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RADIOISOTOPE ELECTRIC PROPULSION OF SCIENCECRAFT TO THE OUTER SOLAR SYSTEM AND NEAR-INTERSTELLAR SPACE

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Abstract

Recent results are presented in the study of radioisotope electric propulsion as a near-term technology for sending small robotic sciencecraft to the outer Solar System and near-interstellar space. Radioisotope electric propulsion (REP) systems are low-thrust, ion propulsion units based on radioisotope electric generators and ion thrusters. Powerplant specific masses are expected to be in the range of 100 to 200 kg/kW of thrust power. Planetary rendezvous missions to Pluto, fast missions to the heliopause (100 AU) with the capability to decelerate an orbiter for an extended science program and prestellar missions to the first gravitational lens focus of the Sun (550 AU) are investigated.

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1. Introduction

Small sciencecraft which encourage more frequent deep-space missions and faster response to changing scientific needs are becoming the preferred platform for space science. Miniaturization is reducing the size and mass of spacecraft instruments so that a payload of tens of kilograms can accomodate several experiments and the associated support systems. It has long been appreciated that high specific impulse, electric propulsion would allow large payload fractions to be delivered to deep-space at high velocities because of the reduced amount of propellant needed. Once thrusting is complete, an abundance of electrical power also enables high data transmission rates to Earth enhancing the mission's scientific return. But electric propulsion and deep-space science are viable only if an adequate space-power source exists.

Radioisotopes have been used successfully for over twenty-five years to supply the heat for thermoelectric generators on various deep-space probes. Extending the use of radioisotopes to primary electric propulsion of small probes has recently been investigated by the author [1][2][3]. Radioisotope electric propulsion (REP) systems are low-thrust, ion propulsion units based on radioisotope electric generators and ion thrusters. Payload size will determine the necessary propulsive power which is selectable in principle by scaling the propulsion unit size or combining a number of smaller, modular units in parallel. Payloads of 50 to 100 kg and a total sciencecraft mass including the REP unit of 200 to 400 kg are envisioned.

The perceived liability of radioisotope electric generators for ion propulsion is their high specific mass (mass per unit power). As shown in ion propulsion texts, the velocity change Δv, which a propulsion unit can produce is proportional to the square root of the thrust time divided by the powerplant specific mass ($\sqrt{\tau/\alpha}$) [4]. A higher specific mass results in a lower velocity change since one is using thrust to accelerate the massive powerplant and not the payload. The only saving grace for high specific mass powerplants is a fact often overlooked in the derivation: to cover a given distance s in field-free space, the thrust time τ is proportional to $s^{2/3}\alpha^{1/3}$ (since $\tau \propto s/\Delta v \propto s/\sqrt{\tau/\alpha}$). This relative insensitivity of thrust time, and implicitly flight time, to specific mass suggests that high specific mass powerplants may be suitable for robotic science missions to the outer Solar System.

Conventional radioisotope thermoelectric generators (RTG) have a specific mass of about 200 kg/kW of electric power [5]. Many development efforts have been undertaken with the aim of reducing the specific mass of radioisotope electric systems [6] [7] [8] [9]. Recent performance estimates suggest that specific masses of 50 kg/kW may be achievable for thermophotovoltaic (TPV) and alkali metal thermal to electric (AMTEC) generators [10] [11]. The need for low-mass, electric generators using less radioisotope fuel on proposed missions like the Pluto Express and Europa Orbiter continues to drive this effort, and candidate generators are being explored for a next-generation Advanced Radioisotope Power System (ARPS).

As we review in the next section, powerplants constructed from these near-term radioisotope electric generators and long-life ion thrusters will likely have specific masses in the range of 100 to 200 kg/kW of thrust power if development continues over the next decade. In earlier studies [1][2][3], it was concluded that heliocentric flight times are indeed a weak function of powerplant specific mass, and that a timely science program of planetary rendezvous and prestellar robotic missions is enabled by primary electric propulsion once the powerplant specific mass is in the range of 100 to 200 kg/kW. Flight times can be substantially reduced by using hybrid propulsion schemes which combine chemical propulsion, gravity assist and electric propulsion [2]. Hybrid schemes are futher explored in this paper to illustrate how the performance of of REP is enhanced for Pluto rendezvous, heliopause orbiter and gravitational lens missions.

2. REP Technology

Various schemes for radioisotope electric generators have been under development for several years, and some may lend themselves to an evolutionary REP program over the next decade. A reduction in the specific mass of radioisotope electric generators from $\alpha_e = 200$ kg/kW to 50 kg/kW of electric power now appears likely if development continues. Most of these devices are intended to produce total electric powers of tens to a few thousand watts, and their modular designs allow them to be easily scaled up or down. Table 1 summarizes the performance and reliability of some candidate technologies for radioisotope electric generators. The information in the first four columns is taken from the references cited below. The last column gives estimated specific masses for hypothetical powerplants using a 30 cm, derated xenon thruster discussed in this section as an example.

Electric generator	α_e (kg/kW)	Lifetime (years)	Limiting component	Powerplant α(kg/kW)
RTG	197	> 20 dem.	Dopant migration	224
MOD-RTG	127	> 8 est., 1.7 dem.	Dopant migration	154
RTPV	60	> 10 est.	Neutron damage to GaSb cells	87
FPSE-DIPS	118	> 10 est.	Two moving parts	145
AMTEC	60	> 10 est., 1.6 dem.	Porous electrode grain growth	87

Table 1: Comparison of radioisotope electric generators and estimated powerplant specific masses using a 0.5 kW, 30 cm xenon thruster.

The first three devices in Table 1 involve direct conversion of heat to electricity: the conventional radioisotope thermoelectric generator (RTG) using SiGe unicouples [5], the modular RTG (MOD-RTG) which uses SiGe/GaP multicouples consisting of 20 thermocouples connected in series [6] and the radioisotope thermophotovoltaic (RTPV) power system which would use GaSb infrared (IR) photovoltaic cells to directly convert radiant heat from the radioisotope to electricity [7]. With the development of improved IR filters, the estimated efficiency of RTPV units has increased from 13% to 23% in recent years. This along with better radiator design has reduced the estimated generator specific mass from 118 to about 60 kg/kW of electric power [10].

The last two devices in Table 1 are thermal engines which use a working fluid to convert radioisotope decay heat to electrical work. The free piston Stirling engine (FPSE) has been proposed as a lightweight dynamic isotope power system (DIPS) for ten to thousand watt space-power applications [8]. The magnet mass in the alternator keeps the specific mass high in these devices. Alkali metal thermal to electric converters (AMTEC) electrochemically convert the isothermal expansion of sodium or potassium vapor to electrical work via a charge exchange of the liquid metal in a beta-alumina solid electrolyte and electron recombination at a porous metal electrode [9]. Recent design improvements have led to slight reductions in the estimated specific mass of hypothetical AMTEC generators [11].

Because of their low specific mass, both the RTPV and AMTEC generators are potential candidates for the ARPS on Pluto Express and the Europa Orbiter. If successfully developed, either of these generators could become the prototypic generator for future radioisotope electric propulsion systems. A reliable ion thruster makes up the other essential part of a propulsive unit. A variety of thrusters has been researched for different applications over the years [12] [13] [14]. Significant progress has been made in reducing the mass and extending the lifetime of low-power, inert-gas (xenon, krypton, argon) ion thrusters in recent years. There are now definite plans to use such thrusters for primary solar electric propulsion, as for example on the Japanese MUSES-C mission for asteroid sample return [15].

The specific mass of a complete propulsive powerplant has contributions from both the electric generator and ion thruster. For example, the mass of a NASA developmental 30 cm thruster (Lewis Research Center) excluding the power processing unit (PPU) is estimated at 7 kg [12]. An added gimbal assembly for maintaining proper thrust vectoring has a mass of 34% of the thruster mass (2.4 kg in this case). The specific mass of present PPU's is $\alpha_{PPU} = 8$ kg/kW of input power, though a reduction by a factor of five to ten may occur in the next decade. The specific mass of the thruster unit will depend on the input power P_0, the total number of thrusters (including spares) N_t, the thruster mass m_t and the gimbal mass m_g. The complete powerplant specific mass is then

$$\alpha = \alpha_e + \alpha_{PPU} + N_t \cdot (m_t + m_g)/P_0. \tag{1}$$

In the last column of Table 1 are listed the estimated specific masses of hypothetical powerplants constructed from a NASA 30 cm thruster with 0.5 kW of input power supplied by the different radioisotope electric generators. These are probably conservative estimates since PPU and thruster specific masses will probably decrease with development. A 30 cm ion thruster operating at 0.5 kW input power has a projected lifetime of about 9 years. Assuming a total thruster efficiency η_t of 75%, all the technologies beyond the standard RTG can give effective powerplant specific masses α/η_t in the range 100 to 200 kg/kW suitable for propelling small sciencecraft.

3. Pluto Rendezvous Missions

Planetary rendezvous missions are very demanding due to the need to carry extra propellant for deceleration at the target planet. Aerobraking is useful for planets with atmospheres, but this option is not available for the Pluto-Charon system. Because of the high transit velocity, electric propulsion is perhaps the only viable near-term solution to achieve fast Pluto rendezvous. Electric propulsion also enables an extended science mission with transfers between Pluto and Charon low-orbits.

To maximize the payload delivered for a mission along a low-thrust trajectory, both the rocket configuration (relative masses of powerplant and propellant) and the thrust program (thrust magnitude and direction versus time) are typically optimized. For small sciencecraft, ion thrusters with constant propellant velocity and thrust magnitude are adopted here because of their simplicity and long-life. The parameters characterizing an ion rocket are the payload mass ratio M_L/M_0, the powerplant to propellant mass ratio $K = M_W/M_P$, the effective powerplant specific mass α/η_t and the total thrust time τ. Here $M_0 = M_L + M_W + M_P$ is the initial ion rocket (sciencecraft) mass after any chemical rockets are jettisoned, and η_t is the total ion thruster efficiency. The ratio M_W/M_P is related to the effective propellant velocity $v'_p = \eta_m v_p$ by $M_W/M_P = (v'_p/v'_c)^2$, where η_m is the mass utilization efficiency of the thruster and $v'_c = (2\tau\eta_t/\alpha)^{1/2}$ is the characteristic velocity of the propulsion system [4].

Previous work on planetary rendezvous missions to the outer Solar System [3] established a relatively simple rocket configuration and constant-thrust program that gives the same flight time for a given payload as programs quoted in the literature but uses a smaller powerplant. This may be advantageous for small sciencecraft where the cost of the radioisotope electric generator may be a large fraction of the total mission cost. The thrust program has a constant thrust directed at a constant angle relative to the velocity vector during each powered phase. A coast period τ_c, during which there is no thrust, is introduced to minimize the flight time for a given payload fraction. The optimal rocket configuration (yielding the maximum payload fraction) for this thrust program is given by the approximate field-free formula [1]

$$K_{opt} = 0.26(1 + 2\ln(1 + 3(M_L/M_0)_{max})). \tag{2}$$

The calculation of trajectories described in this paper was done with a FORTRAN program incorporating a standard, double-precision differential equation solver to evolve heliocentric trajectories in the ecliptic. The simple computer program for two-dimensional motion was not intended for detailed mission analysis but rather to allow a quick parameter study.

Table 2 contains results on flight time τ_F, total thrust time τ and optimal coast period τ_C for Pluto rendezvous starting from Earth orbit. To simplify the calculations, Pluto's orbit was taken as being in the same orbital plane as the Earth with an eccentricity of 0.25. Launch from Earth occurs in 2010 for all cases. Powerplant specific masses α/η_t of 100 and 200 kg/kW are used, characteristic of the range for near-term REP systems. A fixed payload fraction $M_L/M_0 = 0.25$ for the ion rocket is assumed since flight times quickly increase for larger values and do not decrease much for smaller values [3]. Gravitational effects of the Pluto-Charon system are not included in the trajectory analysis so the propellant mass of the ion rocket only refers to that needed on the low-thrust trajectory to achieve rendezvous. Any mission-specific propellant needed for orbital entry and maneuvers is assumed to be included in the payload mass allocation.

Escape from low-Earth orbit (LEO, 320 km) is achieved by chemical propulsion for small probes, and ion thrusting commences immediately after Earth escape. Minimal Earth escape (Δv = 3.2 km/s) results in no hyperbolic excess velocity relative to the Earth ($u_0 = 0$). Providing an excess velocity u_0 with the chemical rocket at Earth escape reduces the amount of time needed by the ion rocket to spiral out of the inner Solar System. Pluto rendezvous can be achieved by small REP sciencecraft in only 10 to 20 years when excess velocities of several km/s are supplied at Earth escape. Note that as the hyperbolic excess velocity is increased, there is proportionally less improvement in flight time with lower specific mass. There is no reason to delay a Pluto science program in the hope of getting very low specific mass powerplants. Table 2 also gives the mass ratio in LEO of the chemical rocket system (oxygen-hydrogen, dry mass of 15 percent) and the REP sciencecraft for the different Δv values applied at Earth escape. Even for a 400 kg sciencecraft (100 kg payload, 100 kg REP unit and 200 kg of propellant) given a 10 km/s excess velocity, the LEO mass is only about 5000 kg.

$\alpha/\eta_t = 200$ kg/kW			
τ_F(yr)	24.8	19.9	12.2
τ(yr)	13.8	10.9	7.7
τ_C(yr)	11.0	9.0	4.5
R(AU)	37.8	36.6	34.7
$\alpha/\eta_t = 100$ kg/kW			
τ_F(yr)	16.8	14.5	9.8
τ(yr)	9.8	9.5	6.3
τ_C(yr)	7.0	5.0	3.5
R(AU)	35.7	35.1	34.0
u_0(km/s)	0	5	10
Δv(km/s)	3.19	4.29	7.08
M_{LEO}/M_0	2.46	3.51	12.4

Table 2: Flight time, thrust time and optimal coast period for Pluto rendezvous corresponding to different powerplant specific masses α/η_t and different hyperbolic excess velocities u_0 supplied at Earth escape. The ion rocket payload fraction is 0.25. Launch occurs in 2010, and Pluto's distance from the Sun at rendezvous is denoted by R(AU). Also given are the velocity increment Δv needed in Earth orbit to provide the excess velocity and the ratio of the chemical rocket mass to ion rocket mass in LEO.

4. Heliopause Orbiter Missions

The Voyager probes launched in the 1970's to explore the outer planets are leaving the Solar System at about 3 AU/year (1 AU/yr = 4.74 km/s) and may reach the heliopause ($\sim$ 100 AU) between 2005 and 2010. Missions have been proposed to send dedicated probes out of the Solar System on fast escape trajectories in order to explore the heliopause and near-interstellar space. Because of the scientific interest in time-dependent phenomena occurring at the heliopause and in extending parallax measurements over longer baselines with an orbiting space telescope, we consider the use of REP for sending a sciencecraft to a 100 AU orbit for an extended scientific mission.

REP enables one to economically decelerate a rapidly moving robotic craft far from the Sun. However for high specific mass REP powerplants, using low-thrust propulsion alone for both the solar escape and deceleration phases can result in long mission times due to the long distances. For example, using Equation (1) in Ref. 3, the rendezvous flight time from Earth to 100 AU is 50 years for a specific mass of 200 kg/kW and 36 years for 100 kg/kW, assuming minimal Earth escape velocity and a payload fraction of 0.25 for the ion rocket. These times seem excessively long for such a mission. To reduce the flight time to 100 AU, we use a hybrid propulsion scheme in which chemical propulsion and gravity assist are applied in the inner Solar System, and electric propulsion is used in the outer Solar System.

Specifically the Earth-Jupiter gravity assist (EJGA) maneuver described in Ref. 2 is adopted here for the 100 AU orbiter missions. Application of low-thrust propulsion occurs only after the Jupiter encounter once the chemical rocket has been jettisoned. Transfer to Jupiter is achieved by placing the craft on a so-called 1/2 resonance orbit (period 2 years, perihelion 1 AU) and performing a small Δv maneuver at aphelion (2.175 AU). This enables an Earth gravity assist (EGA) 1.82 years after launch saving considerable chemical propellant compared to a conventional Hohmann transfer to Jupiter. The EGA is a well established technique which was used for the Galileo mission to Jupiter and will be used by the Cassini mission to Saturn.

The EGA transfer also makes it economical to take along extra chemical propellant to Jupiter for a powered gravity assist maneuver there. Performing a Δv maneuver deep in Jupiter's gravitational potential can result in a large increase in hyperbolic excess velocity. For example, an unpowered JGA results in a 2.7 AU/yr excess velocity while adding a 3 km/s periapsis burn gives a remarkable 6 AU/yr excess velocity [2]. Because a larger periapsis burn at Jupiter requires more storable propellant to be boosted out of Earth orbit, velocity increments greater than 3 km/s result in large mass ratios in LEO ($>$ 25) and excessive LEO masses even for small sciencecraft.

Table 3 contains results on flight time τ_F, total thrust time τ and optimal coast period τ_C for 100 AU rendezvous missions assuming REP powerplants with a specific mass in the 100 to 200 kg/kW range. The payload mass fraction of the REP sciencecraft is 0.25. The cases for no EJGA and EJGA are presented. Even though the Earth-Jupiter transfer takes 3.54 years, the benefit of the EJGA in reducing flight time is substantial. For the EJGA trajectories, it is interesting to note how little reduction in flight time is obtained by reducing the powerplant specific mass by a factor of two or adding a Jupiter periapsis burn ($\Delta v_p = 3$ km/s) alone. Only the combination makes a significant reduction in flight time. For the purpose of carrying out a timely science program, it probably makes little sense to wait for powerplants with specific masses below 100 kg/kW to be developed since the savings in mission time will likely be lost in the development time. With EJGA, near-term REP sciencecraft can reach heliopause orbits within 20 to 30 years.

$\alpha/\eta_t = 200$ kg/kW			
τ_F(yr)	50.0	28.7	25.6
τ(yr)	34.0	16.7	13.6
τ_C(yr)	16.0	8.5	8.5
$\alpha/\eta_t = 100$ kg/kW			
τ_F(yr)	36.0	24.5	21.8
τ(yr)	24.0	14.0	11.8
τ_C(yr)	12.0	7.0	6.5
Δv_p(km/s)	No EJGA	0	3
M_{LEO}/M_0	2.46	6.3	25.2

Table 3: Flight time, thrust time and optimal coast period for heliopause rendezvous (100 AU) corresponding to different powerplant specific masses α/η_t. The ion rocket payload fraction is 0.25. The first column refers to simple low-thrust transfer from Earth to 100 AU with no EJGA, and the latter two columns refer to EJGA with different velocity increments Δv_p applied at Jupiter periapsis. Also given is the ratio of the chemical rocket mass to the ion rocket mass in LEO required for the mission. The flight time includes the 3.54 year period for the Earth to Jupiter transit for the EJGA trajectories.

5. Gravitational Lens Missions

Sending sciencecraft farther out of the Solar System and into near-interstellar space with short mission times becomes increasingly difficult because of the relative insensitivity of flight time to the powerplant specific mass, as mentioned in the Introduction. In Ref. 2 the use of REP for sending robotic probes on fast solar escape trajectories out to several hundred AU was studied. In particular flight times for getting to the first gravitational lens focus of the Sun (550 AU) were presented for REP augmented by powered Jupiter gravity assist. A range of powerplant specific masses from 20 to 200 kg/kW was used in the previous study to determine the dependence of flight time on this parameter. For specific masses of 100 to 200 kg/kW, it was found that large periapsis burns at JGA were required to reduce the time to fly out to 550 AU. The payload fraction of the REP probe was taken as 0.25.

To investigate the tradeoff in payload fraction versus flight time, we present results for the flight time to 550 AU for payload fractions of 0.1, 0.25 and 0.4.

Figure 1 shows the flight time as a function of powerplant specific mass for these payload fractions. A 3 km/s periapsis burn is used during the Jupiter gravity assist, and the flight times include the 3.54 year transit time between Earth and Jupiter. Note that payload fractions above 0.25 are probably too costly in flight time unless the specific mass is below 70 kg/kW. Reducing the payload fraction from 0.25 to 0.1 only reduces the flight time by about 12 percent. For a payload fraction of 0.25, the flight time ranges from 45 to 52 years for specific mass of 100 to 200 kg/kW, and only falls below 30 years for specific masses less than 20 kg/kW.

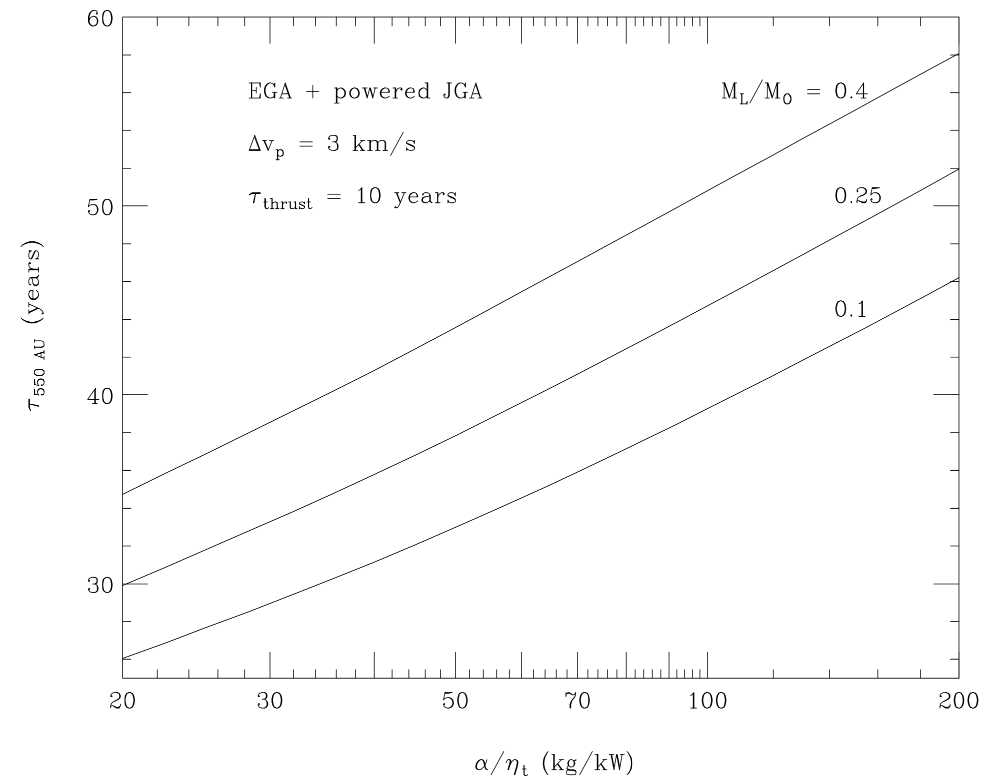

Figure 1: Flight time to reach 550 AU as a function of the powerplant specific mass for different payload mass ratios of the ion rocket. The velocity increment Δv_p at Jupiter periapsis is 3 km/s. The low-thrust period is ten years commencing after the Jupiter encounter. The flight time includes the 3.54 years needed for the Earth to Jupiter transit.

For the results in Fig. 1, the thrust duration of the ion rocket (commencing after the Jupiter encounter) was fixed at 10 years assuming that this was the reliability limit of the propulsion unit. Typically for low-thrust escape from the Solar System, there is an optimal thrust duration to minimize the flight time out to a particular distance. This optimum arises because of the competing effects of initial acceleration and final velocity. For shorter thrust times, the ion rocket's design acceleration would be higher (reducing the initial escape period), while for longer thrust times, the final velocity would be higher (reducing the final coast period). For a payload fraction of 0.25, the optimal thrust time was found to vary from 15 years to 26 years in the specific mass range of 20 to 200 kg/kW for flights to 550 AU. But the actual flight time was in fact quite insensitive to this choice of optimum, being reduced by only three to eight percent relative to that for a ten-year thrust period in this specific mass range.

6. Conclusion

Near-term radioisotope electric propulsion systems based on radioisotope electric generators and long-life ion thrusters are likely to have specific masses in the range 100 to 200 kg/kW of thrust power in the next ten years. When augmented by chemical propulsion and gravity assist, REP enables fast rendezvous missions to the outer Solar System and fast escape trajectories to near-interstellar space. Rendezvous missions to the Pluto-Charon system using chemical propulsion and REP can be carried out with 10 to 20 year flight times. A heliopause orbiter can be placed at a 100 AU parking orbit in 20 to 30 years when REP is combined with Jupiter gravity assist. Finally, REP and powered Jupiter gravity assist enable a sciencecraft to fly out to the first gravitational lens focus of the Sun (550 AU) in 40 to 50 years. In all cases, flight times are relatively insensitive to powerplant specific mass. For the purpose of performing a timely scientific program, the sum of hardware development time and mission time is an important consideration. It may be faster to simply use high specific mass REP systems rather than delay space science programs while waiting for low specific mass propulsion systems to be developed.

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