Design of a Nuclear Electric Propulsion Orbital Transfer Vehicle

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Payload increases of three to five times that of the Shuttle/Centaur can be achieved using nuclear electric propulsion. Various nuclear power plant options being pursued by the SP-100 Program are described. These concepts can grow from 100 kWe to 1 MWe output. Spacecraft design aspects are addressed, including thermal interactions, plume interactions, and radiation fluences. A baseline configuration is described accounting for these issues. Starting the orbital transfer vehicle from an altitude of 300 km indicates no significant additional risk to the biosphere than a reactor powerplant that has run for 7 years at a 300-year orbit.

Introduction

THE mission envisioned here for nuclear electric propulsion (NEP) is that of raising satellites from Shuttle orbit to geosynchronous Earth orbit (GEO). Orbit raising could be performed for one-way missions, where the power supply is also the electric source for the satellite, or as round-trip missions, where a delivery vehicle is reused in future missions.

Nuclear electric propulsion systems are optimized by constructing low-mass nuclear powerplants. Electric power levels from 100 kW to several megawatts are desirable. Desirable goals for the powerplant specific mass range from 20-30 kg/kW at the lower powers to 5-10 kg/kW at the higher powers.

Several different reactor-power-system concepts are candidates for space power systems. Under the Space Nuclear Reactor Systems Technology (SP-100) Program, the field has been narrowed to three concepts for 100 kWe. Growth versions to the megawatt range are possible. The selected candidates include a high-temperature (1500 K) fuel-pin reactor with thermoelectric power conversion, an in-core thermionic reactor powerplant, and a lower temperature (~900 K) fuel-pin reactor with Stirling power conversion.

The design of an NEP spacecraft requires consideration of a wide range of subsystem interactions. In addition to the usual thermal and plume interactions, NEP-OTV (orbital transfer vehicle) designs must address the reactor neutron and gamma fluxes, the electromagnetic fields produced by the electric propulsion subsystem, and the natural radiation associated with the low-thrust trajectory.

Electric thruster interactions with science and telecommunications can be a serious problem and are complicated by the long periods of thruster operation required. The NEP spacecraft thermal design is dominated by the large, high-temperature radiator that is required to reject the power system waste heat.

Studies conducted since the early 1960's have developed a variety of NEP stage configurations. These configurations and a configuration recently developed for the 100-kWe nuclear power system are discussed. The performance of a nuclear NEP system in maneuvering from low Earth orbit (LEO) to GEO as a function of flight times, payload, and reactor power levels has been calculated. Assuming a 120-day orbit transfer, 19,000 kg can be moved from Shuttle orbit to GEO in a single Shuttle trip with a 400-kWe power system. Significant payload gains are achieved over chemical stages, where the inertial upper stage can deliver 2270 kg and the Centaur up to 6360 kg.

Orbit Transfer Between Shuttle Orbit to GEO

The potential application of nuclear power to rocket propulsion was recognized early. The energy available from a unit mass of fissionable material is approximately 10^7 times larger than that available from the most energetic chemical reaction. Approximately $3 billion was invested in solid-core nuclear rocket development in the U.S. prior to 1973. This work was principally directed at the development of large, high-thrust engines based on hydrogen-cooled graphite reactor technology. The subject of this paper is the nuclear electric propulsion OTV (NEP-OTV). In this concept the nuclear reactor is a heat source for one of a variety of thermal-to-electric power conversion processes. The produced electrical power is supplied to one of several electric propulsion systems, examples of which include resistojets, arcjets, ion thrusters, and magnetoplasmadynamic (MPD) thrusters. Candidate propellants for these systems include liquid metals (e.g., Hg and Cs), inert gases (e.g., Ar and Xe), and hydrogen. Nuclear electric propulsion systems provide higher specific impulse and lower thrust than a chemical propulsion system of the same mass. Jet power P (watts) can be expressed as the product of specific impulse, I_{sp} (seconds), and thrust F (newtons),

$$P = gFI_{sp}/2\eta$$

In this equation, the gravitational constant g is 9.8 m/s^2. Chemical rockets produce prodigious quantities of power for short periods of time. For instance, the two RL-10 engines of the Centaur produce a total of 287 MW. The jet power delivered by the NEP system cannot exceed the electrical power output of the nuclear electric power supply (100 kWe to 1 MWe for the OTVs considered in this paper). Therefore, the maximum thrust that can be produced at a specific impulse of

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1000 s is 20-200 N, and at 5000 s is 4-40 N. Nuclear electric propulsion OTVs are, therefore, low acceleration vehicles that require trip times many times greater than those of chemical OTVs but deliver significantly more payload per unit mass of propellant expended. As in the case in terrestrial transportation systems, there will be a requirement for both the rapid delivery of time-schedule-critical materials and the economic delivery of heavy payloads, including bulk materials. It is the latter requirement as well as that of transporting acceleration-limited structures that the NEP-OTV is best suited to serve.

Many previous studies of NEP for orbit-raising applications have been conducted. The objective of this paper is neither to provide an overview of nuclear electric power nor to develop a new approach to the design of NEP-OTV. Instead, the objective is to present an overview of nuclear electric power and propulsion system options for the OTV application. We have examined some of the special problems associated with the integration of NEP into a practical OTV design and aim to illustrate to first order the performance of NEP-OTVs relative to that of chemical OTVs.

Data for selected electric propulsion systems are given in Table 1.[1] The performance of the NEP-OTV presented in Fig. 1 is based on the assumption that the nuclear power supply is part of the payload. This is an appropriate assumption for missions that require the large amount of electrical power provided by the reactor for on-orbit operations. Figure 1 indicates that a transit time of 120 days can be achieved by a 400 kWe OTV with a payload of 19,000 kg.

Table 2 compares the payloads and transit time of the Centaur, including a solar power supply, with that of the 400 kWe NEP-OTV. For Shuttle/Centaur the maximum payload mass is about 6000 kg to GEO. Assuming that the payload mass is assigned to the power system and that advanced solar technology is used to represent future solar power technology, one could deploy a 40 kWe power system. This leaves a balance-of-payload of 3000 kg.

In Table 2, the projected performance of nuclear powerplant systems is taken from Fig. 2 (based on data from Refs. 3 and 4). Nuclear reactor power system mass and specific mass change nonlinearly as a function of power level because 1) reactors must be designed to have a critical fuel mass and small incremental fuel additions will lead to larger than proportional gains in power output (increasing the reactor mass 40% will double power output); 2) shielding is an exponential function of thickness (doubling reactor power leads to about a 33% increase in shield mass); and 3) the mass of thermoelectric converters tends to be linear with power output, but dynamic converters are not. The mass and specific mass curves include radiation attenuation shielding for unmanned payloads (i.e., electronics). Based on Fig. 2, the mass of the 400 kWe powerplant is 6000 kg. For missions not

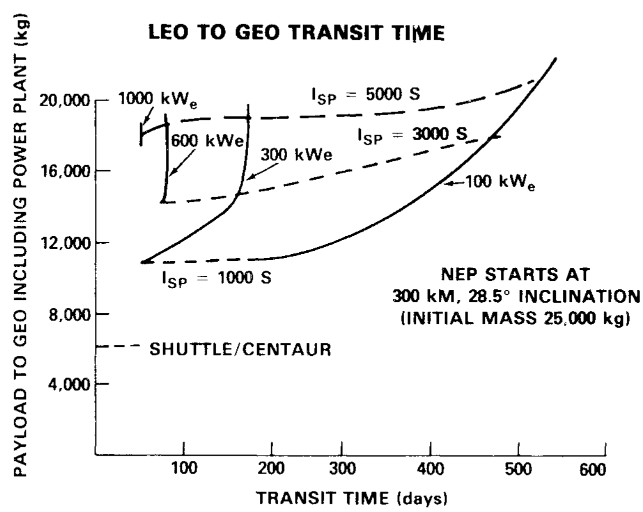

Fig. 1 Shuttle/NEP to GEO (from Ref. 1).

Table 1 Data for electric propulsion vehicles

Device	Specific impulse, s	Total system efficiency, %	Tankage fraction, %	Thruster size	System power, kWe	Propulsion system mass, kg
NH_3 arcjet	1000	40	15	30 kWe	100	500
				30 kWe	300	1500
				30 kWe	600	3000
				30 kWe	1000	5000
Hg ion thruster	3000	68	5	30 cm	100	1360
				30 cm	300	3960
				50 cm	600	3640
				50 cm	1000	5670
Hg ion thruster	5000	78	5	30 cm	100	940
				30 cm	300	2670
				50 cm	600	2860
				50 cm	1000	4500

Table 2 Comparison between Centaur and 400 kWe NEP-OTV performance

		Shuttle/NEP (400 kWe)[a]		
	Shuttle/Centaur	Powerplant part of payload	Powerplant part of propulsion	Powerplant part of propulsion and payload
Total power/payload power	40/40[b]	400/400	400/0	400/40
Balance-of-payload, kg	3,000	19,000	13,000	15,000[c]
Transit time, days	0.25	120	120	120

[a] 400 kWe powerplant mass = 6000 kg. [b] Advanced solar power systems. [c] Assumes spacecraft long-term power need is 40 kWe and nuclear powerplant mass is 2000 kg.

requiring this power, the mass of the powerplant must be subtracted from the payload shown in Fig. 1 to give a payload of 13,000 kg for these missions. However, if the spacecraft needs 40 kWe for the payload (the amount an advanced solar power system was computed to be able to deliver), we can charge the equivalent nuclear powerplant mass to the spacecraft and the balance to the propulsion system. The payload balance is 15,000 kg. The latter payload is five times the payload compared to spacecraft containing an advanced solar power system delivered by a Shuttle/Centaur transportation vehicle. Although 3-4 months are added to the transfer times from Shuttle orbit to GEO, the total mission schedule may not be impacted when one considers that several Shuttle launches and on-orbit assembly are eliminated to transfer such large payloads using the Centaur transfer stage.

In the above, we have limited our discussion to the case of an unmanned NEP-OTV. A manned NEP-OTV does not appear reasonable at this time because of the added radiation shielding required (10,000-15,000 kg) and the relatively long transit times. A unmanned tug would require resolution of such issues as: 1) a possible shielded docking port for manned operations in Shuttle orbit and perhaps GEO; 2) safety related to large reactor fission product inventories on the return to Shuttle orbit; 3) possible separation of the nuclear reactor or power systems during maintenance of the remainder of the vehicle and refueling of the electric propulsion tanks; and 4) long-term scheduling, with perhaps two round trips per year.

Nuclear Power Systems

Following the screening of over a hundred potential space nuclear power system concepts by the SP-100 Program, the field has now been narrowed to three candidate systems that appear to meet the requirements in Table 3 with a reasonable balance of technical risks and development time.

One concept uses a fast spectrum, lithium-cooled, pin-type fuel element reactor coupled to thermoelectrics for power conversion (Fig. 3).[2] The system is made up of a 12-sided cone structure with a 17-deg cone half angle. The reactor, which is a right circular cylinder approximately 1 m in diameter and 1 m high, is at the apex of the conical structure. It is controlled by 12 rotatable drums, each with a section of absorbing material and a section of reflective material to control the criticality level. Control of the reactor is maintained by properly positioning the drums.

Table 3 SP-100 reference goals

Performance	
Power output, net to user,	100 kW_e
Output variable up to 100 kWe	
Full power operation, years	7 years
System life, years	10 years
Reliability	
First system, 2 years	0.95%
Growth system, 7 years	0.95%
Multiple restarts	
Physical constraints	
Mass	3000 kg
Size, length within STS envelope	6.1 m
Interfaces	
Reactor-induced radiation after 7 years of operation, 25 m from forward end of reactor	
Neutron fluence	10^{13} N/cm^2
Gamma dose	5×10^5 rad
Mechanical	STS launch conditions
Safety	Nuclear Safety Criteria and Specifications for Space Nuclear Reactors

A second approach to SP-100 design is an in-core Thermionic system with a pumped sodium-potassium eutectic coolant. The general arrangement of the system design is shown in Fig. 4.[3] The design forms a 5.8-m-long conical frustum with minor and major diameters of 0.7 and 3.6 m, respectively. The reactor-converter subsystem includes the reactor, the reflector/control drums, and the neutron shield. The reactor contains the thermionic fuel element (TFE) converters within a cylindrical vessel that is completely surrounded by control drums. The NaK primary coolant routing to and from the reactor vessel is arranged so that NaK leaves the reactor at the aft end and cold NaK is returned to the forward end, thus helping to minimize differential thermal expansion in the piping. The reactor is also surrounded by an array of long, thin cylindrical reservoirs that collect and retain the fission gases in the reactor core during the operating life of the system. Waste heat is removed

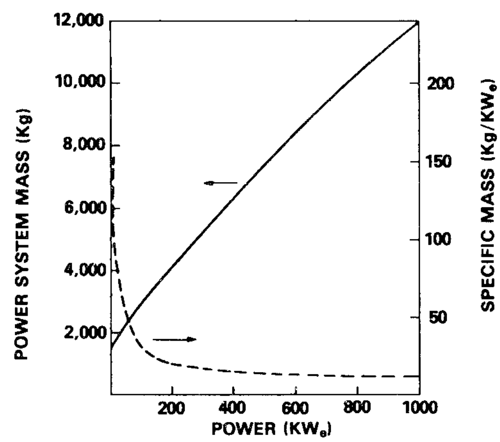

Fig. 2 Space nuclear reactor power system performance projections.

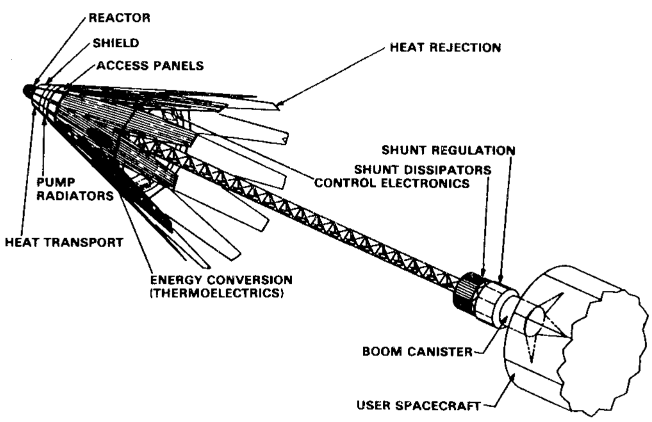

Fig. 3 High-temperature reactor with thermoelectric power conversion concept.

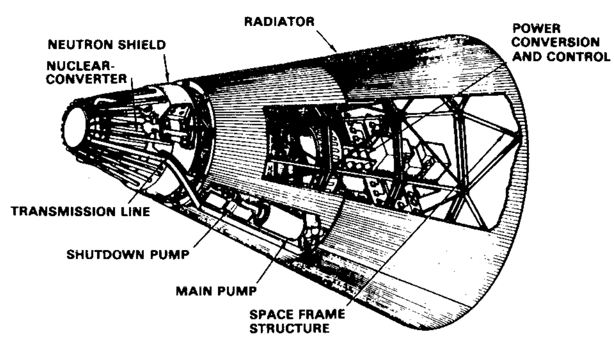

Fig. 4 In-core thermionic powerplant concept.

from the primary reactor loop through the heat exchanger. The energy is transferred through the heat sink exchanger to heat pipes that form the radiating surfaces for rejection of heat to space.

The third approach considered under the SP-100 Program uses a Stirling engine to convert heat from the lower temperature (~ 900 K) fuel-pin reactor design to electricity. This design emphasizes the use of state-of-the-art fuel pins of stainless steel and UO_2 with sodium as the working fluid. Such fuel pins have been developed for the Breeder Reactor Program with 1,059 days of operation and 8.5% burn-up demonstrated. The reactor can be similar in design to the high-temperature reactor, but utilizes lower temperature materials. In Fig. 5,[4] the reactor is constructed as a separate module from the conversion subsystems. Four Stirling engines, each rated to deliver 33 kWe, are included in the design concept, providing some redundance in case of a unit failure. Each engine contains a pair of opposed motion pistons operating 180 deg out of phase. This arrangement eliminates unbalanced linear momentum. Each engine receives heat from a pumped loop connected to the reactor vessel. An alternate arrangement would deliver the heat through an interface heat exchanger with heat pipes between the heat exchanger and engine. The heat is supplied to heater heads integral with the engine. Waste heat is removed from the cooler heads and delivered to a liquid-to-heat-pipe heat exchanger. The heat pipes, in turn, deliver the waste heat to the radiator, where it is rejected to space.

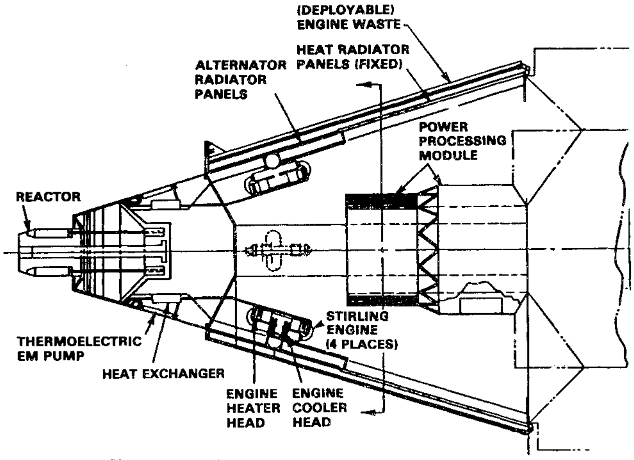

Fig. 5 Stirling engine conversion concept.

Nuclear Electric Propulsion Spacecraft Interactions

Figure 6 identifies the subsystem interactions that distinguish the design of an NEP-OTV from that of a chemical OTV. Special attention must be given to the radiation and thermal environments produced by the power system, and to the electromagnetic fields and propellant fluxes produced by the propulsion system. These effects can be estimated by comparisons with exposures projected for the Galileo spacecraft.

Nuclear electric propulsion OTV designs must address the reactor neutron and gamma fluxes as well as significantly larger natural radiation exposure that results from the increased time spent by the low-thrust NEP-OTV in transition through the Earth's radiation belts. Figure 7 shows the cumulative radiation exposure to the mission during this phase, and compares this exposure to that expected for the Galileo orbiter spacecraft (which will orbit the planet Jupiter). While the Jovian radiation environment is more intense than

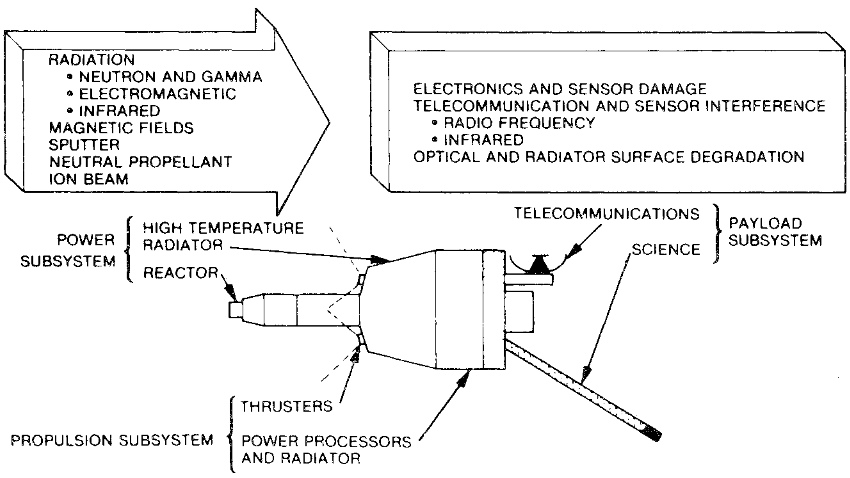

Fig. 6 NEP spacecraft subsystem interactions.

Table 4 Re-entry radiation levels for aborted NEP missions starting from 300 km orbit ($2-MW_t$ reactor)

Boost time, days	Orbit decay time, years			Radiation levels at re-entry (Ci)		
	I_{sp} = 1000 s	3000 s	5000 s	I_{sp} = 1000 s	3000 s	5000 s
			100 k We NEP			
1.1	0.7	0.7	0.6	800	1.0×10^3	1.0×10^3
3.5	3	1	0.6	255	1207	3.0×10^3
11.5	3000	40	7	0.06[a]	115[a]	326
			400 kWe NEP			
1.1	3	0.8	0.7	86[a]	580	800
3.5	700	40	6	0.5[a]	35[a]	110[a]
11.5			3000	[a]	[a]	0.06[a]

[a] Radioactivity dose levels below those after 300 years following 7 years of operation.

the Earth's, the Galileo orbiter spends most of the time outside the radiation belts, and thus the radiation exposures are comparable to those seen by the NEP-OTV in transferring between LEO and GEO. Radiation shielding, roughly equivalent to that required for the Galileo spacecraft (0.5 cm aluminum), must be provided to protect the NEP-OTV avionics and payload from the natural radiation environment. These shielding requirements should be considered when comparing the performance of chemical and nuclear propulsion vehicles.

Reactor-produced neutron and gamma radiation effects on OTV and payload systems are controlled by a combination of shielding and spatial separation between the reactor and critical components. The radiation limits for the Galileo spacecraft are 7.5×10^4 rad and 2.5×10^{10} N/cm^2 (1 MeV). Shield design criteria currently being used in the SP-100 Program (5×10^5 rad and 10^{13} N/cm^2) are significantly higher than these values based on projected advances in radiation-hard electronics technology. If the imposition of such requirements on general-purpose OTV payloads is not practical, additional shielding will be required. The radiation shield is a significant fraction of the power-subsystem mass, approximately 30% for a 100 kWe system and 10% for a 10 MWe system. These estimates are representative of shadow shields, which intercept a small fraction of the reactor fluence. Orbital transfer vehicle systems such as thrusters, booms, and radiators located outside the protected zone are not only themselves subject to radiation damage, but serve to increase the fluence to systems located behind the shield by scattering radiation into these regions. Such configurations should be avoided. A space-based NEP-OTV will require additional shielding on the vehicle or at the depot to permit accesss for maintenance and servicing.

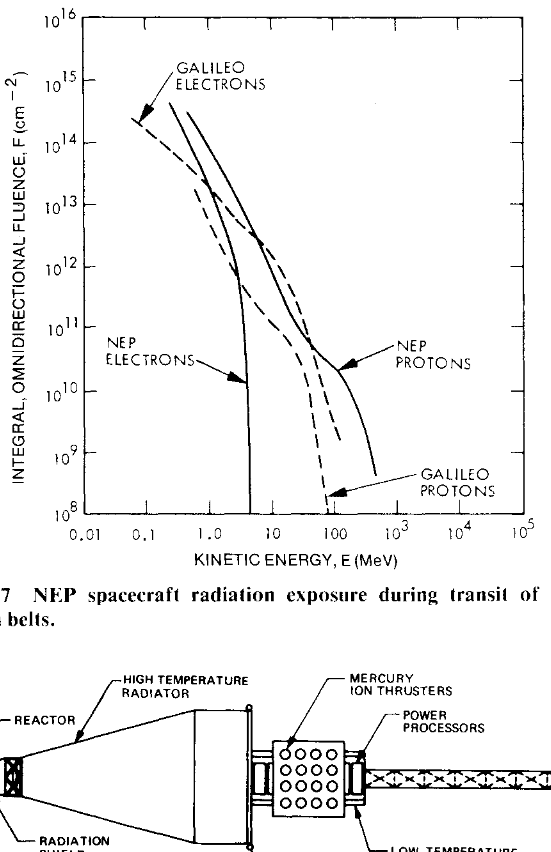

Fig. 7 NEP spacecraft radiation exposure during transit of Van Allen belts.

The thermal design of the NEP-OTV is complicated by the large, high-temperature radiator required for heat rejection by the power and propulsion power processor systems. The thermal power rejected is typically 4-15 times the electrical power supplied to the thrusters. Power system radiators must operate at high temperatures (700-1000 K) to keep the surface area and mass of the radiator within acceptable bounds. Propulsion system power processor heat rejection requirements vary depending on the electric propulsion system used, but are much lower that those of the power system and can be met with radiators operating at lower temperatures (300-400 K). The integration of these large, high-temperature surfaces into the spacecraft configuration requires considerable ingenuity to avoid excessive thermal loads to the OTV propellants and payloads electronics. Advanced lightweight radiator concepts such as the droplet radiator may permit operation at lower temperatures, but these radiators introduce additional design and operation constraints. The use of cryogenic propellants, such as argon or hydrogen, will likely require sophisticated thermal control systems to prevent excessive boil off.

Various interactions can occur between the electric propulsion system and spacecraft surfaces and systems. These interactions vary depending on the type of thruster used (e.g., ion, MPD, arcjet, or resistojet) and the propellant (e.g., mercury, argon, xenon, or hydrogen). The following discussion focuses on the electron bombardment ion thruster, since the characteristics of this thruster are better defined than those of other electric thrusters. Four basic mechanisms characterize

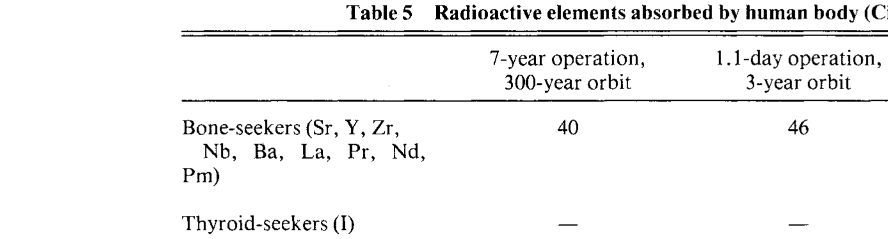

Fig. 8 SP-100 NEP-OTV flight configuration.

Table 5 Radioactive elements absorbed by human body (Ci), 2MWt

	7-year operation, 300-year orbit	1.1-day operation, 3-year orbit	3.5-day operation, 3-year orbit
Bone-seekers (Sr, Y, Zr, Nb, Ba, La, Pr, Nd, Pm)	40	46	138
Thyroid-seekers (I)	—	—	—
Kidney-seekers (Ru)	—	3	8
Muscle tissues (Cs, Ba)	30	14	41
Total all radioactivity	118	86	255

electric propulsion/spacecraft interactions: 1) surface erosion, 2) film deposition, 3) plasma interactions, and 4) electromagnetic interference.

Spacecraft surfaces exposed to the ion thruster beam plasma can be degraded by sputter erosion, which can result in optical degradation of thermal control surfaces and spacecraft contamination by the sputtered material. Thruster location should be selected to avoid impingement of the beam plasma on spacecraft surfaces. The half angle of the energetic beam can be as large as 40 deg for some thrusters, but is typically 15 deg.[5]

The deposition of propellant and sputtered thruster materials on critical spacecraft surfaces can be a serious problem. Such materials can be transported upstream of the thruster by diffusion and electromagnetic field effects. The deposition of these materials on surfaces depends on the arrival rate, the vapor pressure of the materials, and the temperature of the surfaces. High vapor pressure propellants such as argon, xenon, and hydrogen should not present a serious problem; however, low vapor propellants such as mercury and thruster materials such as molybdenum will accumulate on all but the highest spacecraft surface temperature. Thin films of such materials can alter surface electrical conductivity and thermal radiation characteristics. Changes in surface electrical conductivity can impact the performance of photovoltaic arrays, antennas, and electrical insulators; changes in surface thermal properties can impact radiator performance.

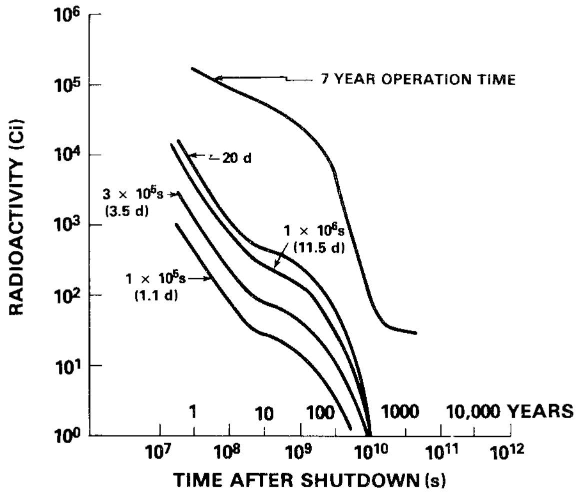

Fig. 9 2-MWt radioactivity decay.

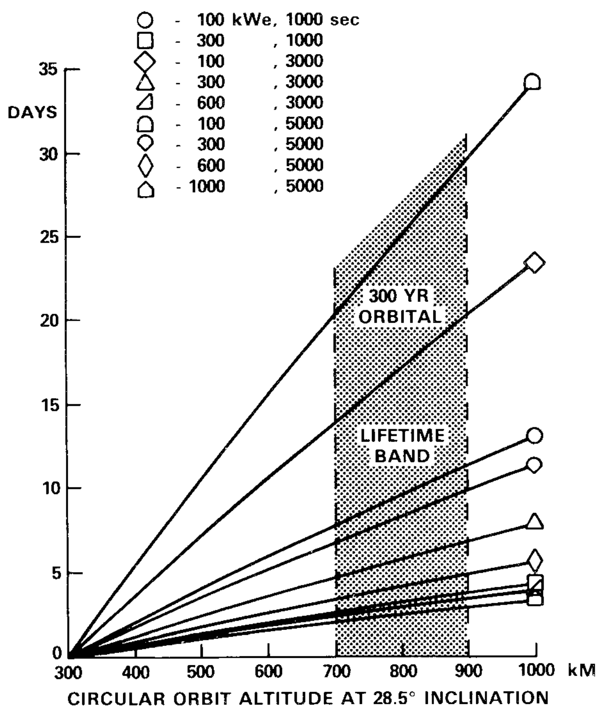

Fig. 10 Transit time from 300 to 1000 km on trips to GEO (from Ref. 1).

Interactions between the NEP-OTV, the ambient space plasma, and the electric propulsion-generated plasma can result in spacecraft charging and arcing that can produce upsets in logic circuits, breakdown of electrical insulation, and enhancement of surface contamination. Such effects are of principal concern with power system thermoelectric elements and interconnects.[5] Spacecraft electrostatic potential will actually be stabilized during thrusting by beam neutralization, as has been demonstrated in the AT 5-6 and SCATHA.[6-8]

Electromagnetic interference is produced by the ion thruster discharge chamber permanent magnets and dynamic electromagnetic fields. The transmission of radio signals through the ion beam has been investigated by Jet Propulsion Laboratories (JPL).[9] S-band transmission through the beam of a 2-A 30-cm mercury ion thruster showed only a small amount of signal attenuation and negligible reflection loss. These experiments were not extended to X-band or inert gas propellants, but the effects at that frequency should be significantly smaller, and the inert gas plasma should (to first order) behave the same way as the mercury plasma in terms of absorption and reflection of RF signals.

NEP-OTV Flight Configuration

Numerous NEP spacecraft configurations have been proposed.[10,11] Figure 8 presents a configuration developed for an early SP-100 power system concept.[12] This configuration serves to illustrate the impact of previously discussed NEP interactions on vehicle design. In this configuration, the thrust vector is orthogonal to the vehicle longitudinal axis and the reactor and payload are located at the opposite ends of the vehicle. The side thrust/end reactor configuration was selected because this design avoids many of the less well defined subsystem interactions (e.g., sputter erosion of hardware downstream of the thrusters, and scattering of nuclear radiation by hardware not protected by the shadow shield). Clear fields of view are provided for the high-temperature power system and thruster system power processor radiators. Thermal control problems are minimized by integrating the spacecraft subsystems along the thermal gradients, i.e., high- and low-temperature systems are located on opposite ends of the spacecraft and intermediate temperature systems are located in between. The design also provides a clear field of view for payload systems.

Ion thruster technology was used for propulsion. The thruster subsystem power processor packages are structurally integrated with two flat plate radiators, which reject processor waste heat at 320 K. The 16 individual gimbaled 30-cm thrusters shown in this configuration are attached to a support structure that is integrated into the power processor radiator strucure. A propellant tank is located at the spacecraft center of gravity (cg) behind the thrusters and between the two banks of power processors. Mercury, xenon, and argon are potential propellants. The inert gases may be preferable for the OTV mission due to potential mercury contamination of the Earth's atmosphere; however, mercury is preferred for the planetary missions due to ease of storage and better thruster performance. Mercury is also effective in reducing the radiation fluence to the payload.

The maximum length of the launch configuration of the spacecraft is limited by shuttle cargo bay dimensions. After leaving the cargo bay, the payload section of the spacecraft is extended 11 m from the end of the thrust module by a lightweight collapsible mast. This is necessary to place the spacecraft center of mass at the center of the main propellant tank (no cg change during burn). The extension also serves to place payload electronics at a distance (25 m) from the SP-100 shield consistent with the shield design requirement.

Relative Biosphere Risk

To protect the Earth's population against undue risk, radiation levels at the time of a nuclear reactor re-entering the Earth's atmosphere should be low. Most fission products decay away, if the orbital lifetime of a satellite in orbit is sufficiently long. A long-lived, high orbit is defined in the reactor safety specification[13] as an orbit with a lifetime of 300 or more years. This corresponds to an initial altitude of between 700 and 900 km; the orbital lifetime is dependent on the re-entry body initial altitude, mass, and shape. Figure 9 plots the radioactivity for a 2MWt reactor as a function of operating times (calculations were performed on the Origen code) and indicates that if the reactor re-enters the biosphere after 300 years in orbit, the fission product activity will have been reduced from approximately 10^7 Ci to about 100 Ci.

Actinides are another source of radiation. Their quantity is proportional to the operating time, fuel enrichment, and reactor spectrum. The dominant actinide is ^{239}Pu, which has a half-life of 24,390 years. At low thermal power and operating times, the actinide levels are very small; but at 2 MWt power operating for 7 years, they represent a 4 Ci radiation source.

Certain designs may use materials that are activated while in the reactor, e.g., Nb – 1% Zr – 0.1% C fuel cladding. Their presence can result in the generation of additional long-lived radioactive isotopes. For the reactor in Ref. 4, activation of the fuel cladding results in an increase of 22 Ci at the end of 300 years because ^{94}Nb is generated (half-life of 2×10^4 years).

The total dose level after 300 years is 118 Ci. It is derived mainly from long-lived isotopes. If the orbit time is increased to 600 years, the dose level decreases to 34 Ci, and in 2000 years to 28 Ci.

To avoid payload penalties with the Shuttle, an initial operational orbit at about 300 km is preferred. A 750-km orbit, corresponding to about a 300-year orbital lifetime, can be reached with the Shuttle by adding two Orbiting Maneuvering Systems (OMS KITS); however, this results in a 38% payload reduction. Biological risk questions associated with starting at 300 km altitude relate to: 1) the quantity of additional fission products present at re-entry if an abort occurs prior to reaching a 300-year orbit, 2) the biological hazards of those fission products, and 3) whether the spacecraft can be powered into the atmosphere. The last condition can be avoided by independent and redundant control of the propulsion and power supply to insure NEP operation termination if the spacecraft trajectory is unacceptable. The first two questions will be addressed.

Nuclear electric propulsion OTV trajectory data are presented in Fig. 10 for a range of powers and specific impulses.[1] Aborts were assumed at various times during orbit transfer and the radiation levels compared with a 300-year orbit (Table 4). It was concluded that for a short time duration the fission products could be greater than those produced by long-term operation followed by a 300-year radioactive decay period. For 100 kWe NEP sytems, the radiation level above those for our reference case is several weeks for a 5000 s specific impulse and about 1 day for 1000 s. The peak level for 1000 s is about 800 Ci.

If a more efficient electrical conversion subsystem is used with the 2 MWt heat source. 400-kWe output power can be achieved. Higher power reduces the time during which radiation levels at re-entry are above the 300-year orbit levels following an abort. For 400 kWe this is less than 1 day for an $I_{sp} = 1000$ s and 3.5 days for $I_{sp} = 5000$ s. The radiation levels are such that we can conclude that reactors designed to disperse on re-entry could be started on a NEP transfer from below the 300-year orbit with little additional risk or damage to the biosphere.

The distribution of radioactive elements at several points in Table 4 were reviewed. The results, shown in Table 5, indicate some buildup in bone-seeker above the 7-year reference, but do not change our conclusions.

Summary

It appears that about 400 kWe would be a desirable choice for NEP-OTV applications. Lower power significantly increases the transit times and higher power decreases the payload. Current 100-kWe designs being pursued in the growth versions of SP-100 Program can deliver powerplants using either in-core thermionics or a high-temperature pin-fuel reactor with Stirling power conversions at 400 kWe.

An NEP spacecraft concept has been developed in which the thrust vector is orthogonal to the vehicle longitudinal axis and the reactor and payload are located at opposite ends of the vehicle. This configuration avoids such issues as sputter erosion and radiation scattering, and takes into account thermal interactions. Also, it allows for a clear view for payload systems.

Biological risk associated with starting NEP operations from 300 km are not significantly different than starting from a 300-year (750-km) orbit.

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