Ion Thrusters and LFAs for Outer Planet Exploration[#]

Muriel Noca[1], James E. Polk[2]

[1]*Deep Space Mission Architecture Group, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Dr., Pasadena, CA 91109, muriel.noca@jpl.nasa.gov*
[2]*Advanced Propulsion Technology Group, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Dr., Pasadena, CA 91109, james.e.polk@jpl.nasa.gov*

Given the recent advancements in power generation, waste heat rejection systems and electric propulsion, a reassessment of the benefits of Nuclear Electric Propulsion (NEP) is provided. Seven different planetary missions are evaluated: a Pluto rendezvous, an Europa rendezvous, a Titan/Saturn sample return, an Europa sample return, a fast Mars piloted mission and a fast Neptune piloted mission. These various missions are evaluated against three major power levels which constitute an evolutionary path for the technology: 1) a 100-kWe relatively near term power system with ion engines, 2) a 0.5 to 1-MWe power system with either ion engines or magnetoplasmadynamic thrusters (MPDTs), 3) a 10 to 100-MWe power system with MPD thrusters. Detailed NEP vehicle mass breakdowns are established and combined with parametric low thrust trajectories. Delivered payload mass as function of trip times for each mission is provided. The analysis shows that NEP has applications over a large spectrum of missions. NEP is especially applicable for short trip time, large launch masses and high-energy missions.

INTRODUCTION

New power and propulsion technology efforts such as the DS-1 ion propulsion system demonstration, the recent funding of solar sail technology after many years of neglect and renewed interest in space nuclear power sources call for a reassessment of the mission types for which each technology is the most applicable. This paper focuses on Nuclear Electric Propulsion (NEP) as a means to transport large payloads to targets in the solar system which are energetically difficult to reach.

The main objective of this study is to assess the benefits of NEP. A large emphasis has been placed in defining the NEP vehicle configuration and corresponding subsystem elements in order to produce an estimate of the vehicle's payload delivery capability which is as credible as possible. Therefore a few design points were studied, and reasonable interpolations were made around those design points.

The study is divided in three parts. The first part describes a 100-kW_e class NEP vehicle and evaluates its delivered payload capabilities for near term robotic planetary missions. The second part describes a 1-MW_e class vehicles with two different propulsion systems and evaluates their performance and relative benefits. These systems are evaluated for robotic

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missions. The third part describes the assumptions and results for a 10 to 100-MW_e vehicle for piloted missions. All detailed power system assumptions and mass breakdowns can be found in [1], from which this paper is vastly inspired.

FIRST STEPS: ROBOTIC EXPLORATION OF THE SOLAR SYSTEM (100-KW CLASS POWER SYSTEMS)

System Design and Vehicle Configuration

The 100-kW_e class propulsion system assumed here is derived from a Kuiper Belt Object Rendezvous study [2] that involved several NASA centers and DOE Laboratories. The overall NEP vehicle configuration is based on the use of a SAFE-300 nuclear reactor and of an ion propulsion system. All subsystem masses assume the vehicle configuration shown in Figure 1. In this configuration, a long boom separates the power and propulsion systems from the other subsystems of the spacecraft. The boom also serves as a structural attachment for the deployable radiators. Every element of the vehicle other than the reactor is located in the reactor shield's shadow, which covers an elliptical 10° x 30° solid angle. The power conversion system, propulsion system fuel tanks, feed system, power processing and thrusters are mounted next to the shield. The very large deployed radiators (about 130 m^2) are unfolded along each side of the main boom. In stowed configuration, the spacecraft fits within a Delta IV launch fairing (5 m diameter by about 14 m long).

Figure 2 shows a system level block diagram of the NEP design, which is common to the 100-kW_e and higher power cases.

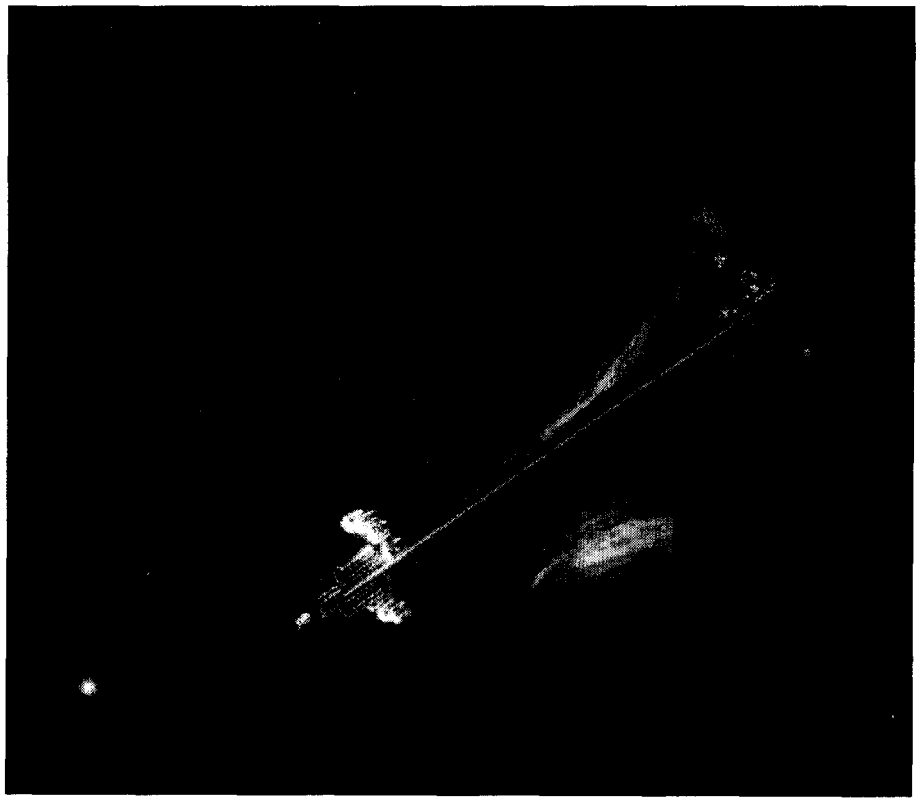

Figure 1. Kuiper Belt Object Rendezvous Mission [2], vehicle configuration.

Power System

The 100-kW_e system is the result of a detailed trade study in which a variety of reactor concepts and conversion systems were evaluated. The baseline system has a NASA/Marshall Space Flight Center (MSFC) SAFE-300 UO_2 fueled, heat-pipe-cooled reactor that has been extensively analyzed by NASA/MSFC and Los Alamos National Laboratories. The 100-kW_e system produces 102.4 kW_e electric and approximately 320 kW_{th} thermal. Table 1 summarizes the reactor and power generation system characteristics. The nuclear reactor is launched cold and is not started until the vehicle has reached an Earth escape orbit.

Note that this power system is relatively light compared to other current designs [3]. It could represent our expectations for a second-generation 100-kW system design.

Table 1: Power System Characteristics.

	Heat Pipe SAFE-300
Power	320 kWth
Electric power	102 kWe
Nuclear fuel	UO_2
Turb. inlet temperature	1300 K
Conversion cycle	Recup. Brayton
Conversion efficiency	32%
Thermal radiator techn.	CPL Al
Radiator specific masses	2.3 kg/kWth 4 kg/m^2
Radiating area	130 m^2
Radiator temperature	360 K
Shield	200 krads @ 12 m /10 yrs
Power system mass (no Struct., no contngncy)	1342 kg
Power syst. spec. mass (no Structures)	13 kg/kWe 17 kg/kWe (30% contingency)

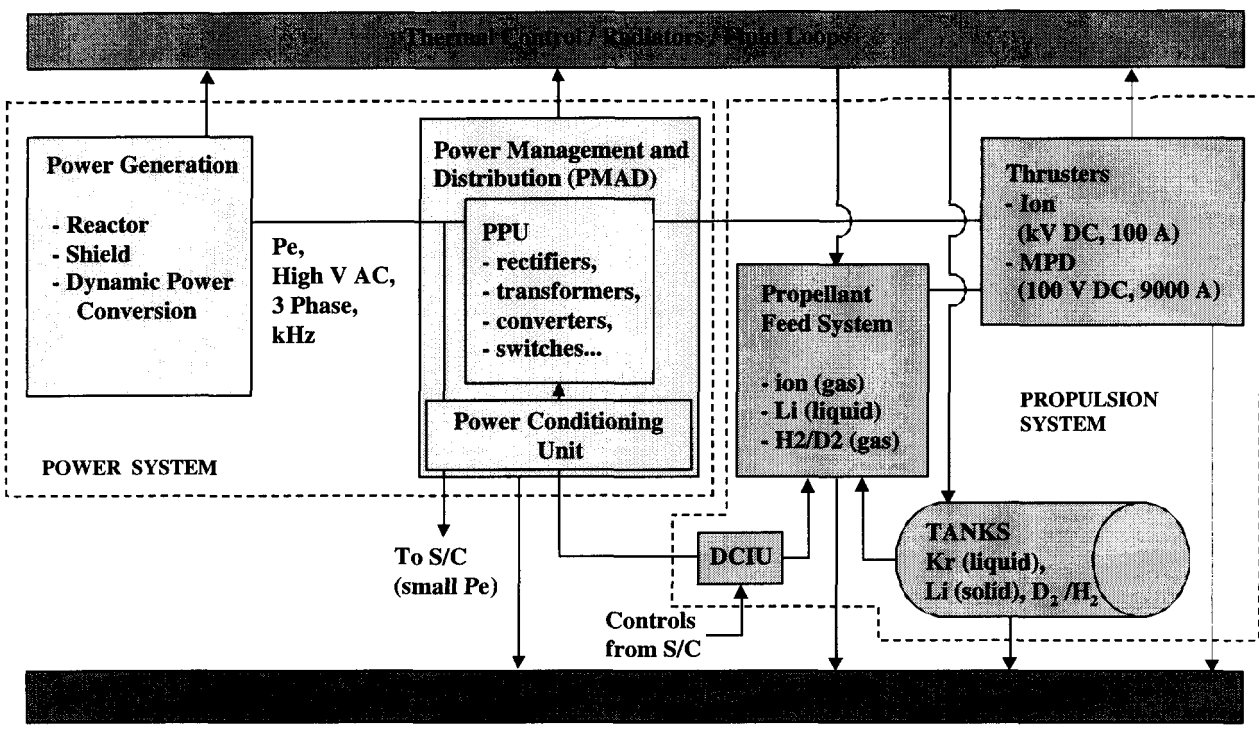

Figure 2. NEP system level block diagram.

Propulsion System

The ion propulsion system (IPS) is composed of 60-cm diameter ion engines that can process 25-kW of electric power and use krypton rather than xenon as propellant. The thruster has an estimated efficiency of 0.67 at a specific impulse (Isp) of 5000 s, 0.77 at an Isp of 10000 s, and 0.77 at 15000 s. At 25 kW and 9500 s of specific impulse, the thruster beam voltage is 5300 V, the beam current is 2 A, the accelerator grid voltage is 935 V, and the discharge propellant efficiency is 0.88. The propellant throughput capability of each engine was estimated to be 500 kg, by scaling the capability of an existing, flight qualified 2.3 kW engine. The thrusters are powered by the PPUs, which convert the power from the turbo-alternator to the voltages and currents required by the engine. The feed system and PPU are controlled by the Digital Control Interface Unit (DCIU), which accepts and executes high-level commands from the spacecraft computer and provides propulsion subsystem telemetry to the spacecraft data system.

The Power Processing Unit (PPU) design was provided by Luis Pinero from NASA Glenn Research Center (GRC) for the Kuiper Belt Object Study [2]. The mass and complexity of the PPU were greatly reduced by tuning the output voltage of the turbo-alternator to a value close to the thruster inputs demand (direct-drive architecture). The design of the turbo-alternator is flexible enough to allow for this tuning. The efficiency of this PPU was estimated at 0.94. Each PPU processes 25 kW of power, so under normal operations four PPUs are operated simultaneously. The PPUs are designed to have a lifetime much greater than the thrusters, and thus only one spare PPU was included.

Although the design of the ion engines would be new, it is based on the experience and technology of the 30-cm, 2.3-kW NSTAR engine that flew on the New Millennium Deep Space-1 mission (DS-1) in 1998. Very little development of high-power ion thrusters has occurred and new technologies will be required, but none appear to have major feasibility issues. NASA is currently developing a 75-cm ion engine with capabilities similar to those assumed in this study [4].

The tankage fraction was calculated assuming two cylindrical composite tanks. Those tanks have a propellant storage efficiency (including tank shell and insulation) of about 0.5% [5] for krypton, when stored as a liquid at 120 K and 1.4 Bar (20 psia). To keep the krypton stored below 120 K, a passive cooling system that uses a Sun shade and a V-Grove isolation system was designed. The Sun shade uses Silver-FEP Teflon on the Sun side, and is sharply angled. The side of the Sun shade facing the tank has a 20 layer Multi-Layer Insulation (MLI) blanket. Also between the tanks and the Sun shade is a double V-Grove radiation shield. With this design at 1 AU, the temperature of the tank is kept well below 120 K. The mass of this system is dependant on the total propellant mass and tank size. For 10,000 kg of Krypton, the total mass is 75 kg, including the Sun shade, V-Grove isolation, MLI for tanks and support structure. This mass was scaled to accommodate different propellant loads.

Feed system design is similar to the Comet Nucleus Sample Return [6] feed system. It is an improved version of the DS-1 feed system (no plenum tanks) and involves some new components (variable regulator, flow control devices) that should be available by 2010.

System Masses and Efficiencies

Table 2 summarizes the system masses, efficiencies and assumptions for the 100-kW_e class missions assuming 2000 kg of deterministic ΔV propellant mass. Since each mission trajectory we considered has a different propellant mass requirement and thus a different number of engines, the system masses have been adjusted for each trajectory case. The mass list includes all the subsystems involved in the NEP vehicle but not the spacecraft bus-related subsystems such as telecommunications and avionics.

Table 2: NEP Propulsion Module Mass Breakdown

Subsystem	Mass (kg)	Comment
Attitude Control	65	Cold gas
Power System	1342	
Reactor / Shield	720	
Power conversn	121	
Heat rejection	500	
Thrust,PPU,DCIU	174	5 thrusters,5PPU
Tanks	28	TF = 0.5%
Feed System	10	
Total Structure	635	
Cabling	95	
Thermal	122	
Total:	**2471**	
Contingency	741	30%
Propellant	2200	10% conting.
NEP Total:	**5412**	

One spare ion engine for every four operating ion engines and one spare PPU and digital control interface unit (DCIU) are included for single-fault tolerance.

The structures/cabling masses are not based on a specific design but are a percentage of the subsystems to which the structures apply (typically 26% of the propulsion system and 16% of the power system for structures). These percentages) are based on historical data and are consistent with the design guidelines of the JPL integrated project design center (Team X).

The 30-m deployable boom is a similar design to that of the AEC-Able FASTmast, which was designed for the International Space Station. FASTmast is a 35-m mast system that supports the Space Station large blanket solar arrays. The total mass of this mast including canister, deployment system and mast is about 300 kg. The mass used in this study was 80% of the FASTmast value, under the assumption that this technology would improve within the next decade.

Pitch and yaw control of the vehicle is assumed to be performed with the ion engines. Roll will be achieved by a separate system (cold gas). Also, a two-degree-of-freedom gimbaled momentum wheel will be used to cancel momentum from the turbine in the power conversion system.

This mass breakdown varies with the propellant mass and number of engines. When necessary, the power system was scaled as the square root of the ratio of the desired power to 100 kW_e. This scaling was done close to the 100 kW_e point design.

Mission Results

The 100-kW_e class NEP system was evaluated for a Pluto orbiter and a Europa Orbiter mission. All trajectories were computed by Carl Sauer at JPL. They assumed the efficiency profile described above of the 25-kW ion thrusters and PPU, a tankage fraction of 10% and a duty factor of 100%. Each trajectory was optimized for specific impulse (Isp) although the maximum Isp was constrained to 16000 s to be consistent with the assumed thruster technology. The calculations of the net delivered mass assumed the launch from either a Delta IV M+ [5,4] (similar in performance to the Atlas V 521) or a Delta IV Heavy. With 10% derating and a C3 slightly above 0 km^2/s^2, the Delta IV M+ injected mass was assumed to be 4060 kg and the Delta IV Heavy injected mass 8325 kg. At 2500-km Low Earth Orbit (LEO), the Delta IV M+ capability is 9800 kg with 10% derating.

The net delivered mass (which is the total arrival mass at the target minus the dry NEP mass described in Table 2) is provided in Figures 3 and 4.

The Pluto orbiter launch date was assumed to be 2010. The orbiter approaches Pluto with an excess velocity very close to 0 km/s. However, in the trajectory results, the vehicle does not get captured by the planet. An additional small ΔV would be required to capture and potentially circularize around Pluto. Here, the NEP ΔV is typically between 40-55 km/s. Propellant mass varies between 1000 and 4000 kg, leading to systems with 5 to 10 engines. As shown in Figure 3, a Delta IV M+ [5,4] can deliver to

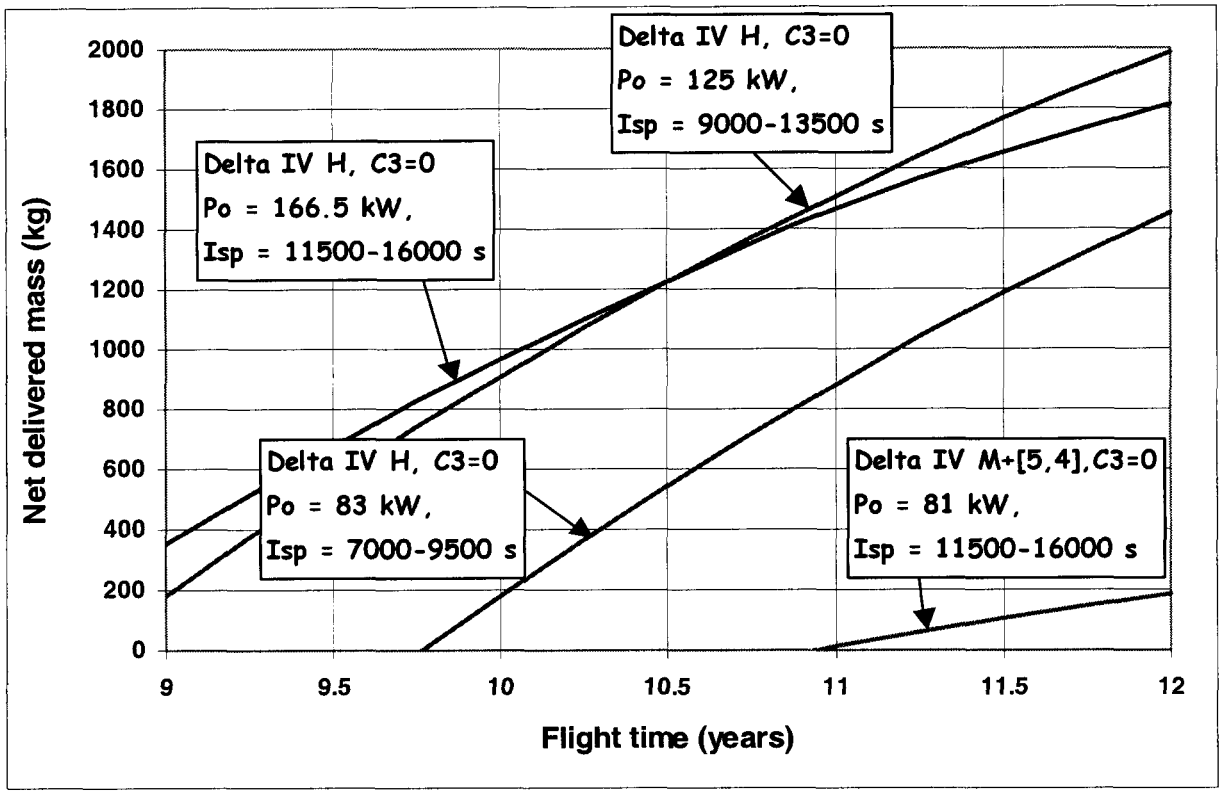

FIGURE 7. Net delivered mass for the Pluto rendezvous mission as a function of flight time for a launch at C3 = 0 km^2/s^2.

Pluto (rendezvous) a net mass slightly less than 200 kg in 12 years, while a Delta IV Heavy can deliver a net mass slightly above 300 kg in 9 years and about 2000 kg in 12 years.

The Europa orbiter mission assumes a launch in 2014. The vehicle reaches Jupiter with a low excess velocity and the NEP system is used to capture to a circular orbit around Europa at 100 km altitude. Spiral-out times from Earth are between 160 and 400 days and spiral in to Europa between 130 and 550 days. The NEP ΔV is typically between 34-36 km/s. Propellant mass varies between 3000 and 6000 kg, leading to systems with 7 to 15 engines. As shown in Figure 4, a Delta IV M+[5,4] can deliver a net mass between 500 and 2500 kg in 4 to 7 years.

SECONDS STEPS: OUTER PLANET SAMPLE RETURNS (0.5-1 MW CLASS POWER SYSTEM)

With 0.5-1 MWe of power, fast, high energy robotic missions can be accomplished. The vehicle and subsystem assumptions are summarized here and mission results as a function of flight time are provided.

System Design and Vehicle Configuration

For the 0.5-1 MW_e class missions, two propulsion systems were studied. The first one uses the same size ion engines as the 100-kW_e class, but operated at 60 kW per engine. Nine of these engines clustered together forms a 480-kW engine equivalent (1 engine is redundant). The description of this system is provided below. For this system, the same vehicle configuration as

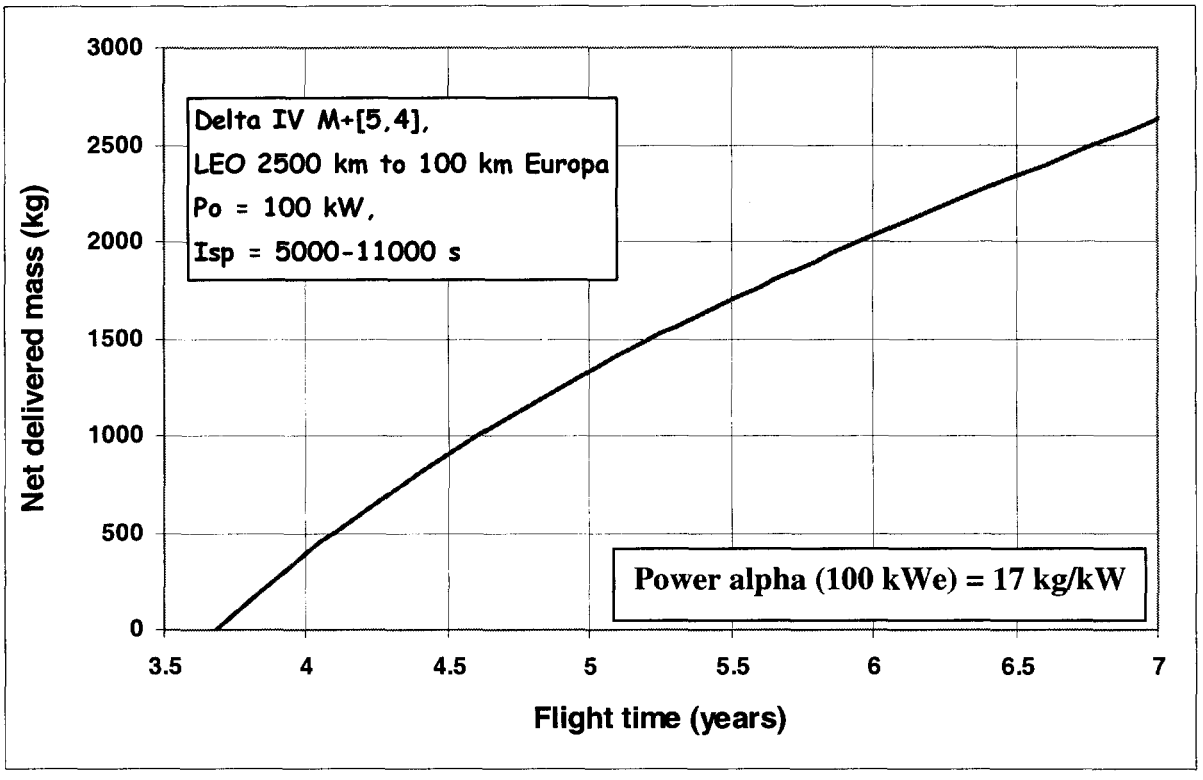

FIGURE 4. Net delivered mass for the Europa rendezvous mission as a function of flight time.

the 100-kW_e case was assumed. The other propulsion system considered uses lithium-fuelled Lorentz Force Accelerators (LFA's), also known as Magnetoplasmadynamic thrusters (MPDT's). To avoid contamination of "cold" surfaces by condensation of lithium from the engines, a slightly different vehicle configuration was used, which isolates sensitive spacecraft surfaces from the engines, as shown in Figure 5. A plume shield was also added to reduce lithium contamination. This configuration was inherited from a detailed Mars cargo vehicle study performed in the early 90s [7]. Experimental measurements of lithium backflow [8] and numerical simulations of lithium thruster plumes [9] indicate that a combination of physical separation and plume shields will prevent spacecraft contamination. Other configurations might be possible where the thrusters could be located close to the reactor system (as for the ion engines).

The NEP power system is based on the SP-100 reactor design with dynamic power conversion. Both vehicles have a boom of 50-m total length. Here again, this deployable boom is similar to the AEC-Able FASTmast with an assumed 30% mass reduction by 2016.

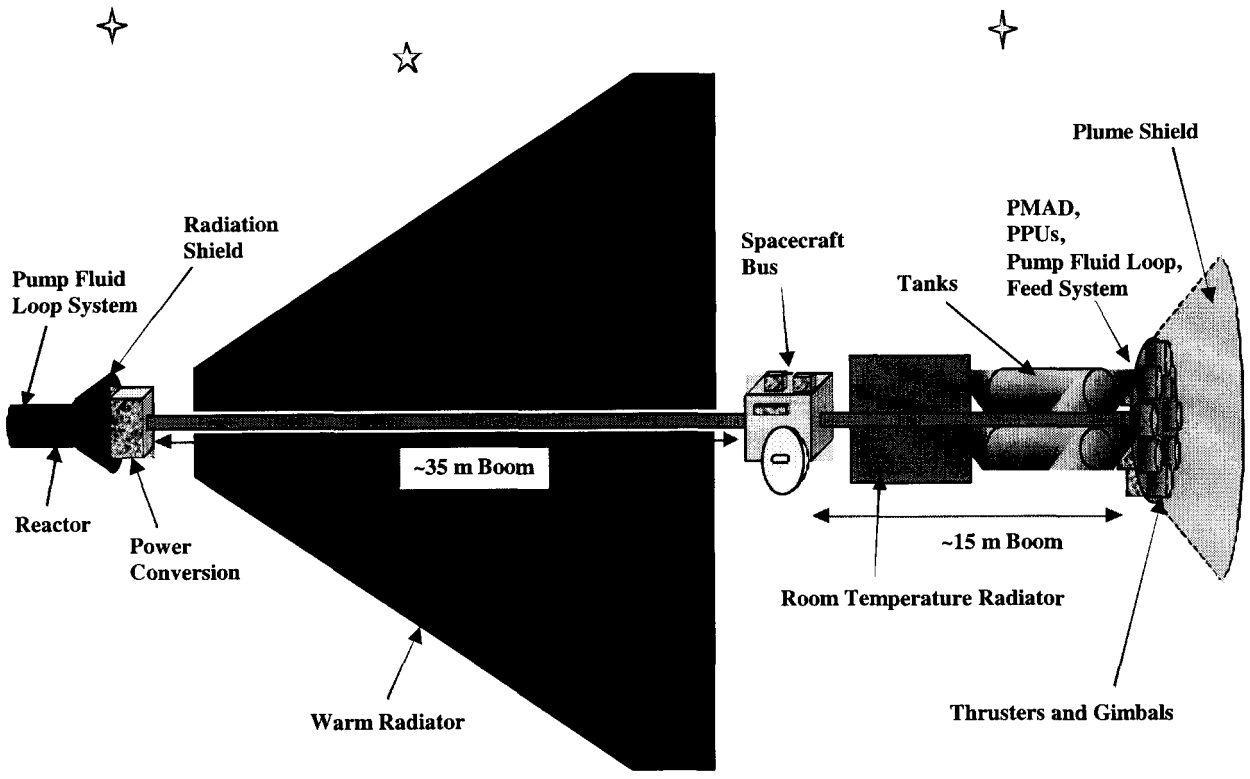

FIGURE 5. 1-MWe Vehicle configuration assumed for an MPDT based propulsion system.

Power system

The direct gas-cooled concept was selected for the 1 MW_e class system. It is particularly attractive for its lighter weight at these thermal power levels, and it is a good match between required thermal power and upper temperature limits. The mass of shielding has increased to accommodate the increased reactor power level. The direct gas cooled system has a potential to be scaled to at least the 10-MW_e class of power.

The power generation system assumes a Brayton cycle power conversion. For this 1-MW_e class system the turbine speed was increased and the size was also increased over the previous 360-kW_e design point of the study. The alternator design appears to be at the crossover point for incorporating high temperature superconducting stator windings. It does not appear feasible to increase alternator efficiency above about 97%; this means that 30 kW_{th} will have to be rejected through the primary radiator. Detailed analyses of harness weight will also have to be traded to ascertain the utility of incorporating high temperature superconducting harnesses at this power level.

The primary difference between the radiator for the 1-MW_e concept and the 100-kW_e class vehicle is the incorporation of lightweight, high thermal conductivity panels in the Capillary Pump Loop (CPL) radiator. Since the turbo-machinery is larger, the efficiency increases, so less fractional thermal power must be radiated. Given the power radiated, the specific mass of the radiator should be reduced to about 0.55 kg/kW_{th}.

Propulsion Systems

MPDT Propulsion System

Lorentz Force Accelerators (LFA's) are the only type of electric thruster with a demonstrated capability to process steady state power levels up to several MWe in a relatively compact

device. In these engines a very high current is driven between coaxial electrodes through an alkali metal vapor (e.g., Li) or gaseous propellant. Lithium propellant yields very high engine efficiency because it has low frozen-flow losses. Because it has a very low first ionization potential and a high second ionization potential, very little power is expended in creating the plasma. The current interacts with a self-induced or externally-generated magnetic field to produce an electromagnetic body force on the gas. LFA's can operate efficiently at power levels from 150 kWe up to tens of MWe and are therefore ideally suited for a number of ambitious future in-space applications which require high power. Table 3 shows the projected MPDT performances.

Each 500-kW MPDT is able to process 4500 kg of lithium propellant. Lithium is stored as a solid and thus minimal tank strength is required. Waste heat from the thrusters is used to melt lithium at a temperature of 181°C. Two cylindrical tanks are used to store propellant on either side of the boom. The tanks and thruster waste heat transfer system have a tankage fraction of 2.8% [5]. The feed system masses were estimated based on current MPDT feed system work done at Princeton University and JPL [10].

Lithium-fed thrusters are also a stepping-stone to more advanced hydrogen-fueled thrusters, which can operate at many MWe per engine with Isp's ≥10,000 s and efficiencies ≥60 %. These engines could enable the use of multi-hundred megawatt nuclear electric vehicles, as discussed in the third part of this paper.

TABLE 3. Projected LFA thruster performance and lifetime.

Power per engine (MW)	**0.5**	**1**	**2**	**5**
Efficiency	0.55	0.65	0.65	0.65
Lifetime (hrs)	5000	5000	3000	2500
Thruput (kg)	4500	6300	16000	30000

The 1-MW PPU and cabling design for the LFA thrusters is a scaled version of the PPU design proposed in the early 90's for a Mars cargo mission [11]. This design assumes that the PPUs are located close to the thrusters such that it minimizes the cable mass between the turbo-alternator and the PPU. The transmission lines between the turbo-alternator and the Power Processing Module (PPM) are low-current, high-voltage (10000 V, 3-phase voltage of 5773 V, 5 kHz) lines. A transformer located in the vicinity of the PPUs and thrusters steps down the voltage to a 60-100 V level. The PPU then includes one input switch array that drives three 500-kW controlled rectifiers and filters (including one spare) and two power conditioning units (one spare) for supplying the spacecraft bus with Housekeeping power. A controlled rectifier and filter output switch array feeds the thrusters with 60 V and 9000 A via large bus bars (current leads) that also serve as structural supports. Each thruster also includes a non-load break electromechanical switch.

Ion Propulsion System

The 1-MW ion propulsion system is composed of two 480-kW ion engines. Each ion engine includes eight 60-cm, 60-kW ion sources plus one redundant source configured as a square 3 by 3 array. This approach is called a "segmented ion engine" in which multiple discrete ion sources are integrated together to form a single large ion engine with a large effective total grid area. A significant advantage of such a design is that the ground facilities and pumping requirements needed for testing are much relaxed (and feasible) relative to testing an equivalent single 480-kW ion engine. The 60-kW ion sources are essentially the same design as the previously described 25-kW engines. Each 60-kW source is assumed to have an efficiency between 0.65 and 0.75 at specific impulses between 5000 s and 16000 s. At 60-kW and 10000 s of specific impulse, the thruster beam voltage is about 3800 V, beam current for the segmented ion engine as a whole is 120 A, acceleration grid voltage is 400 V, and discharge current for each ion source is about 110 A. It is recognized that these characteristics are still preliminary since no test data are currently available. The engine lifetime has been estimated to be 240 kg per ion source, resulting in a total propellant throughput capability of

about 1900 kg of krypton for the segmented engine. Each engine sits on a structure approximately 2.4 m wide and 3.0 m long. To fit within the launch vehicle fairing (about 4-5 m diameter), a deployment system has been added to deploy the two engines.

The PPU design for the ion engines was provided by Luis Pinero from NASA GRC. It assumes the engine cluster configuration described in the Propulsion section and one PPU per cluster (each PPU processes 500 kW of power). Here again, the PPU is designed to be "direct-drive". The PPU total efficiency is about 98%. Note that in this design, the assumed frequency is 5000 Hz.

The krypton tankage assumptions along with the thermal control (Sun shade) for the ion propulsion option are the same as for the 100-kW_e case but scaled to the larger propellant loads for the 1-MW_e system.

System Masses and Efficiencies

Table 4 summarizes the vehicle and subsystem mass breakdown. As can be seen from the tables for a 1-MW_e system, the vehicle mass with ion engines is less than that for the MPDTs. Although MPDTs have a higher power density than ion engines, the mass of cabling internal to the PPU or external to route the power largely offsets their advantage (MPDTs require around 100 V and 9000A as inputs). An effort needs to be made in the power routing and processing to reduce the mass of cabling, possibly with high temperature superconductors, to make it an equally attractive solution to ion engines at this power level.

However, for missions that require higher power (> 2-3 MW_e), ion propulsion becomes less and less attractive due to the large number of engines required and their deployment, and their limited propellant throughput capability, which is lower than that of the MPDTs. Thus typically for human missions, MPDTs were a preferred option as a propulsion system.

The mass breakdown shown in Table 4 varies with the propellant mass and number of engines. When necessary, the power system was scaled as the square root of the ratio of the desired power to 1 MW_e. This scaling was done in the range of 0.3-1 MW_e. Also, for some 1-MW_e trajectories, 3 to 4 ion engines were used as demanded by the propellant loads, but no consideration was given to the packaging within the launch vehicle fairing.

Table 10: NEP Propulsion Module Mass Breakdown for an ion engine and an LFA based system

Subsystem	Ion engines mass (kg)	LAFs mass (kg)
Attitude Control	130	130
Power System	2990	2990
Reactor / Shield	1915	1915
Power conversn	300	300
Heat rejection	770	770
Thrust,PPU,DCIU	640	1445
Tanks	20	110
Feed System	75	25
Total Structure	1525	1310
Cabling	175	1390
Thermal	255	235
Total:	**5810**	**7635**
Contingncy (30%)	1740	2290
Propellant (+10%)	4200	4200
NEP Total:	**11750**	**14130**

Mission Results

All trajectories were calculated parametrically as a function of the ratio of initial power in kW_e to the initial mass in kg. They assumed the efficiency profile of the 60-kW ion propulsion system or of the MPDT propulsion system provided in the Propulsion section, a tankage fraction of 10% and a duty factor of 100%. Here again, each trajectory was optimized for specific impulse (Isp) although the maximum Isp was constrained to 16000 s for ion engines or to 8000 s for MPDTs to be consistent with thruster technology. All trajectories start with a C3 slightly above 0 km^2/s^2.

The 1-MWe system with ion engines was very well-suited for high-energy robotic missions.

Here a Titan/Saturn sample return and a Europa sample return mission were considered. Power levels above 1 MW_e appeared to be more applicable to large payload deliveries, as for a piloted mission. All trajectories presented below assume an ion engine propulsion system.

As with the 100-kW_e case, the Titan/Saturn sample return mission leaves Earth in 2013, and in about half the total trip time enters into a highly elliptical orbit around Saturn with apoapsis at Titan's altitude. There is a 5-month stay time in this elliptical orbit. The spiral in and out times to the elliptical orbit are very similar in both cases and are between 30 and 60 days depending on the trajectory. This mission assumes no mass drop at Titan, meaning that the NEP carrier brings back the same inert mass as it carried on the way there. On the way back, the NEP vehicle performs an Earth fly-by. The trajectories were constrained by a specific impulse of 16000 s. Figure 6 shows the net delivered mass (which is the total arrival mass at the target minus the dry NEP mass described in Table 4) for various initial masses. The NEP ΔV is between 30-60 km/s. Propellant mass varies between 2000 and 9000 kg, leading to systems with 1 to 5 engine clusters. As shown in Figure 6, a Delta IV Heavy can return to Earth a net mass slightly over 1000 kg in 10 years.

The Europa sample return mission assumes a launch in 2008. The vehicle reaches Jupiter with a low excess velocity and the NEP system is used to capture to a circular orbit of Europa's radius. Spiral in and out times to this orbit are between 100 and 180 days. No mass drop at Europa was assumed, which is conservative. The return trajectory terminates with a fly-by of the Earth. The NEP ΔV is typically between 40-55 km/s. Propellant mass varies between 2000 and 3000 kg, leading to a system with 2 engine clusters. As shown in Figure 7, a Delta IV Heavy can return to Earth a net mass of about 850 kg in 5.2 to 5.6 years.

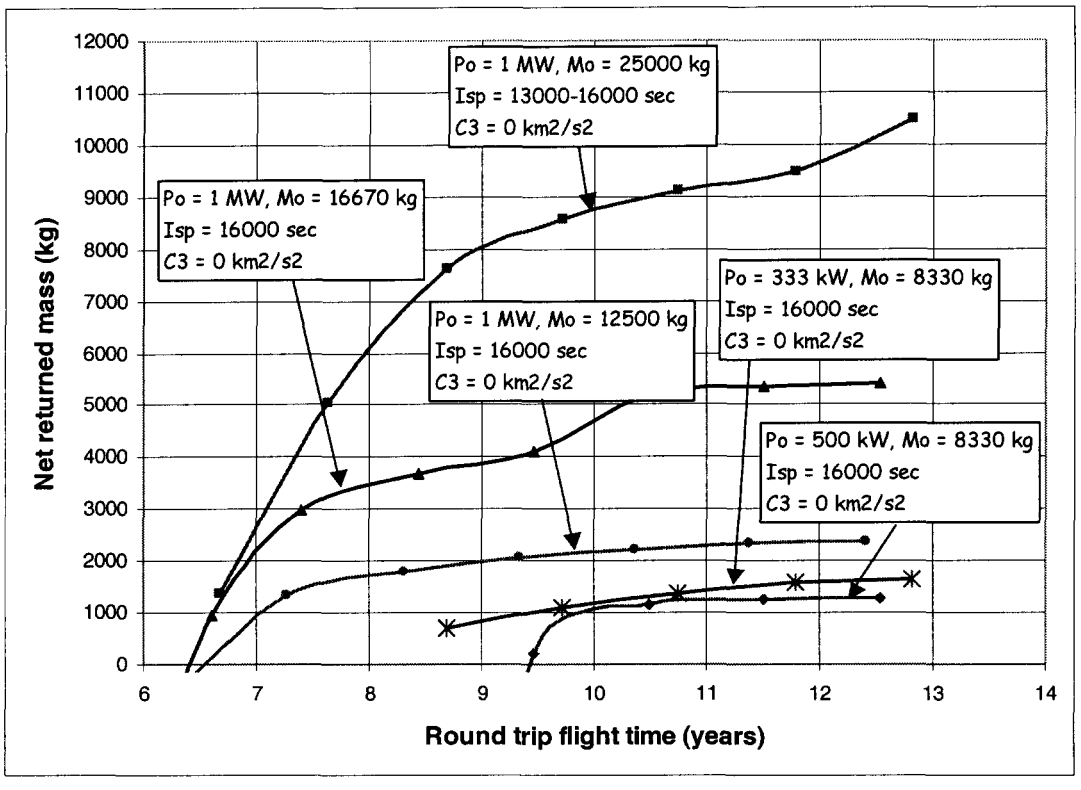

FIGURE 6. Net returned mass for the TITAN/SATURN Sample Return as a function of flight time. 1 MWe vehicle.

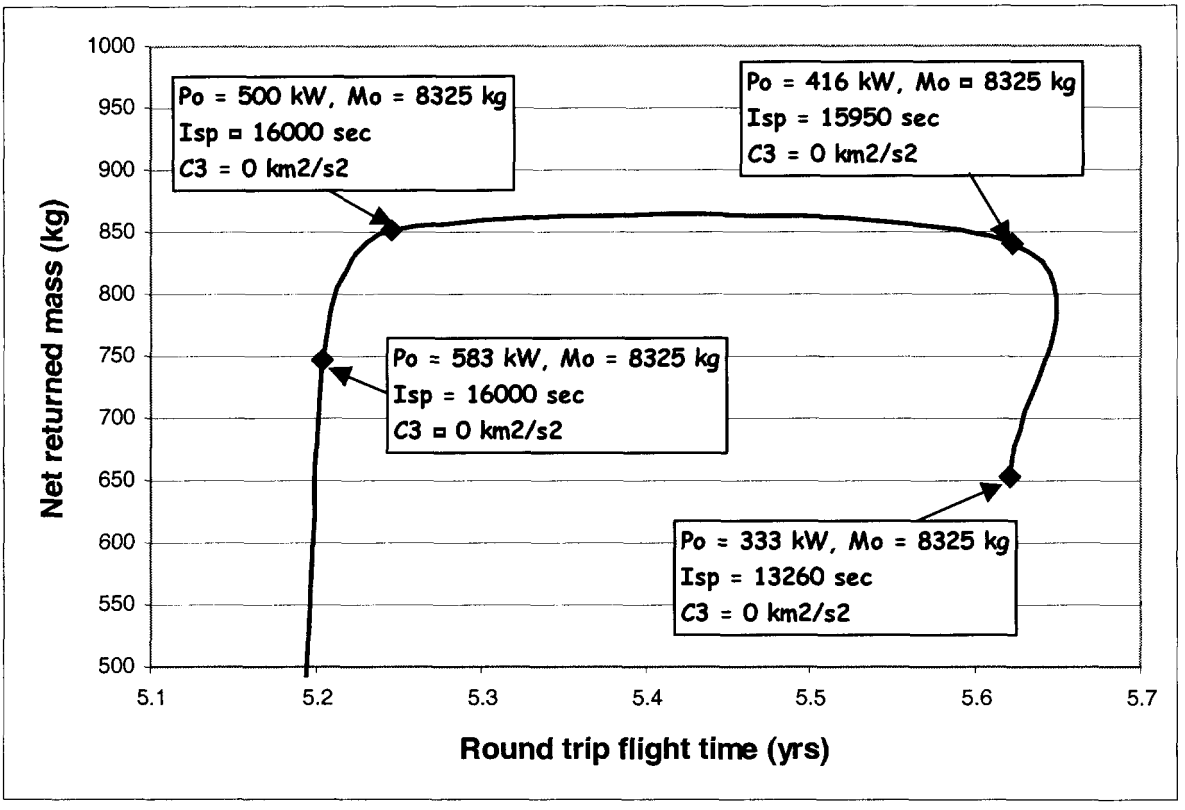

FIGURE 7. Net returned mass for the Titan/Saturn sample return as a function of flight time. 1 MWe vehicle.

THIRD STEPS: HUMAN EXPLORATION OF SOLAR SYSTEM PLANETS (10-100 MW CLASS POWER SYSTEM)

With 10-100 MW_e available to the propulsion system, piloted missions are enabled. Two missions have been looked at: a Mars piloted mission and a Triton/Neptune piloted mission.

System Design and Vehicle Configuration

Mars Piloted Missions

The system for the Mars piloted mission was assumed to be a scaled version of the 1 MW_e system. The power system was scaled up as the square root of the power ratio. Thus the technology assumed is a gas-cooled nuclear reactor, a Brayton conversion cycle and LFAs, since they provide a simpler and lighter propulsion system than the ion engines at that power.

For robotic missions, a reactor shield mass of 700 kg was assumed whereas for piloted missions, a reactor shield mass of 1400 kg was used based on a Manned Mars Mission study [12] to account for the lower radiation environment required by human presence.

Neptune Piloted Missions

To reach Neptune and come back in reasonable trip times, very high power levels (100-MW_e) were required. For those power levels, no attempt was made to define a vehicle configuration or to develop a detailed mass breakdown. Rather a set of requirements is provided in order to perform a piloted outer planet mission.

The power system could potentially be based on a Fissioning Plasma Core Reactor with Magnetohydrodynamic power conversion system [13]. This power system could provide a power specific mass (with reactor, shield, radiators, structure, pumps and generator) as low as 0.5 kg/kW$_e$ at 100 MW$_e$.

The propulsion system would be based on the used of hydrogen or deuterium MPD thrusters. These engines are predicted to be 24% efficient at an Isp of 5000 s, 47% efficient at 10000 s and 65% efficient at 15000 s [14].

Results for a Piloted Triton/Neptune Mission

Mars Piloted Missions

A possible architecture for a Mars piloted mission is to send from LEO a piloted vehicle that would deliver humans to an orbit around Mars and bring them back to Earth. Multi-MW NEP could be suitable for fast trip times of both a piloted vehicle and possibly cargo vehicles.

The trajectories were run by Carl Sauer at JPL. They assume a 2018 launch, start at 400-km LEO, and end in a 1-SOL elliptical orbit at Mars. Stay time at Mars is about 30 days. The specific impulse was kept constant at 5000 s (previous analysis showed that trajectories with an Isp of 8000 s were not feasible for trip times of 1 year). They also assume a constant thruster efficiency of 0.6 and PPU efficiency of 0.95. The return trajectory concludes with an Earth fly-by with an excess velocity of 9.4 km/s, which leads to atmospheric entry of a crew return vehicle of about 14 km/s. Round trip ΔVs are between 30 and 50 km/s. Propellant (lithium) masses were around 70 MT for the 10 MWe case, 160 MT for the 20 MWe case and 370 MT for the 50 MWe case.

The power levels considered varied between 10 and 50 MWe. Only 11 LFA thrusters (one redundant) with the capability to process between 1 and 5 MW each were assumed for each power level. The assumed propellant throughput capability per engine is shown in Table 3. Dry vehicle masses were computed using the mass list provided in Table 4 for MPDTs by scaling the power system as the square root of the ratio of the power needed to 1 MW. Dry vehicle masses were about 40 MT for the 10-MW$_e$ case, 60 MT for the 20-MW$_e$ case, and 100 MT for the 50-MW$_e$ case.

Figure 8 shows the net returned mass at Earth fly-by. The result of this study is that a 20-MW$_e$ NEP vehicle could carry a 41 MT habitat to Mars in about 1 year. Its total initial mass in LEO (IMLEO) is about 236 MT.

Triton/Neptune Piloted Missions

The Triton/Neptune piloted mission leaves Earth in the 2020 time frame from a 1000 km altitude LEO. In about half the total trip time it enters into a highly elliptical orbit around Neptune with apoapsis at Triton's altitude. There is a 5-month stay time in this elliptical orbit. The spiral in and out times to the elliptical orbit are between 3 and 50 days depending on the trajectory. This mission assumes no mass drop at Triton. The return trajectory assumes an Earth spiral-in back to a 1000-km LEO. The NEP ΔV is between 130-170 km/s. Propellant mass varies between 300 and 1400 MT.

Trajectories assumed the D_2 MPDT efficiency profile and a 94% efficient PPU. Net returned mass results are provided in Figure 9 for a Neptune piloted mission as a function of trip time and total vehicle specific mass (excluding tank mass). IMLEO masses varied between 500 and 2000 MT. A tankage fraction of 16% was assumed to calculate the net returned mass.

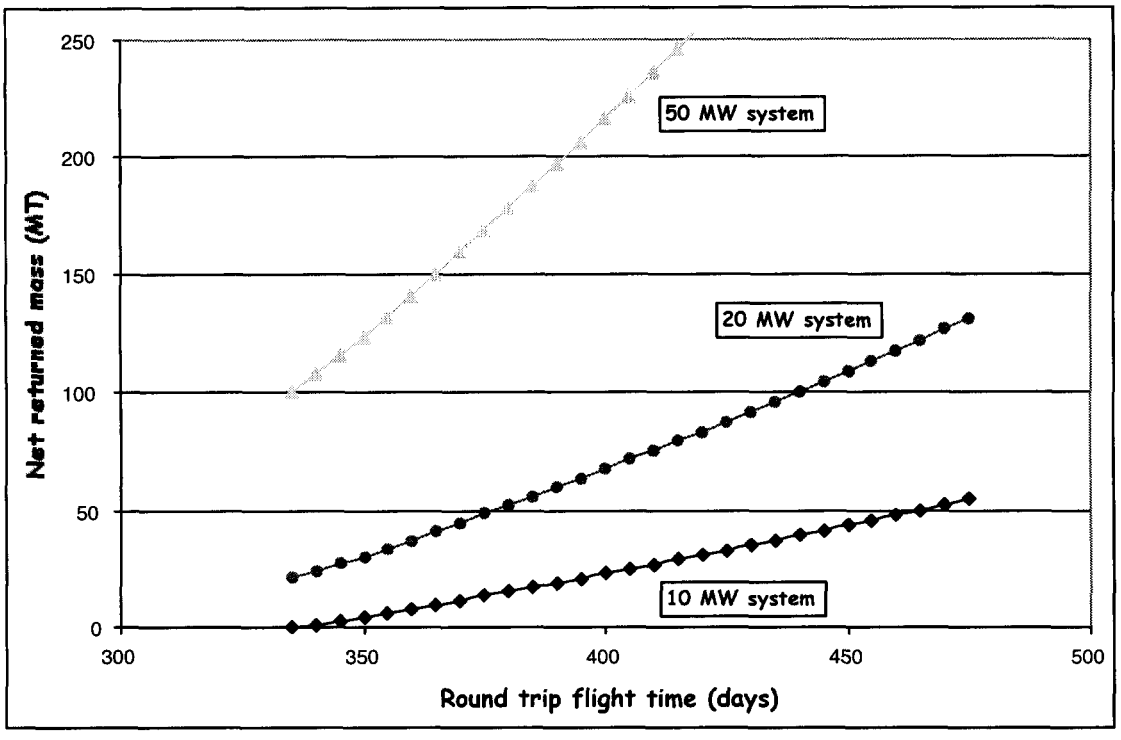

FIGURE 8. 1 MWe Mars piloted mission results – net returned mass as a function of round trip time and power level.

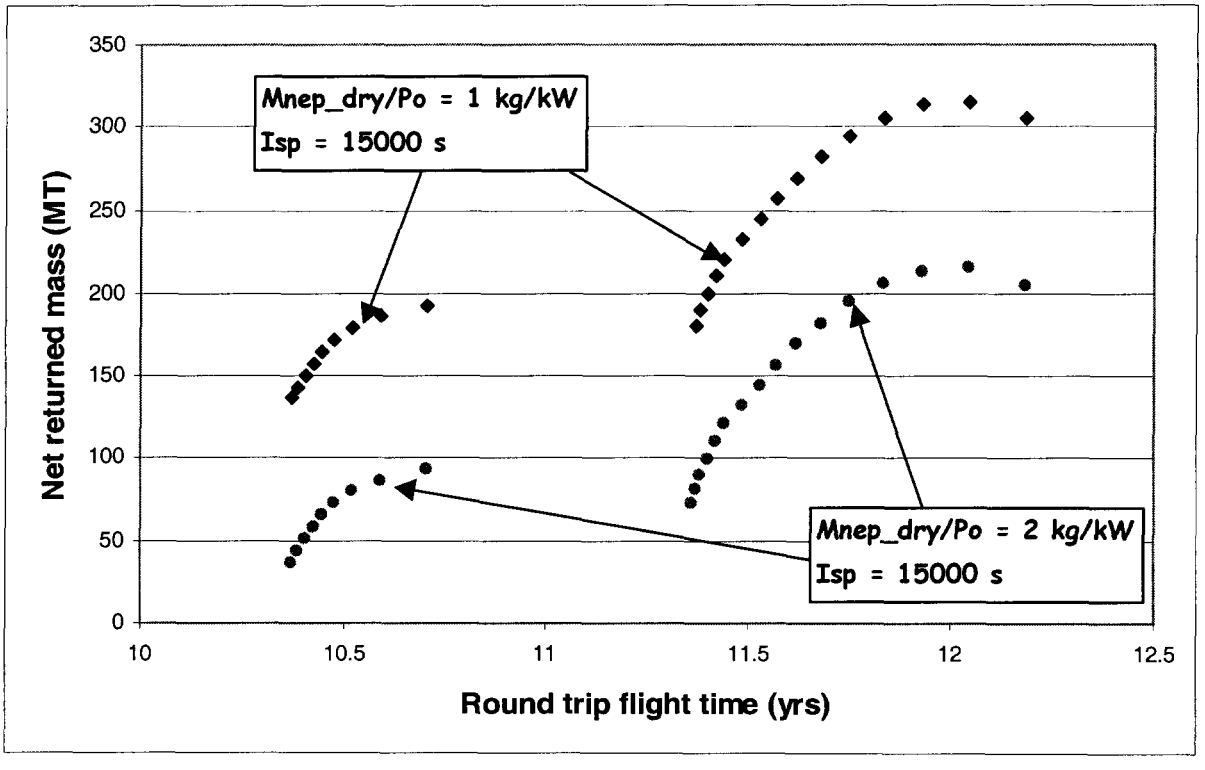

FIGURE 9. Neptune piloted mission results for a 100 MWe vehicle

CONCLUSIONS

This study reviews the potential applications of a range of NEP power and propulsion systems. It is found that 100-kW_e power and ion propulsion systems are applicable for a 9-12 year Pluto rendezvous, and a 3-4 year Europa orbiter mission. Net delivered masses varied between 500 and 2000 kg. As the total dry mass of the 100-kW_e NEP is quite large (around 3200 kg), the benefit of NEP only shows for a Delta IV Heavy (or equivalent) launch vehicle. The mission trajectories started either from a slightly positive C3 or from 2500-km LEO.

A 500-kW_e to 1-MW_e class NEP vehicles enabled a Europa sample return in 5-6 years and a Titan/Saturn sample return in 10 years.

Piloted missions, such as a Mars piloted mission in 1 year or a Neptune/Triton piloted mission in 11 years, required power systems from 10 to 100 MW_e since the mass of the transit habitats are large (several tens of MT). Probably higher Isp would enhance mission results. For most high energy trajectories an Isp of 20000 to 40000 s would be more appropriate (for the same trip times). The piloted missions started from low Earth orbit and assumed an Earth fly-by for the Mars case and a return to LEO for the Neptune case.

Detailed mass estimates of the 100-kW_e, 1-MW_e (with ion engines or with MPD thrusters), and 10-50-MW_e vehicles are provided. The NEP vehicle specific mass as function of power level is summarized in Table 5. $M_{NEP(veh\text{-}tk)}$ represents the mass of the NEP vehicle (not including the spacecraft mass) minus the tank mass. Po is the total power of the vehicle. Tank fraction is also included since it varies as a function of propulsion systems chosen.

TABLE 5. NEP vehicle specific mass.

Power level	0.1 MWe	1 MWe	10 MWe	20 MWe	50 MWe	100 MWe
$M_{NEP(veh\text{-}tk)}$ /Po (kg/kW)	31.7	7.5	3.8	2.7	1.8	1.0
Tank fraction %	0.5	0.5-2.8	2.8	2.8	2.8	16

Ion engines are the most attractive propulsion system (at 25 kW to 500 kW per engine) for robotic missions with system power levels up to a few MWe, while MPD thrusters (at 500 kW to 5 MW per engine) are very well-suited for 5-100 MWe piloted missions. A large fraction of the mass for the MPD thruster system resides in the power processing and power routing cable mass. Lowering this mass (perhaps with superconducting cables) would significantly improve the net delivered mass.

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