41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit AIAA 2005-3892
Tucson, Arizona, 10-13 July 2005.

The Nuclear-Electric Pulsed Inductive Thruster (NuPIT): Mission Analysis for Prometheus

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The Nuclear-Electric Pulsed Inductive Thruster (NuPIT) is an electromagnetic plasma accelerator that could demonstrate attractive performance for a variety of high-power Nuclear Electric Propulsion (NEP) missions under consideration by NASA. Both robotic and piloted interplanetary missions may benefit from NuPIT technology, as well as LEO-to-GEO transfers and escape trajectories. The thruster uses abundant propellants (such as NH_3), as well as propellants that could be derived from extraterrestrial resources (such as H_2O, CO_2, etc.). Also, the NuPIT is subject to less erosion than other thrusters at high powers due to its electrodeless nature. This paper presents the results of mission analyses that expose the advantages and disadvantages of the NuPIT for missions considered under the Prometheus Project. The analyses include comparison studies of the NuPIT technology relative to state-of-the-art Ion propulsion systems and quantify the unique benefits of the technology such as long-life, *in-situ* propellant utilization, and high I_{sp} at relatively fixed efficiency.

I. INTRODUCTION

The Pulsed Inductive Thruster (PIT) is an electromagnetic plasma accelerator that has demonstrated, in single-shot or burst mode, efficiency of greater than 50%, and a specific impulse (I_{sp}) range of 2,000 to 9,000 lb_f-s/lb_m at nearly constant efficiency. It uses plentiful and cheap propellants such as ammonia or water and is subject to less erosion than other thrusters due to its electrode-less nature. In 2004, NASA released a competitive NASA Research Announcement (NRA) to develop Advanced Electric Propulsion (AEP) technologies.[1] The ultimate goal of this activity is defined by the NRA Scope of Program description:

> "The goal of this AEP Technologies program is not to develop flight-qualified hardware, but to promote and advance the development of very high power, AEP thruster technologies that result in reduced AEP system mass and complexity and that may enable future missions that might otherwise not be considered credible and to deliver conceptual AEP system designs. As spacecraft power levels become very high, building high powered gridded ion or Hall thrusters (>100 kWe) or clustering large numbers of moderately powered (~25 kWe) thrusters becomes massive, voluminous, and complicated. The proposed AEP thruster system must offer advantages at a system level over an equivalently performing gridded ion or Hall thruster systems, as well as improvements in component and system lifetimes and performance over the current state-of-the-art (SOA) of AEP systems."[1] (Emphasis added)

NASA selected Northrop Grumman Space Technology (NGST) and its team members, the NASA Glenn Research Center (GRC), Arizona State University (ASU) and the NASA Jet Propulsion Laboratory (JPL), to develop the Nuclear-Electric Pulsed Inductive Thruster (NuPIT) in support of Project Prometheus. NuPIT will be a significant advancement over the previous version of the thruster, the PIT MkVI as shown in Figure 1. The main goal of the program is to develop a model of the thruster capable of sustained operation at power level of 200 kW_e, and at efficiency of 70% or higher while retaining I_{sp} between 3,000 and 10,000 lb_f-s/lb_m. Four key development advancements of the existing laboratory thruster are planned during the NuPIT program: 1) changeover of the electrical pulse switches from spark gaps to solid state switches (thyristors) capable of the nearly 10^{10} operations that are likely to be required in an actual mission (assuming 50 pulses/second for as much as 6 years of actual run time), 2) modification of the propellant injection valve for similar life capability, 3) provision of active cooling of components in which energy dissipation is likely to occur, and 4) design and operation improvements based on insights gained from numerical/theoretical analysis.

The ultimate goal of the NuPIT program is to develop a compact thruster that could benefit several high-power missions considered by Project Prometheus. In this paper we present mission analysis for several such candidate missions, and quantify the benefits of employing NuPIT technology as the primary onboard propulsion.

II. DESCRIPTION OF THE NuPIT PROPULSION SYSTEM

The proposed specifications of the NuPIT system are listed in Figure 1. The thruster is an electromagnetic plasma accelerator concept that has been under investigation at TRW, and more recently NGST, since the late

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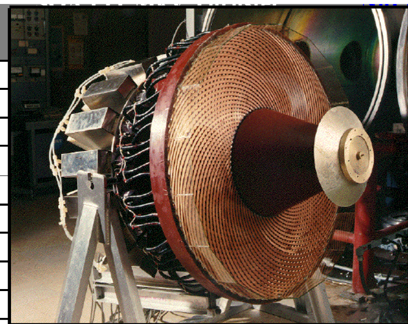
1960s.[2] The thruster design has evolved through a number of stages, of which the most relevant for NuPIT were built and tested after 1985.

The PIT creates its plasma by inductive breakdown of a layer of gaseous propellant transiently puffed onto the surface of a flat induction coil, as shown in Figure 2 (left). At the instant of optimum placement of the propellant along the inductor coverglass ring, energy stored in a bank of capacitors is switched into the coil. The azimuthal electric field E_θ produces rapid ionization of the gas, and establishes a flat ring of current that provides a piston against which the rising magnetic field acts, entraining and ionizing the balance of the propellant, and ejecting it along the thruster axis (Figure 2, right).



Performance Metric		Range
Specific Impulse (sec)		3000-10000
Impulse Bit (N-sec)		0.06-0.2
Thrust Efficiency %		>70
Energy per pulse (kJ)		4
Mass per pulse (mg)		0.6-6.67
Charging voltage (kV)		15
Thruster Size (m) (DxL)		-
Propellant		NH_3, H_2O, CO_2
Thruster mass (thruster+ps)(kg)		-
Equivalent Steady State Performance at Optimum Point		
Input power (kW)	40**	200
Average Thrust (N)	1	5
Total flow rate (mg/sec)	20	83.5
Pulses per second (pps)	10**	50
Switch lifetime	10^{5**} pulses	10^{10} pulses (>6yr@50 pps)
Valve lifetime	10^{3**} pulses	10^{10} pulses (>6yr@50 pps)
Capacitor lifetime	10^{7**} pulses	10^{10} pulses (>6yr@50 pps)
EP System alpha (kg/kW)	6.25 (thruster+ps only)	2.75 (thruster+ps only)

Figure 1. The Nuclear-Electric Pulsed Inductive Thruster (NuPIT).

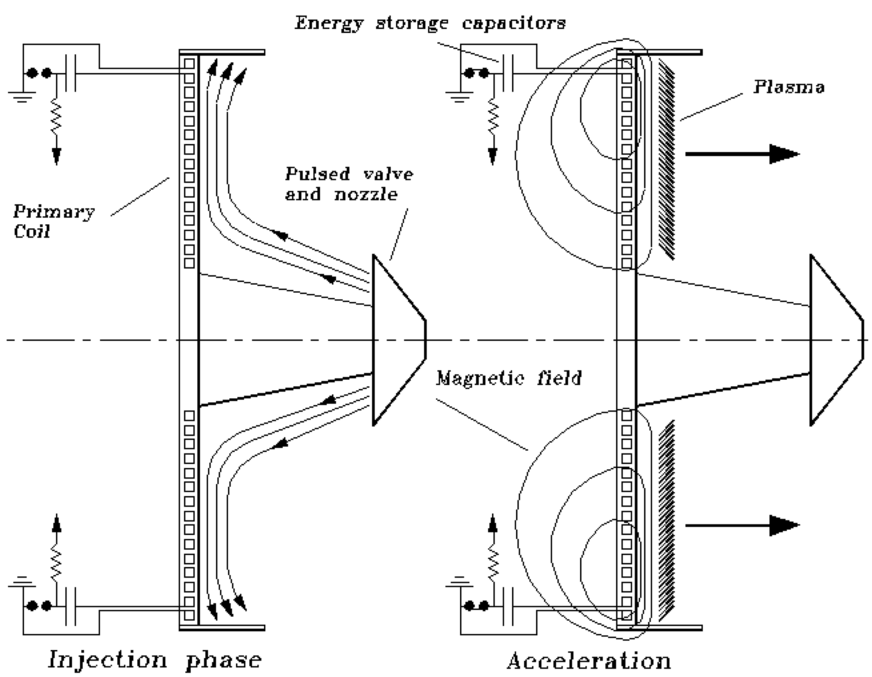


Figure 2. Fundamental Operation Sequence of the Pulsed Inductive Thruster Concept.
Left: Gas injection. Right: Breakdown and electromagnetic acceleration.

Specific impulse (I_{sp}) and thrust efficiency (η), are determined on single discharges by measurement of the injected propellant increment (Δm), the impulse (I) delivered to the thruster (measured as deflection of the thrust balance), and capacitor bank energy (E), with $E=CV^2/2$ (where C=capacitance and V=applied voltage):

$$I_{sp} = I / (g_c \, \Delta m) \quad \text{and} \quad \eta = I^2 / (C \, V^2 \, \Delta m)$$

with g_c being the appropriate unit conversion term for I_{sp} in m/s or lb_f-s/lb_m. When operated at a given rate in pulses per second (pps) ν, the equivalent steady-state thrust (T) and power (P) delivered to the thruster by the power supply are determined by $T = \nu I$ and $P = \nu E$.

III. MISSION ANALYSIS RESULTS FOR ROBOTIC PLANETARY MISSIONS

A. Introduction

A number of outer Solar System Nuclear Electric Propulsion (NEP) robotic planetary missions have been under consideration as follow-on missions beyond the Jupiter Icy Moon Orbiter (JIMO) mission. Typically, because of the need for short trip times to these distant destinations, the anticipated mission ΔVs and NEP total or "bus" power levels are significantly higher than those for the more near-term JIMO mission. However, Ion thrusters, such as the Herakles ion thruster proposed for use in the JIMO mission, are inherently low power density electric propulsion devices; at high vehicle power levels, the low power-per-thruster of an Ion thruster can result in the need for many thrusters with a corresponding increase in system mass and complexity. Thus, we see the potential advantage of the NuPIT thruster with its almost order-of-magnitude increase in power-per-thruster over the Ion thruster.

For these mission analyses, we selected two extreme cases of possible JIMO follow-on missions: the first is a Saturn Orbiter with Moon Tour with a somewhat higher mission ΔV than JIMO (e.g., for departure from Earth escape [C_3=0], the total mission ΔV is ca. 33 km/s for the Saturn mission vs only ca. 31 km/s for the JIMO mission),[3] and an Interstellar Precursor mission, with ΔVs ranging from about 28 to 53 km/s (for C_3=0) depending on the final Solar System escape velocity (V_{inf}).[4]

B. Nuclear Electric Propulsion Robotic Planetary Vehicle Assumptions

The NEP vehicle consists of a nuclear-electric power system, a main boom that is used both to support the power system's radiators and to separate the spacecraft systems and payload from the reactor's radiation, a power management and distribution (PMAD) system (high-power cabling between the power system and the spacecraft bus), and the spacecraft bus and payload. The spacecraft bus contains the reaction control system (RCS), various miscellaneous spacecraft systems (e.g. telecommunications, etc.), and the electric propulsion system.

1. NEP Vehicle Configuration

A conceptual schematic of the NEP vehicle is shown in Figure 3. Note that some electric propulsion options may require the use of a plume shield to protect sensitive spacecraft surfaces from the thrusters' exhaust plume. For example, spacecraft surfaces can be subjected to thermal contamination or physical erosion from the high-energy plume. Also, there is the risk of material contamination by condensable propellants. Material contamination can be a serious concern for optically-sensitive systems like camera lenses, and especially for radiators where the contaminant material can change the radiator's emissivity. Fortunately, all the propellants used in the electric propulsion systems described below have a high enough vapor pressure at typical spacecraft surface temperatures (e.g., room temperature and above) so as to not represent a serious material contamination issue. Also, placing the electric thrusters at the far end of the vehicle, with a 180^o field of view to space, ensures that no spacecraft surfaces will experience thermal contamination or erosion impact from the thruster plumes. Thus, we did not include plume shields in our analyses.

2. Nuclear Electric Power System

Mass scaling estimates for the nuclear-electric power system were provided by Lee Mason (GRC).[5] The power system consists of a reactor, shield, heat exchanger, dynamic (Brayton) thermal-to-electric power conversion system, and waste heat rejection system (radiators, pumps, fluid loops, etc.). The overall system specific mass (kg/kW_e) is shown in Figure 4 as a function of total or "bus" power level. Also shown are the specific masses of the main boom and PMAD systems discussed below. As is commonly observed in space-based NEP power systems, there is a significant economy-of-scale at higher powers. This often tends to drive the power requirement to high power levels in order to achieve short trip times, because the overall vehicle specific mass is inversely proportional to overall vehicle acceleration for a given I_{sp}. Also, higher powers reduce the effective specific mass of fixed-mass vehicle elements (e.g., payload), again favoring increased power for short trip times.

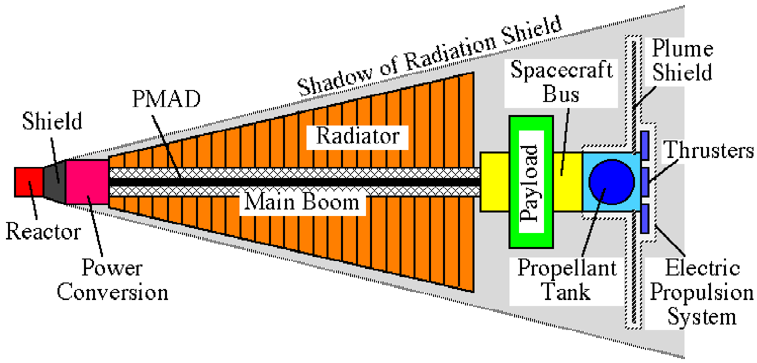


Figure 3. Conceptual Schematic of an NEP Vehicle.

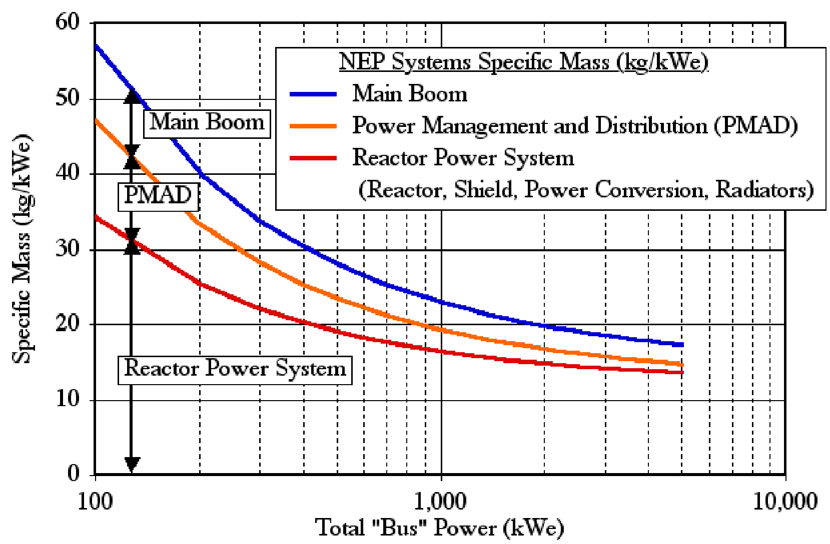


Figure 4. Assumed NEP Power System, Main Boom, and Power Management and Distribution (PMAD) System Specific Mass as a Function of Total "Bus" Power.

3. Main Boom

In an NEP vehicle, the main boom runs from the power system to the main spacecraft bus, as illustrated in Figure 3. The boom serves to both support the power system's radiators, as well as provide standoff distance to reduce the radiation load on the spacecraft systems and payload. Mass scaling estimates for the main boom were provided by Muriel Noca (JPL).[6] The mass of the main boom includes the forward (power-system end) equipment structure that attaches the boom system to the power system, the boom deployment canister (including drive motors, actuators, etc.), the boom structure itself, its mounting structure, cabling (PMAD) attachment hardware, thermal control, and micrometeoroid protection. As shown in Figure 4, there is only a slight economy-of scale due to the need to strengthen the boom structure to prevent buckling as the boom length increases with increased power.

For calculation purposes, the mass per unit length of the main boom was calculated based on the radiator length, which in turn is a function of the radiation shield's shadow half-angle (assumed to be 10^{o}).[5] An additional 10-m length of boom was then added to represent separation distance between the hot radiator and the spacecraft bus.

4. Power Management and Distribution (PMAD)

The power management and distribution (PMAD) system consists of high-power cabling between the power system and the spacecraft bus. Mass scaling estimates for the NEP PMAD system were provided by Lee Mason (GRC).[5] As with the main boom, the mass per unit length of the PMAD cabling was estimated based on the boom length, and an additional 10 m of PMAD was added to correspond to the extra 10 m of boom length.

As shown in Figure 4, there can be a substantial economy of scale for the PMAD system. Also note that in an NEP vehicle, the electric power produced by the turboalternators in the dynamic power system is in the form of high-voltage AC. By contrast, the electric power from the photovoltaic arrays in a Solar Electric Propulsion (SEP) vehicle is in the form of relatively low-voltage DC. Thus, for a given power level, the PMAD for an SEP "bus" will be somewhat heavier than for an NEP system, although the mass penalty for a low-voltage SEP system is somewhat offset by the additional electrical insulation and isolation required for a high-voltage NEP system. In fact, the greatest PMAD impact is in the electric propulsion system power processing units (PPUs) that convert the "bus" voltage to the form required by the electric thruster. For example, both an Ion and PIT thruster require very high-voltage DC. Thus, the PPU for an NEP system requires only a transformer (to convert the "bus" voltage level to that required by the thruster) and rectifiers (to convert the "bus" AC to DC). However, a PPU for an SEP vehicle requires an additional initial DC-to-AC invertor, along with a transformer and rectifier as in the NEP PPU, so an SEP-PPU is generally heavier than an NEP-PPU.[8] The topology of these various options is illustrated in Figure 5.[7]

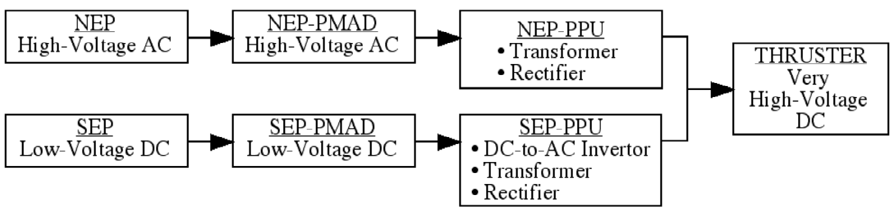


Figure 5. NEP and SEP Power System, PMAD, PPU, and Thruster Topology.

5. General Structural Mass Overhead, and Electric Power and Dry Mass Contingencies

The power system, main boom, and PMAD masses include their own structural overhead. For the other various spacecraft systems described next, we have assumed a general structural overhead of 26% of the dry mass of the various components based on JPL system design study practices.[6] Additionally, we have assumed a general structural overhead of 4% of the mass of propellants.[6] These structural overheads represent miscellaneous structure required to tie the various components together and to the NEP vehicle. Also, for those systems requiring electric power, we have assumed a power contingency of 10%. The total power is subtracted from the total "bus" power in determining the power available to the electric propulsion system.

Finally, all dry masses (other than the payload) have an additional 30% dry mass contingency.[6] This contingency, although large, is representative of current baselines for estimating spacecraft mass growth. Note that this is a relatively recent addition to mission analyses; many previous studies omitted this quantity.[8]

6. Chemical Hydrazine (N_2H_4) Reaction Control System (RCS)

A sequentially-recharged blowdown chemical hydrazine (N_2H_4) reaction control system (RCS) is used to maintain attitude control during all phases of the mission. The thrusters have an I_{sp} of 220 lb_f-s/lb_m; the mass of hydrazine is determined based on a 50-m/s ΔV for the fully loaded "wet" vehicle (with payload). The RCS components have a fixed mass of 18 kg and a tankage factor of 8.92% (including 0.73% for residuals and holdup). The RCS electric power requirement is 51 W_e (primarily to power the hydrazine thruster catalyst bed heaters); a 10% margin is added to this for a total of 56 W_e. Finally, as described above, an overall structural factor of 26% is applied to the dry mass of the RCS (including tankage), and an additional 4% structure is applied to the mass of hydrazine propellant.

7. Miscellaneous Spacecraft Systems

A number of "generic" robotic spacecraft systems[6] are included in the NEP vehicle's mass. These systems include attitude control systems (ACS), command and data handling (C&DH) systems, telecommunications (Telecom), vehicle cruise/emergency power (an allocation of 230 kg for some combination of batteries, radioisotope thermoelectric generators [RTGs] for deep-space missions, or solar arrays for inner Solar System missions), thermal control, and component-specific structure (108 kg). The total mass of the miscellaneous systems is 387 kg (without component-specific structure); 26% of this mass is added to represent miscellaneous structure, resulting in a total dry mass of 596 kg. The total power is 328 W_e; a 10% margin is added to this for a total of 361 W_e.

8. Science Payload

For these mission analyses, we have assumed a net science payload of 2,500 kg with a power requirement of 1 kW_e during NEP operation and cruise. This represents a significant mass of payload that could include landers and probes for the Saturn Mission, or an independent, self-contained spacecraft for the Interstellar Precursor Mission that could be jettisoned after the NEP vehicle's propulsive burn to continue on its own out of the Solar System and beyond. Also, a structural adaptor of 2.5% of the payload's mass[6] is added to tie the payload to the NEP vehicle. Additionally, the 30% dry mass contingency is added to the structural adaptor, but not the payload. Finally, a 10% margin is added to the payload's electric power requirement, such that the total power requirement is 1.1 kW_e.

9. Ion (Herakles) Propulsion System

The Ion thruster used as a baseline for comparison with the NuPIT system is based on the Herakles thruster proposed for the JIMO mission. We developed a series of scaling equations to calculate mass, power, efficiency, etc. for this engine based on a theoretical model developed by Thomas Randolph (JPL), Doug Fiehler (QSS Group), and Kurt Hack (GRC).[9] The thruster characteristics assumed for our analyses are illustrated in Figure 6. The thruster mass is 58 kg (including an 8-kg cable between the thruster and the PPU) with a beam diameter of about 69 cm (the overall outside diameter is roughly 94 cm). (The flattening out of thruster power above an I_{sp} of about 7,000 lb_f-s/lb_m is due to limiting of the beam current density.) Finally, the Herakles thruster is assumed to have a propellant throughput of 5,500 kg of Xe propellant.

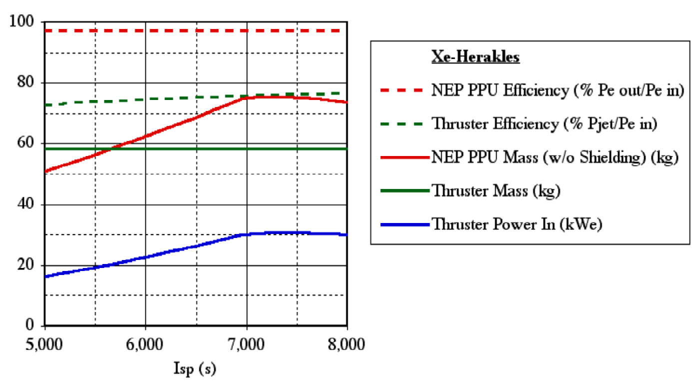


Figure 6. Ion (Herakles) Thruster and PPU Characteristics.

A mass list for a Xenon propellant storage and feed system was developed based on a design by Gani Ganapathi (JPL).[10] For the Xe-propellant Ion thruster, we can store the propellant as high-pressure, room-temperature supercritical gas and feed the high-pressure Xe to the thrusters. Also, as shown in the feed system schematic below, a Xenon Recovery System (XRS) using a sorption compressor is used to scavenge residual Xe in the main tank as it nears depletion. This has the effect of dramatically reducing the amount of residual Xe that would otherwise be unavailable as pressure drops in the main tank. A schematic of the Xe-Ion thruster propellant storage and feed system is shown in Figure 7. A summary of the various components is given in Table 1. Note that some electric heater power is required to maintain the system at room temperature to compensate for heat lost due to radiation in deep space far from the sun,[11] as well as additional electric power corresponding to the heat of desorption of xenon in the Xenon Recovery System (XRS). Although this desorption power would only be needed near the end of the mission when the XRS is operating, we have assumed that it would be required at all times; thus, the various heater powers and XRS desorption power (equal to the XRS heat of desorption [440 J/g] multiplied by the mass flow rate [g/s] of propellant into the Ion thrusters) are subtracted from the total "bus" power. Also, note that there are both fixed-mass (or power) terms in the mass and power scaling relationships, as well as terms dependant on propellant mass (M_p) and on surface area ($M_p^{2/3}$). Finally, a large number of ion thrusters are required on a MW_e-class vehicle. To facilitate vehicle integration and packaging in the launch vehicle, the Ion thrusters are collected into two clusters or "pods," with each pod having a gimbal and associated flex lines. The Ion thrusters are mounted directly on the pod structure (i.e., not individually gimbaled).

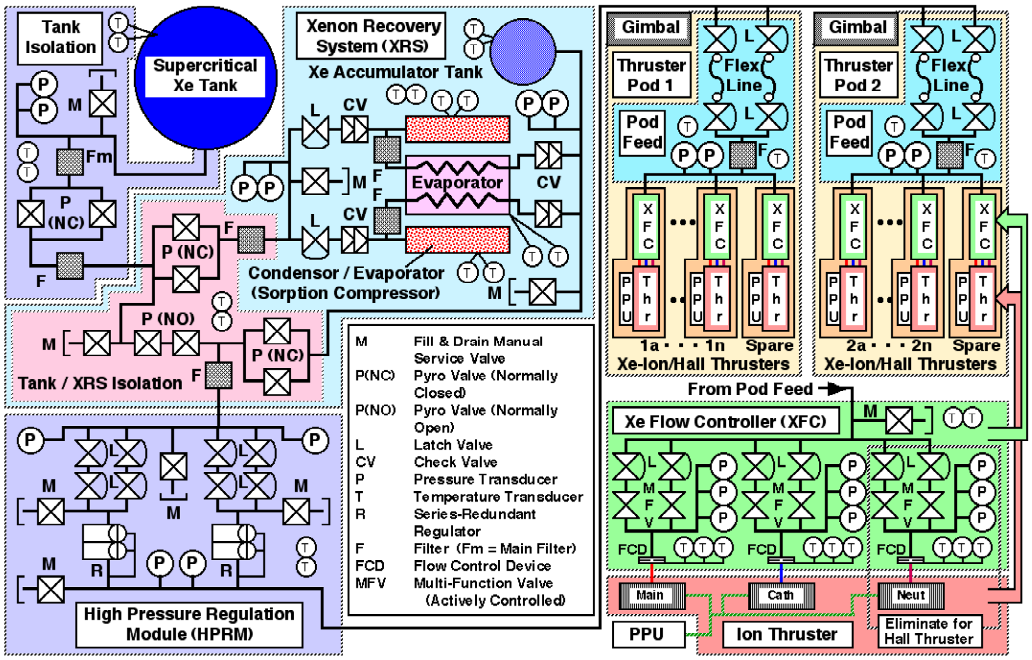


Figure 7. Xe-Propellant Ion (Herakles) Thruster Storage and Feed System Schematic.

Table 1. Summary of Propellant Storage and Feed System Mass and Power Scaling for the Xe-Ion (Herakles) Thruster System. (All masses in kg and powers in W_e.)

Component	Fixed Mass, Power, or Parts Term	% of M_p Term	% of $(M_p)^{2/3}$ Term
Storage & Feed System up to Pod Feed			
Mass	63.85	2.5770%	1.5232%
Power	6.3		7.7371%
Parts Count	55		
Pod Feed System (1 per Pod)			
Mass (Includes Gimbal)	13.90		
Power	0.0		
Parts Count	12		
Thruster Feed System (1 per Thruster)			
Mass	14.22		
Power	0.0		
Parts Count	26		

10. Thruster Number Calculation Methodology

For both the Ion and NuPIT propulsion systems, the number of thrusters (and their corresponding PPUs, feed systems, etc.) required in each pod will be a function of the total "bus" power and the power-per-thruster of each thruster. Specifically, we first calculate the net total power available to the thrusters by subtracting all the various system electric powers (e.g., systems, payload, RCS catalyst bed heaters, propellant storage and feed system heaters, etc.) from the total "bus" power. We then determine the power-per-PPU for the thrusters, which is equal to the thruster's power-per-thruster (typically a function of the thruster's I_{sp}) divided by the PPU's efficiency. The number of operating thrusters is then simply the rounded-up integer of (available power)/(power-per-PPU). For the Ion thruster system with two pods, it may be necessary to add one additional thruster so that each pod has the same number of operating engines so as to ensure overall thrust balance (e.g., left/right symmetry). Also, for thrusters with short lifetimes (i.e., low propellant throughput-per-thruster), it may be necessary to add extra complete sets of thrusters so as to consume all of the propellant for the mission. Finally, one extra thruster (and its PPU) is added to each pod as a redundancy spare. As a specific numerical example, if we have a 6,000-lb_f-s/lb_m I_{sp} Ion propulsion system for a 1,000-kW_e total "bus" power vehicle, the power available to the thrusters is 998.2 kW_e. The power-per-PPU is 23.9 kW_e, so there are (998.2/23.9)=41.7 operating, which is rounded up to 42 thrusters. (This corresponds to each of the 42 thrusters operating at 99.3% of their full rated power.) For the Ion system, we have two pods, which implies a need for an even total number of operating thrusters. In this case, we already have an even total number (i.e., 42 total, with 21 in each pod), so we do not need to add an additional thruster for thrust balance. (If there were only one pod, then an even or odd number of operating thrusters would be allowed based on the assumption that any gimbaling required to compensate for thrust imbalance would be small, and thus would not impact overall performance, because all the thrusters would be relatively near the vehicle's thrust centerline. By contrast, thrusters in separate pods would be far from the vehicle thrust centerline, and thus produce a thrust imbalance moment-arm that could not be corrected by gambling without unduly impacting performance.)

The Ion thrusters have a long lifetime, so only one set of operating thrusters is required to consume all of the Xe propellant. Otherwise, it would be necessary to add additional sets of 42 thrusters (again distributed evenly between the pods) until a cumulative total throughput (lifetime) was reached that consumed all of the required propellant. In this case, the number of sets (with 42 thrusters in each set) would be the rounded-up integer value of (total propellant mass)/(total throughput of 42 thrusters). Again, one extra spare thruster (and PPU) would be added to each pod for redundancy. Finally, when we consider MW_e-class electric propulsion vehicles, systems using thrusters with modest power-per-thruster and modest lifetimes (throughput) will only require one set of thrusters because the large total number of thrusters required to consume the available MW_e of power naturally results in sufficient numbers of thrusters to consume the available amount of propellant. Typically, the need for additional sets of thrusters arises only when we have the combination of high power-per-thruster and low throughput-per-thruster, as can be the case with magnetoplasmadynamic (MPD) thrusters.

11. NuPIT Propulsion System

The NuPIT thruster was described in detail above. For these analyses, we have assumed a thruster mass of 550 kg, which includes the thruster and gas injection valve, as well as the electric switches and capacitor banks for the thruster. The thruster has a diameter of about 1.5 m. The NuPIT nominal (baseline) I_{sp} assumed for these analyses is 6,000 lb_f-s/lb_m, with an efficiency of 72.0% at 6,000 lb_f-s/lb_m I_{sp}. We also evaluate the NuPIT operating at I_{sp} values ranging from 4,000 to 8,000 lb_f-s/lb_m, with efficiencies ranging from 68.7% to 75.3%, respectively. The lifetime of the thruster is based on an assumed electric switch and gas valve life of 10^{10} pulses at 50 pulses per second, or 6.34 years. The total propellant throughput of the thruster is based on its lifetime and average propellant mass flow rate, which is in turn based on the thruster's I_{sp} and average "jet" power (thruster input electric power multiplied by the thruster's efficiency); for example, the throughput is 16,660 kg per thruster at an I_{sp} of 6,000 lb_f-s/lb_m.

However, the thruster mass quoted above does not include a power processing unit (PPU) that converts the NEP "bus" electric power (e.g., high-voltage AC) to the 15 kV DC power required to charge the capacitor bank. Based on mass and efficiency scaling models for Ion thruster PPUs, we have estimated the mass of a NuPIT NEP PPU as 368 kg with an efficiency of 97.4%. Lastly, we have added a "generic" shielding mass for the Ion and $ALFA^2$ PPUs that corresponds to 0.92% of the NEP PPU mass.[10] This shielding is intended to protect the PPU electronics from the general space environment, and not the much more severe Jupiter radiation environment encountered in a JIMO-type mission.

Finally, a mass list for a propellant storage and feed system was developed based on the Ion thruster schematic shown above. For the NH_3-propellant NuPIT, we can store the propellant as liquid ammonia and feed the high-pressure ammonia vapors (ammonia vapor pressure is about 10 bar at room temperature) to the thrusters. We also added gas plenum tanks so as to minimize ripples in the upstream pressure due to the pulsed nature of the NuPIT system. Finally, the high power-per-thruster of the NuPIT makes it possible to have a small number of individually-gimbaled NuPIT engines in a single (non-gimbaled) thruster pod. A schematic of the NH_3-NuPIT propellant storage and feed system is shown in Figure 8. A summary of the various components is given in Table 2. Note that some electric power is required to maintain the system at room temperature (as with the Xe system), as well as additional electric power corresponding to the heat of vaporization of ammonia (i.e., the power is equal to the heat of vaporization [1,368 J/g] multiplied by the mass flow rate [g/s] of propellant into the NuPIT thrusters).

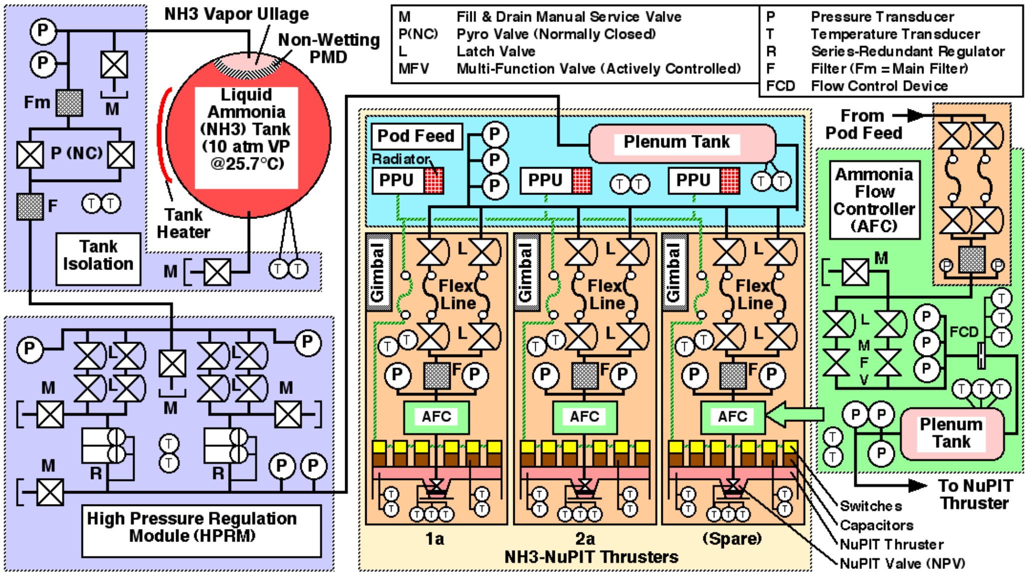


Figure 8. NH_3-Propellant NuPIT Storage and Feed System Schematic.

Table 2. Summary of Propellant Storage and Feed System Mass and Power Scaling for the NH_3-NuPIT System. (All masses in kg and powers in W_e.)

Component	Fixed Mass, Power, or Parts Term	% of M_p Term	% of $(M_p)^{2/3}$ Term
Storage & Feed System up to Pod Feed			
Mass	31.22	2.5547%	2.8758%
Power	0.0		14.7446%
Parts Count	29		
Pod Feed System (1 per Pod)			
Mass	3.18		
Power	0.0		
Parts Count	5		
Thruster Feed System (1 per Thruster)			
Mass (Includes Gimbal)	20.9		
Power	0.0		
Parts Count	25		

C. Mission Analysis Results for NEP Robotic Planetary Missions

We evaluated two robotic missions in these analyses. The first was a Saturn Orbiter Mission that includes an extensive tour of Saturn's moons. The second mission was an Interstellar Precursor Mission to 200 Astronomical Units (AU) with a Solar System escape velocity (V_{inf}) of either 5 or 10 AU/year. For each of these missions, we considered two primary factors in evaluating the mission benefits of the NuPIT propulsion system:

1. Total Initial Mass in Low Earth Orbit (IMLEO) and Trip Time
2. Propulsion System Storage and Feed System Complexity (as represented by a parts count)

1. Initial Mass in Low Earth Orbit (IMLEO) vs Trip Time for the Saturn Orbiter with Moon Tour Mission

Figure 9 illustrates the IMLEO versus trip time performance for the Saturn Orbiter with Moon Tour mission using the Ion (Herakles) and NuPIT thruster options over a range of I_{sp} from 6,000 to 8,000 lb_f-s/lb_m. These analyses show the importance of a high I_{sp} for high-ΔV missions; for example, there is a pronounced difference between the 6,000- lb_f-s/lb_m I_{sp} curves and the 7,000 or 8,000- lb_f-s/lb_m I_{sp} curves. For this high-ΔV (ca. 41 km/s total leaving from a 1,000-km low Earth orbit [LEO][12]) mission, the NuPIT system is very competitive with the Ion system, having a mass only about 10% higher than the Ion system, which is probably within the uncertainty of the mass estimates for the two systems. Nevertheless, the superior thruster efficiency and I_{sp} potential of the Ion system provides superior performance in terms of initial launch mass and trip time.

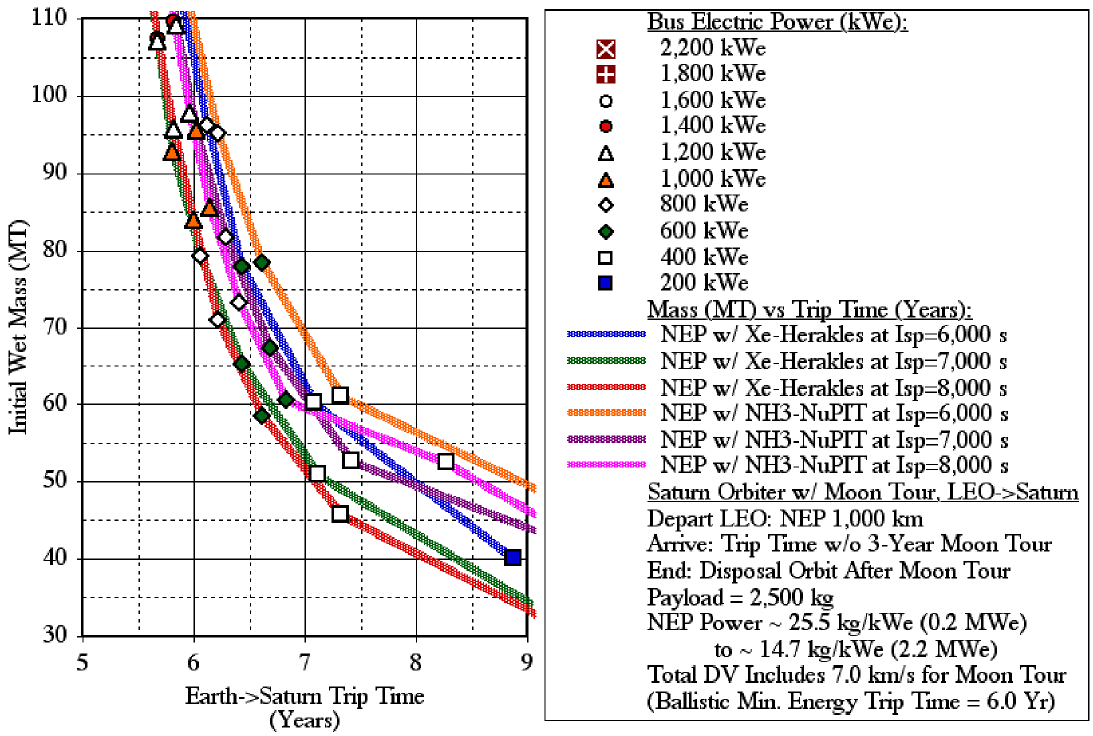
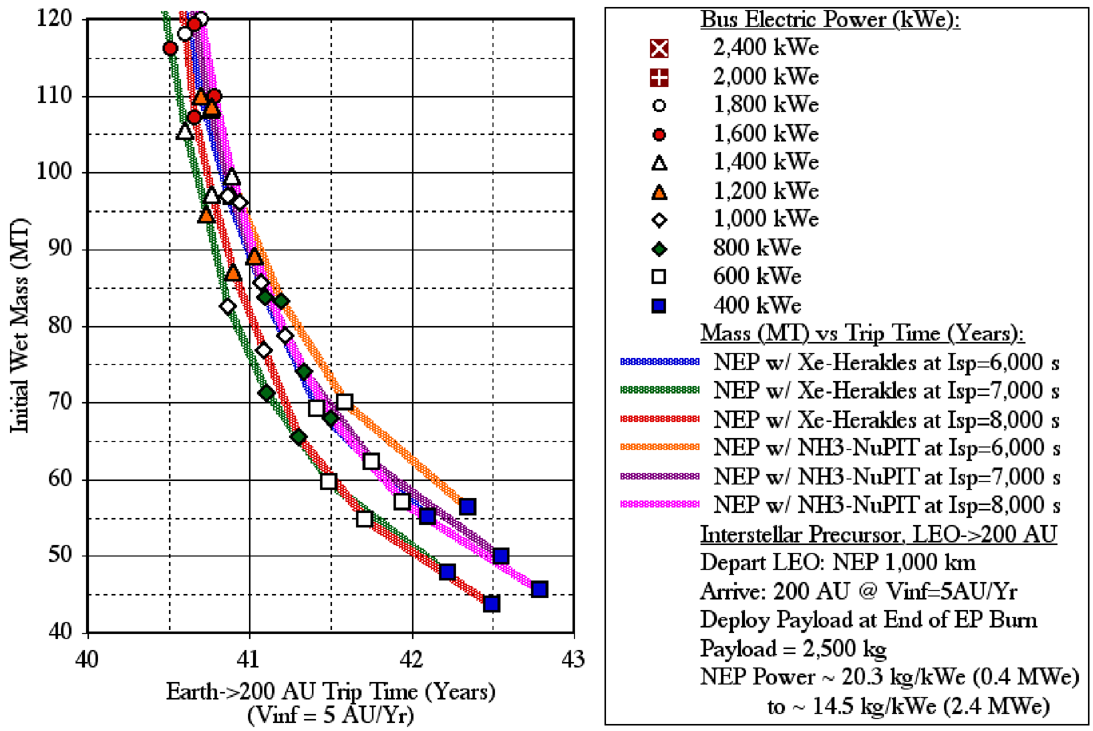


Figure 9. Variation in IMLEO and Trip Time for the Saturn Orbiter with Moon Tour Mission.

2. Initial Mass in Low Earth Orbit (IMLEO) vs Trip Time for the Interstellar Precursor Mission

Figure 10 illustrates the IMLEO versus trip time (to 200 AU) performance for Interstellar Precursor mission with the Ion (Herakles) and NuPIT thruster types over a range of I_{sp} from 6,000 to 8,000 lb_f-s/lb_m. Performance for two cases is given; the first is for a slower final velocity (V_{inf}) of 5 Astronomical Units (AU) per year, and the second is for a faster V_{inf} of 10 AU/year. Not surprisingly, the slower mission is less demanding in overall mass and required power. As with the Saturn Mission, these analyses show the importance of a high I_{sp} for high-ΔV missions; for example, there is a modest difference between different I_{sp} values for the 5 AU/Yr mission which has a ΔV on the order of 35 km/s leaving from a 1,000-km LEO (comparable to the Saturn Mission total ΔV); by contrast, there is a strong I_{sp} dependence for the 10 AU/year mission with a ΔV around 60 km/s (leaving from a 1,000-km LEO).

Again, as with the Saturn Mission, we see that the NuPIT system is quite competitive with the Ion system for the lower ΔV case (i.e., 35 km/s at 5 AU/year). However, for the very high-ΔV 10 AU/year mission with its large propellant requirement, the NuPIT is at a significant disadvantage because of its limited lifetime (throughput). This is because the Ion thruster system has a very large number of engines operating at MW_e power levels (e.g., 16 at an I_{sp} of 7,000 lb_f-s/lb_m), so that even with a modest assumed lifetime or throughput of 5,500 kg per thruster, all of the propellant can be consumed with a single set of Ion thrusters. By contrast, the NuPIT, with its high power-per-thruster, only requires 5 engines operating at 1 MW_e. In this case, the NuPIT, even with a large throughput (16,660, 12,523, and 9,805 kg per thruster at an I_{sp} of 6,000, 7,000, and 8,000 lb_f-s/lb_m, respectively) corresponding to the assumed electric switch and gas valve life of 10^{10} pulses, has insufficient cumulative throughput to eliminate the need for two thruster sets at all the power levels shown in Figure 10. Thus, the need for additional complete sets of NuPIT engines results in a significantly heavier vehicle, with correspondingly poorer performance, than the Ion thruster system.



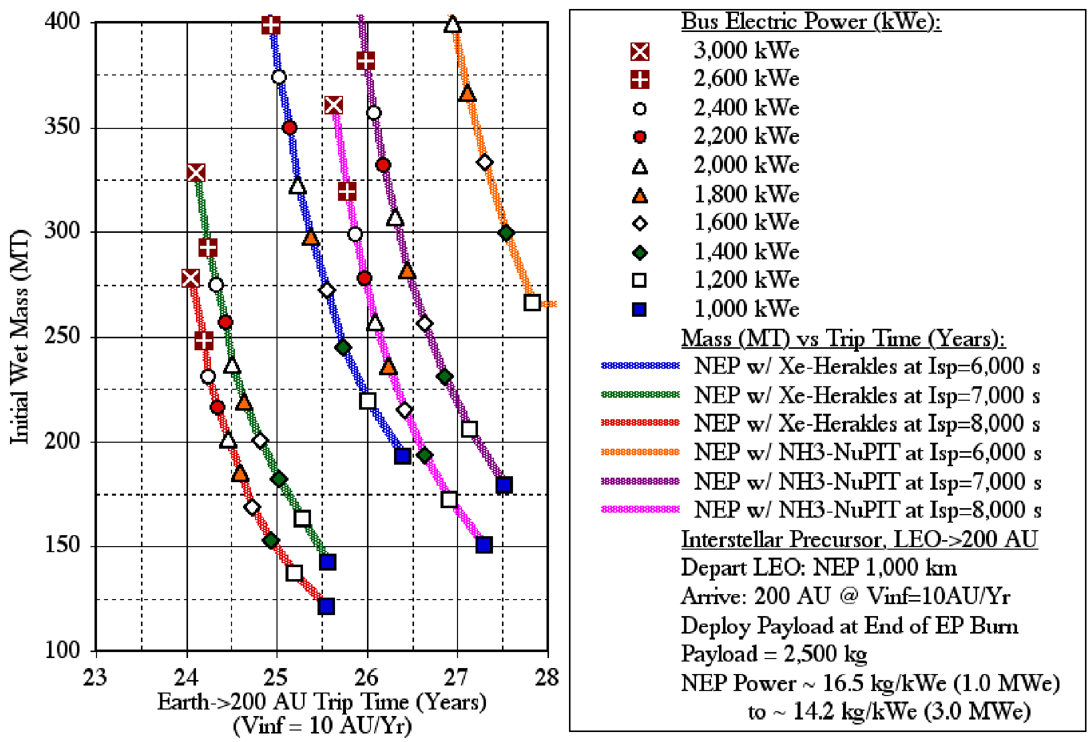


Figure 10. Variation in IMLEO and Trip Time for the 200 AU Interstellar Precursor Mission. (Upper Figure V_{inf} = 5 AU/Year; lower Figure V_{inf} = 10 AU/Year.)

3. Propulsion System Complexity

Thus far we have concentrated on mass and trip time as the traditional figures of merit in determining the benefit (i.e., feasibility) of the new propulsion technologies embodied in the ALFA[2] system. However, another element of mission feasibility is the overall system "complexity." In these high-level system analyses, "complexity" is quantified as a measure of propulsion system parts count. No matter how it is quantified, high "complexity" is generally considered undesirable because of its perceived impact on decreasing system reliability and increasing flight system integration and test costs.

Figure 11 illustrates the propulsion system parts count (e.g., thrusters and propellant storage and feed system components [valves, regulators, filters, etc.]) as a function of the total or "bus" power level for the 5 AU/year 200 AU mission. (The situation for the other missions at the same power level would be similar because the parts count is driven by the number of thrusters required, which is in turn a function of the power-per-thruster.) In this case, the parts count is used as a measure of system "complexity." For example, propulsion systems like the high-power (e.g., 200 kW_e per thruster) NuPIT have a relatively small parts count, and thus ultimately "complexity," for components like the number of thrusters, valves, etc. By contrast, the Ion system, with its relatively low power-per-thruster (e.g., 30.4 kW_e per thruster at 7,000- lb_f-s/lb_m I_{sp}) has an enormous parts count. Also note the increase in parts count for the Ion thruster at 6,000 lb_f-s/lb_m I_{sp}; this is due to the dependence of power-per-thruster on I_{sp} in Ion thrusters, as illustrated in Table 3.

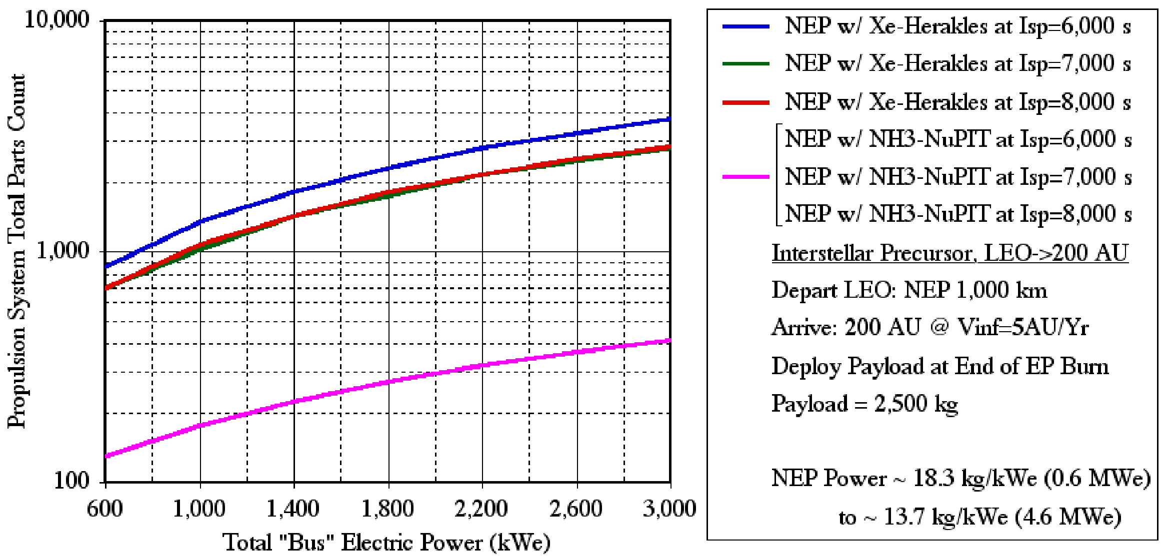


Figure 11. Electric Propulsion System Parts Count vs Total "Bus" Electric Power for the NEP 200 AU Interstellar Precursor Mission (V_{inf} = 5 AU/Year).

Table 3. Power-per-Thruster and Number of Thrusters Required for a 1-MW_e NEP System.

Thruster	Ion (Herakles)			NuPIT
I_{sp} (lb_f-s/lb_m)	6,000	7,000	8,000	6,000
Power into Thruster (kW_e)	22.7	30.4	30.1	200.0
Power into PPU (kW_e)	23.3	31.2	30.9	205.3
Number Running at 1 MW_e Total "Bus" Power	43	32	33	5
Number per Pod	22 + 1 Spare	16 + 1 Spare	17 + 1 Spare	5 + 1 Spare
Number of Pods	2	2	2	1
Total Number of Thrusters	46	34	36	6

Also, although not explicitly considered in detail in this study, there is the non-trivial issue of packaging a large number of thrusters in the Earth-launch vehicle launch shroud. For example, even though the Herakles Ion thruster has roughly 2/3 the diameter of the NuPIT, when we take into account the need to have roughly 6.6 times as many Herakles as NuPIT thrusters (e.g., due to the difference in power-per-thruster), we find that there is a significant packaging and integration challenge for Ion thrusters in MW-class electric propulsion systems, as illustrated in Figure 12 below.

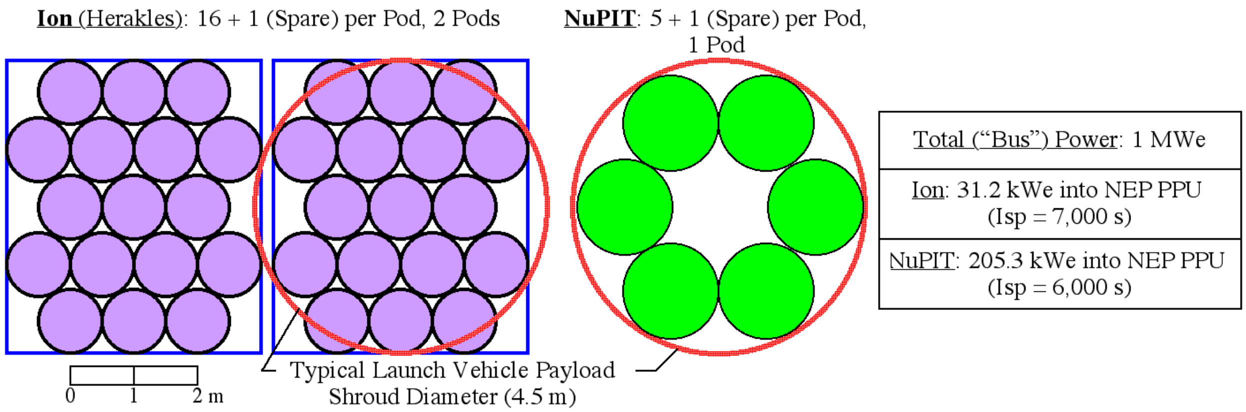


Figure 12. Size Comparison Between Ion (Herakles) and PIT (NuPIT) Thrusters for a 1-MW_e Total "Bus" Power.

D. Parametric Investigation of NuPIT Performance for NEP Robotic Planetary Missions

As described above, we found that the performance of the NuPIT system for high-ΔV missions is somewhat less than that found for the Ion system for the nominal NuPIT thruster characteristics even at high I_{sp} (e.g., up to 8,000 lb_f-s/lb_m I_{sp} for the NuPIT). In this section, we will investigate approaches that could improve the performance of the NuPIT system in order to make it comparable to or superior to the Ion system.

1. Variation in NuPIT Thruster and PPU Specific Mass

For example, at a nominal I_{sp} of 6,000 lb_f-s/lb_m, the specific masses of the NuPIT thruster and PPU are 2.750 and 1.778 kg/kW_e, respectively, for a total of 4.528 kg/kW_e. Figure 13 illustrates the performance impact that reductions in the thruster and PPU specific mass would have on the Saturn Mission described above. In this case, both the thruster and PPU specific mass were reduced proportionally to the total shown in the Figure. We see here that the NuPIT system has a mass and trip time comparable to the Ion system for a NuPIT system with a combined thruster and PPU specific mass of 2 kg/kW_e for a NuPIT at an I_{sp} of 6,000 lb_f-s/lb_m, or 3 kg/kW_e at an I_{sp} of 7,000 lb_f-s/lb_m and 3.5 at 8,000 lb_f-s/lb_m. This suggests that considerable performance leverage can be obtained by attacking the overall thruster and PPU mass.

2. Variation in Thruster Efficiency

Similarly, the nominal NuPIT efficiency is 72% at an I_{sp} of 6,000 lb_f-s/lb_m. (For comparison, the Ion thruster has an efficiency of 76.7% at an I_{sp} of 8,000 lb_f-s/lb_m.) Figure 14 illustrates the performance impact that improvements in the thruster efficiency would have on the Saturn Mission. In this case, improvements in the NuPIT efficiency have modest impact on performance. In contrast to the specific mass analysis above, this evaluation of thruster efficiency suggests minimal benefits for aggressively pursuing major improvements in NuPIT efficiency.

3. Variation in Thruster Lifetime (Throughput)

As discussed above, the NuPIT requires two sets of thrusters for the very high-ΔV 10 AU/year Interstellar Precursor mission, with its large propellant requirement, because of the NuPIT's limited throughput (e.g., 12,523 and 9,805 kg per thruster at an I_{sp} of 7,000 and 8,000 lb_f-s/lb_m, respectively). Figure 15 illustrates the performance for this mission with the NuPIT throughput increased sufficiently to allow operation with one set of thrusters at the lowest power level (i.e., with the fewest number of thrusters required for power). For this mission, increasing the nominal throughput by only around 50% (i.e., to $1.5x10^{10}$ pulses) makes it possible to operate on only one set of NuPIT engines, and thus have performance comparable to the Ion thruster system.

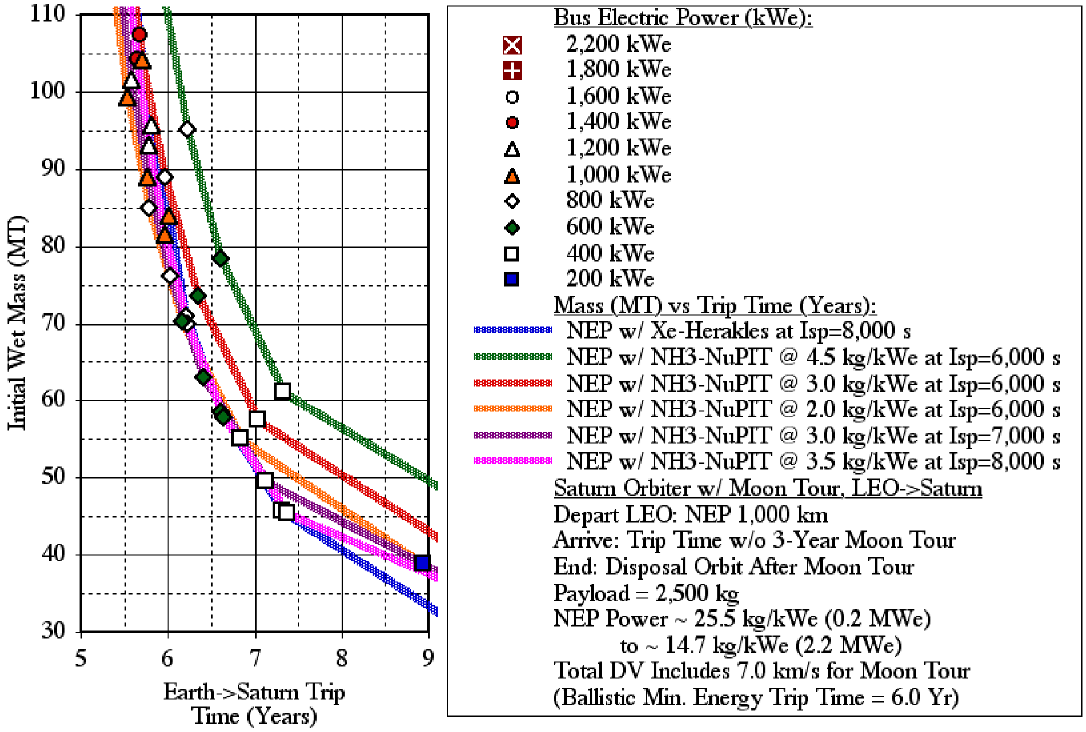


Figure 13. Variation in IMLEO and Trip Time for the Saturn Orbiter with Moon Tour Mission with NuPIT Thruster and PPU Specific Mass Variations.

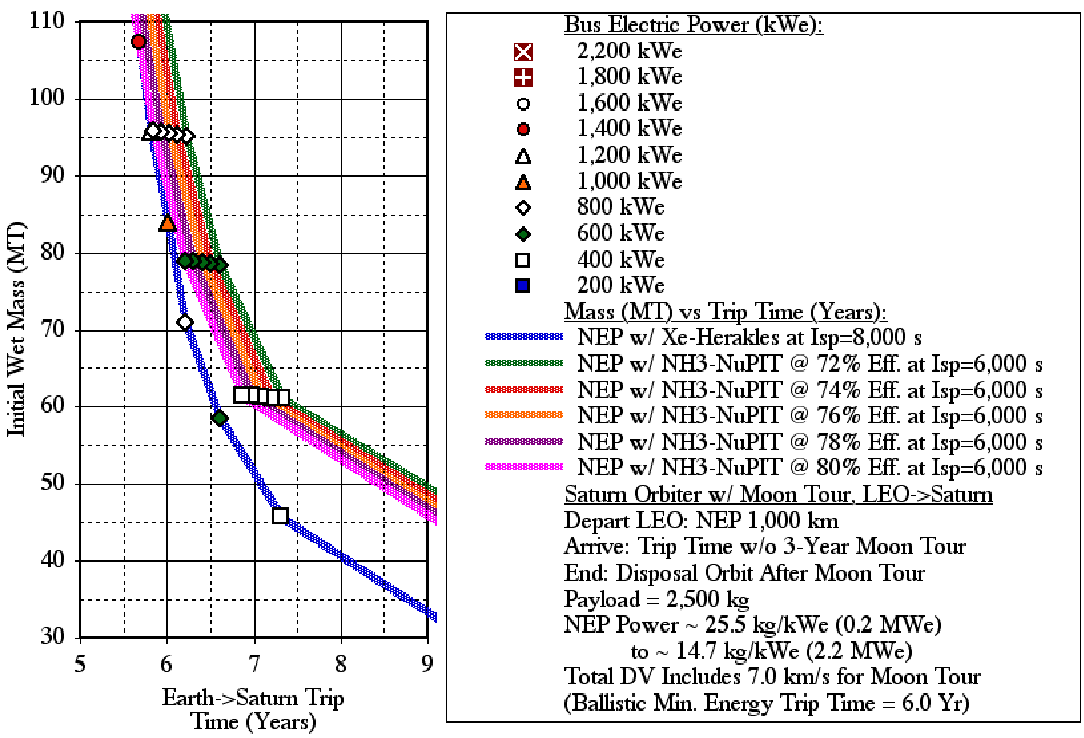


Figure 14. Variation in IMLEO and Trip Time for the Saturn Orbiter with Moon Tour Mission with NuPIT Thruster Efficiency Variations.

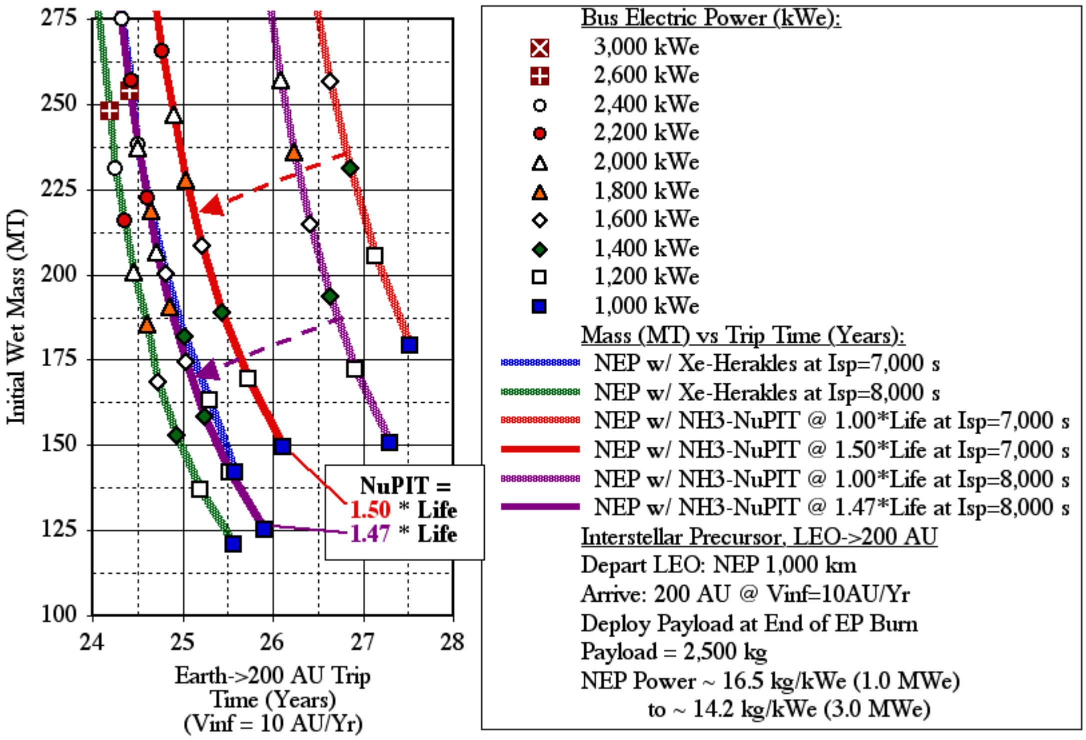


Figure 15. Variation in IMLEO and Trip Time for the 200 AU Interstellar Precursor Mission at V_{inf} = 10 AU/Year with NuPIT Thruster Lifetime (Throughput) Variations.

E. Summary of Mission Analysis Results of NuPIT for NEP Robotic Planetary Missions

As shown above, these high-ΔV outer Solar System missions, where the vehicle spends most of its flight time in heliocentric space, tend to optimize towards high values of I_{sp}, with an optimum in the 7,000 to 8,000 lb_f-s/lb_m I_{sp} range. This may represent a significant challenge for the NH_3-NuPIT system; however, as will be shown below, missions within the inner Solar System tend to favor lower I_{sp} values, because a greater fraction of the time is spent in planetary gravity wells. In these cases, the lower I_{sp} produces more thrust (at a given power level), so as to potentially reduce the trip time. Nevertheless, even if limited to an I_{sp} of 7,000 lb_f-s/lb_m, the NH_3-NuPIT system is still quite competitive with a high-performance Ion thruster system, with the very significant advantages of easier spacecraft integration within the launch vehicle (i.e., smaller thruster pods), and especially reduced system complexity as evidenced by a nearly order-of-magnitude reduction in the number of system components for the NuPIT system as compared to the Ion system.

IV. MISSION ANALYSIS RESULTS FOR THE NEP MARS CARGO MISSION

A. Introduction

Although intended primarily for high-power NEP robotic planetary exploration applications, high-power-per-thruster systems like NuPIT can also be used for high-power electric propulsion (EP) Cargo Missions supporting Human exploration of the Moon or Mars. In these “split” mission scenarios, time-insensitive cargo (e.g., propellants, landers, surface habitats, etc.) is transported by a high-I_{sp} (i.e., fuel-efficient), although slow (i.e., low-T/W), EP vehicle from LEO to an orbit about the target body. A fast (i.e., high-T/W) vehicle is used to transport the crew from LEO to a rendezvous with the Cargo vehicle, where the pre-delivered supplies are then used for exploration of the target. The crew then returns to Earth; in some scenarios this may be accomplished by using propellants delivered by the Cargo vehicle. For lunar missions, the Cargo vehicle is typically re-used; by contrast, Cargo vehicles for Mars missions are typically left in Mars orbit.

For the Mars Cargo Mission, we have assumed the use of a one-way (expendable) megawatt-class NEP Cargo Vehicle for transport of payload from LEO to a 6,000-km altitude low Mars orbit (LMO). Typical ΔVs for low-T/W LEO-to-LMO transfers are on the order of 16 km/s.[4] This LMO is at the same altitude as the inner, and larger moon of Mars, Phobos. This high-altitude LMO, rather than a low-altitude (e.g., 400-km altitude) LMO was chosen to make it possible to support exploration of Phobos, with special emphasis on Phobos as a potential extraterrestrial resource of water that could be processed to produce chemical (O_2/H_2) propellants. For example, after delivery of the cargo payloads, the EP Cargo Vehicle could land on Phobos and use its power to support mining, ore processing, water electrolysis, and so on. (Note that the chemical RCS thrusters might be needed for the landing, however, because the vehicle acceleration from the electric thrusters might be too small even for the micro-gravity surface gravity of Phobos.) Finally, the mass of Phobos could provide shielding mass to prevent radiation from the NEP vehicle from damaging other assets in Mars space.[8]

Also, we have chosen an Earth-to-Mars trip time goal of 2.2 years to match the Earth-Mars synodic period. This makes it possible to launch the Cargo Vehicles during one trans-Mars injection (TMI) opportunity, travel to Mars, perform Mars orbit insertion (MOI), and check out all the payload systems prior to launching the crew during the next Mars TMI opportunity.

B. Nuclear Electric Propulsion Mars Cargo Vehicle Assumptions

1. NEP Systems

The assumptions made for the NEP vehicles described above are again used in this section.

2. Mars Cargo Payload

As with the Human missions to the Moon, architectures for Human exploration of Mars are still under study. For our EP Cargo Vehicle analyses, we have assumed a 63.892 metric ton (MT) payload derived from the NASA Human Exploration of Mars Design Reference Mission (DRM) Version 3.0.[13] This payload corresponds to delivery of an Earth Return Vehicle (ERV) into Mars orbit. In the nominal DRM 3.0 Mission scenario, a Nuclear Thermal Propulsion (NTP) stage is used for Earth escape and trans-Mars injection (TMI); the NTP stage is then jettisoned, and the total payload (74.072 MT for the ERV and entry Aeroshell) is aerocaptured for Mars orbit insertion (MOI). For the electric propulsion options, the Aeroshell (10.180 MT) required for the nominal NTP ERV Cargo Mission is removed, because the EP Cargo Vehicle places the ERV directly into Mars orbit. There is also a second NTP Cargo Vehicle launch with a payload consisting of a 66.043-MT Cargo Lander (CL) (to place an Ascent Vehicle and other elements on the surface) that is aerobraked directly to the martian surface. Finally, the Crew Vehicle (with landers) is sent by NTP on a fast trajectory to Mars where the Crew Vehicle aerocaptures into Mars orbit.

Interestingly, if a slow, minimum-energy (Hohmann) trajectory is acceptable for the crew, an aerobraked chemical (O_2/H_2) propulsion Crew Vehicle can provide comparable IMLEO to the NTP Crew Vehicle. However, the real advantage of NTP is its combination of high-T/W and high-I_{sp} (projected[13] to be in the range of 940-960 lb_f-s/lb_m); this makes it possible to fly fast, high-energy trajectories that have much shorter flight times than that for the minimum-energy trajectory (e.g., 130-180 days versus the ideal 259 days, respectively, for the Earth-to-Mars step) without suffering from an excessive IMLEO.

C. Mission Analysis Results for the NEP Mars Cargo Mission

As with the Lunar Cargo Mission, we considered the following factors:

1. Total Initial Mass in Low Earth Orbit (IMLEO) and Trip Time
2. Propulsion System Complexity (as represented by a parts count)
3. Vehicle Power Level Required for a Given Earth-to-Mars Trip Time (nominally 2.2 year)

1. Initial Mass in Low Earth Orbit (IMLEO) vs Earth-to-Mars Trip Time

As with the NEP robotic missions, we again see the similarity in performance between the NuPIT and Ion thruster systems for an NEP Cargo Mission, as shown in Figure 16. For this mission, with a ΔV typically on the order of 16 km/s, the optimum I_{sp} is around 6,000-7,000 lb_f-s/lb_m; higher I_{sp} values result in the need for a higher power to achieve a given trip time such that the increase in power system mass (and corresponding increase in thruster, PPU, etc. mass) essentially negates any propellant (and propellant tankage) mass savings afforded by the higher I_{sp}. Also, it is interesting to note that in the NuPIT NEP system, the optimum I_{sp} is around 6,000 lb_f-s/lb_m rather than the 7,000 lb_f-s/lb_m of the Ion system; this effect is probably due to the interaction of the thruster efficiency and thrust. More specifically, vehicle thrust (and thus trip time) is proportional to the propulsion system's exhaust or "jet' power divided by I_{sp}; thus, the NuPIT, with its lower efficiency, needs a lower I_{sp} to have the same thrust as an Ion thruster at a higher efficiency and I_{sp}.

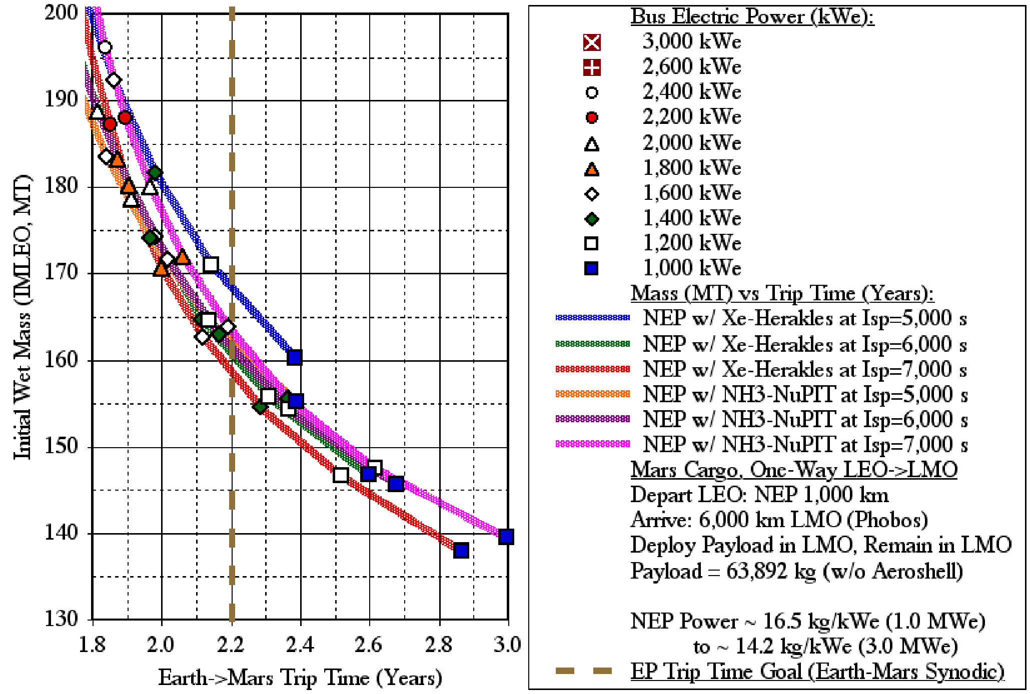


Figure 16. Variation in IMLEO and Trip Time for the NEP Mars Cargo Mission.

2. Propulsion System Complexity

Figure 17 shows the general trends seen previously where the low power-per-thruster of Ion thrusters results in almost an order-of-magnitude increase in propulsion system storage and feed system parts count and thus complexity over the NH_3-NuPIT system. Also shown in Figure 17 are the power levels required to achieve the target Earth-to-Mars trip time of 2.2 years. Finally, because we have kept the NuPIT power-per-thruster constant, independent of I_{sp}, the parts count curves for the three NuPIT I_{sp} cases fall on top of each other.

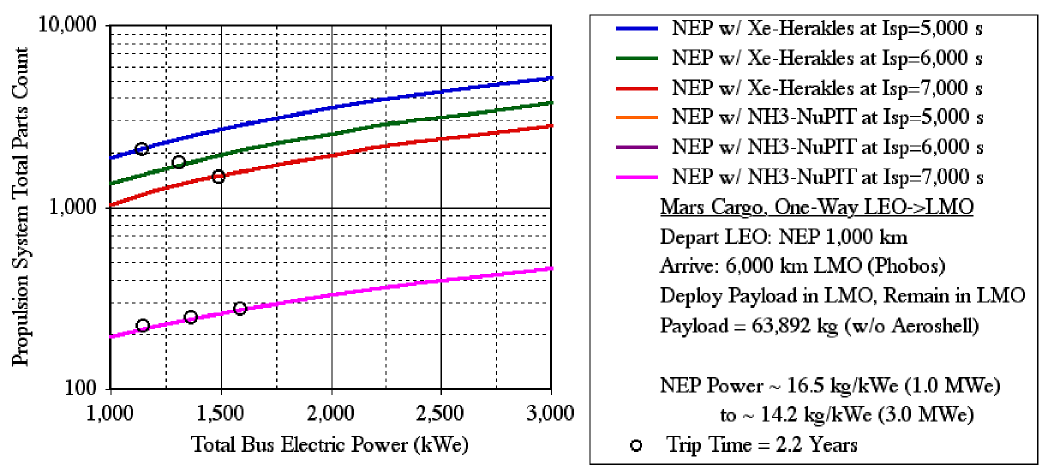


Figure 17. Electric Propulsion System Parts Count vs Total "Bus" Electric Power for the NEP Mars Cargo Mission.

3. Vehicle Power Level Required for a 2.2-Year Earth-to-Mars Trip

As discussed above, we chose an Earth-to-Mars trip time goal of 2.2 years to match the Earth-Mars synodic period. Figure 18 illustrates the general trend of requiring higher power at higher I_{sp} values in order to achieve a desired trip time. We also see the similarity in IMLEO and power between the NH_3-NuPIT and the Ion thruster systems. Finally, it is worth noting the different contributions to dry mass in each vehicle. For example, the NEP systems have a significant fraction of their dry mass tied up in the electric power system. However, the actual propulsion system (e.g., thrusters, tankage, etc.) is relatively modest. By contrast, a Chemical or NTP Cargo Vehicle would a much higher propellant load, and correspondingly high propulsion system dry mass, even though they would have a minimal power system (for vehicle "housekeeping"). Finally, as mentioned previously, the NTP (or Chemical) option would require that the net payload be aerocaptured directly into Mars orbit; thus, the NTP or Chemical option would require inclusion of a payload Aeroshell for Mars orbit insertion. By contrast, the EP vehicles deliver the payload directly into Mars orbit, so an Aeroshell is not needed.

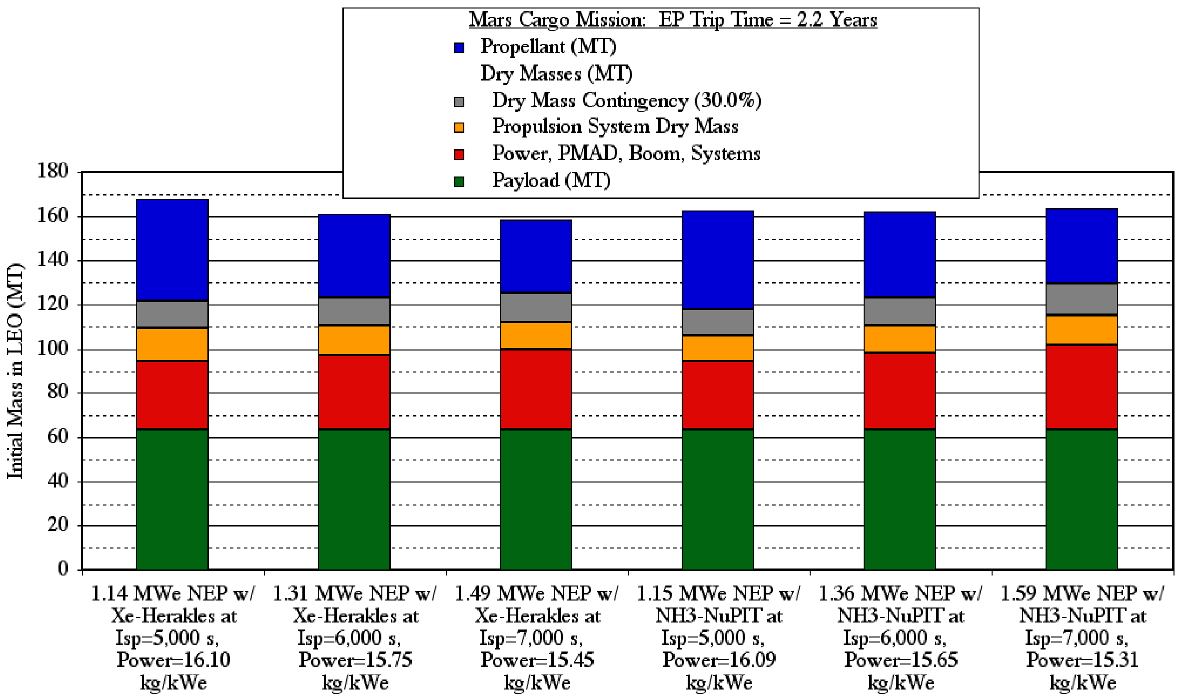


Figure 18. Mass Breakdown for NEP Mars Cargo Vehicles with a 2.2-Year Earth->Mars Trip Time.
(Nuclear Thermal Propulsion Mars Cargo Vehicle one-way trip time is 0.7 years.)

4. Impact of Extraterrestrial Resources on Initial Mass in Low Earth Orbit (IMLEO) and Earth-to-Mars Trip Time

One of the most unique features of the NuPIT propulsion concept is its ability to operate on a number of propellant gasses, such as ammonia (NH_3), water (H_2O), or carbon dioxide (CO_2). This suggests the possibility of extraterrestrial resource utilization (ETRU) as a means of reducing IMLEO for round-trip missions. For example, Mars is a potential source of water or carbon dioxide, although the infrastructure requirements of launching large quantities of propellants from the martian surface into orbit could negate the benefits of "free" propellants directly from Mars. (This same argument would also apply to launching water from the Moon into lunar orbit.) However, Mars has two moons, Deimos and Phobos, which spectroscopically appear to be captured asteroids of the carbonaceous chondrite class. This suggests the availability of a resource of water already in Mars orbit, with minimal propulsive accessibility requirements (e.g., Phobos escape velocity is only 10 m/s). Figure 19 shows a comparison of IMLEO and trip time for several NEP Cargo Vehicle options, including the use of an H_2O -propellant NuPIT system that obtains water propellant for a return trip to Earth from Phobos. For comparison, one-way (expendable) NEP Cargo Vehicle options are show with NH_3-NuPIT or Xe-Ion thrusters, along with round-trip NEP vehicles with NH_3-NuPIT and Xe-Ion (with all the propellants for the round trip from Earth), and H_2O-NuPIT (with H_2O propellant for the return trip from Phobos). (At this time of these analyses, we have only round-trip trajectory data for NEP vehicles at an I_{sp} of 5,000 lb_f-s/lb_m; we hope to be able to evaluate round-trip trajectories at different I_{sp} values at a later date.)

Figure 19 illustrates the remarkable performance benefit in both IMLEO, trip time, and power that can be obtained from the use of extraterrestrial resources. For example, the round-trip H_2O-NuPIT system is significantly lighter than the other non-ETRU round-trip systems. In fact, the use of an H_2O-NuPIT with ETRU results in a reusable, round-trip vehicle with mass, trip time, and power performance similar to that of an expendable, one-way vehicle!

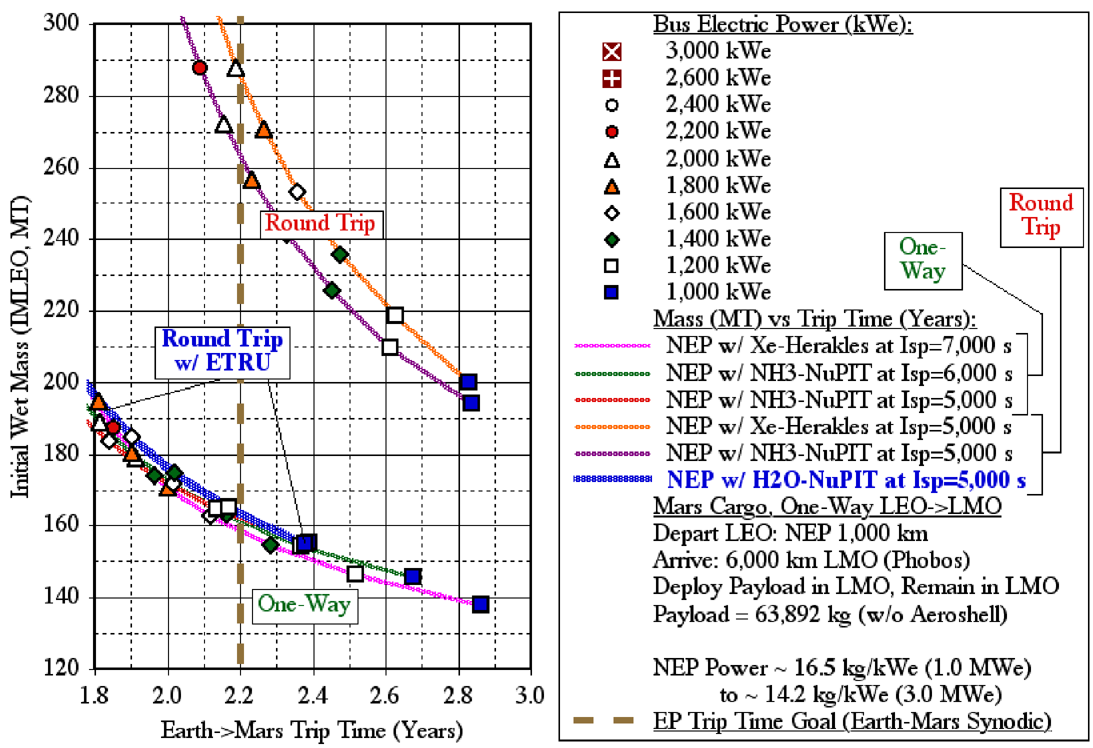


Figure 19. Variation in IMLEO and Trip Time for a One-Way NEP, Round Trip NEP, and Round Trip NEP with Extraterrestrial Resource Utilization (ETRU) for the Mars Cargo Mission. (ETRU water assumed available at Phobos for the Mars->Earth transfer.)

E. Summary of Mission Analysis Results of NuPIT for the NEP Mars Cargo Mission

Electric propulsion in general and NuPIT electric propulsion in particular holds the promise of providing significant mass savings for Cargo Missions in support of Human missions to Mars. Also, NuPIT propulsion systems using water propellant are unique in being able to use extraterrestrial water from low-gravity comets, asteroids, and small moons without needing the additional launch vehicle infrastructure required to supply water or other propellants from the surface of large moons (e.g., Earth's Moon, Europa, etc.) or planets. In fact, this ability to use essentially "free" water from Phobos makes it possible to operate a reusable NuPIT NEP Cargo Vehicle for only slightly more mass than an expendable vehicle.

V. SUMMARY AND CONCLUSIONS

In these analyses, based on preliminary estimates of future NuPIT technology capabilities, we found that the NuPIT propulsion system has performance comparable to that of an advanced Ion (Herakles) system for a variety of NEP robotic and Cargo missions. Modest improvements in NuPIT thruster and PPU mass, and thruster lifetime can provide significant benefits with respect to Ion thrusters. Also, in response to one of the primary goals of the original NRA solicitation,[1] the inherently high power-per-thruster of the NuPIT engine can result in almost an order-of-magnitude reduction in the number of thrusters as compared to the inherently low power-per-thruster Ion engine, with a corresponding reduction in the parts count of a NuPIT propellant storage and feed system. This reduction in the NuPIT system complexity may ultimately prove more attractive than any mass or trip time benefits of this technology by allowing the implementation of a more reliable propulsion system. Finally, the unique ability of the NuPIT thruster to use a variety of propellants opens up the potential of using propellants derived from extraterrestrial resources. For example, use of a water-propellant NuPIT, with thruster performance projected to be comparable to the NH_3-NuPIT, would make it possible to operate a Mars Cargo vehicle in a reusable mode with the same initial mass as a one-way (disposable) vehicle if water for the return trip is available in Mars orbit. This unique capability could allow the H_2O-NuPIT propulsion system to dramatically reduce the cost of delivering cargo by reducing the number of Cargo Vehicles needed to for the Human exploration of Mars.

VI. ACKNOWLEDGEMENTS

The work described in this paper was carried out at the Jet Propulsion Laboratory (JPL), California Institute of Technology, under a contract with the National Aeronautics and Space Administration (NASA).

The authors would like to thank John Warren of the Prometheus Advanced Systems and Technology Office for support of this work. Additionally, we would like to thank Daniel Parcher at JPL and Jim Gilland at the Ohio Aerospace Institute (OAI) for their support on system modeling and trajectory analysis.

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