse of 11,000 s optimum power level is approximately 400 kWe.

Thruster and Propellant Selection for NEP

Two electric thrusters are presently under development. The ion thruster using mercury propellant has been strongly supported for the last 15 yr. These thrusters are now approaching flight status in the 30 cm grid configuration operating at a specific impulse of 3000 s. The magneto-plasma dynamic (MPD) thruster operating in argon is in the initial stages of development and feasibility of application to long missions is yet to be demonstrated.

Selection of a propellant for the ion thrusters rests primarily in the specific impulse range of operation. Operation with mercury or cesium is expected to be limited to a specific impulse less than 6400 s. Operation with argon is expected to be limited to specific impulses greater than 5000 s.

Concern about mercury contamination of the atmosphere exists although more than 90% of the mercury is expelled by the thruster at Earth escape velocity and will not re-enter the biosphere. Availability of mercury in the quantities needed for NEP missions is questionable. Argon is available in large quantities at a much lower cost. Storage of liquid argon for the mission times of interest appears feasible.[2] The storage method would be a nonvented low-pressure two-phase cryogenic with temperature maintained by regenerative refrigeration.

Propellant selection for the MPD thruster is expected to be limited to argon at specific impulses below 5000 s. Above 5000 s a lighter propellant is required which would probably be helium. Should this technology advance to the point of

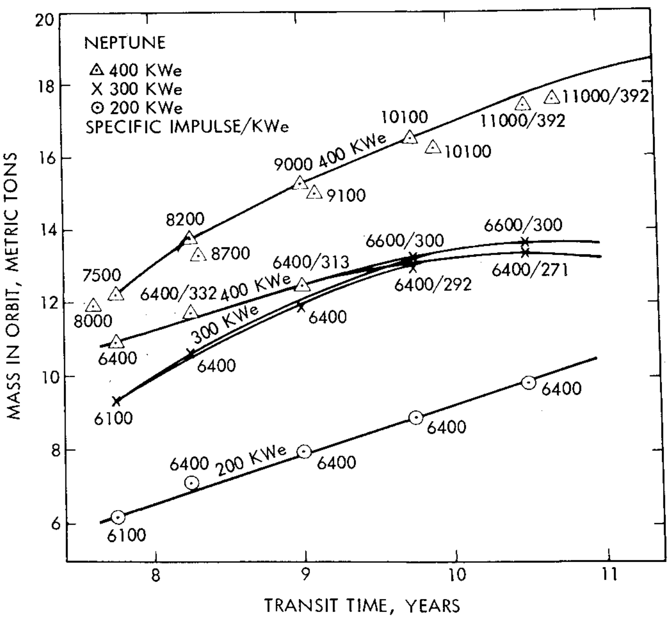

Fig. 4 Total spacecraft mass in orbit about Neptune a function of mission time, system power level, and thruster specific impulse.

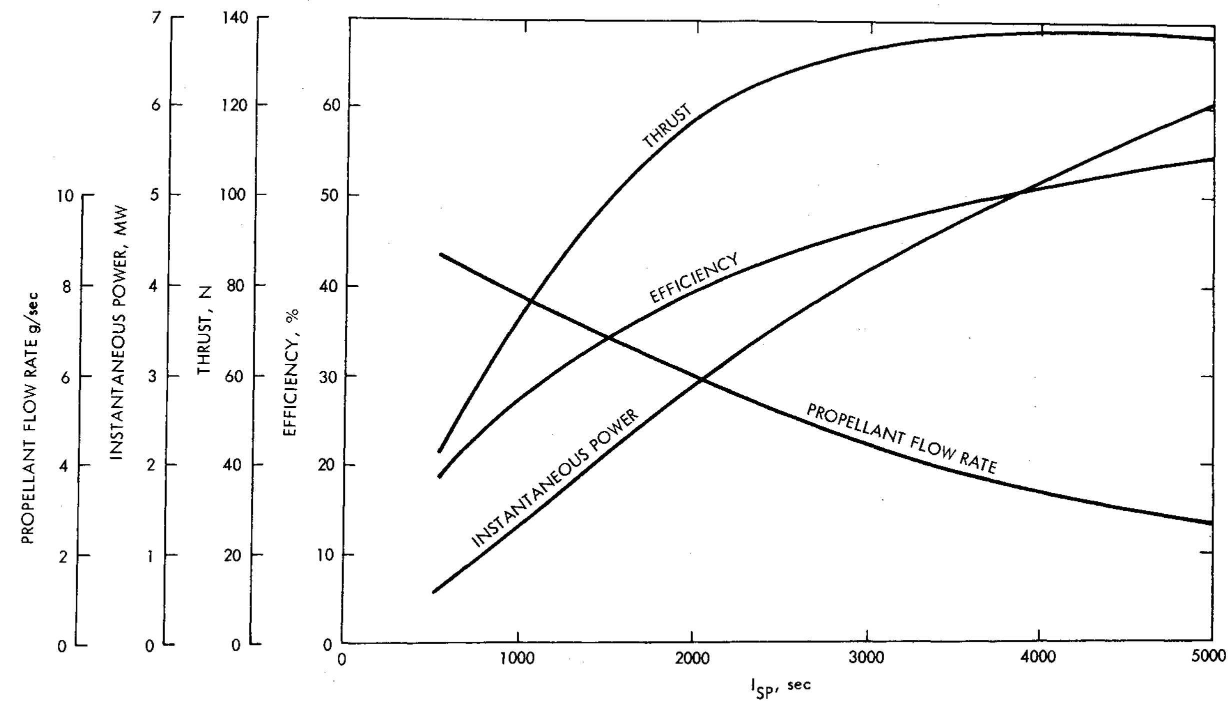

Fig. 5 Characteristics of MPD thrusters as a function of mission time, system power level, and thruster specific impulse.

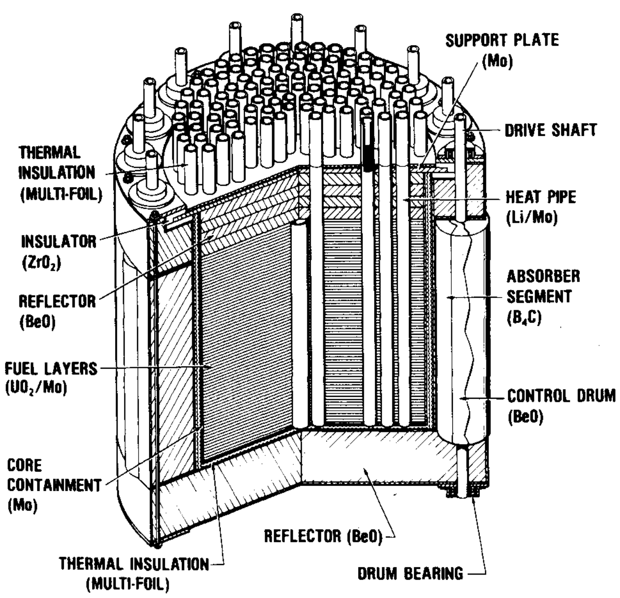

Fig. 6 Heat-pipe-colled 3 MWth fast reactor.

allowing consideration of these thrusters propellant requirements should not pose significant problems.

Key characteristics of MPD thrusters as a function of specific impulse are shown in Fig. 5.[3] The major problem area associated with MPD thrusters is the lower efficiency as compared to ion thrusters. This will require larger input powers, hence large power systems of greater mass to obtain the jet power required. This in turn means increased specific mass in terms of total system mass/jet power. Some of this mass increase can be offset through the ease of integration of an MPD thruster into the spacecraft.

The power conditioning subsystem approach will be entirely changed from that for ion thrusters. MPD thrusters operate in the quasi-steady-state mode. Instantaneous power levels are in the multimegawatt range. The power source supplies hundreds of kilowatts. Hence the thruster must be pulsed and energy storage must be provided in the power conditioning subsystem. The on/off ratio is determined by the ratio of power available to instantaneous requirements while pulse rate is determined by a combination of thruster design and power conditioning system design. Preliminary evaluations have indicated that mass savings can be achieved in the power and power conditioning subsystems which will also offset the effects of MPD thruster efficiency.[4] Thus MPD remains a candidate for NEP application if progress continues to be made in its development.

NEP System Design Approach

The objective of evolving an NEP system design is to direct technology development. As the technology evolves and the national requirements for NEP become better defined these can be iterated with the proposed system design. At present the primary application (outer planet intensive study) and the earliest use date (1990 and beyond) can be estimted from Fig. 1. The thrusters can be assumed to be 30-cm-grid-diam ion thrusters, each operating at 10 kWe and requiring a 50-cm-diam mounting area. Mounting of these thrusters in an array and protecting the balance of the spacecraft from the sputtering erosion effects of the exhaust plume is one of the major constraints on the system design and determines overall spacecraft diameter. Physical size limitations are imposed by the shuttle bay. Power conversion efficiency has been assumed to be either 10% (near term technology) or 15% (extrapolation of thermionic technology). This defines the thermal power needed from the nuclear reactor heat source and the amount of waste heat to be rejected from the primary radiator.

The design concept for the NEP spacecraft has been based on previous efforts.[5-7] The heat source for the NEP system is a fast spectrum nuclear reactor.[8] The concept allows the number of heat pipes used to cool the reactor to be varied within limitations imposed by maximum fuel temperature. The NEP systems will use molybdenum/lithium heat pipes approximately 15-20 mm in outside diameter.

The reactor design power flattens the core by appropriate heat pipe spacing within the core. The fuel is layers of UO_2 which is alternated with layers of molybdenum to conduct heat to the heat pipes and minimize peak fuel temperatures. Beryllium BeO reflectors and segmented BeO/B_4C control drums within the reflector are used for reactor control. A sectional view of a 3 MW thermal reactor is shown in Fig. 6.

The electronics and power conditioning for the NEP system are shielded from reactor neutrons by 0.85 m of LiH while

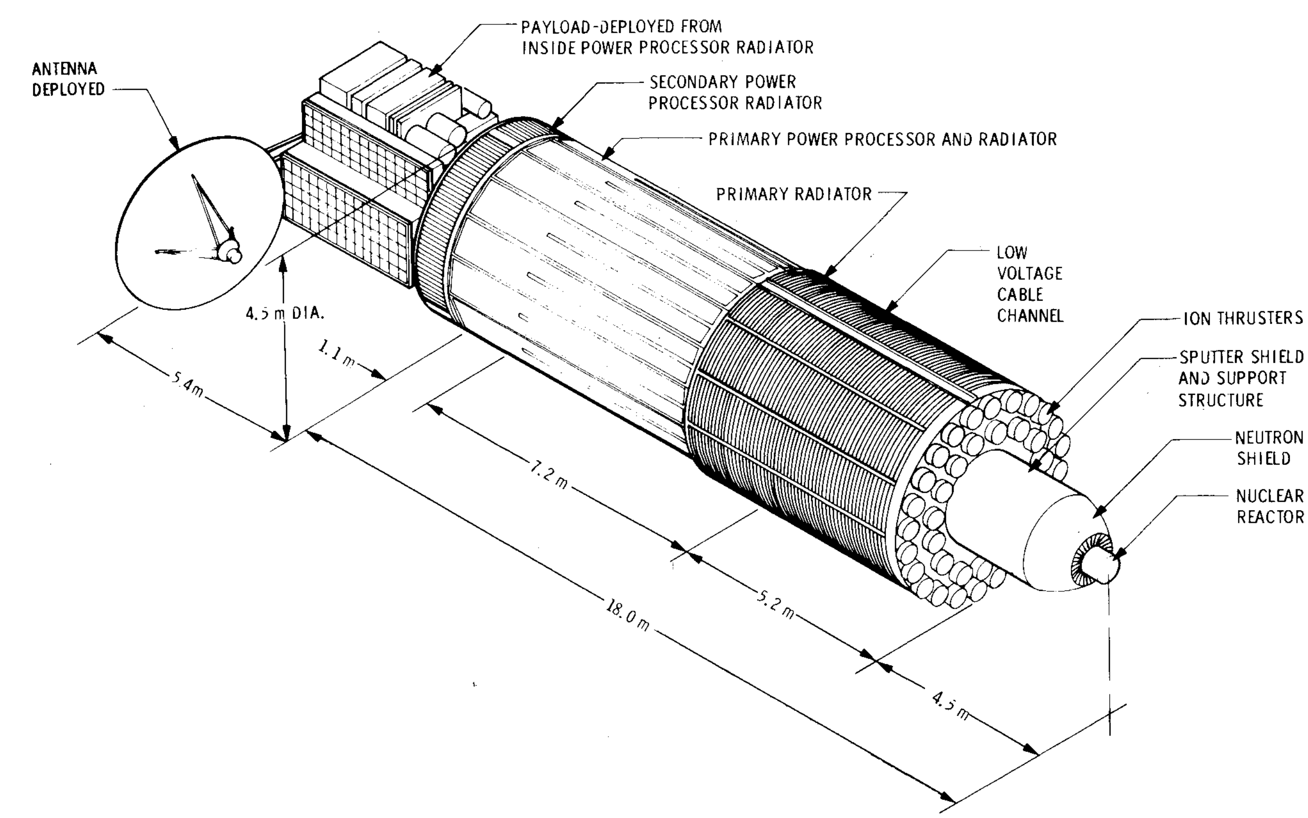

Fig. 7 Integration of power system into an NEP spacecraft.

Table 1 Effect of electrical power level and thermal-to-electric conversion efficiency η on system characteristics

System power level			No. operating thrusters	System dimensions, m			System mass, kg					System specific mass, kg/kWe	
kWe	kWth $\eta = 15\%$	kWth $\eta = 10\%$		Length $\eta = 15\%$	Length $\eta = 10\%$	Diameter	Thrust subsystem	Power subsystem $\eta = 10\%$	Power subsystem $\eta = 15\%$	Total mass $\eta = 10\%$	Total mass $\eta = 15\%$	$\eta = 10\%$	$\eta = 15\%$
400	3040	4560	36	11.4	15.3	4.20	2000	12280	8885	14280	10885	34.7	27.2
378	2875	4310	34	11.4	15.1	4.05	1890	11750	8520	13640	10410	36.1	27.5
355	2700	4050	32	11.2	14.9	3.90	1775	10875	7975	12650	9750	35.6	27.5
333	2540	3810	30	11.0	14.7	3.75	1665	10260	7615	11925	9280	35.8	27.9
310	2375	3560	28	11.0	14.0	3.75	1550	9515	7195	11065	8745	35.7	28.2
290	2230	3340	26	10.8	13.4	3.75	1450	9000	6875	10450	8325	36.0	28.7
267	2055	3080	24	10.5	12.8	3.75	1335	8465	6345	9800	7680	36.7	28.8
245	1895	2840	22	10.0	12.1	3.75	1225	7810	6075	9035	7300	36.9	29.8
234	1815	2720	21	8.6	10.7	4.36	1170	7525	5940	8695	7110	37.1	30.4
222	1720	2580	20	8.6	10.8	4.20	1110	7330	5800	8440	6910	38.0	31.1
211	1640	2460	19	8.6	10.7	4.05	1055	6965	5530	8020	6585	38.0	31.2
200	1560	2340	18	8.5	10.5	3.90	1000	6645	5270	7645	6270	38.2	31.4
189	1475	2210	17	8.5	10.6	3.75	945	6325	5075	7270	6020	38.5	31.9
178	1395	2090	16	8.5	10.6	3.55	890	6100	4825	6990	5715	39.3	32.1
167	1315	1970	15	8.5	10.6	3.40	835	5890	4670	6725	5505	40.3	33.0
156	1235	1850	14	8.4	10.4	3.25	790	5470	4415	6250	5195	40.1	33.3
145	1150	1720	13	8.3	10.3	3.10	725	5285	4275	6010	5000	41.4	34.5
134	1070	1600	12	8.3	10.1	2.95	670	4955	4045	5625	4715	42.0	35.2
122	980	1470	11	8.1	10.1	2.80	610	4680	3900	5290	4510	43.4	37.0
111	895	1340	10	8.1	9.9	2.65	555	4500	3715	5055	4270	45.5	38.5

gamma ray shielding is provided by a combination of power conversion system mass, propellant and tankage, and an estimated 200 kg of additional shielding. The calculated neutron dose at the power conditioning electronics over a 10 yr mission is 10^{12} nvt accompanied by 10^{6} rad of gamma radiation. These levels are within the capabilities of selected solid state electronics presently available.[9] A very limited number of solid state devices will withstand 10^{7} rad and 10^{14} nvt. Improvements in electronics may allow relaxation of radiation fluxes to these higher levels.

The overall NEP system layout (Fig. 7) is essentially identical for either thermionic or thermoelectric power conversion methods. Essentially no changes in supporting technology is required to interchange the two power conversion methods.[10] Thermionic conversion of the reactor heat to electricity has been assumed to size the system and obtain system weights.

Either power conversion approach will use a high-temperature insulator between the heat pipe which transports heat from the nuclear reactor and the hot side of the converter. Sialon, sapphire, and beryllia are being evaluated for this application. Two thermionic converters per molybdenum/lithium heat pipe were used in the design since this allows maximum flexibility in the layout of the heat rejection system, easiest access to welds during system assembly, and therefore is expected to maximize system reliability. Each converter uses one columbium/sodium or potassium heat pipe for heat rejection. The conversion system uses four converters connected in parallel electrically for redundancy.

Each heat rejection heat pipe is a continuous member from the converter to the heat rejection radiator. The radiator armor is beryllium to produce a 90% survival of heat pipes after ten years using the interplanetary meteoroid flux. If a specific mission requires an extended period of operation around a planet where meteoroid focusing is occurring the armor thicknesses and masses must be modified accordingly.

The thermionic converters are designed to operate at an emitter temperature of 1650 K and a collector temperature of 950 K. Temperature drops through the heat rejection system result in a radiator temperature of approximately 920 K. The 1650 K hot-side temperature is limited by fuel swelling. The 950 K rejection temperature is nearly optimum for present thermionic converters. As additional data becomes available on advanced thermionic converters this temperature may be modified to minimize system weight.

This design approach was utilized to evaluate the effect of conversion efficiency on system mass as a function of power level. Results of this study are shown in Table 1.

The electrical power levels were determined by the number of 10 kWe thrusters, plus 10 kWe housekeeping power, heat losses, and power conditioner efficiency (90%). The layout of the operating thrusters plus 10% or more spare thrusters determined the diameter of the system. Radiator area requirements, propellant tankage, and conversion system space requirements determined system length. The primary conclusion to be drawn from Table 1 is that specific weight is virtually unaffected from 400 kWe down to nearly 200 kWe where specific weight starts to rise rapidly. Thus, selection of a power system power level in the 200-400 kWe range can be made on the basis of factors other than specific weight.

Baseline System Selection

Without an in-depth study of a specific mission it is not possible to define a science payload other than its probable mass (several metric tons) based on the trends in Fig. 1. The second constraint is mission duration which impacts component lifetime requirements and mission costs in terms of maintaining ground operations. The third constraint is cost of development of the NEP system, i.e., a more advanced or sophisticated NEP system will deliver a greater payload or the same payload with a shorter mission time but at greater development cost and time.

Thus, the selection of a baseline system at this time should be to direct technology development. Flexibility should be maintained since the system design can be expected to evolve as specific mission requirements become better defined and applicable technology is evolved.

The initial power conversion system evolved will be a lower efficiency system, i.e., the 10% efficient system (Table 1). Improving the conversion efficiency of the system to 15% will reduce system weight allowing increased payload or reduced mission time. The cost of achieving such an improvement can be traded off with the need for increased payload or the cost of increased mission time.

The minimum payload requirement for intensive study of the outer planets is estimated to be 3 metric tons. The payload delivery capability of various NEP systems is readily estimated from the mission study results (Figs. 2-4) combined with system mass (Table 1). This was done at three power levels—200, 300, and 400 kWe—and the results plotted in Figs. 8-10 for Saturn, Uranus, and Neptune, respectively.

The mission studies show that if a 400 kWe system were to be built, it would be operated at lower powers to maximize payload under some mission time and thruster specific impulse constraints. For example, 400 kWe is usable for a Saturn mission; Fig. 2, at a mission time of 3¾ yr with thrusters of 5600 s specific impulse to place a total mass of 14 metric tons in orbit including the NEP power system. If mission time were to be increased to 6 yr and thrusters limited

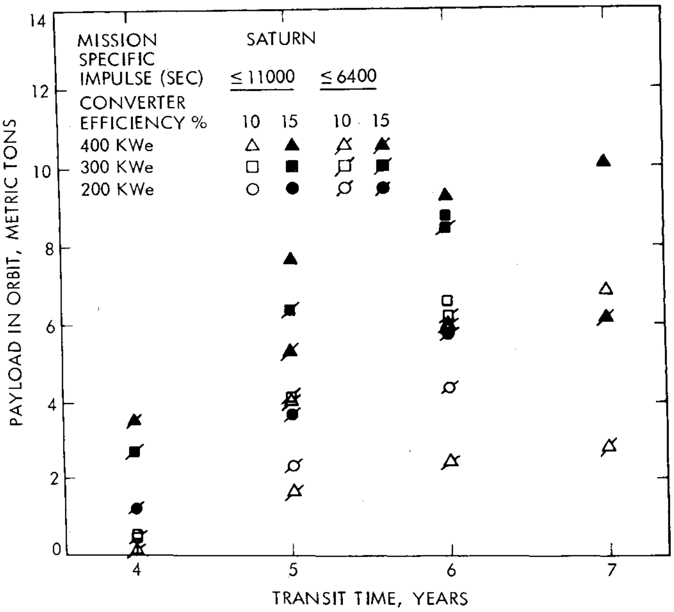

Fig. 8 Net science payload in orbit about Saturn as a function of mission time, system power level, thruster specific impulse, and conversion system efficiency.

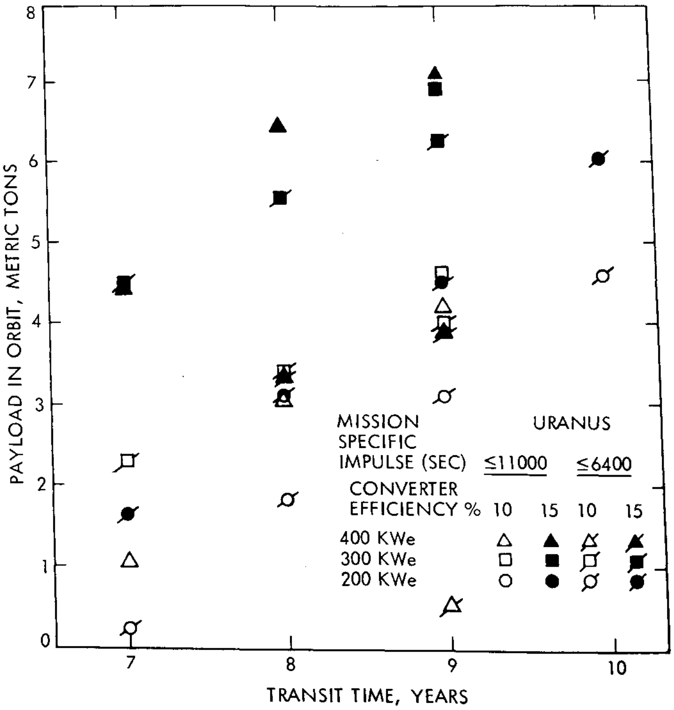

Fig. 9 Net science payload in orbit about Uranus as a function of mission time, system power level, thruster specific impulse, and conversion system efficiency.

to 6400 s specific impulse, 17 metric tons could be placed in orbit about Saturn, however, the system would be operated 292 kWe and not its 400 kWe capability.

For the next payload in orbit curves (Figs. 8-10) the masses of both the 300 or 400 kWe systems were retained as constant although the operating power could be far less than maximum capability. This leads to the result that a 400 kWe system under some constraints will deliver less payload than a 200 kWe system.

The following conclusions can be drawn based on the data of Figs. 8-10:

1) 400 kWe is optimum if the power conversion system is developed to a level of 15% conversion efficiency and ion thrusters with a specific impulse of 11,000 s are also developed.

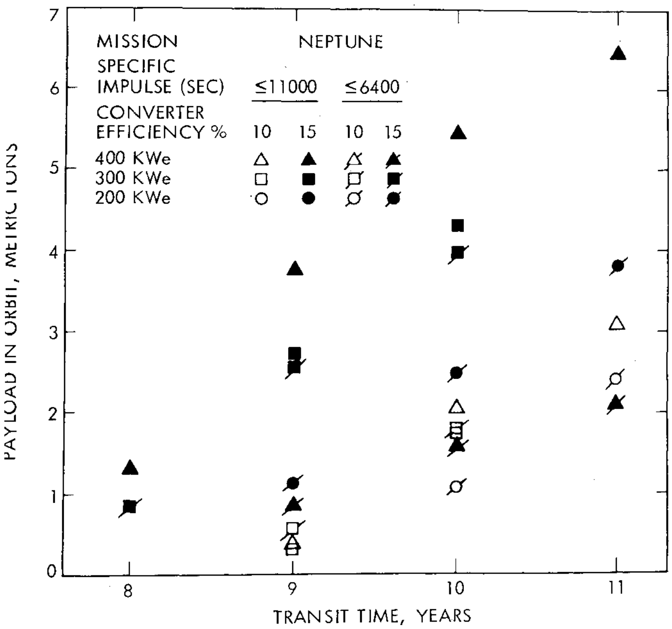

Fig. 10 Net science payload in orbit about Neptune as a function of mission time, system power level, thruster specific impulse, and conversion system efficiency.

2) If thruster development is limited to a maximum specific impulse of 6400 s, optimum power is approximately 300 kWe. The penalty for doing so is relatively small for a 15% efficient power conversion system.

3) If development of the power conversion system is limited to an efficiency of 10%, the payload reduction penalty is greater, however, the 3 metric ton minimum payload of Fig. 1 can still be met for Saturn and Uranus with a mission time of less than 10 yr using a 200-300 kWe system. A Neptune mission will require a mission time greater than 10 yr.

4) Imposing both the 10% efficiency limit and the 6400 s specific impulse limit has little effect compared to conclusion 3.

The design approach used for an NEP spacecraft in the 200-300 kWe range coupled with 10% efficient powr conversion and ion thrusters limited to a specific impulse of 6400 s or less will accomplish the projected requirements for exploration and intensive study of the outer planets. Selection of a power level which produces a system mass less than the full capability of the shuttle has advantages. Among these are payload capability to incorporate a chemical propulsion stage with the NEP system in a single shuttle launch. This will allow the NEP system to be started in a higher orbit with a longer decay time. Requirements for a minimum orbital altitude for startup of a nuclear reactor is expected to be resolved in the United Nations.[11]

Selection of an initial level near 200 kWe coupled with early power conversion techniques of lower efficiency (10%) allows growth potential to higher electrical power levels, if desired. If power conversion technology improves to higher efficiencies, the same thermal power nuclear reactor and same radiator heat rejection system can be retained for the more efficient system operating at a higher electrical power level. Initial mission applications of the NEP system are expected to be Saturn where a 10% efficient system operating at 200 kWe can deliver a payload in excess of 3 metric tons. A later mission would be Uranus where the 10 yr or shorter mission requirement for a payload of 3 metric ton could also be met by a 10% efficient system operating at 200 kWe. Neptune would be a yet later mission for NEP allowing time for power conversion system efficiency improvements, reduced payload requirements, improved system lifetime allowing a longer mission time, or a combination of these to be utilized.

Conclusion

Nuclear electric propulsion spacecraft studies, design and technology development are resulting in power and propulsion subsystem concepts which are scalable over a wide range of power levels. These systems, using near-term conversion technology, provide a capability to deliver payloads to the outer planets which is an order of magnitude greater than solar electric propulsion systems. Advances in conversion technology can increase the payload delivery capability from 3 metric tons to either Saturn or Uranus to more than 8 metric tons if needed. These advances allow utilization of the same nuclear reactor heat source and heat rejection system developed for early lower-efficiency systems to be applied to later higher-efficiency systems.

Electric thruster technology compatible with these systems is based on ion thrusters using argon and operating in the 5000-6400 s specific impulse range. Development of magnetoplasmadynamic thrusters offers the possibility of simpler thruster subsystem integration into the spacecraft coupled with reduced power conditioning system mass.

Acknowledgments

This paper presents the results of one phase of research carried out at the Jet Propulsion Laboratory, California Institute of Technology, under Contract NAS7-100, sponsored by NASA and work performed at the Los Alamos Scientific Laboratory under the auspices of the U.S. Energy Research and Development Administration.

References

[1] "Space Transportation Systems User Handbook," NASA TM-74831, July 1977.

[2] "Cryogenic Propellant Feed Systems for Electro Thermal Engines," Garrett Research Rept. AE1952R, Nov. 1961.

[3] Rudolph, L.K. and Pawlik, E.V., "MPD Thruster Development," AIAA Paper 79-2050, 14th Electric Propulsion Conference, Princeton, N.J., Oct. 1979.

[4] Clarke, K.E. and Pawlik, E.V., "Inductively Coupled TI-MPD Spacecraft Electric Propulsion," Paper 769254, 11th IECEC Conference, Stateline, Nev., Sept. 1976.

[5] Pawlik, E.V. and Phillips, W.M., "A Nuclear Electric Propulsion Vehicle for Planetary Exploration," AIAA Paper 76-1041, AIAA Electric Propulsion Conference, Key Biscayne, Fla., Nov. 1976.

[6] Phillips, W.M., Estabrook, W.C., and Hsieh, T.M., "Nuclear Thermionic Power System for Spacecraft," Paper 769256, 11th IECEC Conference, Stateline, Nev., Sept. 1976.

[7] Helms, I.L. (ed.), *Advanced Propulsion Comparisons Study,* Space Nuclear Systems, Division of USAEC and NASA, Washington, D.C., April 1973.

[8] Phillips, W.M. and Koenig, D.R., "Reactor Design and Integration into a Nuclear Electric Spacecraft," Paper 789564, 13th IECEC Conference, San Diego, Calif., Aug. 1978.

[9] Klass, P.J., "New Microcircuits Resist Radiation," *Aviation Week and Space Technology,* Oct. 10, 1977, pp. 58-61.

[10] Breslow, M. (ed.), *Reactor Technology,* Los Alamos Scientific Lab., LA-7878-PR, Jan.-Mar. 1979.

[11] "Report of the Working Group on the Use of Nuclear Power Sources in Outer Space," United Nations Doc. A/AC.105 /C.1/L.111, 16, Feb. 1979.