IEPC-2003-0137

RADIOISOTOPE ELECTRIC PROPULSION FOR NEW FRONTIERS CLASS MISSIONS

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Abstract

With the initiation of the New Frontiers Class of space science missions, small radioisotope powered spacecraft for outer planet exploration will become reality. In order for these missions to co-orbit various primitive objects and moons of interest, a highly efficient electric propulsion system is needed. The use of such a radioisotope electric propulsion system is enabled by a new direct trajectory using a medium class launch vehicle which provides most of the acceleration for the spacecraft. The electric propulsion system then provides the deceleration at the target. Key to the development, in addition to light sub-kilowatt electric propulsion and radioisotope power systems, is light spacecraft and science payload technologies. This work further explores these new radioisotope electric propulsion direct trajectories by applying them to New Frontiers Class missions which seek to answer specific questions about the solar system. Verification of past trajectories and development of new target trajectories (e.g. Jupiter and a demonstrator mission) are included.

Introduction

With the inclusion of radioisotope power systems in the recently initiated New Frontiers Class of space science missions, science targets beyond Mars can be reached without a large and expensive Flagship mission. The presence of this constant, but low level, power source has been studied by various authors. [1-5] A more recent work has shown that such radioisotope electric propulsion (REP) spacecraft can *orbit or co-orbit* various large and small science targets beyond Mars with transit times similar to large nuclear electric vehicles (Many of these targets are not accessible to chemical, solar electric or aerocapture vehicles.) [6] This recent work discovered that using a medium class launch vehicle with an upper stage can reduce the REP trip times 50% from past estimates by using the launch vehicle to provide the Earth escape and acceleration while the REP (generally) only has to decelerate the vehicle.

Besides the outer planets and their moons many other targets of scientific interest exist including the Jupiter Trojans, the Centaurs, other asteroids, comets, and Trans-Neptunian objects.[7] Exploration of these bodies can answer many of the scientific questions posed in the recent National Research Council Decadal Study. [8]

Recent studies have shown that power and propulsion specific masses of 100 to 150 kg/kW are needed to provide reasonable mission times and performance.[1-6] Existing radioisotope thermoelectric generators (RTGs) combined with off-the-shelf ion propulsion systems (e.g. the 30 cm ion propulsion system flown on Deep Space 1 and capable of 500 W operation) would provide a combined specific mass of almost 300 kg/kW. Current RTGs also use many plutonium bricks due to the low efficiency of the thermoelectric conversion system. Use of advanced radioisotope power systems, such as the Stirling convertor, promise much lighter power systems. Specifically, the Stirling convertor promises an almost four-fold improvement in electric conversion efficiency, thus reducing the number of required plutonium bricks by the same factor. Ion propulsion with long life and optimized for efficient operation at low power is also needed to reduce the thruster system mass required for the extended burn times.

The final requirement to make the REP concept feasible is a small but capable spacecraft, with a science package, but not including power and propulsion, of around 100 to 300 kg. While the technologies and a baseline design for such a light-weight REP spacecraft are still under evaluation, the potential New Frontiers Class mission opportunities for such a spacecraft are explored in this work.

The previous work[6] is continued here with several objectives:

- Verify the past trajectories using alternate optimization tools
- Explore the applicability of REP to New Frontiers Class missions
- Develop potential technology demonstrator targets

A brief review of the necessary technologies and a description of the REP trajectory are presented first.

REP Technologies

The three key technologies needed for an REP spacecraft are small, advanced ion thrusters, lightweight radioisotope power systems, and a small spacecraft (<500 kg) that can perform valuable science. This scoping study assumed ion thrusters with an operational power range of 100-500 W, Stirling radioisotope generators (representative of the advanced systems under development) that can supply constant power of 100-900 W to the ion propulsion system and lightweight spacecraft bus technologies that enable revolutionary 100-300 kg spacecraft bus designs. Each will be discussed in turn.

Sub-kilowatt Ion Propulsion

NASA Glenn Research Center is developing a lightweight (< 3.0 kg combined mass, representing a 80% reduction from state-of-the-art), sub-kilowatt xenon ion thruster (figure 1) and power processor. Performance goals include 50% efficiency at 0.25 kW, representing a 2x increase over the state-of-the-art.

The sub-kilowatt ion propulsion activity includes both an in-house hardware development element for the thruster and power processor, as well as a contracted system element. At NASA GRC, the fabrication and performance assessment of a small (0.25 kW class) laboratory model thruster with an 8 cm beam diameter has been completed,[9-12] and the fabrication of a second-generation lightweight engineering model thruster with a 100-500 W power throttling envelope has also been completed. (ref) Also at NASA GRC, first- and second-generation breadboard power processors have been fabricated and successfully integrated with the 8 cm thruster.[13-15]

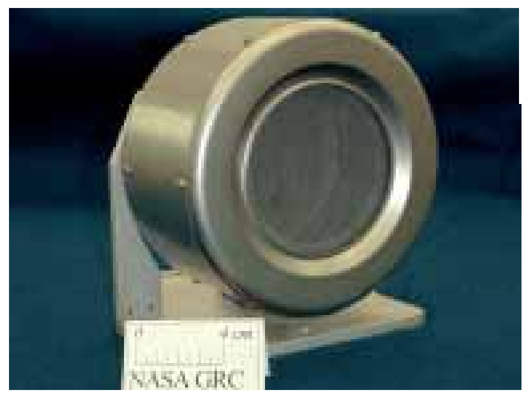

Figure 1. NASA Sub-Kilowatt Ion Thruster

The second-generation breadboard power processing unit (PPU) (Figure 2) was fabricated with a maximum output power capability of up to 0.45 kW at a total efficiency of up to 90 percent. Four power converters were used to produce the required six electrical outputs which resulted in significant mass reduction for the PPU. The component mass of this breadboard is 0.65 kg and the total power convertor mass is 1.9 kg. Integration tests with the thruster included short circuit survivability, single and continuous recycle sequencing, and beam current closed-loop regulation.

Figure 2. Power Processing Unit

General Dynamics, under contract, developed a conceptual design for the low-power ion propulsion system.[16] The objectives of this effort were to develop a system that improved performance and reduced system mass compared to existing state-of-the-art systems. The resulting design was tailored to the meet the needs of the satellite and spacecraft integration community as identified in an extensive user survey performed by General Dynamics. The basic characteristics of the system are as follows:

- up to 20 mN thrust
- 100-500 Watts input power
- 1600-3500 seconds I_{sp}
- thruster mass: 0.95 kg
- PPU mass: 2.0 kg
- Central Xenon Feed System mass: 3.1 kg (excluding tank)

Recently, an 8 cm pyrolitic graphite grid set was tested at GRC. Initial results showed operational performance similar to that of molybdenum grids. [17] Grid lifetime estimates using such materials predicts improvements over molybdenum of a factor of 5 or more.

Stirling Radioisotope Generator (SRG)

An advanced radioisotope electric power generator is currently being developed for use on deep space missions, as well as for Mars surface applications. It is based on the high efficiency free-piston Stirling power convertor (Stirling engine coupled to a linear alternator). The Department of Energy (DOE) has responsibility for developing the SRG for use on NASA missions. GRC is supporting DOE in this effort, drawing upon its many years of experience in developing Stirling power conversion technology. The SRG is a high-efficiency alternative to the Radioisotope Thermoelectric Generators (RTGs) that have been used to power past missions. The Stirling efficiency, in excess of 25%, leads to a factor of 4 reduction in the inventory of plutonium required to heat the generator. The spacecraft power system will be comprised of one or more generators, based on the power requirements of the mission.

Figure 3. Stirling Technology Demonstrator Convertor

The SRG will be based on a Stirling power convertor known as the Technology Demonstration Convertor (TDC). The TDC was developed as a laboratory device to validate free-piston Stirling technology for the radioisotope generator application (figure 3.) A joint government/industry committee developed a set of criteria that was used to determine the readiness of Stirling technology for transition to flight.[18] Having successfully passed these tests, the TDC is now being transitioned from a laboratory device to flight application. As a part

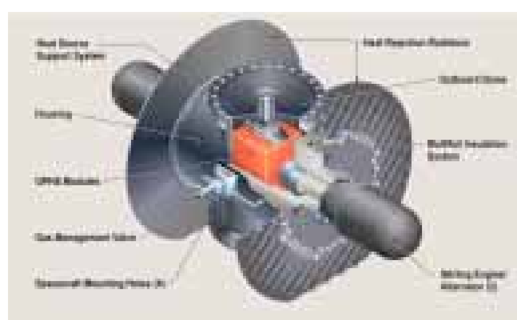

Figure 4. Stirling Radioisotope Generator Concept

of this process, DOE conducted a competitive procurement for a System Integration Contractor to design, develop, qualify and supply SRG units to NASA for the future missions. Lockheed

Martin Aeronautics of Valley Forge, PA was selected as the System Integration Contractor. Figure 4 shows an early concept of the SRG however the unit being developed differs significantly from this. The present schedule would produce an engineering unit in about two years. The follow-on effort would produce a qualification unit and then flight units for missions in the later half of the decade.

The SRG will be heated by plutonium housed inside of two General Purpose Heat Source (GPHS) modules. Each module will provide approximately 250 W_{th} at beginning of mission (BOM). The initial SRG, based on the laboratory TDC transitioned to flight, will have a mass of about 27 kg with contingency and produce approximately 114 Wdc. This results in specific power of 4.2 W/kg. Analysis performed at GRC projects that an advanced SRG could increase the specific power to nearly 10 W/kg with the major advance being in a low mass Stirling convertor along with modest advances in the controller and thermal systems.

Long life with no degradation has been accomplished through the use of non-contacting operation to virtually eliminate wear of the moving components. The present design of the Stirling convertor for the SRG has been designed for a 100,000 hour life (11.4 year) however the life could be extended through a design modification of the heater head or possibly through the operating methodology chosen. Three components are critical to achieving long life; the flexure bearing system, the permanent magnets in the linear alternator, and the heater head. Although the flexure technology has its origins in engines, it has gained more wide-spread acceptance for long-life cryocoolers. Long-life Stirling cryocoolers are presently flying on spacecraft, with the most recent launch being the RHESSI spacecraft. Flexures are designed and qualified for the design life, and are then operated at significantly derated conditions to achieve essentially infinite life. For the SRG, creep of the heater head is the life-limiting component. The life can be extended multifold by an engineering trade to reduce heater head stress and creep rate with in exchange for reduced performance. These issues are presently being addressed with analysis and tests at GRC.[19] As demonstration of the long-life capability, a free-piston Stirling convertor continues to operate after approximately 70,000 hours (8.0 years) with no degradation.[20]

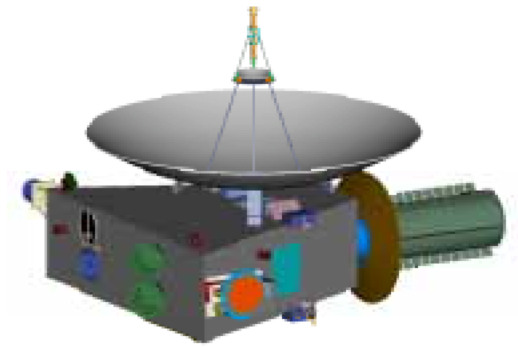

Figure 5. New Horizons Spacecraft

Lightweight Spacecraft Bus and Instrument and Bus Technologies

Advanced microelectronics/lightweight spacecraft bus development has been underway at the JHU/APL and will be leveraged toward the outer planet mission opportunities. This analysis is ongoing but has not been updated from past works.[1,2]

A recent spacecraft design that is of a similar class mission to that of an REP orbiter is the New Horizons Pluto Flyby mission. It has a payload mass of only 24 kg with a launch mass of 412 kg. This design represents a conservative, near-term design, and includes power and chemical propulsion.[21]

Since the spacecraft bus is still undefined the analyses in this work traded the delivered spacecraft bus and payload mass with the propulsion parameters and trip time. When the spacecraft & science analysis is complete it will be integrated with this analysis.

Systems Analyses

For the sample outer planetary object missions, the previous technology descriptions were modeled for mass and performance analyses. A launch date of 2011 was chosen to allow sufficient technology advancement, but earlier or later launch dates should have similar results. The assumed performance of the power and propulsion system is shown in Table 1. The 750 W point was chosen based on past mission analysis iteration as near-optimal.[6] Using the information in the table, a fixed specific mass (alpha) of 150 kg/kW was assumed for the trajectory runs for this scoping study. The tankage was set to 10% of the required fuel mass. An additional 30% contingency, commensurate with mission scoping practices, was assessed to the power and propulsion system. The rest of the spacecraft: bus, science and margin, (BSM) was varied from 127 kg to 267 kg. This BSM includes the contingencies and margins for the bus and science but not the power and propulsion system.

Table 1. Outer Planet Orbiter Assumptions

Outer Planet Exploration Subsystem Options	Unit	Total (150 kg/kW)
	Mass/Power	Mass/Power
Complete SRG System	19 kg / 162W (avg.)	5 Units 94 kg / 810 W
8 cm Ion Propulsion System		8 Thrusters, 3 PPUs 18.1 kg / 750 W
Thruster (w structure, feed & gimbal)	1.5 kg	
PPUs	2.1 kg	
Feed Sys.		3.1 kg
DCIU	2.5 kg	
Cable (per thruster)	0.2 kg	
Thermal	0.4 kg	
Tankage	10%	
Net Spacecraft Bus (Launch Mass less Science, Power, Wet Propulsion)		127 to 267 kg
Science		20 – 50 kg
Required Fuel Throughput / Thruster	20 – 30 kg xenon	
Ion Thruster Isp (sec)	Single Isp (2600 s to 3700 s) optimized by trajectory code	375 W power into Thruster
Estimated Ion Propulsion System Efficiency (based on test data)	Relative to Optimal Isp (48% to 53%)	

For the ion thruster system, lifetime was assumed possible using advanced grid technologies including thick molybdenum, titanium, or carbon based technologies (pyrolitic graphite).[17] Specific impulse was optimized in the analysis to guide future development. Total propulsion system performance (efficiency) was varied based on required I_{sp} by the function: Efficiency = $(bb * I_{sp}^2) / (I_{sp}^2 + dd^2)$ where bb=.764693 and dd = 2195.36. This trend is representative of 8 cm test data at similar power levels.[9-12] Masses for the thruster and components include gimbal, structure and thermal control masses. A spare PPU was assumed to ensure that two are operational so that roll control can be provided by the ion thrusters during their operation. A digital control interface unit [DCIU] is added to control the thrusters, PPUs, and the feed system. The DCIU interfaces with the spacecraft computer. The Stirling system technology is based upon nickel-based super alloys and temperatures of 923K.

Shown in Table 1 are the system assumptions for the outer planetary target orbiters. The housekeeping power was limited to 60 W during thrusting. Spacecraft communications were restricted to ion thruster off-times when more power is available. Two thruster operation is assumed, where possible, to allow for attitude control of the spacecraft during cruise with the ion thrusters. Eight thrusters were carried on the spacecraft. Seven of the eight thrusters are expected to handle the required fuel throughput in case of engine-out. Improvement of thruster lifetime by using advanced materials (such as pyrolitic graphite) and longer life cathodes could reduce the number of thrusters.

Mission Analyses

Past works used the trajectory optimization code VARITOP (developed by Carl Sauer of JPL) to assess actual trajectories for REP systems for outer planet missions.[19] The VARITOP code was also be used to optimize I_{sp} and power level given the appropriate thruster and mass models. Specific launch vehicle performance was also an input to the code and optimal excess velocities were found.

Verification of 'Fast' REP Trajectories

Work reported recently showed the ability of REP to provide fast trajectories to outer planet targets.[22] In most cases the previous and current analyses assumed an Atlas V 551 medium launch vehicle using a Star 48 upper stage.[23] This combination also provided the up to 400 kg launch mass to an excess velocities up to of 14.14 km/s ($C3 = 200\ km^2/s^2$).

While the VARITOP code has been verified, its solutions are sometimes a local minimum. As such another tool using a different optimization method – the Direct Trajectory Optimization Method (DTOM) code - was utilized to check the Varitop results and provide evaluation of other targets.

DTOM is a direct method for obtaining optimal, low-thrust, interplanetary trajectories.[24] The DTOM numerically integrates the equations of motion using modified equinoctial orbital elements to accommodate circular orbits (e = 0).[25] The parameterized continuous-time control, thrust and coast lengths, launch date scaling factor, and Earth-escape parameters define the generic design space. More specialized problems can be defined with planetary gravity assists, loiter periods at the target body (used for sample-return missions), optimization of power level and specific impulse (either single value or parameterized continuous-time profile), and specialized thruster system models.

Table 2. Comparison of VARITOP and DTOM Results

	VARITOP	DTOM
Saturn and Vicinity		
Launch C_3 (km^2/s^2)	154	152
Launch Mass (kg)	610	628
Power & Propulsion System Dry Mass (kg)	130	130
Propellant Mass (kg)	178	197
Trip Time (yr)	6.0	5.9
Optimal I_{SP} (s)	2610	2496
REP ΔV (km/s)	8.9	9.2
Uranus and Vicinity		
Launch C_3 (km^2/s^2)	146	151
Launch Mass (kg)	686	637
Power & Propulsion System Dry Mass (kg)	137	137
Propellant Mass (kg)	248	203
Trip Time (yr)	10.1	10.1
Optimal I_{SP} (s)	2956	3498
REP ΔV (km/s)	13.0	13.2
Neptune and Vicinity		
Launch C_3 (km^2/s^2)	145	145
Launch Mass (kg)	702	694
Power & Propulsion System Dry Mass (kg)	139	139
Propellant Mass (kg)	262	255
Trip Time (yr)	13.7	13.7
Optimal I_{SP} (s)	3423	3503
REP ΔV (km/s)	15.7	15.7
Pluto/Charon and Vicinity		
Launch C_3 (km^2/s^2)	145	144
Launch Mass (kg)	696	704
Power & Propulsion System Dry Mass (kg)	138	138
Propellant Mass (kg)	257	264
Trip Time (yr)	14.7	14.6
Optimal I_{SP} (s)	3624	3504
REP ΔV (km/s)	16.3	16.2

Bus, Science, and Non-Power/Propulsion margins: 267 kg
Power & Propulsion System Contingency

Several of the previous VARITOP runs from the previous work were compared with runs from the DTOM code. Table 2. compares the results of the two codes. Payloads are targeted to 267 kg for all of the runs. Trip time performance compares favorably as do the trajectories (not shown). Both programs were tasked with optimizing I_{sp}. This parameter accounts for the greatest differences in each program's results. However, these differences are small enough that the use of either program for these early systems trades is adequate.

Due the success of the DTOM code compared to the VARITOP code, as well as the increased ease of use the DTOM code was used for the other targets explored in this work.

New Frontiers Class Missions

The New Frontiers Class (NFC) missions will be funded to a $650 million dollar level and launched every three years. [26] New Frontiers has been described as a Discovery-Plus mission since its funding and allowable launch vehicle are roughly twice that of the Discovery Class. (The Atlas V 551 medium launch vehicle used in this study is one of the assumed NFC launchers.) The science that the NFC missions will address were recently defined by the National Research Council Decadal Study which distilled its solar system exploration requirements into four scientific themes for the next decade.[8]

>The first billion years of solar system history
>Volatiles and Organics: the stuff of life
>The origin and evolution of habitable worlds
>Processes; How planet systems work

A list of list of 12 key scientific questions were generated for these four themes and five strawman missions were consequently suggested to address those questions. The science questions were as follows.

1. What processes marked the initial stages of planet formation?
2. Over what period did the gas giants form, and how did the birth of the ice giants (Uranus, Neptune) differ from that of Jupiter and Saturn?
3. How did the impactor flux decay during the solar system's youth, and in what way(s) did this decline influence the timing of life's emergence on Earth?

4. What is the history of volatile compounds, especially water, across our solar system?
5. What is the nature of organic material in our solar system and how this matter evolved?
6. What global mechanisms affect the evolution of volatiles on planetary bodies?
7. What planetary processes are responsible for generating and sustaining habitable worlds, and where are the habitable zones in our solar system/
8. Does (or did) life exist beyond the Earth?
9. Why have the terrestrial planets differed so dramatically in their evolutions/
10. How do the processes that shape the contemporary character of planetary bodies operate and interact?
11. What does the solar system tell us about the development and evolution of extrasolar plantetary systems, and vice versa?

The strawman missions were

>Kuiper Belt Pluto Mission (KBP)
>Lunar South Pole Aitken Basin Mission (SPA-SR)
>Jupiter Polar Orbiter with Probes (JPOP)
>Venus In Situ Explorer (VISE)

These strawman missions were suggested since they in some way answer the 12 scientific questions. However, it was clearly stated that other missions may be proposed as long as they address some of the scientific questions.

So what solar system objects (appropriate for exploration with REP) can answer these questions?

One of the main classes of targets which can answer several of the aforementioned questions are the many small bodies, whether in high orbits around the gas giants or in solar centered orbits of their own, which scientists hope hold the key to what materials, especially water and organics, were present at the formation of the solar system.[27] These primitive objects are thought to include various outer planetary objects. The following are of list of these objects in order of their progression towards the inner solar system (orbital distance from the Sun):

- Kuiper Belt Objects (including Pluto and Charon) also called Trans-Neptunian Objects [estimated 70,000 objects, >100 km diameter]
- Neptune's moon Triton (and perhaps other moons of the outer Gas Giants) which is thought to be a captured
- Centaur Objects (between Saturn and Uranus) [>100,000 objects, 13 to 25 AU, >40 km diameter]
- Jupiter's Trojan Asteroids [>400 objects]
- Comets

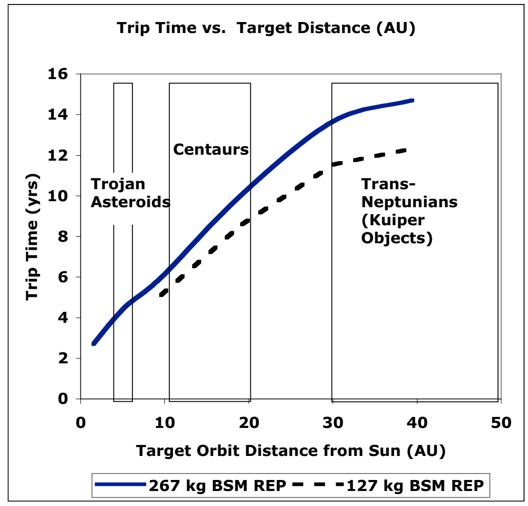

Figure 5. Trip Times for REP Spacecraft to Co-orbit Primitive Bodies

These objects are all thought to start in the furthest reaches of the solar system and progressively work their way into the Sun due to orbit perturbations from the planets such as Neptune. As such these objects should have primitive compounds in various states.

Scientific investigation of such bodies may in some form help to answer all the questions except perhaps 8-10. REP's ability to reach these targets with the payloads previously mentioned is shown in figure 5. As seen in the figure, transit times are reasonable for all of the objects (<15 years). With advances in technology (127 kg BSM spacecraft) even quicker missions are possible. All of these missions use a medium class launch vehicle and upper stage and the REP system, primarily for deceleration and capture (or co-orbiting) at the target. Propulsion alternatives to the REP system for reaching these classes of targets are discussed later.

A sample REP trajectory for reaching Pluto or objects at similar distances is shown in figure 6. This figure shows the main attributes of the REP trajectory – a high C3 escape from Earth and a direct trajectory to the science target. Note that REP is used to provide some additional acceleration at the beginning of the mission while providing all of the deceleration to co-orbit with the target, in this case Pluto. Once at the target, the REP will be able to maneuver to other nearby targets or spiral down to moons of interest in the case of planets.

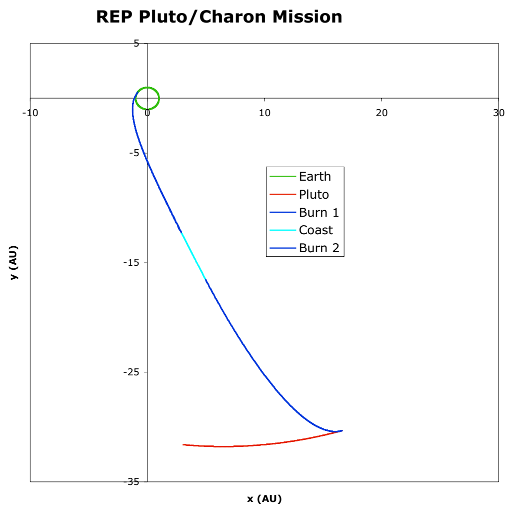

Figure 6. Sample REP Trajectory

Based on past analyses targets appropriate for REP can be generalized for missions with the following aspects:

- Destinations which require power sources other than solar, normally those beyond Mars, which will necessarily have a radioisotope power system on-board
- Science packages whose required power level and mass are not large (<1kW and <100 kg) whether due to the level of the science gathering requirements, the size of the target, the level of specificity of the science or the available mission funding (NFC)

In addition to the multitude of small objects in distant solar orbits are the many moons of Jupiter, Saturn, Uranus, and Neptune. The outer planets' many moons (over 40 in all), as well as the many as well as the solar orbiting objects are shown in figure 7.[21] Scientists believe that some of these moons – including Triton at Neptune – are in fact captured Kuiper objects. Investigations of these moons would also be valuable for answering questions about the planet.

The resulting trip times and payloads found to get to these planets is representative of the times and payloads to the other objects in the vicinity. An additional spiral-in time will be needed to reach the outer planet's moons. This time was estimated using the Edelbaum-Fimple closed form method.[19]

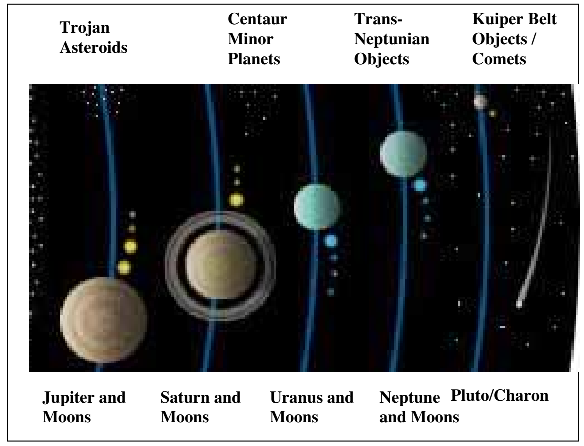

Figure 7. Potential Outer Planetary Targets

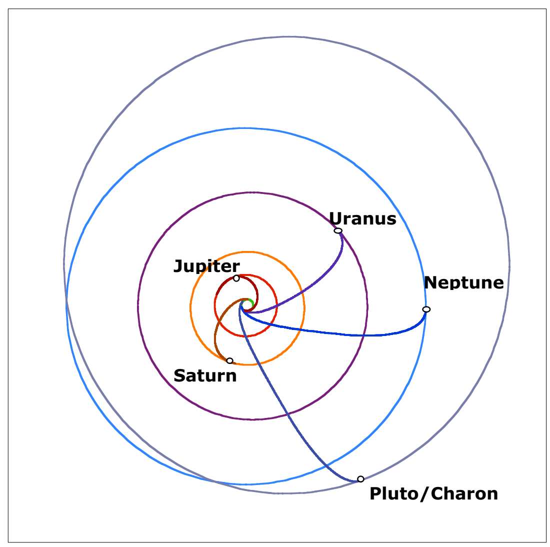

Figure 8. Direct REP Trajectories

A compilation of the trajectories found by DTOM for the various outer planetary distances (noted by Jupiter, Saturn, Uranus, Neptune, and Pluto) are shown in figure 8. Note that the trajectories provide almost straight paths to the target with a circularization at the end. These results closely matched those from VARITOP. It should be noted that no third body effects are used by VARITOP or DTOM to determine these results. Thus the mass of the planet has no impact of the trajectory and no flybys with gravity assists are used. This greatly simplifies the trajectory and reduces launch window constraints.

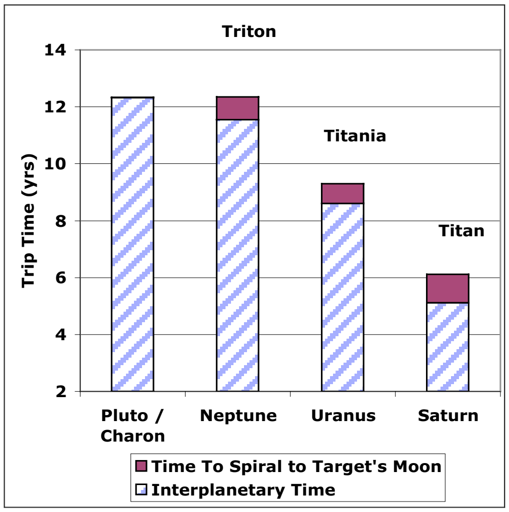

Figure 9. Orbiter Trip Time vs. Object [127 kg BSM]

Initial results are shown in figure 9 for the lightest spacecraft bus, science, and margin (BSM) currently conceived. For the BSM of 127 kg the trip times to the outer planetary targets are surprisingly quick with Pluto distance targets being close to 12 years from launch. Since the moons of the outer planets are also of great scientific interest an estimate of the time to spiral down from the high capture orbit (somewhere below the sphere of influence) was made for sample moons of the outer planets: Titan (Saturn), Titania (Uranus), and Triton (Neptune). Results (also shown in figure 9) showed that the trip times were on the order of a year for all but Charon which was less than a month. This is due to the very low mass of the Pluto/Charon system. Spiral times for the heavier 267 kg BSM spacecraft are commensurately longer.

The required propellant throughputs and optimal I_{sp}s were also found in each analysis. These parameters are key to guiding the propulsion technology development. Table 3. shows these data, respectively. It is clear that further targets require more throughput per thruster or more engines. The baseline included eight engines with three power processors (two engine operation). For most of the mission cases the engine throughput is around 30 kg /engine. In the case of engine out (only 7 engines available) around 35 kg throughput on each engine would be required for the heavier BSM masses. This equates to required burn times of three to four years for each engine. The GRC developed NSTAR 30 cm thruster, with which the 8-cm ion thruster draws heritage, has currently been tested for more than three years in a ground-based life test. The optimal, single set-point I_{sp}s were determined by DTOM to be in the 2600 sec to 3700 sec range which is commensurate with the 8 cm ion engines current design. A summary of the 750 W REP cases is shown in Table 3.

Table 3. REP Mission Summary

Jupiter and Vicinity	
Launch C_3 (km^2/s^2)	158
Launch Mass (kg)	575
Power & Propulsion System	
Dry Mass (kg)	130
Propellant Mass (kg)	147
Trip Time (yr)	4.5
Optimal I_{SP} (s)	2589
REP ΔV (km/s)	7.5
Saturn and Vicinity	
Launch C_3 (km^2/s^2)	152
Launch Mass (kg)	628
Power & Propulsion System	
Dry Mass (kg)	130
Propellant Mass (kg)	197
Trip Time (yr)	5.9
Optimal I_{SP} (s)	2496
REP ΔV (km/s)	9.2
Uranus and Vicinity	
Launch C_3 (km^2/s^2)	151
Launch Mass (kg)	637
Power & Propulsion System	
Dry Mass (kg)	137
Propellant Mass (kg)	203
Trip Time (yr)	10.1
Optimal I_{SP} (s)	3498
REP ΔV (km/s)	13.2
Neptune and Vicinity	
Launch C_3 (km^2/s^2)	145
Launch Mass (kg)	694
Power & Propulsion System	
Dry Mass (kg)	139
Propellant Mass (kg)	255
Trip Time (yr)	13.7
Optimal I_{SP} (s)	3503
REP ΔV (km/s)	15.7
Pluto/Charon and Vicinity	
Launch C_3 (km^2/s^2)	144
Launch Mass (kg)	704
Power & Propulsion System	
Dry Mass (kg)	138
Propellant Mass (kg)	264
Trip Time (yr)	14.6
Optimal I_{SP} (s)	3504
REP ΔV (km/s)	16.2

Bus, Science, and Non-Power/Propulsion margins: 267 kg
Power & Propulsion System Contingency

(30% of 750 W Power & Propulsion System): 34 kg

Other Options to Outer Planetary Targets

The REP outer planetary orbiter missions showed relatively fast transit times for small payloads. Other technologies can also reach the other planets and will now be compared to determine REP's role in outer planetary exploration.

Using only state-of-art chemical systems to capture at Jupiter, Saturn, Uranus and Neptune would require the largest of planned launch vehicles (e.g. Delta IV Heavy) and/or planetary flybys and equivalent trip times for each orbiter. Adding aerocapture systems can improve the delivered payload but requires technology development and imposes risk for the first mission to the planet. For Pluto/Charon or any of the other small objects (Trojans, Centaurs, Trans-Neputnian Objects, Kuiper Objects, and various asteroids and comets) chemical capture requires much longer trip times at best and aerocapture is not possible.

Combining aerocapture technologies with a solar electric propulsion (SEP) stage has shown better results. This concept uses an SEP stage and a Venus flyby to send a payload quickly to an outer planet where an aerocapture system captures into orbit about one of the large outer planets (Jupiter, Saturn, Uranus, or Neptune). The SEP system is separated before arriving at the target planet. Once captured in orbit small chemical maneuvers and time can allow transfers to a planet's moons with the appropriate planet/moon gravity flybys. The SEP / aerocapture propulsion system can deliver respectable payload spacecraft ~500 kg to these planets using medium launch vehicles and trip times similar to the REP system. Aerocapture at Pluto or the other above mentioned objects is not viable so the SEP/Aerocapture method is not available.

The other approach currently of interest is nuclear electric propulsion (NEP). The NEP system carries a reactor with powers of 100 to 500 kW. Since the spacecraft is fairly large (>8000 kg) the NEP vehicle must be launched with a heavy launch vehicle to a low earth orbit and spiral out. The NEP vehicle then accelerates out to and decelerates into an outer object. Quick spiraling at the target is then possible. Payloads from 500 kg and up are possible with power available at the target of >100 kW. Trip times are similar to the REP system. The main difference is the size of the vehicle, payload and power level. The NEP system is more appropriate for flagship type missions with the REP being a cheaper solution for the emerging New Frontiers Class missions (similar to the Discovery Class). Thus the REP is perhaps more appropriate for smaller targets with more focused science.

Demonstration Missions

The combination low power, long-life electric propulsion, light, high-efficient radioisotope power systems and light spacecraft components may require a demonstration mission to prove the readiness of these technologies for New Frontiers Class proposers.

Such demonstration missions should exercise the technology's capabilities but not embody the relative long trip times (>5 years) which will be required for actual missions. In addition, some valuable science portion of the mission is thought to be important based on previous demonstrator missions such as Deep Space One.

One possible demonstrator target which would not require a long trip time but would provide valuable science returns would be a science target at Mars. Two good candidates are Mars' Asteroid Eureka or Mars' high orbiting moon Deimos. Eureka is located at the Sun/Mars lagrange point similar to Jupiter's Trojans. Data on the moon Deimos is sparse, most coming from very distant investigation by the Viking missions.[7] In fact, whether Demios was formed with Mars or captured later has not been answered. A verification (and comparison) of Eureka and Deimos's composition, possible with an REP suite of instruments, may help answer the question. Perhaps a single REP spacecraft could reach both targets in succession.

	Comb. Solar Array/Radioisotope Power	2.5 kW
Interplanetary Trajectory	Launch Date	October 8, 2011
	Arrival Date at Mars Orbit (Eureka)	October 26, 2013
	Trip time (years)	2.05
	Trip time (days)	748
	Payload (kg)	267
	Launch Mass (kg)	500
	Final Mass (kg)	437
	Propellant Mass (kg)	62
	Launch Vehicle	Delta II 7925
	C_3 (km^2/s^2)	47
	Burn 1 length (days)	374
	Burn 2 length (days)	374
	Coast Arc (degrees)	5.73E-05
	Initial Power @ 1 AU (kW)	2.5
	Isp (s)	4000
	Duty Cycle	0.9
	Tankage Fraction	0.1
Spiral	Spiral Time (days)	211
	Additional Propellant Consumed (kg)	13
Totals	Trip Time (years)	2.63
	Trip Time (days)	959
	Final Mass (kg)	424

Figure 10. Potential REP Demonstrator Summary

From a spacecraft and trajectory standpoint even Deimos is relatively easy to reach due to Mars' nearness to Earth and the moon's very high orbit. Such investigation by a chemical system, which by necessity would capture with a low perigee would be difficult. Preliminary performance of an REP demonstrator spacecraft to Eureka and Deimos, using a smaller Delta 7925 is shown in figure 10 (transit time from Eureka to Mars and science times are not included). In this case the demonstrator spacecraft was assumed to have only one advanced radioisotope system and two 8 cm thrusters; a 2.5 kW solar array and NSTAR engine would provide the rest of the needed thrust. This approach will reduce costs and ensure success. Work is underway to better define the actual mass of the demonstrator spacecraft. The trajectory is shown in figure 11. It is not a direct trajectory due to Mars' proximity to Earth; there is insufficient time and distance for a low thrust REP spacecraft to decelerate sufficiently. The use of the REP system for Mars is consequently only for a quick demonstrator; Mars is too close both in terms of the vehicle's ability to slow down with low thrust levels and to compete with solar powered thrusters.

Other potential demonstrator targets exist and need to be evaluated but the Eureka/Deimos example shows targets similar to those in the outer solar system but can be reached relatively quickly in order to demonstrate the important technologies and return valuable science.

Further Work

The analyses performed so far show great promise for the use of REP for New Frontiers Class missions. Future analysis work will concentrate on two areas: spacecraft point designs and refining the required technology challenges. Spacecraft point designs will be made to obtain a better idea of the potential mass all the subsystems as well as the impact of other launch systems.

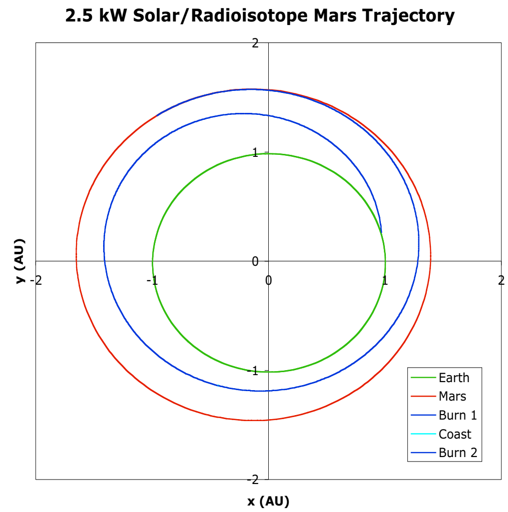

Figure 11. REP Demonstrator Trajectory

Conclusions

Studies were undertaken to focus on what a radioisotope electric propulsion system could do for New Frontiers Class Missions. It was found that for scientific targets past Mars, especially small bodies such as Jupiter's Trojans, various Trans-Neptunian Objects and Pluto and other Kuiper-belt objects, REP could orbit these targets in reasonable trip times (4.5 to 15 years). On-going work in small ion thrusters, advanced radioisotope power systems such as the Stirling convertor, and small planetary science spacecraft point toward the possibility of a viable REP spacecraft for outer planetary exploration. The new direct trajectory previously found was verified with the DTOM code.

Acknowledgments

The authors wish to thank Dr. Robert Noble for his previous work on and enthusiasm for the radioisotope electric propulsion concept.

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