SLAC-PUB-13663
July 2009

Radioisotope Electric Propulsion for Deep Space Sample Return

Robert J. Noble[1]
SLAC National Accelerator Laboratory, Stanford University, Menlo Park, California 94025 USA

The need to answer basic questions regarding the origin of the Solar System will motivate robotic sample return missions to destinations like Pluto, its satellite Charon, and objects in the Kuiper belt. To keep the mission duration short enough to be of interest, sample return from objects farther out in the Solar System requires increasingly higher return velocities. A sample return mission involves several complicated steps to reach an object and obtain a sample, but only the interplanetary return phase of the mission is addressed in this paper. Radioisotope electric propulsion is explored in this parametric study as a means to propel small, dedicated return vehicles for transferring kilogram-size samples from deep space to Earth. Return times for both Earth orbital rendezvous and faster, direct atmospheric re-entry trajectories are calculated for objects as far away as 100 AU. Chemical retro-rocket braking at Earth is compared to radioisotope electric propulsion but the limited deceleration capability of chemical rockets forces the return trajectories to be much slower.

Nomenclature

AU	=	Astronomical Unit = 1.496 × 10^8 kilometers, the radius of the Earth's orbit around the Sun
EP	=	electric propulsion
g	=	acceleration of gravity at Earth's surface = 9.8 m/sec^2
I_{sp}	=	specific impulse = rocket's exhaust velocity/g
K	=	ratio of powerplant mass to propellant mass
M_L	=	payload mass
M_O	=	initial rocket mass (sum of payload, powerplant, and propellant masses)
R	=	starting distance from Sun for sample return trajectory
REP	=	radioisotope electric propulsion
α	=	powerplant specific mass = electric propulsion system mass/input power to thrusters
Δv	=	velocity change
η_t	=	electric rocket's total efficiency
τ	=	powered flight time (time during which thrusters are operating)
τ_{return}	=	sample return flight time (includes all powered thrust and coast durations)

I. Introduction

The primary methods for investigating extraterrestrial objects are remote sensing, either from Earth or a spacecraft, in-situ analysis, and sample return. Remote sensing can provide an overview of an object but at the expense of missing local details. In-situ analysis can explore these details, but the science is limited by the number of experiments that can be carried on-board a robotic lander, as was the case with the Viking missions to Mars in search of life. Because of the wide array of tests that can be performed on a sample in a terrestrial or space-station laboratory, sample return would be preferred if it could be done in a timely fashion and at an acceptable cost The desire to answer basic questions regarding the origin of the Solar System will motivate robotic sample return missions to increasingly distant objects.

Propulsion and power have always limited the feasibility of sample return missions. This type of mission inherently involves twice the interplanetary velocity change of a one-way rendezvous mission. Because the ratio of initial to final mass of a rocket increases exponentially with the ratio of velocity change to exhaust velocity,

[1] Staff Physicist, Advanced Accelerator Research Dept., 2575 Sand Hill Road, Mail Stop 07, AIAA Senior Member.

Presented at the 45th AIAA Joint Propulsion Conference, 08/02/2009 -- 08/05/2009, Denver, CO

propellant masses can become prohibitively large for velocity changes greater than a few times the rocket's exhaust velocity. Chemical propellants produce exhaust velocities of only a few kilometers per second and are impractical for outer Solar System missions requiring tens of kilometers per second velocity change. More distant targets demand ever-higher spacecraft velocities to keep the mission duration short enough to be of scientific interest. Fast sample return from the outer planets and the Kuiper belt would open an entirely new avenue for space science, but the vast distances make this a daunting task.

Low-thrust, electric propulsion (EP), which expels ions at high velocity (~10 to 100 km/sec) to attain a large momentum transfer with less propellant, is a recognized technology for high velocity space missions. Anticipating new developments in radioisotope electric generators and long-life ion engines, various authors have studied the use of radioisotope electric propulsion (REP) for robotic science missions in the outer Solar System[1-8]. Radioisotope electric propulsion refers to systems in which the heat source for generating electricity is an onboard radioisotope inventory. Several technologies for generating electricity from radioisotope heat have been explored over the years. These include standard semiconductor thermoelectric cells, thermo-photovoltaic cells, alkali-metal thermal-electric cells, and Stirling-cycle, free-piston alternators. Due to the modularity of these generators, REP scales gracefully to lower powers and is well suited to small robotic probes with masses of tens to hundreds of kilograms and propulsive power requirements of order kilowatts or less. The effective specific mass α/η_t (powerplant mass per unit input power to thrusters divided by rocket's total efficiency) of proposed, near-term REP systems is estimated in the range of 100 to 200 kg/kW, although lower values may be possible with development. REP is ideal for the sample-return leg of a deep-space robotic mission since the return vehicle is small (weighing perhaps a few hundred kilograms total and carrying kilogram-size samples), the radioisotope power system has few or no moving parts with low risk of start-up problems after years of storage, and the radioisotope provides a natural heat source for the dormant craft on the outbound journey.

In a previous work, we explored the use of radioisotope electric propulsion for returning samples from the planets of the outer solar system[5]. In this paper we extend the REP parameter study and trajectory calculations for objects out to 100 Astronomical Units (AU) in the Kuiper belt. High transit speeds are essential to make these missions feasible. A sample return mission involves several complicated steps to reach an object and obtain a sample, but only the interplanetary return phase of the mission is addressed in this paper. The generic scheme for a deep-space sample return mission is that a parent craft flies out from Earth to rendezvous with a target body. This craft or a special lander descends and collects a sample. The retrieved sample is loaded onto a special, dedicated return vehicle, which has its own propellant and optimized REP unit. This sample-return vehicle is the only object to fly back to Earth, while the parent craft continues its extended science mission at the body or moves on to study other objects.

On the return journey, a low-thrust rocket can accelerate to a high velocity in just a few years of thrusting, with the Sun's gravity helping the acceleration as the vehicle falls inward. REP produces very high-energy return trajectories with hyperbolic excess velocities approaching 7 AU/year. The problem of course is that in approaching the inner Solar System, the rocket with its extraterrestrial sample must shed much of this energy as it approaches Earth. Conventional braking with a practical sized, chemical retro-rocket is only suitable for a low-energy return when the velocity change is a few kilometers per second. Electric propulsion has the needed energy content for interplanetary deceleration, but the rocket can require significant time in the inner Solar System to decelerate due to the low thrust.

Two deceleration options are explored in this paper. If direct atmospheric entry must be avoided, a smooth orbital match and rendezvous with the Earth, moving at 30 km/sec on its 1 AU orbit, is required. The REP rocket must decelerate in a long, inward spiral trajectory (essentially the reverse of an escape spiral), as shown by the solid curve of Figure 1, until it reaches the Earth's sphere of influence, where it can be retrieved in a high orbit by another vehicle sent from Earth. Unfortunately, several years of the sample-return mission are spent spiraling inward within just a few AU of the Sun. Typically about *one-third* of the return trip time is spent in this low-thrust deceleration phase approaching Earth. The return times can be greatly reduced if a high-speed, direct re-entry to the Earth's atmosphere is used, like the returning Apollo spacecraft from the Moon and recent comet sample-return missions. For this option much less velocity change is required from the rocket, and the sample return vehicle can maintain a high in-fall speed over a shorter return path (dashed curve in Fig. 1). Although one might consider a chemical retro-rocket for the final braking step at Earth, we will show that there is a huge penalty in propellant mass unless the return trip from deep-space is slow and hence uninteresting.

For fast sample return with atmospheric re-entry, REP should be used for all acceleration and deceleration maneuvers. The REP rocket will be jettisoned from the sample return capsule just prior to Earth encounter and sent on a slightly deflected course to pass Earth. Only the sample capsule re-enters the Earth's atmosphere. Return trip times are reduced by more than *one-quarter* compared to the slower Earth rendezvous option. For example we find that with REP rockets of 100 to 200 kg/KW specific mass, sample return from Pluto can be accomplished in a remarkably short 9 to 11 years and from 100 AU in only 20 to 25 years. Future robotic craft sent to explore the outer Solar System and Kuiper belt objects will be major investments. Radioisotope electric propulsion will enable fast sample return vehicles to be included on these missions, vastly increasing the scientific return by bringing primordial material from distant worlds back to Earth.

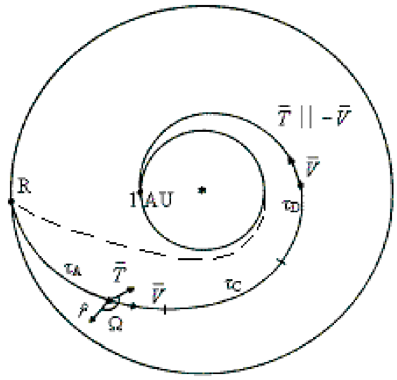

Figure 1. Return trajectories to an Earth orbit rendezvous (solid) and direct atmospheric re-entry (dashed).

II. Sample Return with Earth-Orbit Rendezvous

In this initial parametric study, flight times for sample-return missions using electric propulsion with Earth rendezvous and direct atmospheric re-entry are compared. To elucidate the REP technology requirements, a wide range of the powerplant specific mass is considered. We use the same FORTRAN program developed for our previous trajectory studies in References 1-3 and 5.

To simplify the analysis, only constant exhaust velocity, constant thrust ion propulsion is assumed for these calculations. The use of ion engines that have variable thrust and specific impulse is expected to yield somewhat improved flight times, typically at the level of ten percent. The simplified, constant thrust trajectory we use for our studies is shown in Figure 1. Orbits are approximated to lie in the ecliptic plane, which although an imprecise model for Pluto's inclined orbit, is perfectly adequate for our flight time estimates. The sheer distances to objects in the outer Solar System basically determine the return times, and plane changes introduce minor corrections.

For interplanetary rocket transfer, four quantities must be specified to determine a final orbit relative to an initial orbit in the same plane. Typically these are the semi-major axis, the eccentricity, the argument of perihelion, and the epoch of perihelion. Four mission parameters must then be selected to match the rocket's trajectory with the desired target orbit. The departure date from the initial body provides one parameter to insure that the rocket meets the destination orbit at a particular time. The acceleration thrust time τ_A, the thrust angle Ω (relative to the Sun's radial vector **r**, for example), and the deceleration thrust time τ_D can be used as the other three matching parameters. The deceleration thrust vector **T** is fixed exactly opposite to the velocity vector **V** during the braking maneuver. With these choices, the coast duration τ_C (the period of no thrusting between the acceleration and deceleration phases) is the only free parameter in our thrust program to minimize the flight time. This simplified thrust program fully describes the trajectory matching constraints, but not the rocket configuration, i.e. the choice of mass fractions for the electric rocket's powerplant and propellant.

In a complete optimization to maximize the payload delivered in the shortest flight time, ideally all trajectory and rocket parameters are varied simultaneously. However the optimal solution is not equally sensitive to all parameters. In Reference 2, it was found that for constant-thrust rockets which spend a significant fraction of time in the weak gravitational fields beyond a few AU from the Sun, the rocket configuration that maximized the payload mass for a given flight time was roughly the same as for a rocket in field-free space, nearly independent of the exact trajectory. In this case, the optimal ratio K of the powerplant mass M_W to propellant mass M_P is given approximately by the empirical formula $K_{opt} = 0.26(1+2\ln(1+3\ M_L/M_O))$, where M_L is the desired maximum payload mass, and M_O is the initial rocket mass (sum of payload, powerplant and propellant). This simplified rocket configuration is adopted for all calculations in this paper. The reader is reminded that for a specified powered flight time τ, the optimal propellant velocity $v'_p = \eta_m v_p = K_{opt} v'_c$ where $v'_c = (2\tau\ \eta_t/\alpha)^{1/2}$ is the powerplant's characteristic velocity, α is the powerplant specific mass, η_t is the electric rocket's total efficiency, η_m is the thruster's mass utilization efficiency, and v_p is the actual exhaust velocity. Note that the effective powerplant specific mass α/η_t is the relevant figure of merit for the propulsion system and determines the velocity change it can produce.

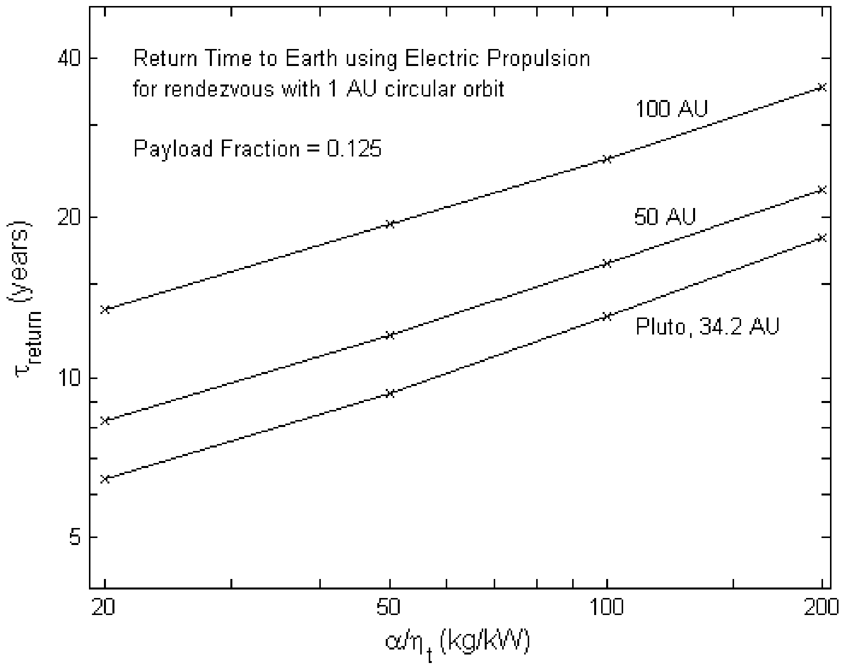

Figure 2. Return time to Earth orbit rendezvous as a function of powerplant specific mass for different distances from the Sun.

Figure 2 shows the return flight time to rendezvous with the Earth's 1 AU circular orbit from the distances of Pluto (34 AU), 50 AU, and 100 AU as a function of the powerplant effective specific mass α/η_t. Here the payload fraction of the returning electric rocket is $M_L/M_O = 0.125$, and the powerplant to propellant mass ratio is $K_{opt} = 0.43$. Near-term REP systems are anticipated to have specific masses in the range 100 to 200 kg/kW, but we show results for specific masses down to 20 kg/kW for comparison with hypothetical, advanced nuclear electric propulsion. Table 1 lists the thruster specific impulses $I'_{sp} = v'_p/g = \eta_m v_p/g$ for the missions illustrated in Figure 2.

Table 1: Effective specific impulse I'_{sp} as a function of powerplant specific mass for sample return missions using Earth orbit rendezvous and a payload fraction of 0.125.

I'_{sp} (sec)	Pluto	50 AU	100 AU
α/η_t = 20 kg/kW	7395	8416	10694
α/η_t = 50 kg/kW	5606	6423	8280
α/η_t = 100 kg/kW	4834	5485	6917
α/η_t = 200 kg/kW	4114	4622	5838

The scaling of flight time with the starting distance R from the Sun is roughly proportional to $R^{0.7}$, which is similar to the two-thirds power law for a low-thrust rocket in free-space. The flight time scales approximately with powerplant specific mass as $(\alpha/\eta_t)^{0.45}$, which not surprisingly is about the same scaling found in Ref. 2 for *outbound* rendezvous trajectories starting at 1 AU. Very low specific-mass propulsion (≤ 20 kg/kW) must be developed before significantly shorter trips are possible with Earth orbit rendezvous. One reason for the long return times is the large fraction of time spent by the rocket decelerating at the *inner* Solar System to smoothly match to the Earth's orbit. Tables 2, 3, and 4 present the fractions of the return flight time used for acceleration, optimal coast, and deceleration, respectively, for the missions illustrated in Figure 2. Approximately *one-third* of the return time is spent in the final deceleration spiral approaching the Earth, and this fraction of time increases as the specific mass of the powerplant increases.

Table 2: Acceleration fraction of flight time as a function of powerplant specific mass for sample return missions using Earth orbit rendezvous and a payload fraction of 0.125.

$\tau_{accel}/\tau_{return}$	Pluto	50 AU	100 AU
α/η_t = 20 kg/kW	0.39	0.38	0.38
α/η_t = 50 kg/kW	0.36	0.36	0.36
α/η_t = 100 kg/kW	0.34	0.35	0.36
α/η_t = 200 kg/kW	0.32	0.32	0.35

Table 3: Optimal coast fraction of flight time as a function of powerplant specific mass for sample return missions using Earth orbit rendezvous and a payload fraction of 0.125.

$\tau_{coast}/\tau_{return}$	Pluto	50 AU	100 AU
α/η_t = 20 kg/kW	0.39	0.39	0.39
α/η_t = 50 kg/kW	0.40	0.38	0.37
α/η_t = 100 kg/kW	0.36	0.34	0.34
α/η_t = 200 kg/kW	0.34	0.33	0.31

Table 4: Deceleration fraction of flight time as a function of powerplant specific mass for sample return missions using Earth orbit rendezvous and a payload fraction of 0.125.

$\tau_{decel}/\tau_{return}$	Pluto	50 AU	100 AU
α/η_t = 20 kg/kW	0.22	0.23	0.23
α/η_t = 50 kg/kW	0.24	0.26	0.27
α/η_t = 100 kg/kW	0.30	0.31	0.30
α/η_t = 200 kg/kW	0.34	0.35	0.34

To illustrate the trade-off between flight time and increased payload mass, sample return from Pluto is compared for payload fractions of 0.125 and 0.25. The return time versus specific mass is shown in Figure 3, with an increase of about 30% in flight time for twice the payload. Flight times are not reduced much for payload fractions below 0.125. Return times go up quickly for payload fractions above 0.25 so higher fractions are not recommended.

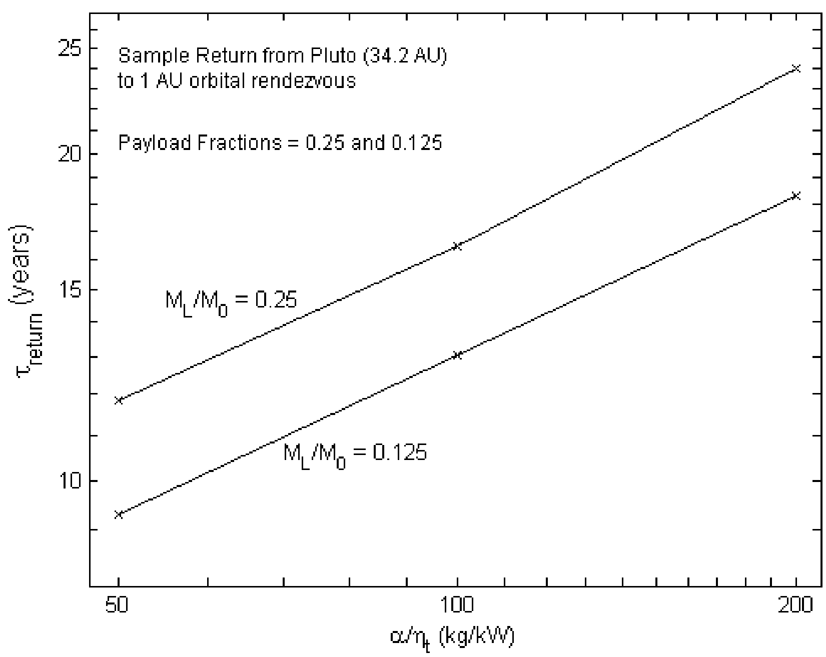

Figure 3. Return time from Pluto to an Earth orbit rendezvous for two different payload mass fractions.

III. Sample Return with High-Speed Atmospheric Re-entry

For the Earth orbit rendezvous option of Section II, the REP rocket must decelerate all the way down to 30 km/sec to smoothly match the Earth's 1 AU orbit around the Sun. The entire return spiral actually involves one or more revolutions around the Sun, lengthening the flight time. The return times can be greatly reduced if a high-speed, direct re-entry to the Earth's atmosphere is used, like the returning Apollo spacecraft from the Moon and comet sample-return missions. Much less velocity reduction is required, and the sample return vehicle can maintain a high in-fall speed over a shorter interplanetary return path. In this option, after completing its job of deceleration, the REP rocket will be jettisoned from the sample return capsule just prior to Earth encounter and sent on a slightly deflected course to pass Earth. Only the sample capsule re-enters the Earth's atmosphere.

The trajectory of the return craft approaches the Earth tangentially from behind in its orbit about the Sun. This approach allows the maximum arrival velocity since only the relative difference in velocity between the Earth and vehicle is relevant for atmospheric impact. State-of-the-art heat shield technology limits the maximum atmospheric entry speed to 15 km/sec relative to Earth. But this relative speed is made up of the arrival speed plus the final velocity gain from falling through the Earth's gravitational potential. A simple calculation shows that the vehicle can approach the Earth at a maximum relative speed of about 10 km/sec (40 km/sec heliocentric speed) so as to result in a 15 km/sec atmospheric impact speed.

Since the returning sample capsule now must only be decelerated down to a heliocentric speed of 40 km/sec rather than 30 km/sec, one might consider a chemical retro-rocket for this final braking maneuver if it involves only a few kilometers per second of deceleration. The velocity change Δv which we can practically achieve is limited because storable chemical propellants have an exhaust velocity v_p of only about 3 km/sec, and the ratio of initial rocket mass M_i to final mass M_f is driven up exponentially according to the familiar rocket equation $M_i/M_f = \exp(\Delta v/v_p)$. Furthermore, all this propellant mass must be transported from Earth by the parent craft at a high cost. REP is still assumed to accelerate the sample capsule and its chemical retro-rocket toward Earth from deep-space because to do this with a chemical rocket would make the propellant masses completely unrealistic.

We present two instructive examples where the chemical retro-rocket supplies a decelerating velocity change of either 2.5 km/sec or 5 km/sec to reduce the heliocentric speed to 40 km/sec. The REP payload ratio is kept at 0.125 in these examples, but the payload now includes the sample capsule, retro-rocket, and mostly chemical propellant. Because of this added propellant, the REP rocket must actually accelerate a mass of at least 2.3 or 5.3 times the sample capsule mass, respectively, for these two examples. For the same payload ratio then, the actual size of the REP stage is increased several times, making it much more expensive. Tables 5 and 6 list the return times from the distances of Pluto (34 AU), 50 AU, and 100 AU for different specific masses of the accelerating REP stage. The return times are very insensitive to the specific mass because we are limiting the acceleration energy to produce the same arrival speed at Earth (42.5 or 45 km/sec), and in fact we are not using REP to its full acceleration capability. The corresponding hyperbolic excess velocities of these orbits are 5.8 km/sec and 15.8 km/sec, respectively. This results in rather long flight times increasing linearly with distance. Shorter flight times come at the expense of an exponentially increasing propellant mass so chemical braking is not suitable for fast sample return from deep-space.

Table 5: Return time when the chemical rocket Δv at Earth is 2.5 km/sec as a function of powerplant specific mass with an REP rocket payload fraction of 0.125.

τ_{return} (years)	Pluto	50 AU	100 AU
α/η_t = 50 kg/kW	14.1	22.7	54.5
α/η_t = 100 kg/kW	14.5	23.1	54.9
α/η_t = 200 kg/kW	15.1	23.6	55.6

Table 6: Return time when the chemical rocket Δv at Earth is 5 km/sec as a function of powerplant specific mass with an REP rocket payload fraction of 0.125.

τ_{return} (years)	Pluto	50 AU	100 AU
α/η_t = 50 kg/kW	9.44	13.9	28.5
α/η_t = 100 kg/kW	10.1	14.5	29.4
α/η_t = 200 kg/kW	11.0	15.5	30.7

Radioisotope electric propulsion is the preferred method for decelerating the sample capsule prior to atmospheric re-entry because its performance surpasses chemical braking. The sample return trajectory calculations described in Section II were repeated for REP deceleration in which the final heliocentric speed at 1 AU was set to 40 km/sec, resulting in 15 km/sec atmospheric impact speed at Earth. The return trajectory to 1 AU is greatly shortened in length compared to the Earth rendezvous option. The vehicle now traverses less than one solar revolution over the entire path. These are high-energy return orbits with hyperbolic excess velocities in the range 16 km/sec to 32 km/sec for REP specific masses in the range 200 to 100 kg/kW. Figure 4 shows the return flight time to the Earth from the distances of Pluto (34 AU), 50 AU, and 100 AU as a function of the REP effective specific mass α/η_t when the payload fraction is 0.125 and $K_{opt} = 0.43$. Table 7 lists the thruster specific impulses $I'_{sp} = v'_p/g = \eta_m v_p/g$ for the missions illustrated in Figure 4. The optimal specific impulses are lower compared to those for the Earth rendezvous option because the total powered flight times are less for these trajectories.

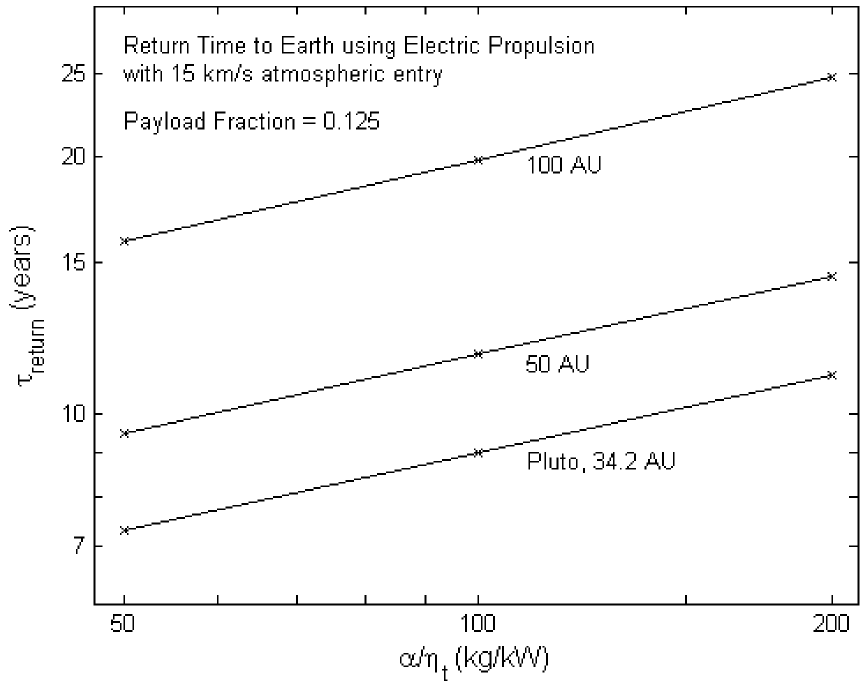

Figure 4. Return time to Earth with direct atmospheric re-entry as a function of powerplant specific mass for different distances from the Sun.

Comparison of the return times in Figure 4 with those in Tables 5 and 6 shows that the return time with chemical rocket braking only approaches the REP flight time for the heaviest REP system at 200 kg/kW in the case of Pluto. This is at the expense of a huge chemical propellant mass for the 5 km/sec velocity change. In all other cases, REP is far superior for fast, deep-space sample return. The REP advantage increases with distance. The scaling of flight time with the starting distance from the Sun is approximately $R^{0.7}$, which is again similar to the two-thirds power law for a low-thrust rocket in free-space. The flight time scales with specific mass roughly as $(\alpha/\eta_t)^{0.32}$ which is a weaker scaling than the Earth rendezvous option but the same as the one-third power law for a low-thrust rocket in free space. Sample return time is not significantly shortened by making small reductions in specific mass. One can commit to REP sample return once the specific mass is in the 100-200 kg/kW range since the time lost in developing a slightly improved propulsion system will not be made up by a reduced flight time.

Table 7: Effective specific impulse I'_{sp} as a function of powerplant specific mass for sample return missions using atmospheric re-entry and a payload fraction of 0.125.

I'_{sp} (sec)	Pluto	50 AU	100 AU
α/η_t = 50 kg/kW	4720	5500	7241
α/η_t = 100 kg/kW	3665	4272	5664
α/η_t = 200 kg/kW	2854	3332	4427

Tables 8, 9, and 10 list the fractions of the return flight time used for acceleration, optimal coast, and deceleration, respectively, for the missions illustrated in Figure 4. Only ten to fifteen percent of the return time is spent in the deceleration phase. Return trip times are reduced by more than *one-quarter* compared to the slower Earth rendezvous option. Finally we note that increasing the payload fraction from 0.125 to 0.25 for these fast trajectories results in about a 30% increase in flight time, the same increase as found for the slower rendezvous trajectories of Section II. With this doubling of the payload fraction, there is a significant increase in scientific return for a slightly longer flight time.

Table 8: Acceleration fraction of flight time as a function of powerplant specific mass for sample return missions using atmospheric re-entry and a payload fraction of 0.125.

$\tau_{accel}/\ \tau_{return}$	Pluto	50 AU	100 AU
α/η_t = 50 kg/kW	0.43	0.42	0.42
α/η_t = 100 kg/kW	0.42	0.42	0.41
α/η_t = 200 kg/kW	0.42	0.42	0.41

Table 9: Optimal coast fraction of flight time as a function of powerplant specific mass for sample return missions using atmospheric re-entry and a payload fraction of 0.125.

$\tau_{coast}/\ \tau_{return}$	Pluto	50 AU	100 AU
α/η_t = 50 kg/kW	0.45	0.43	0.41
α/η_t = 100 kg/kW	0.47	0.44	0.42
α/η_t = 200 kg/kW	0.48	0.45	0.43

Table 10: Deceleration fraction of flight time as a function of powerplant specific mass for sample return missions using atmospheric re-entry and a payload fraction of 0.125.

$\tau_{decel}/\ \tau_{return}$	Pluto	50 AU	100 AU
α/η_t = 50 kg/kW	0.12	0.15	0.17
α/η_t = 100 kg/kW	0.11	0.14	0.17
α/η_t = 200 kg/kW	0.10	0.13	0.16

IV. Conclusion

Near-term radioisotope electric propulsion with specific mass in the range of 100 to 200 kg/kW yields relatively short flight times for deep space, sample return missions. REP is ideal for small vehicles of only a few hundred kilograms total mass returning kilogram-size samples. REP produces fast, high-energy return trajectories with hyperbolic excess velocities approaching 7 AU/year. Chemical rockets are unsuitable for these transfers because of their limited velocity capability and the huge mass of propellant which would have to be transported from Earth. Using REP, the shortest return times to Earth are accomplished with trajectories ending in a high-speed, direct re-entry to the Earth’s atmosphere. The sample-return time from Pluto is 9 to 11 years and from 100 AU it is only 20 to 25 years for specific masses in the range of 100 to 200 kg/kW and a payload fraction of 0.125. For these types of missions, the payload fraction of the REP vehicle is recommended to be in the range 0.125 to 0.25 since the flight time increases quickly for larger payloads. The weak scaling of flight time with the powerplant specific mass (proportional to the one-third power) means that advanced propulsion is not a necessary precursor for sample return capability to be added to future robotic missions. One can commit to REP sample return once the specific mass is in the 100-200 kg/kW range since it is very difficult to reduce the flight time enough to offset the development time of an improved propulsion technology.

Acknowledgments

The author thanks John L. West at NASA/JPL and Gary L. Bennett for their continued interest, encouragement, and discussions. This work was supported by Department of Energy contract DE-AC03-76SF00515.

References

[1] Noble, R.J., "Radioisotope Electric Propulsion of Small Payloads for Regular Access to Deep Space," AIAA Paper No. 93-1897, *29th Joint Propulsion Conference*, Monterey, CA, 1993.

[2] Noble, R.J., "Radioisotope Electric Propulsion for Robotic Science Missions to Near-Interstellar Space", *J. British Interplanetary Soc*., Vol. 49, pp. 322-328, 1996; "Radioisotope Electric Propulsion for Small Robotic Space Probes", *J. British Interplanetary Soc*., Vol. 49, pp.455-468, 1996.

[3] Noble, R.J., "Radioisotope Electric Propulsion of Sciencecraft to the Outer Solar System and Near Interstellar Space", *Acta Astronautica*, Vol. 44, Nos. 2-4, pp.193-199, 1999. An adapted article of the same title also appeared in *Nuclear News*, pp. 34-40, November 1999 (American Nuclear Society).

[4] Oleson, S., Benson, S., Gefert, L., Patterson, M., Schreiber, J., "Radioisotope Electric Propulsion for Fast Outer Planetary Orbiters," AIAA 2002-3967, *38th Joint Propulsion Conference*, Indianapolis, IN, 2002.

[5] Noble, R.J., "Combining Electric and Sail Propulsion for Interplanetary Sample Return", *Proc. of the 28th International Electric Propulsion Conference*, Toulouse, France, 2003.

[6] Williams, R., Gao, Y., Kluever, C.A., Cupples, M., Belcher, J., "Interplanetary Sample Return Missions Using Radioisotope Electric Propulsion," AIAA 2005-4273, *41st Joint Propulsion Conference*, Tucson, AZ, 2005.

[7] Oh, D., Bonfiglio, E., Cupples, M., Belcher, J., Witzberger, K., Fiehler, D., Artis, G., "Evaluation of Radioisotope Electric Propulsion for Selected Interplanetary Science Missions," IEPC-2005-181, *Proc. of the 29th International Electric Propulsion Conference*, Princeton, NJ, 2005.

[8] Schmidt, G.R., Manzella, D.H., Kamhawi, H., Kremic, T., Dudzinski, L.A., "Radioisotope Electric Propulsion: A Near-term Nuclear Propulsion System," AIAA 2008-5099, *44th Joint Propulsion Conference*, Hartford, CT, 2008.