Outer Planet Exploration with Advanced Radioisotope Electric Propulsion

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Abstract

In response to a request by the NASA Deep Space Exploration Technology Program, NASA Glenn Research center conducted a study to identify advanced technology options to perform a Pluto/Kuiper mission without depending on a 2004 Jupiter Gravity Assist, but still arriving before 2020. A concept using a direct trajectory with small, sub-kilowatt ion thrusters and Stirling radioisotope power systems was shown to allow the same or smaller launch vehicle class as the chemical 2004 baseline and allow a launch slip and still flyby in the 2014 to 2020 timeframe. With this promising result the study was expanded to use a radioisotope power source for small electrically propelled orbiter spacecraft for outer planet targets such as Uranus, Neptune, and Pluto.

Introduction

Outer planet exploration is experiencing new interest with the open competition for a Pluto flyby mission. Voyager 2 conducted flybys of all the outer planets from Jupiter outward, except for Pluto, giving us a short glimpse of these mysterious planets and their many moons. At the request of the NASA Deep Space Exploration Technology Program, an examination of advanced power and propulsion technologies to allow a post 2004 launch of a fast Pluto flyby (missing the 2004 launched Jupiter gravity assist opportunity) was undertaken at Glenn Research Center (GRC). It was found that with the use of small, advanced 8 cm ion thrusters and Stirling radioisotope power systems, both under development at GRC, it was possible to launch the Pluto/Kuiper mission as late as 2012.[1] With the promising results of this analysis, a look at other outer planet missions using this concept was undertaken, specifically, orbiting science spacecraft for Uranus, Neptune, and Pluto.[2]

In several past works, Robert Noble of Fermi labs has noted the potential advantages of using radioisotope-powered ion propulsion for outer planet exploration.[3,4,5] Advantages of radioisotope electric propulsion (REP) include a long-life

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power source, not reliant on the sun, which provides propulsion power to reach the target and then provides relatively higher power levels for science payloads (since more power is needed for the ion propulsion system as opposed to past all chemical RTG spacecraft). REP also provides a propulsion system which uses much less fuel than chemical systems and therefore allows the use of smaller launch vehicles. The primary disadvantage to the REP system is its limited propulsion power, (hundreds of watts); which limits the reasonable payload spacecraft size (without power or propulsion) to around 100 kg for REP missions of reasonable duration. If larger payloads are required a nuclear reactor powered system would be needed.

While the past studies noted the advantages of the combining radioisotope and ion propulsion technologies, the technologies to provide a light weight power and propulsion system did not exist. Specific masses of 100 to 150 kg/kW are needed to provide reasonable mission times and performance. 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 more plutonium bricks due to the low efficiency of the thermoelectric conversion system. Use of 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. Long life, low power ion propulsion 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 science package, but not including power and propulsion, of around 100 kg. The Johns Hopkins University Applied Physics Laboratory (JHU/APL) has built or is building several interplanetary spacecraft of this class including NEAR, Contour and Messenger.

Both the technologies needed for an REP spacecraft and the potential mission opportunities for such a spacecraft are explored in this work.

REP Technologies

The three key technologies needed for an REP spacecraft are small, advanced ion thrusters, lightweight radioisotope power systems, and small spacecraft which can perform valuable science. This study assumed ion thrusters with an operational power range of 100-500 W, Stirling radioisotope power systems that can supply constant power of 100-500 W to the ion propulsion system and lightweight spacecraft bus technologies that enable revolutionary 100-200 kg spacecraft bus designs. Each will be discussed in turn.

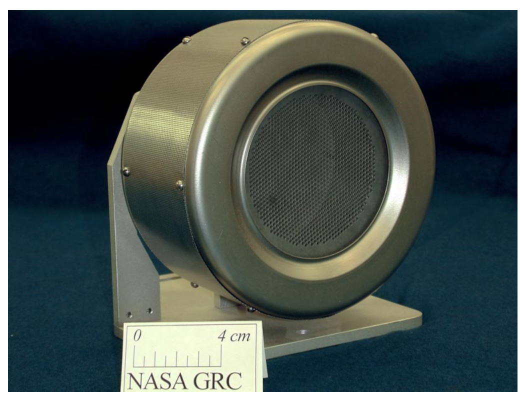

Figure 1. NASA 8 cm Ion Thruster

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 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,[6-9] and the fabrication of a second-generation lightweight engineering model thruster with a 100-500 W power throttling envelope has also been completed. Also at NASA GRC, first- and second-generation breadboard power processors have been fabricated and successfully integrated with the 8 cm thruster.[10-12]

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 convertors were used to produce the required six electrical outputs which resulted in large 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.[13] 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 Isp
- thruster mass: 0.95 kg
- PPU mass: 2.0 kg
- Xenon Feed System mass: 3.1 kg (excluding tank)

Stirling Radioisotope Power System

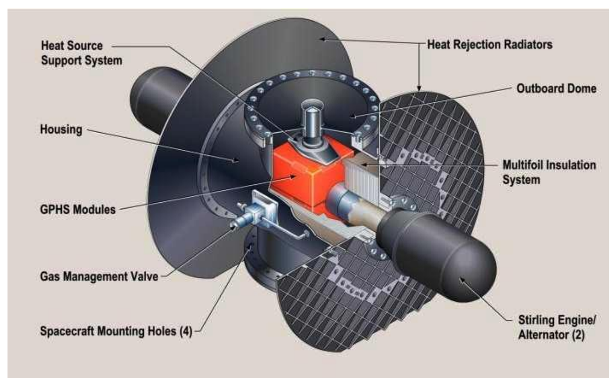

Figure 3. Stirling Radioisotope Power Concept

An advanced radioisotope electric power generator is being developed for use on deep space missions, as well as for Mars surface applications. A concept is shown in figure 3.It is based on the high efficiency free-piston Stirling power convertor. The Department of Energy (DOE) has responsibility for developing the SRPS. GRC is supporting the DOE in this effort, drawing on the its many years of experience in developing Stirling power conversion technology. The SRPS is a high-efficiency alternative to the Radioisotope

Thermoelectric Generators (RTGs) that have been used on many previous missions. The Stirling efficiency, well in excess of 20%, leads to a factor of 3 to 4 reduction in the inventory of plutonium required to heat the generator. The power system will be comprised of one or more generators, based on the power required for the mission.

The SRPS will be based on a free-piston Stirling power convertor (Stirling engine coupled to a linear alternator) 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 4.) A joint government/industry committee developed and agreed upon a set of criteria used to determine the readiness of the Stirling technology for transition to flight.[14] Having passed these tests, the TDC is now being transitioned from a laboratory device to flight application. As a part of this process, DOE has conducted a competitive procurement for a System Integration Contractor to design, develop, qualify and supply SRPS units to NASA for the future missions. Selection of a System Integration Contractor should be announced in late 2001. The present system integration schedule would complete the design and development of the SRPS and be able to provide flight qualified generators to support missions as soon as 2007.

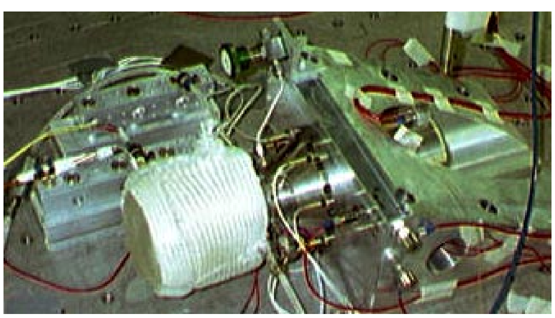

Figure 4. Stirling Technology Demonstrator Convertor

The SRPS will be heated by plutonium housed inside of two General Purpose Heat Source modules. Each module will provide approximately 250 W_{th} at beginning of mission (BOM). The initial SRPS, based on the laboratory TDC transitioned to flight, will be able to offer mass savings and increased specific power compared to the RTG. Analysis performed at GRC projects each generator having a mass of 25 kg, power output of 112 Wdc with specific power of 4.2 W/kg at BOM.[15] With engineering development, but without the need for basic technology development, a future generation of the generator could offer improvements to 20 kg mass, power output of 120 Wdc, and specific power of 7.8 W/kg at BOM. A more advanced version that would require technology development that makes use of high temperature refractories to increase the temperature ratio and is projected to achieve a 20 kg generator, power output of 170 Wdc, and specific power of 8.6 W/kg at BOM.

One of the benefits of this system is the elimination of degradation in efficiency of the power conversion unit. The plutonium heat source is based on the standard General Purpose Heat Source (GPHS) modules. The heat generated by the decay of plutonium, with an 88 year half life. A small RTG (one half of the previously used RTG) would use nine GPHS modules and produce 139 Wdc at BOM. The combined effect of decay of the radioisotope heat source and degradation in the conversion efficiency of the thermoelectric unicouples would lead to 119 Wdc after 6 years (86% of BOM) and 99 Wdc after 14 years (71% of BOM). The conversion efficiency of the Stirling convertor should generally remain unchanged and this results in the power supplied by each generator being reduced over time at roughly the same rate as the decay of the heat source. Based on the GRC study, the first generation SRPS with two GPHS modules is anticipated to produce 112 Wdc at BOM. The SRPS would then produce 107 Wdc after 6 years (96% of BOM) and 100 Wdc after 14 years (89% of BOM).

Long life with no degradation has been accomplished with the use flexure supports for the moving components to virtually eliminate contact between the moving components. The present design of the Stirling convertor for the SRPS has been designed for a 100,000 hour (11.4 year) life. However there is margin in the design that allows it to operate beyond this point. Three components are critical to achieving long life; the flexures, 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 machines in the cryocooler application. Long life Stirling cryocoolers are presently flying on spacecraft. The flexures are designed and qualified for the design life, and are then operated at significantly derated conditions to essentially achieve infinite life. For the SRPS, 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 a minor loss in conversion efficiency. These issues are presently being addressed with analysis and tests at GRC.[16] A free-piston Stirling convertor has been operated for over 66,000 hours (7.5 years) to demonstrate the life and lack of degradation.[16]

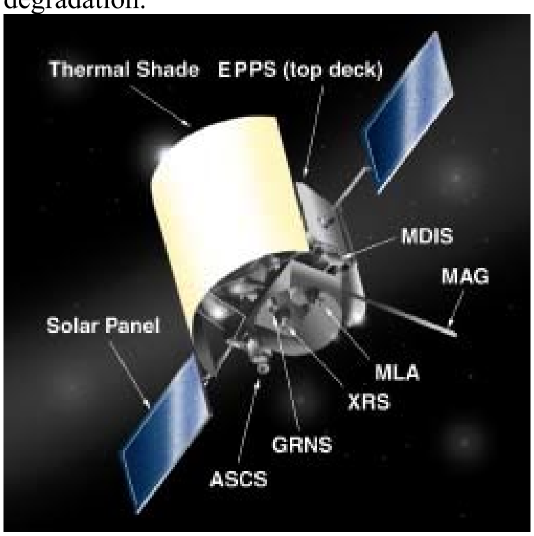

Figure 5. Messenger Spacecraft

Lightweight Spacecraft Bus and Instruments Technologies

Advanced microelectronics/lightweight spacecraft bus development has been underway at the JHU/APL and will be leveraged toward the outer planet mission opportunities.

Two strategies that help reduce the science instrument package mass are instrument integration and spacecraft-directed instrument pointing. A notional integrated science package for an orbiter could consist of a camera, spectrometer, and laser altimeter constrained to 10 kg mass and 25 W power. This assumes that the spacecraft points the camera and altimeter without use of scanning optics. NEAR, which orbited and landed -on asteroid 433 Eros, carried a camera, spectrometer, and laser altimeter that did not use scanning optics. The 5 kg altimeter used an average of 16 W. Another notional science package might consist of particle, plasma, and magnetic field sensors massing at 5 kg and requiring 6 W.

Based upon JHU/APL's design experience with both the MESSENGER mission (figure 5) to orbit Mercury and the CONTOUR mission to multiple comets, 20 kg will provide sufficient mass for a capable science package. For example, a flyby mission (planet or comet) could contain an imaging system, an ultraviolet/visible/infrared spectrometer, a magnetometer on a boom, and either an energetic particle spectrometer or a plasma spectrometer, for a combined mass of 10.5 kg. An orbiter could have all the flyby instruments plus a laser altimeter, X-ray spectrometer, neutron spectrometer, and the plasma spectrometer or energetic particle spectrometer left off the flyby design – all for a mass of 22.5 kg. Avoiding use of scanning or gimbaled instruments will reduce the mass of these notional instrument packages. For missions launching more than five years from now, further mass reduction will likely allow the use of scanning systems. The primary electric power source will supply both the science and housekeeping functions.

Integration is important to realizing a 100-120 kg spacecraft that will carry the science package into deep space. The primary integration goals are to eliminate boxes and combine functionality where possible. This mass includes mechanical, communication, control and data handling, and guidance and control systems, but not the power and propulsion systems.

The mechanics of such a spacecraft will require multi-functional systems and structures. For example, propulsion and electronics may be more than bolt-on packages. They might be integrated into the structure of the spacecraft to realize a bus that is about 50% of the mass (excluding power and propulsion systems). Other mass/volume reduction technologies include the Micro Tool Kit and inflatable structures. These components are based upon shape-memory materials. The tool kit consists of actuators and release mechanisms with masses from 0.07 to 5 grams. Inflatable technologies also allow mass and volume savings by making lightweight, compact booms that deploy to meters in length. Savings may also be realized in the RF communication system by using an inflatable dish antenna.

RF communications in deep space have relied upon a transponder architecture that leaves little room for savings in mass and power. So, JHU/APL has developed a non-coherent RF transceiver architecture that allows significant reductions in mass and power. CONTOUR will be the first deep space flight to use this architecture, but the real savings depend upon advancing the underlying technology. JHU/APL has developed a 22% lower mass breadboard X-band receiver through the NASA ATD (Advanced Technology Development) program. This receiver has also significantly cut the power drawn from 21 W for a two-transponder system to 4 W. Further advances in this technology in the next two years can reduce the power further to less than one watt. Transmitter power can vary from 35 W DC while radiating 5-7 W of RF to 60 W DC while radiating 5-30 W.

APL has reduced the electronics mass by packing as much function into as few chips as possible, and packaging those few chips as compactly as possible. Spacecraft electronics are often realized in many subsystems, each housed in a separate chassis. This approach further increases mass because of the harness requirement to connect all the boxes. JHU/APL's architecture is the Integrated Electronics Module (IEM) wherein each subsystem resides on a single card, or slice. Each slice is four inches square, 1 cm thick, and weighs about 105 g. All the slices are then packaged into one module and they communicate using an IEEE 1394 high-speed, low-power bus. The design is modular to allow any number of slices in the module. Using this integrated approach, and through NASA/GSFC sponsorship, JHU/APL has developed a small, radiation-hard Command and Data Handling module that weighs only 0.5 kg, and operates on 3.3 V electronics. By contrast, the recent NEAR C&DH (command and data handling) unit was 5 kg and used 5 V electronics.

Table 1. Concept Science Spacecraft Mass/Power Breakdown

System	Mass	Power
Bus & Mechanical	60 kg	---
Electronics, Processors, & RF	10 kg	40-50 W
Star Tracker	3 kg	35 W
Reaction Wheels	6 kg	16 W
Flyby Science, or	10.5 kg	21 W
Orbiter Science	22.5 kg	60 W
Flyby Total (less power and propulsion)	89.5 kg	112-122 W
Flyby & Orbiter Total (less power and propulsion)	112 kg	172-182 W

The IEM approach can be used to house the instrument, C&DH, and guidance and control (G&C) processors. This allows the use of a single processor for more than one function. APL already uses its current instrument processor as a C&DH system, executing many C&DH functions like command execution, macro execution, macro storage, telemetry gathering, and CCSDS protocol. The IEM architecture already includes RF slices, so adding uplink, downlink, and mass storage slices is not difficult.

There has been little advance in star trackers beyond the 3 kg, 10 W system used by NEAR. Also, the NEAR Inertial Measurement Units (IMU) weighed 7 kg and required 25 W. Further improvements in IMU mass and power are possible. Reaction wheels are required for fine pointing when scanning optics are not used with the instruments. Honeywell has developed a small reaction wheel that weighs about 1.5 kg and requires 3-4 W. By contrast, NEAR used 3 kg wheels that consumed 7-9 W. The Honeywell technology represents a savings of 6 kg and 20 W for a redundant, three-axis system.

These technologies, and integration make a 100-120 kg class craft for deep space possible (table 1). This mass includes 60 kg for the bus and mechanical components, 20 kg for the IEM and RF communications, 10 kg for the star tracker and IMU, and 6 kg for reaction wheels. The IEM contains the up/down link electronics and the processors for the instruments, C&DH, and G&C. The mass does not include the 20 kg science package, the power, nor the propulsion systems. This accounting also shows that the100 W available for housekeeping during powered flight is adequate. In this notional design, the IEM and RF communications will require 40-50 W while the G&C sensors and reaction wheels take another 37 W.

Systems Analyses

For the sample outer planet missions, the previous technology descriptions were modeled for mass and performance analyses. In the case of the Pluto Flyby mission 'technology windows' of '06, '09, and '12 were used. The assumed performance of the each of the subsystems is shown in table 2.

Table 2. Pluto Flyby Technology Assumptions

Outer Planet Exploration Subsystem Options	Pluto Flyby 2007 Launch	Pluto Flyby 2009 Launch	Pluto Flyby 2012 Launch
	Mass/ Power	Mass/ Power	Mass/ Power
Complete SRPS System (each)	20 kg / 124 W	20 kg / 172 W	18 kg / 172 W
8 cm Ion Propulsion System	7.0 kg	7.0 kg	7.0 kg
Thruster (each)	1.3 kg	1.3 kg	1.3 kg
PPUs (each)	2.1 kg	2.1 kg	2.1 kg
Feed Sys	3.1 kg	3.1 kg	3.1 kg
Cable (per thruster)	0.2 kg	0.2 kg	0.2 kg
Thermal	0.4 kg	0.4 kg	0.4 kg
Tankage	10%	10%	10%
Xenon Fuel Throughput / Thruster	12 kg	20 kg	30 kg
Ion Thruster Isp (sec)	2800 s	2800 s	3300 s
Ion Propulsion System Efficiency	49%	49%	56%

For the ion thruster system, improvement in fuel throughput (lifetime) was assumed using advanced grid technologies including thick molybedenum, titantium, or carbon based technologies. Efficiency and specific impulse were improved for the '12 thruster technology by assuming potential propellantless cathode technology or higher voltage operation. Masses for the thruster and components include gimbal and structure masses. A spare PPU and thruster were assumed in each case.

The Stirling system technology for the '06 opportunity, is based upon nickel-based super alloys and temperatures of 925K. Advances for the '09 Stirling system consist of raising operating temperatures to 1400 K using refractory metals. The '12 opportunity seeks to reduce the mass of the '09 refractory metal system.

Table 3. Outer Planet Orbiter Assumptions

Outer Planet Exploration Subsystem Options	Outer Planets Orbiter
	Mass/Power
Complete SRPS System (each)	18kg / 172W
8 cm Ion Propulsion System	7.0 kg
Thruster (each)	1.3 kg
PPUs (each)	2.1 kg
Feed Sys	3.1 kg
Cable (per thruster)	0.2 kg
Thermal	0.4 kg
Tankage	10%
Net Spacecraft (Launch Mass less Science, Power, Wet Propulsion)	100 kg / 60W
Science	20 kg
Fuel Throughput / Thruster	50 kg xenon
Ion Thruster Isp (sec)	3300 s
Ion Propulsion System Efficiency	56%

Shown in Table 3 is the system breakdown for the outer planet orbiters. It assumed the '12 launch parameters plus an improved throughput of 50 kg of fuel per engine to handle the large fuel loadings. 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.

Mission Analyses

In order to assess REP's viability for outer planet missions several tools were used. Complex, higher order codes such as VARITOP were used to assess actual trajectories. A simple closed form relationship developed by Zola was used to explore the system and mission trade space more easily.[17] This method assumes an equivalent path length which the REP system must fly with a constant thrust/ coast/constant thrust trajectory. Since the payload mass is know one can estimate the launch mass to escape based on available launch vehicles. The closed form relationship from Zola can then be rewritten to determine the trip time given the ΔV we can supply (based on the rocket equation):

$$T = ((2a_oL-2v_j^2(1-e^{-\Delta V/2vj})^2)/ \Delta V +v_j(1-e^{-\Delta V/vj}))/ a_o$$

Where T is the trip time (including any coasts), a_o is the thrust / initial mass, v_j is the thrust / mass flow rate, and L is the equivalent path length.

All the mission analyses include a comparison with state-of-art chemical systems. Launch vehicles for all the missions assumed existing or planned launch vehicles.[18]

Pluto Flyby with REP

In an effort to show how advancing technology can improve Pluto-flyby missions, technology "Launch Windows" were assumed using representative launches in '06,'09,'12 corresponding to available technology. Arrival date was set at 2020 or earlier. A range of existing and projected expendable launch vehicles was considered. Projections of the 8cm ion propulsion and Stirling convertor programs, underway at the NASA GRC, were made to create the ’06,’09, and ’12 baselines. Since this phase of the study was previous to the APL 100 kg class spacecraft study, the trajectories were designed which provided net spacecraft masses (spacecraft less propulsion system) of 150 to 400 kg depending upon launch vehicle and launch date.

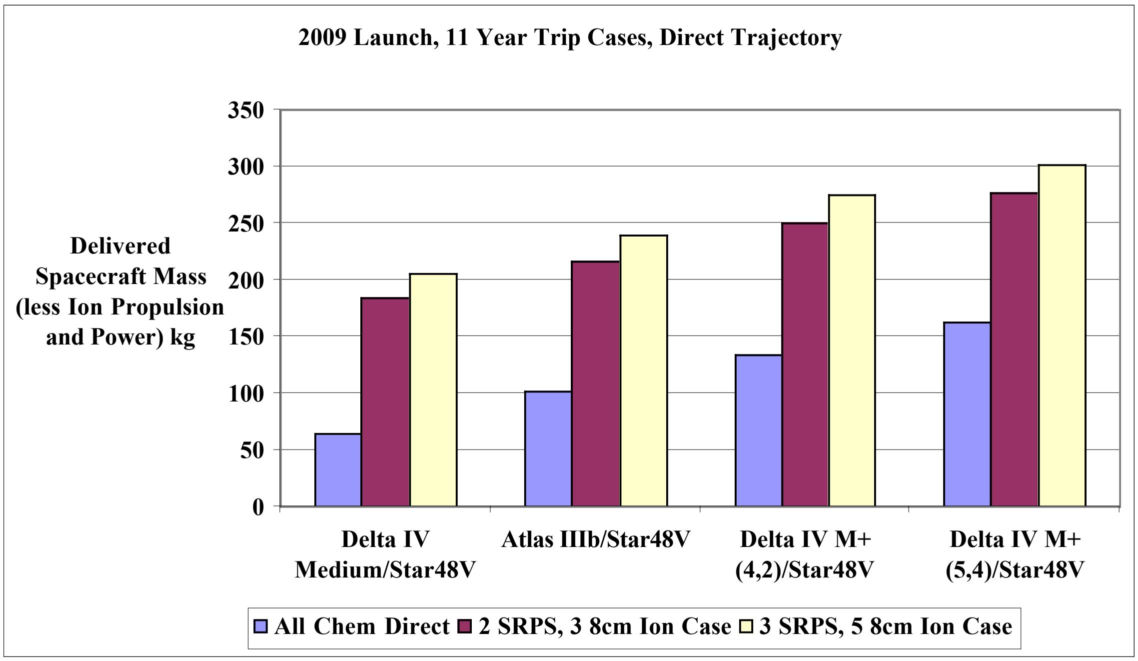

Figure 6. Pluto Flyby Net Mass vs Launcher

The optimal trajectory for using REP for a Pluto flyby consisted of REP constant thrusting starting from the high excess velocity escape condition of the launch vehicle out to about the orbit of Uranus. At this point the REP system is shut down and only used for small correction maneuvers if needed.

Various launch vehicles were used to show their impact on the REP Pluto flyby mission (see Figure 6.) The mission timeline includes a 2009 launch date with a 2020 flyby. It is clear from the figure that the use of REP can at least double the performance of the all-chemical option.

The study varied several parameters to answer specific questions. The first question was 'can one wait for better technology and still arrive on the same date?' Figure 7 shows the variation in spacecraft mass versus trip time. Three curves show the 2006, 2009, and 2012 technology available at launch. For a given spacecraft mass, the earliest launch date provides the earliest arrival date, although at maximum trip time. In other words, the 2006 technology, although less capable gives the best performance since the available trip time is 3-6 years longer than the later launched technologies. However, the 2012 launch date can still provide a 150 kg science spacecraft by 2020.

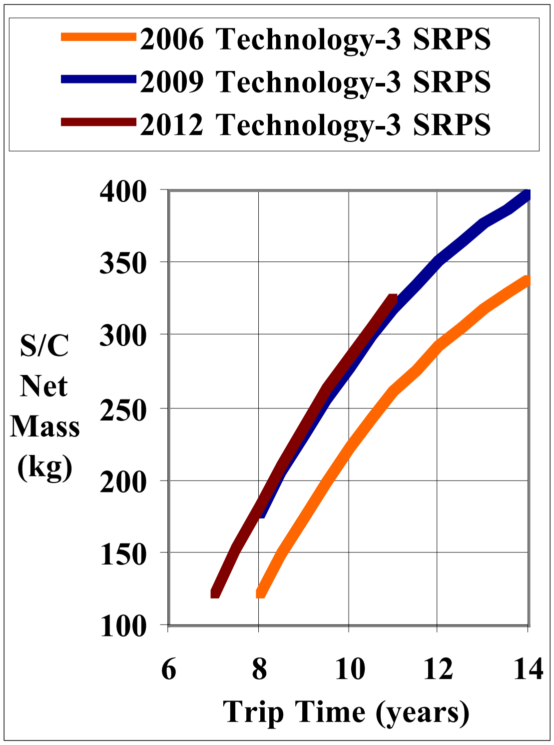

Figure 7. Pluto Flyby Net Mass vs. Technology Launch Window

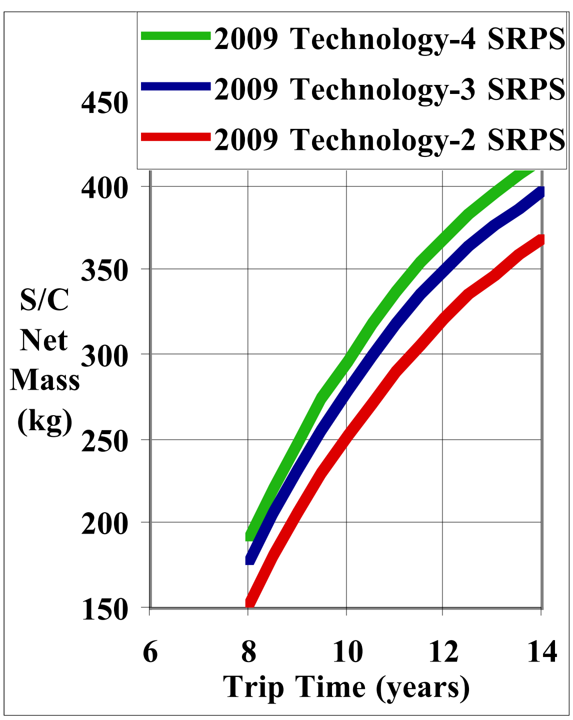

Figure 8. Pluto Flyby Net Mass vs. Number of SRPS

The other question was how power level, and thus thrust, can provide more payload. Figure 8 compares two, three and four SRPS module options for the 2009 launch. It is evident that adding a module can provide up to 30 kg additional delivered science spacecraft mass for a given trip time or can reduce the trip time for the same delivered mass.

In order to show the relative contribution of the electric propulsion system and the Stirling radioisotope power system each was run separately for the Pluto flyby mission. For the electric propulsion only case, off-the-shelf RTGs were used, while for the Stirling radioisotope power system only case the flyby velocity was supplied chemically. This particular case assumed a Delta IV Medium launch vehicle with a Star 48V upper stage. The mission assumed a 2009 launch and a 2020 flyby. Figure 9 and Table 4 shows the relative performance.

Table 4. Pluto Flyby Parameters vs. Technology

Impact of Technology for Pluto Flyby	SOA	Adv EP Only	Adv Stirling Only	Adv EP and Stirling
Power (BOM)	290 W	474 W	250 W	474 W
Power (Flyby)	230 W	376 W	230 W	435 W
EP Propellant		84 kg		84 kg
Power mass	56 kg	92 kg	29 kg	60 kg
Propulsion mass		29 kg		29 kg
Net S/C Mass	121 kg	212 kg	148 kg	243 kg
# of Pu Modules	27	43	6	10

It is clear that the large payload gain comes mostly from the electric propulsion system. However, using SOA RTG technology with electric propulsion will require twice the RTGs of the chemical case, (almost twice as many plutonium bricks, 43 versus 27). The Stirling system greatly reduces the required number of plutonium bricks to only 6 for the Stirling only case and only 10 for the combined electric propulsion and Stirling case. Thus the addition of the highly efficient Stirling system could greatly reduce the cost of the plutonium fuel for an equivalently powered spacecraft.

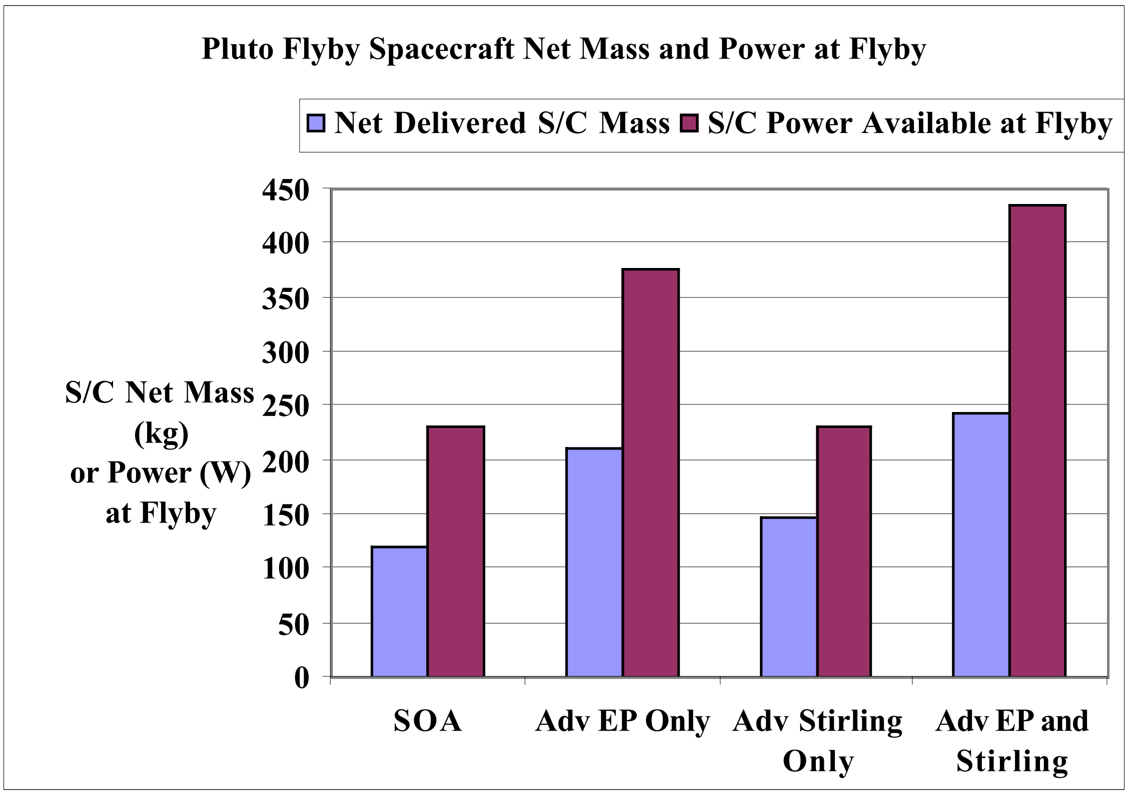

Figure 9. Pluto Flyby Parameters vs. Technology

Outer Planet REP Orbiters

The Pluto Flyby mission showed the advantage of REP for outer planet missions: eliminating the need for gravity assists (and constrained launch windows) and significantly increasing payload mass. With such capability could REP be used as the primary propulsion for outer planetary *orbiters*?

Since both Jupiter and Saturn have had or will soon have orbiting spacecraft, focus for the REP orbiter was set on the furthest outer planets: Uranus, Neptune, and Pluto and their moons. Using state-of-art chemical systems to capture at Uranus and Neptune would require the largest planned launch vehicles (Delta IV M+ or larger) for each orbiter and is not even possible for Pluto. While aerocapture technologies would reduce the required chemical capture stage, their use in an unknown planet's upper atmosphere is deemed to be a risky maneuver at best. Aerocapture at Pluto is not viable. The use of REP to provide the complete interplanetary and near planetary maneuvers would remove this risk and may allow the use of smaller launch vehicles. A top-level look at flying three outer planetary REP orbiters was made to determine the relative flight times required as well as the launch requirements.

Using the method from Zola, REP orbiter mission parameters for Uranus, Neptune and Pluto were determined. Representative equivalent path lengths (L) for rendezvous with the planet were assumed from Zola for Uranus, Neptune, and Pluto as 2.5E12 m, 4.2E12 m, and 5.5E12m, respectively. [17] .Each orbiter mission assumed launch to escape, an outbound acceleration burn, a coast and a final capture burn. Each mission also included a spiral into the target planet after capture. Assuming the REP system from table 3, figure 10 was generated to

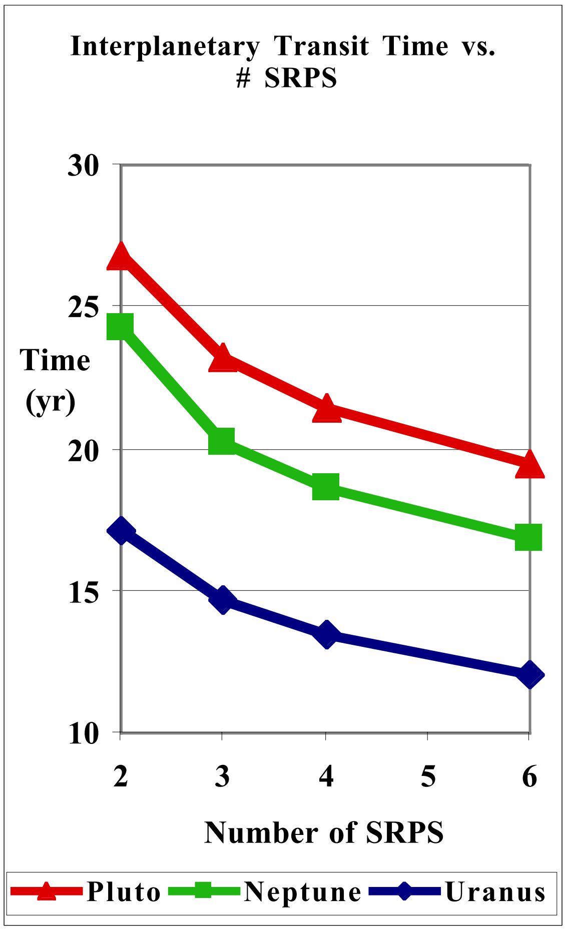

Figure 10. Interplanetary Transit Time vs. Number of SRPS

show the impact of the number of SRPS systems (power) on interplanetary trip time. Ion propulsion systems were added as needed to utilize the power or provide the required fuel throughput. By using three SRPS power systems the transfer times are 15, 20, and 23 years from launch to planetary capture for Uranus, Neptune, and Pluto, respectively. Each spacecraft is roughly identical in terms of spacecraft, payload, and propulsion system. These trip times are high but all three spacecraft (~400 kg total mass each) can be easily launched by *only one* Delta 7925. Trip times may be reduced by 2-3 years by doubling the number of SRPS. In this case only two could be launched on one Delta 7925. Launch of these orbiters, even singly, on the Delta 7925 is not feasible using only chemical means. While use of the Delta IV M+ and state-of-art chemical systems would allow sending these orbiters to Uranus and Neptune (one orbiter per launch vehicle), a Pluto orbiter is not possible using only chemical means.

Consequently, with one small Delta launcher and three small identical REP spacecraft the outer planets could be explored with *orbiting* probes. By using autonomous trajectory profiles of weeks to months, one mission team could keep track and control all three spacecraft. While the wait for data from Pluto would be at least 20 years, Uranus would be visited ~8 years earlier, and Neptune ~3 years earlier; allowing one science team to perform outer planet exploration in succession for a 10 year period after a wait of just over 10 years.

While these trajectories and system designs need to be analyzed with more accurate methods the potential for outer planet orbiters is enticing using REP. While not a sprint, faster trip times for orbiters, at least to Pluto are probably only possible with a much high power nuclear reactor powered system.

Conclusions

Studies were undertaken to show what a radioisotope electric propulsion system would look like and what it could do for outer planetary exploration. On-going work in small ion thrusters, Stirling radioisotope power systems, and small planetary science spacecraft point toward the possibility of a viable REP spacecraft for outer planetary exploration. While a reactor powered system would provide quicker trip times and more science payload mass and power, the REP system alleviates the need for a reactor and large launch vehicles.

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