

Identification of Mission Sensitivities for High-Power Electric Propulsion Systems

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The Advanced Lithium-Fed, Apply-field Lorentz Force Accelerator (ALFA²) project is developing a high-power magnetoplasmadynamic thruster that could demonstrate attractive performance for a variety of high-power electric propulsion missions under consideration by NASA. Both robotic and Human Cargo interplanetary missions may benefit from ALFA² technology, as well as LEO-to-GEO transfers and escape trajectories. The thruster uses lithium (Li) propellant to achieve an efficiency of about 60% at a nominal specific impulse (I_{sp}) of 6,200 lb_r-s/lb_m, with other specific impulses in the range of 4,000 to 7,000 lb_r-s/lb_m under consideration. This paper presents the results of mission analyses that expose various mission performance sensitivities and system advantages of the ALFA² technology for a small but representative subset of nuclear electric propulsion (NEP) missions considered under NASA's Project Prometheus. Multiple technology parameters (e.g., thruster efficiency, throughput, and specific mass) are examined to determine the overall mission sensitivity to these performance metrics. The analyses include comparison studies of the ALFA² technology relative to state-of-the-art Ion propulsion systems and quantify the significant benefits of the technology such as high power-per-thruster and high I_{sp} in a simplified, steady-state, low mass, and small volume propulsion system.

I. INTRODUCTION

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 kW_e) or clustering large numbers of moderately powered (~25 kW_e) 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)

As one of the awards granted by the NRA, the Advanced Lithium-Fed, Apply-field Lorentz Force Accelerator (ALFA²) project combines a Princeton University-led team in collaboration with the Jet Propulsion Laboratory (JPL), NASA, industry, and academia to develop a next-generation lithium-fed, applied-field magnetoplasmadynamic thruster (AF-MPDT). Such a device, also called a "Lorentz Force Accelerator" (LFA), will be designed to optimize and demonstrate its performance and life at 245-250 kW_e and efficiency around 60-63%. A specific impulse (I_{sp}) of 6,200 lb_r-s/lb_m is the nominal design point, although a range of 4,000 to 7,000 lb_r-s/lb_m is under consideration. ALFA² leverages research conducted over the past two decades at the Moscow Aviation Institute (MAI), Princeton, and JPL. The ultimate goal of the ALFA² project is to develop a robust and compact steady-state thruster that could benefit various high-power missions considered by Project Prometheus.

This leap forward in high-power electric propulsion thruster technology will give NASA new robotic exploration capabilities and is also a major step toward MW_e-class systems for supporting human exploration. In this paper we present mission analyses for several such candidate nuclear electric propulsion (NEP) missions, and quantify the mission and system benefits of employing ALFA² technology as the primary onboard propulsion. Parametric variation of key technology performance metrics (e.g., thruster efficiency, throughput, and specific mass) is also used to determine the mission sensitivities of ALFA² in NEP systems. Comparisons are made to relative to

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state-of-the-art Ion propulsion systems (the Herakles Prometheus design) as a baseline. The results quantify the key benefits of the ALFA² technology in reducing system complexity due to the high power-per-thruster and high I_{sp} in a simplified, steady-state, low mass, and small volume propulsion system.

II. DESCRIPTION OF THE ALFA² PROPULSION SYSTEM

Under NASA Project Prometheus NRA funding, the ALFA² project is currently in the first phase of its program to develop an applied-field magnetoplasmadynamic thruster (AF-MPDT) operating with lithium (Li) propellant relevant high-power electric propulsion mission applications. The current base period of study will produce a conceptual design and development plan. A simplified cutaway view of the thruster can be seen in Figure 1. Subsequent phases will result in a laboratory demonstration of an AF-MPDT meeting the NRA performance objectives and validated models of performance and life-limiting phenomena.

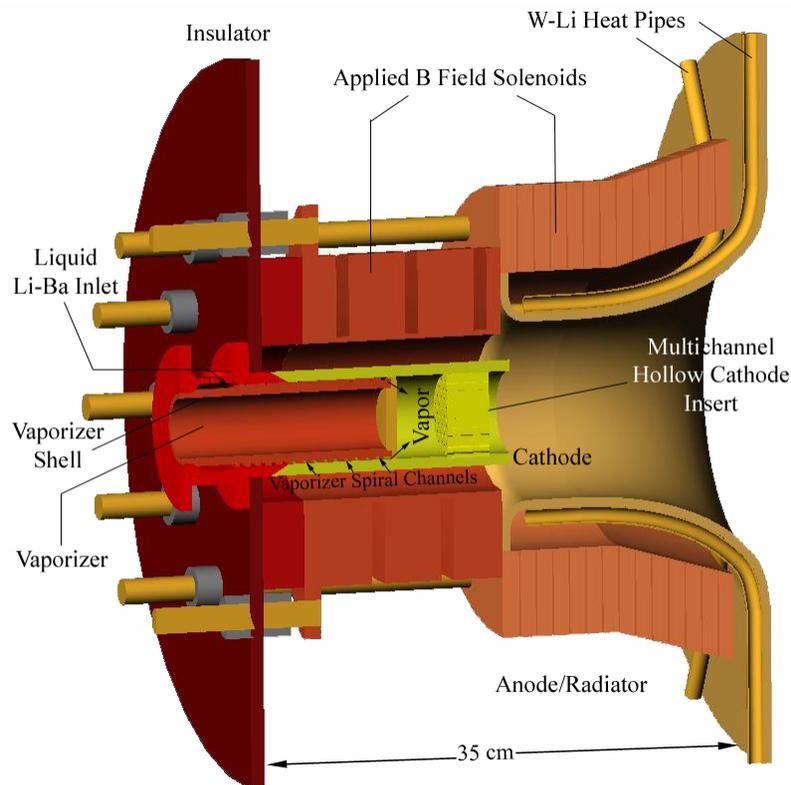


Figure 1. Cutaway View of ALFA² Thruster.

MPDTs utilize the $\mathbf{J} \times \mathbf{B}$ electromagnetic Lorentz force to accelerate plasma. By operating in steady-state with high currents within the thruster plasma discharge between an inner cathode and concentric anode, the plasma discharge produces the necessary current (\mathbf{J}) and self-induced azimuthal magnetic field (\mathbf{B}) to produce the required acceleration mechanism from the Lorentz force for thrust. As shown in Figure 2, an AF-MPDT like ALFA² adds additional performance enhancements by introducing an externally-applied poloidal magnetic field (with radial and axial components) as an additional design parameter.^{2,3}

The ALFA² design is presently optimized for an efficiency of 60 to 63% and a nominal I_{sp} of 6,200 lb_f-s/lb_m at 245 kW_e total thruster power, which includes power to the thruster, the solenoid for generating the applied magnetic field, and the lithium vaporizer. The project will demonstrate electrode geometries, materials and thermal management schemes that mitigate erosion processes, yielding life ≥ 3 years. ALFA² will significantly advance the state-of-the-art (SOA) in lithium-fed AF-MPDTs, and such a thruster would greatly reduce the mass, volume, and complexity of a potential high-power electric propulsion (EP) system. Table 1 compares the proposed ALFA² engine with the state-of-the-art (SOA) resulting from past research. To achieve these goals, the ALFA² project will optimize the design using the detailed analysis models of the project team and the natural benefits of Li as a

Advantages of Lithium:

- Li has uniquely low frozen flow losses as a propellant because the ionization energy is so low (5.39 eV) yet the first excited state and second ionization potential energies are high. Thus, little power is consumed in ionization and maintaining the discharge.
- Also, as a significant benefit to high-power ground testing facilities, lithium condenses on inexpensive, water-cooled vacuum chamber surfaces and does not need to be pumped out of the chamber. In contrast to noncondensable gases, which require very high pump speeds. This reduces facility pumping needs by orders of magnitude. For future long lifetime system tests on the ground, the Li propellant can easily be recycled with closed loop Li purification cycles, similar to those already done with other existing alkali metal (e.g., sodium) closed loop test facilities for power conversion.

III. MISSION ANALYSIS RESULTS FOR ROBOTIC OUTER SOLAR SYSTEM MISSIONS

A. Introduction

A number of outer Solar System Nuclear Electric Propulsion (NEP) robotic exploration 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 ΔV s 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 higher vehicle power levels (multi-100 kW to MWs), 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 ALFA² 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),⁵ and an Interstellar Precursor mission, with ΔV s ranging from about 28 to 53 km/s (for $C_3=0$) depending on the final Solar System escape velocity (V_{inf}).⁶

B. Nuclear Electric Propulsion Robotic Outer Solar System Exploration 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, such as the lithium used in the ALFA² thruster. 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. Also, placing the electric thrusters at the far end of the vehicle, with a 180° field of view to space, ensures that no spacecraft surfaces will experience thermal contamination or erosion impact from the thruster plumes. Fortunately, plume contamination with ALFA² can be mitigated by use of a plume shield. Thus, we included a 10-m diameter plume shield in our analyses of Electric Propulsion (EP) vehicles using ALFA² thrusters.

2. Nuclear Electric Power System

Mass scaling estimates for the nuclear-electric power system were provided by Lee Mason (GRC).⁷ 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) 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.

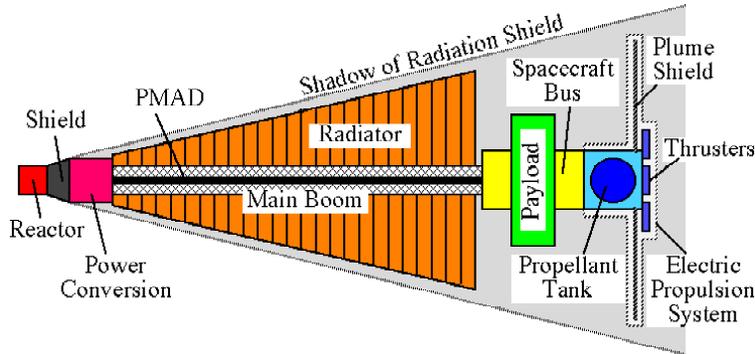


Figure 3. Conceptual Schematic of an NEP Vehicle.

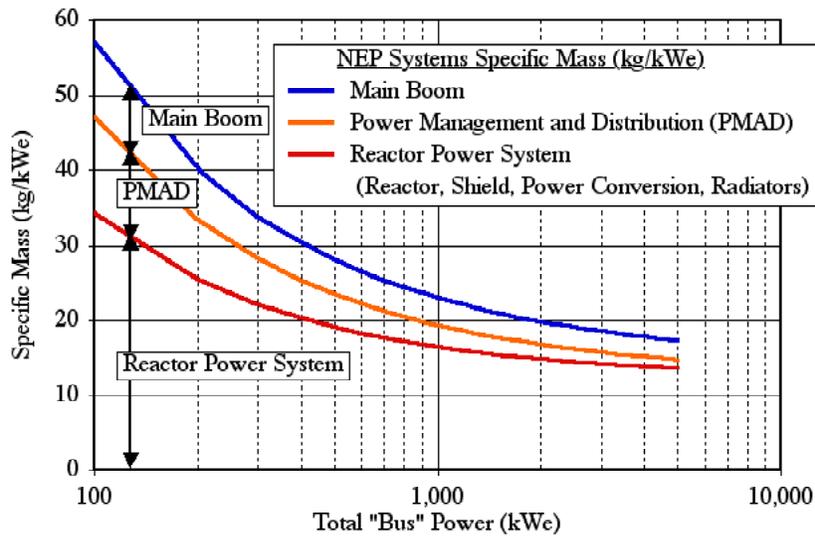


Figure 4. Assumed NEP Power System, Main Boom, and Power Management and Distribution (PMAD) System Specific Mass as a Function of Total "Bus" Power.

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).⁸ 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°).⁷ 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).⁷ 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, an Ion thruster requires very high-voltage (typically many kV) DC. By contrast, an MPDT/LFA engine requires only modest voltage (typically ca. 100 V) 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 inverter, along with a transformer and rectifier as in the NEP PPU, so an SEP-PPU is generally heavier than an NEP-PPU.¹⁰ The topology of these various options is illustrated in Figure 5.⁹

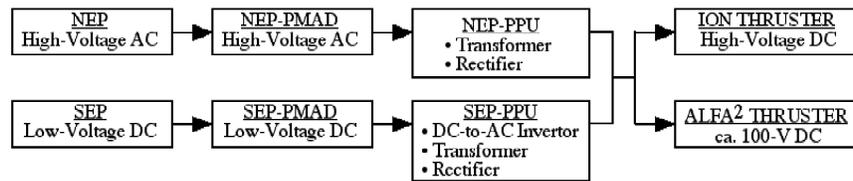


Figure 5. NEP and SEP Power System, PMAD, PPU, and Thruster Topology.

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.⁸ Additionally, we have assumed a general structural overhead of 4% of the mass of propellants.⁸ 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.⁸ 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.¹⁰

6. Chemical Hydrazine (N₂H₄) Reaction Control System (RCS)

A sequentially-recharged blowdown chemical hydrazine (N₂H₄) 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 systems⁸ 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 startup/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 mass⁸ 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 ALFA² 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).¹¹ 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.

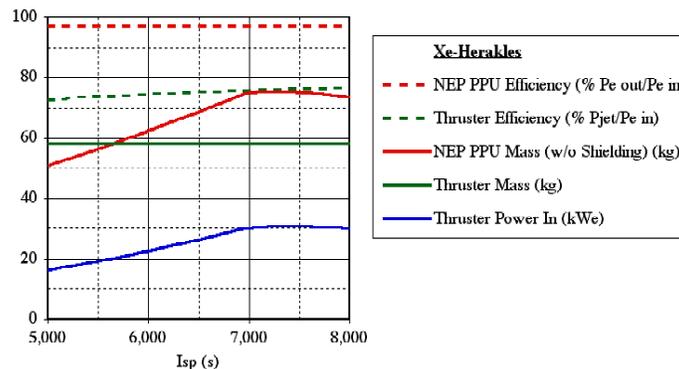


Figure 6. Ion (Herakles) Thruster and PPU Characteristics.

A mass list for a Xenon propellant storage and feed system was developed based on a design by Gani Ganapathi (JPL).¹² 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 2. 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,¹³ 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).

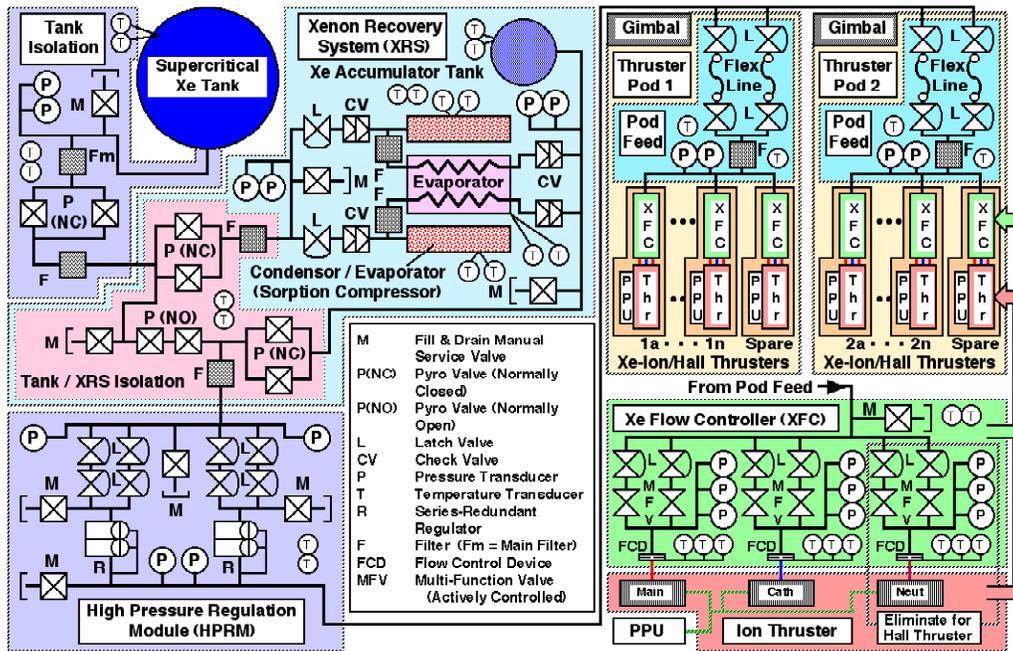


Figure 7. Xe-Propellant Ion (Herakles) Thruster Storage and Feed System Schematic.

Table 2. 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 ALFA² propulsion systems, the number of thrusters (and their corresponding PPU, 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 gimbaling 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. ALFA² Propulsion System

The ALFA² thruster was described in detail above. For these analyses, we have assumed a thruster mass of 129 kg (including the applied-field magnets and Li vaporizer of the ALFA² engine system). The thruster has a diameter of about 0.3 m. The ALFA² nominal (baseline) I_{sp} assumed for these analyses is 6,000 lb_f-s/lb_m, with an efficiency of 60.0% at 6,000 lb_f-s/lb_m I_{sp}. We also evaluate the ALFA² operating at I_{sp} values ranging from 4,000 to 7,000 lb_f-s/lb_m, with efficiencies assumed constant at 60.0%, over this range. The throughput (lifetime) of the thruster is assumed to be 8,300 kg per thruster independent of I_{sp}.

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 low-voltage DC power required by the ALFA² thruster. Based on mass and efficiency scaling models provided by Alexander Kristalinski (Aerojet, Redmond WA),¹⁴ we have estimated the mass of an ALFA² NEP PPU as 360 kg with an efficiency of 98.3% (including power for the applied-field magnets on the thruster counted as a “loss” in determining PPU efficiency). Lastly, we have added a “generic” shielding mass for the Ion and ALFA² PPUs that corresponds to 0.92% of the NEP PPU mass.¹² 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 lithium propellant storage and feed system was developed based on a design by Joseph Lewis (JPL).¹⁵ For the Li-propellant ALFA², we can store the propellant before launch as a room-temperature solid in the propellant tank, and then melt the Li once in space. (This step can occur at an arbitrarily slow rate so as to not represent a major electric power impact, even though the full energy [enthalpy] of melting would need to be provided.) For calculation purposes, the thermal control (heating) power requirement for maintaining the propellant tank and feed system was determined by Robert Miyake (JPL)¹³ based on an assumed storage temperature of 230 °C. (Li melts at 179°C.) Also, the high power-per-thruster of the ALFA² makes it possible to have a small number of individually-gimbaled ALFA² engines in a single (non-gimbaled) thruster pod.

A schematic of the Li-ALFA² propellant storage and feed system is shown in Figure 8. A summary of the various components is given in Table 3. In this system, a low-pressure (few psi) sequential-blowdown helium pressurization system is used to pressurize the molten Li; this ensures that any Li vapors that could diffuse into the pressurization system cannot cause a contamination-induced failure because the vapors are allowed to “see” only already-discharged He pressurization modules (and are isolated from un-used units by normally-closed pyro valves). With sufficient head pressure, the liquid metal is then forced into the electromagnetic pumps, which are used to push the liquid through the feed system and into the thrusters where the liquid is vaporized. Interestingly, the electromagnetic pumps serve the same general purpose as pressure regulators and valves in the Xe system; similarly, Li flow sensors are analogous to gas pressure gages in a Xe system. Note that significant electric power is required to maintain the system at an elevated temperature, as well as additional electric power corresponding to the heat of

vaporization of lithium (i.e., the power is equal to the heat of vaporization [21,279 J/g] multiplied by the mass flow rate [g/s] of propellant into the ALFA² thrusters).

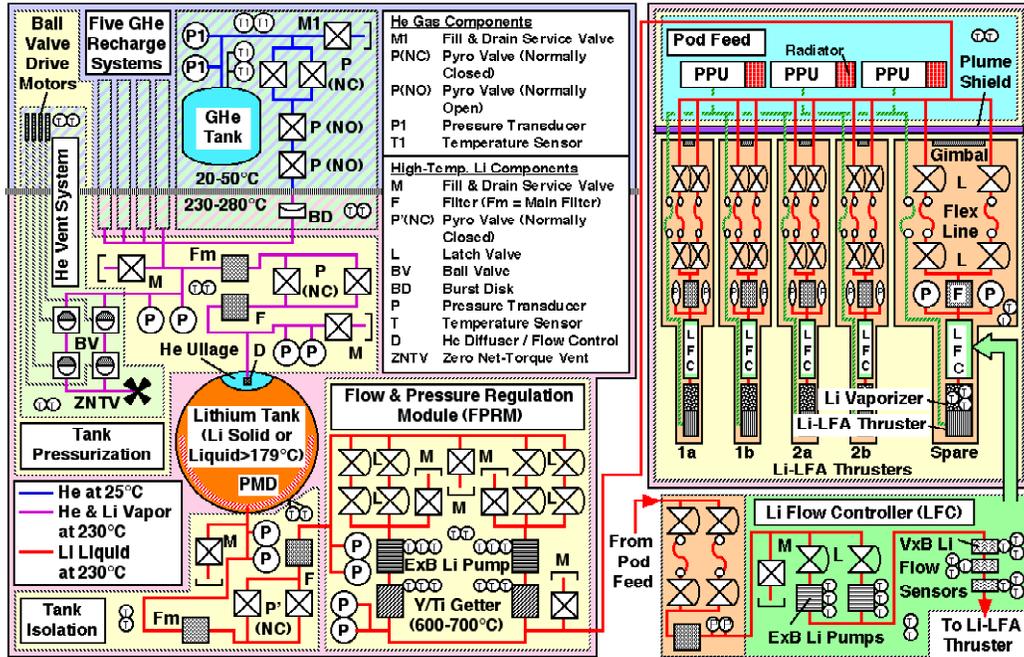


Figure 8. Li-Propellant ALFA² Storage and Feed System Schematic.

Table 3. Summary of Propellant Storage and Feed System Mass and Power Scaling for the Li-ALFA² 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	66.10	3.6341%	7.5425%
Power	69.0		155.3613%
Parts Count	98		
Pod Feed System (1 per Pod)			
Mass	1.67		
Power	17.3		
Parts Count	1		
Thruster Feed System (1 per Thruster)			
Mass (Includes Gimbal)	30.77		
Power	17.3		
Parts Count	20		

C. Mission Analysis Results for NEP Robotic Outer Solar System Exploration 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. In each case, the trajectory data used for the analyses did not include any gravity assists. Gravity assists could potentially reduce flight time for a given mass but would limit the applicability of the analyses to very specific launch date opportunities. Thus, the non-gravity-assisted results here are broadly applicable to approximating many potential future launch dates. Also, the analyses for these missions were generated assuming a starting low-Earth orbit (LEO) at 1000 km altitude.¹⁶ The associated propulsive “spiral out” from this orbit to $C_3=0$ added approximately 7.35 km/s to the total mission ΔV .

For each of these missions, we considered two primary factors in evaluating the mission benefits of the ALFA² 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 Ion (Herakles) system over a range of I_{sp} from 6,000 to 8,000 $lb_f\text{-s}/lb_m$, and the ALFA² thruster option over a range of I_{sp} from 5,000 to 7,000 $lb_f\text{-s}/lb_m$. These analyses show the importance of a high I_{sp} for high- ΔV missions; for example, there is a pronounced difference between the 5,000- or 6,000- $lb_f\text{-s}/lb_m$ I_{sp} curves and the 7,000- or 8,000- $lb_f\text{-s}/lb_m$ I_{sp} curves. However, for this high- ΔV (ca. 41 km/s total leaving from a 1,000-km low Earth orbit [LEO]¹⁶) mission, the ALFA² results in slightly longer trip times than the Ion system, having a roughly 1-year longer trip time than the Ion system at a fixed initial mass. In this case, the superior thruster efficiency, I_{sp} , and especially cumulative throughput of all the running thrusters of the Ion system provide superior performance in terms of initial launch mass and trip time. In particular, the ALFA² system is severely penalized because of its modest throughput and small number of running thrusters. For example, at an I_{sp} of 7,000 $lb_f\text{-s}/lb_m$, the ALFA² points in Figure 9 represent a system with two sets of thrusters (each set run in series) to accommodate the total propellant throughput requirement; at lower values of I_{sp} (i.e., with even more propellant), as many as four sets of ALFA² thrusters may be needed at the lowest power levels.

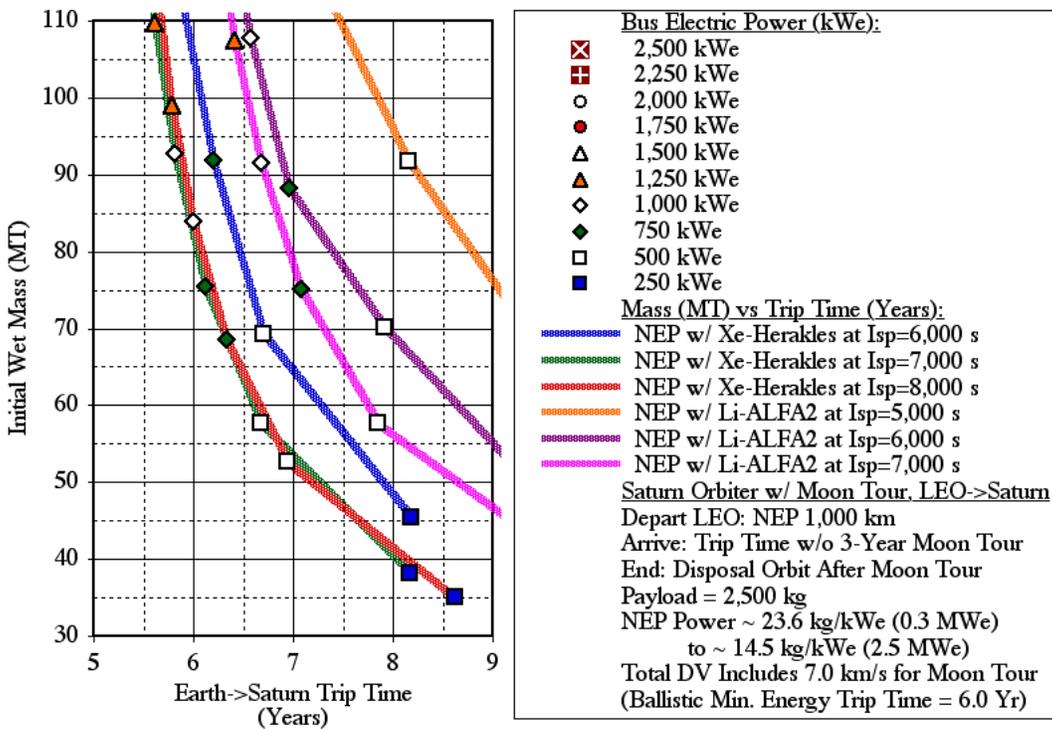


Figure 9. Variation in IMLEO and Trip Time for the Saturn Orbiter with Moon Tour Mission.

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 ALFA² thruster types over a range of I_{sp} from 6,000 to 8,000 $lb_f\text{-s}/lb_m$, and 5,000 to 7,000 $lb_f\text{-s}/lb_m$, respectively. 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} (and high cumulative throughput) for high- ΔV missions; for example, there is a modest difference between the higher 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).

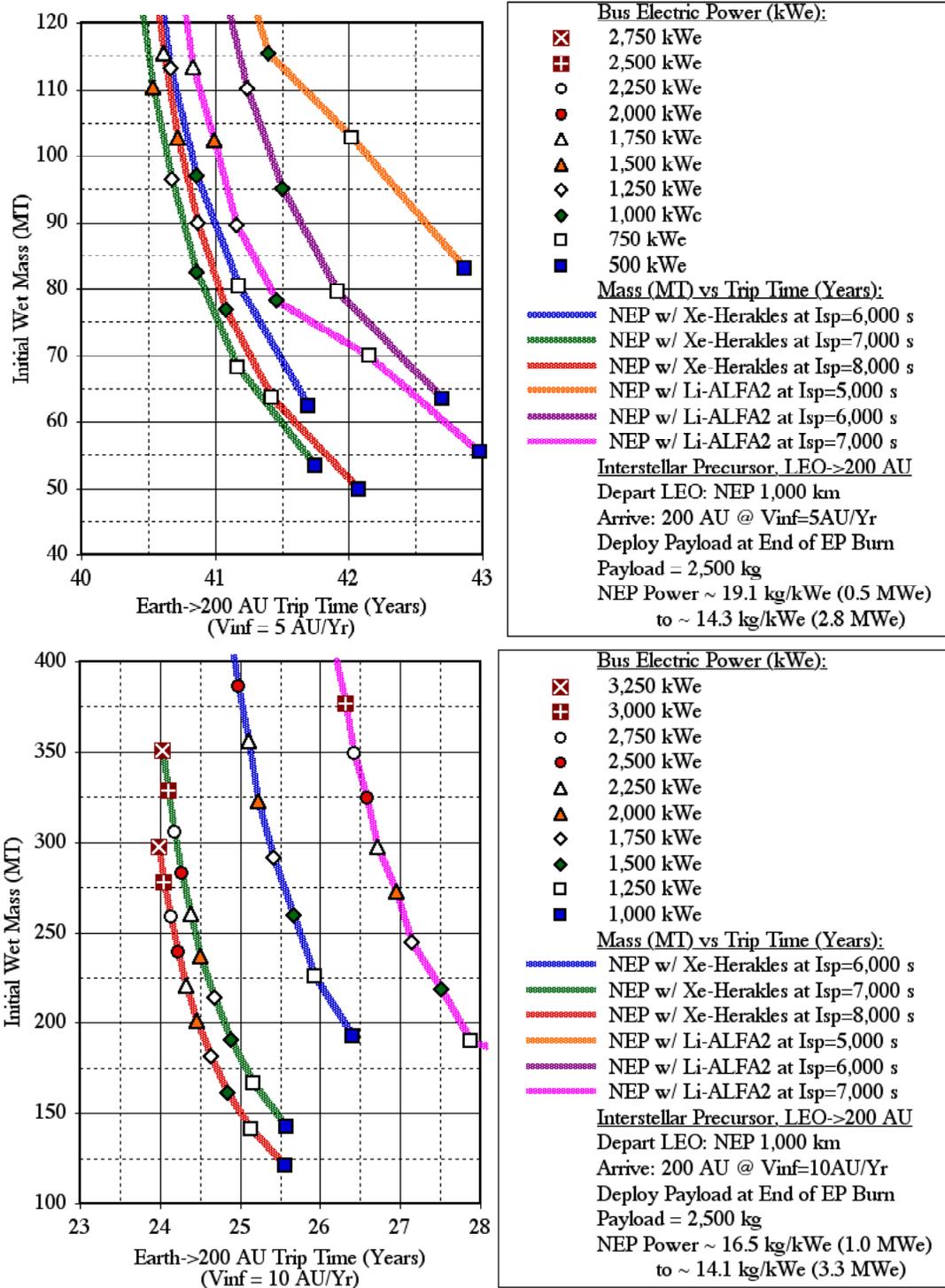


Figure 10. Variation in IMLEO and Trip Time for the 200 AU Interstellar Precursor Mission. (Upper Figure $V_{inf} = 5$ AU/Year; lower Figure $V_{inf} = 10$ AU/Year.)

Also worth noting is the “jog” in the ALFA² curve for the 5 AU/Year mission at an I_{sp} of 7,000 lb_f-s/lb_m between the 750 and 1,000 kW_e points. This is due to the limited lifetime (throughput) of the ALFA² thruster. In this case, at a power level of 1,000 kW_e or more, the number of thrusters required to consume the available power (e.g., four ALFA² thrusters at a total “bus” power of 1,000 kW_e) is sufficient to consume the total amount of propellant.

However, at 750 kW_e, the three running ALFA² thrusters are insufficient to consume all of the propellant, so an extra set of three thrusters (along with their corresponding PPU, feed systems, gimbals, etc.) must be added so as to accommodate all the propellant, resulting in a total of 7 thrusters (including 1 spare). Similarly, at an I_{sp} of 5,000 lb_f-s/lb_m, two or more sets of ALFA² thrusters are needed throughout the range of powers, whereas at 6,000 lb_f-s/lb_m, two sets of thrusters are needed at power levels less than 2,250 kW_e.

The results show that, for a given initial mass, the ALFA² system achieves very comparable trip times (within 1 to 2% difference) to the Ion system for the lower ΔV case (i.e., 35 km/s at 5 AU/year) at higher power levels and I_{sp}. However, for the very high-ΔV 10 AU/year mission, the superior thruster efficiency, I_{sp}, and especially cumulative throughput (i.e., number of running thrusters * throughput per thruster) potential of the Ion system provides clearly superior performance (approximately 10% reduction in trip time for a given initial mass).

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 ALFA² 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. (However, as discussed above, the situation for other missions at the same power level could be different depending on the number of sets of thrusters required for total throughput.) In this case, the parts count is used as a measure of system “complexity.” For example, propulsion systems like the high-power (e.g., 245 kW_e per thruster) ALFA² 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 4.

Note also that the “jog” in the 7,000-lb_f-s/lb_m I_{sp} curve for the ALFA² points between 750 and 1,000 kW_e in Figure 11 corresponds to the same “jog” in the mass and trip time shown in Figure 10. As discussed above, this is due to the need to add additional sets of thrusters to accommodate the total propellant throughput requirement. The situation is even worse at lower values of I_{sp}, with a correspondingly larger total propellant mass. In these systems, many sets of thrusters are required to consume the required propellant mass, although at higher powers, the larger numbers of thrusters needed to consume the available total “bus” power can result in sufficient numbers of running thrusters to consume the available propellant. For example, the 6,000-lb_f-s/lb_m I_{sp} curve for the ALFA² shows that for “bus” powers less than 2,250 kW_e, two sets of thrusters are required, but only one set is needed at 2,250 kW_e and above.

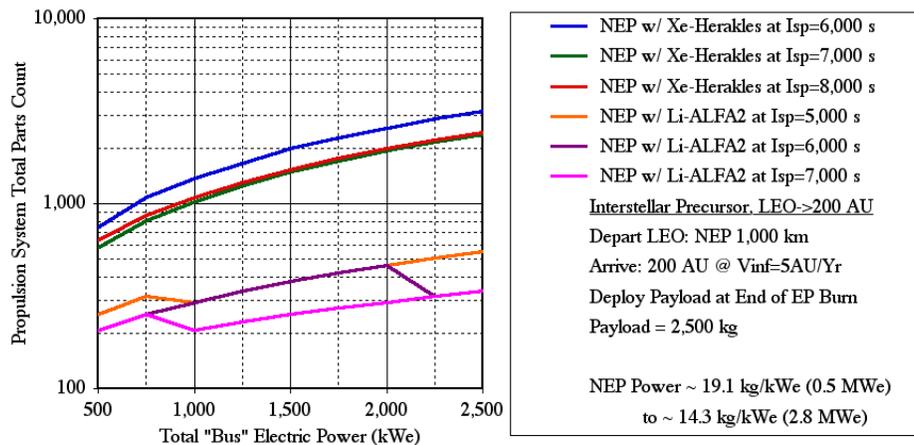


Figure 11. Electric Propulsion System Parts Count vs Total “Bus” Electric Power for the NEP 200 AU Interstellar Precursor Mission (V_{inf} = 5 AU/Year).

Table 4. Power-per-Thruster and Number of Thrusters Required for a 1-MW_e NEP System.

Thruster	Ion (Herakles)			ALFA ²
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	245.0
Power into PPU (kW _e)	23.3	31.2	30.9	249.2
Number Running at 1 MW _e Total “Bus” Power	43	32	33	4
Number per Pod	22 + 1 Spare	16 + 1 Spare	17 + 1 Spare	4 + 1 Spare
Number of Pods	2	2	2	1
Total Number of Thrusters	46	34	36	5

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, the Herakles Ion thruster system requires approximately 40 times the surface area of the ALFA² system; when we take into account the need to have roughly 8 times as many Herakles as ALFA² 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.

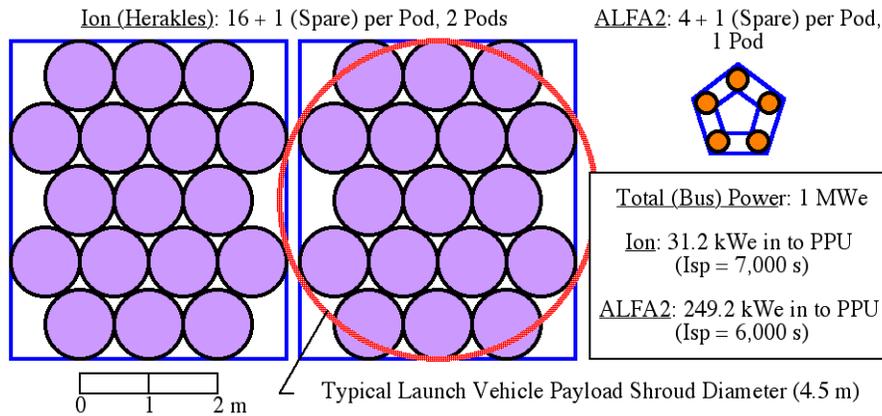


Figure 12. Size Comparison Between Ion (Herakles) and LFA (ALFA²) Thrusters for a 1-MW_e Total “Bus” Power.

D. Parametric Investigation of ALFA² Performance for NEP Robotic Outer Solar System Missions

As described above, we found that the performance of the ALFA² system for high-ΔV missions is somewhat less than that found for the Ion system for the nominal ALFA² thruster. In this section, we will investigate approaches that could improve the performance of the ALFA² system in order to make it comparable to or superior to the Ion system trip time performance, while maintaining the dramatic benefits in reduced packaging volume and parts count associated with the ALFA² system.

1. Variation in ALFA² 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 ALFA² thruster and PPU are 0.527 and 1.444 kg/kW_e, respectively, for a total of 1.971 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 even with a thruster and PPU having zero mass, the ALFA² system does not surpass the performance of the Ion system. In part, this is due to the need for multiple sets of feed systems for each thruster, even when the thruster and PPU have no mass. Also, the propellant tankage for the low-density, high-temperature Li propellant is higher than that for Xe in the Ion system, adding even more mass to the ALFA² system. This suggests that only limited performance leverage can be obtained by attacking only the overall thruster and PPU mass, and this is fundamentally due to the lower assumed thruster efficiency (60%) of the ALFA² system as compared to the high-efficiency Ion system.

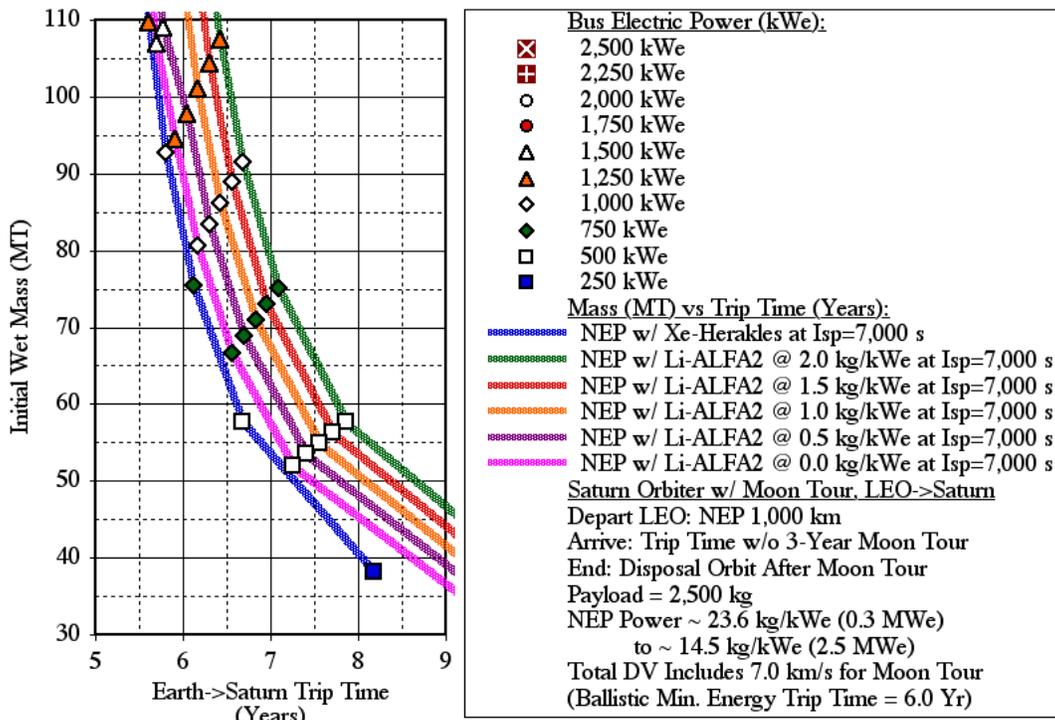


Figure 13. Variation in IMLEO and Trip Time for the Saturn Orbiter with Moon Tour Mission with ALFA² Thruster and PPU Specific Mass Variations.

2. Variation in ALFA² Efficiency

The nominal ALFA² efficiency is 60% independent of I_{sp} . (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 ALFA² efficiency have modest impact on performance. In contrast to the specific mass analysis above, this evaluation of thruster efficiency suggests significant benefits for aggressively pursuing major improvements in ALFA² efficiency.

3. Variation in ALFA² Lifetime (Throughput)

As discussed above, the ALFA² system requires multiple sets of thrusters for the various robotic outer solar system missions because of the ALFA²'s limited throughput (e.g., 8,300 kg per thruster) and small number of running thrusters. For example, at 1 MW_e total power, the cumulative throughput of the four running ALFA² engines is only 33,200 kg as compared to 176,000 kg for the 32 running 7,000-lb_f-s/lb_m I_{sp} Ion (Herakles) engines at 1 MW_e. Figure 15 illustrates the performance for the Saturn and Interstellar Precursor missions with the ALFA² 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). We see the general trend that the required lifetime increases with increasing propellant mass (i.e., with increasing mission ΔV and decreasing I_{sp}). For example, the 5 AU/year Interstellar Precursor mission, with the smallest ΔV (ca. 35 km/s), only needs a 45% increase in ALFA² throughput at an I_{sp} of 7,000 lb_f-s/lb_m, as compared to the 10 AU/year case, with the largest ΔV (ca. 60 km/s), that needs a 2.75-fold increase in ALFA² throughput (at 7,000 lb_f-s/lb_m).

4. Combinations of Variation in ALFA² Throughput, Thruster and PPU Specific Mass, and Thruster Efficiency

Thus far we have treated ALFA² specific mass, efficiency, and throughput as independent variables. In Figure 16, we assume that the ALFA² throughput is increased by a factor of 2.0 for the Saturn mission (at an I_{sp} of 7,000 lb_f-s/lb_m) so that only one set of thrusters is required. We then vary specific mass or efficiency to evaluate the benefits in mission performance that might be realized by advancing these technologies.

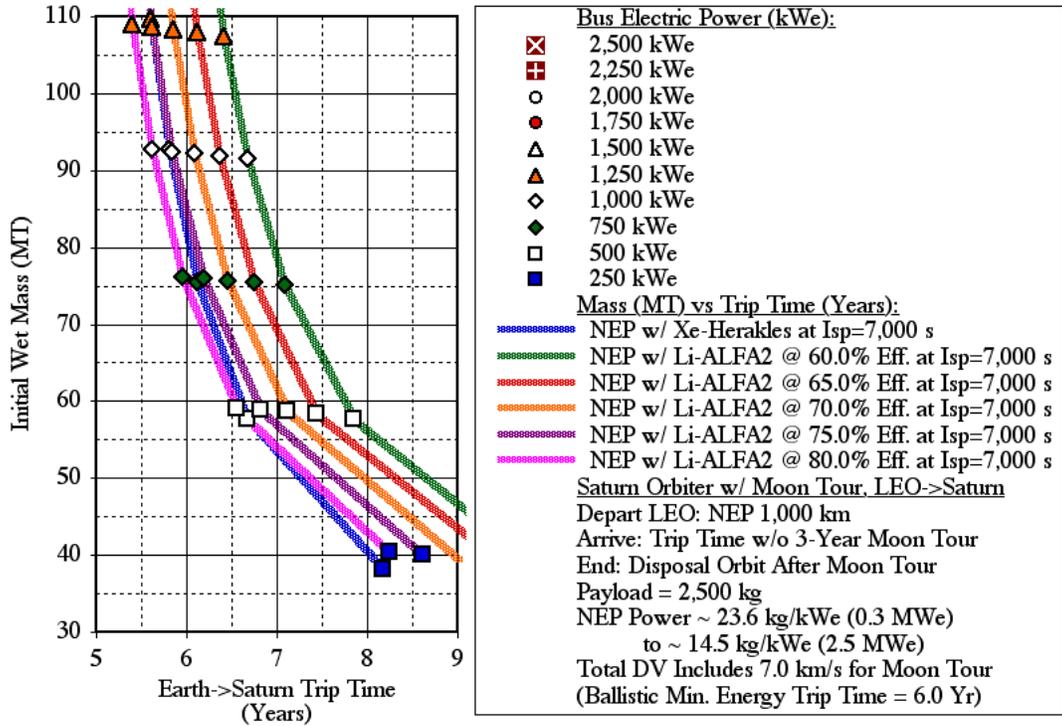


Figure 14. Variation in IMLEO and Trip Time for the Saturn Orbiter with Moon Tour Mission with ALFA² Thruster Efficiency Variations.

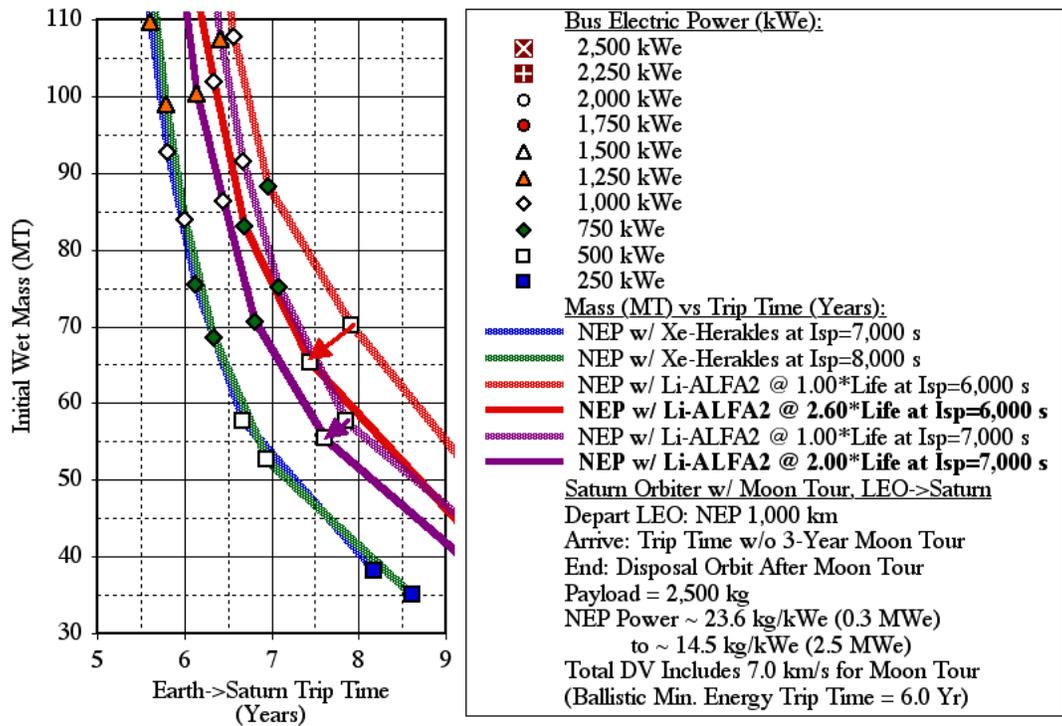


Figure 15 (A). Variation in IMLEO and Trip Time for the Saturn Orbiter Mission with ALFA² Thruster Lifetime (Throughput) Variations.

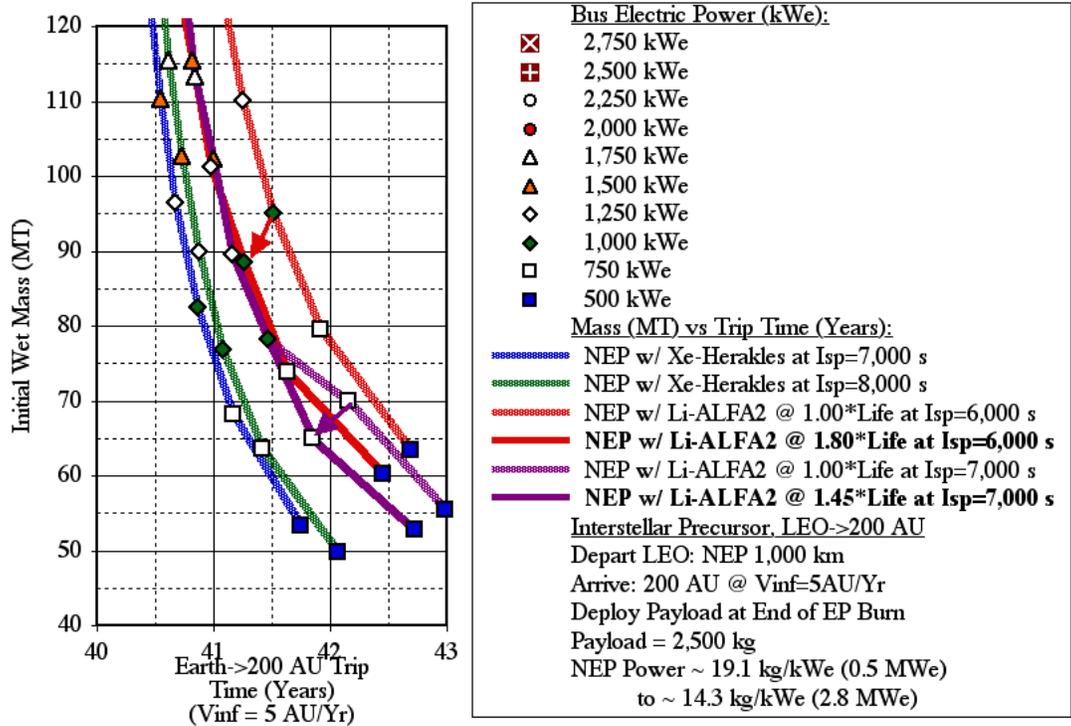


Figure 15 (B). Variation in IMLEO and Trip Time for the 200 AU Interstellar Precursor Mission ($V_{inf} = 5 \text{ AU/Year}$) with ALFA² Thruster Lifetime (Throughput) Variations.

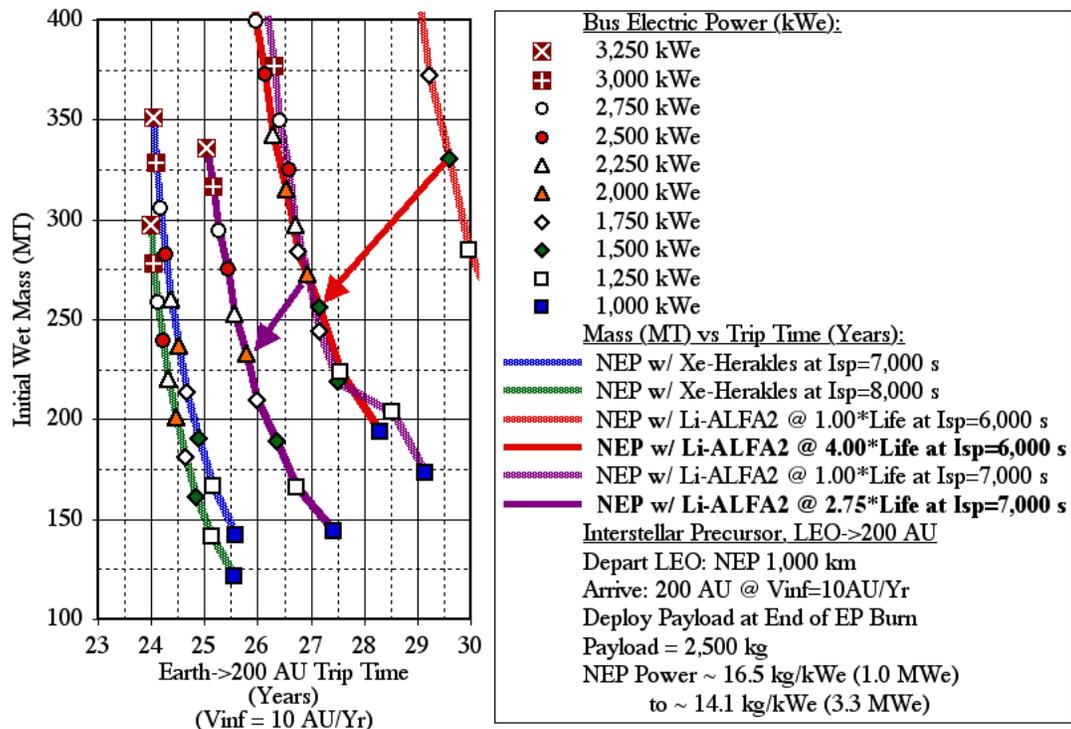


Figure 15 (C). Variation in IMLEO and Trip Time for the 200 AU Interstellar Precursor Mission ($V_{inf} = 10 \text{ AU/Year}$) with ALFA² Thruster Lifetime (Throughput) Variations.

Assuming all the systems shown in Figure 16 have adequate throughput, we see that decreasing the ALFA² thruster and PPU specific mass by a factor of two has only modest benefit. By contrast, increasing ALFA² efficiency from its nominal 60% to 70% results in mass and trip time performance comparable to the Ion system, suggesting that a combination of improvements in ALFA² throughput and efficiency can have major mission performance benefits.

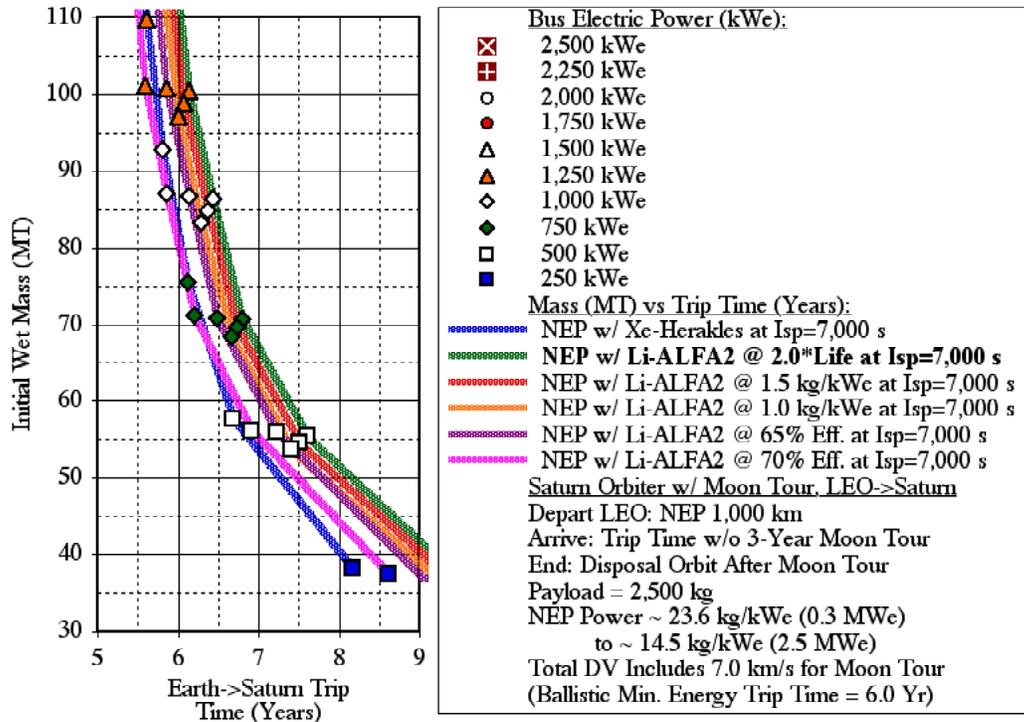


Figure 16. Variation in IMLEO and Trip Time for the Saturn Orbiter with Moon Tour Mission with ALFA² Thruster and PPU Specific Mass, and Thruster Efficiency Variations for a Thruster Throughput Large Enough to Enable a Single Set of Thrusters.

E. Summary of Mission Analysis Results of ALFA² for NEP Robotic Outer Solar System 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} and favor longer mission life (thus higher thruster throughput). 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 and less time is available for thrusting in transit to the target, thus driving up the need for higher thrust acceleration. 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 Li-ALFA² system is still 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 ALFA² system as compared to the Ion system. Finally, there are several Li-ALFA² technology improvements that could significantly enhance mission performance by combinations of reduced thruster and PPU specific mass, increased thruster efficiency, or increased thruster propellant throughput.

IV. MISSION ANALYSIS RESULTS FOR THE NEP MARS CARGO MISSION

A. Introduction

Although often considered for high-power NEP robotic planetary exploration applications, high-power-per-thruster systems like ALFA² 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 ΔV s for low-T/W LEO-to-LMO transfers are on the order of 16 km/s.⁶ 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.¹⁰

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-MT payload derived from the NASA Human Exploration of Mars Design Reference Mission (DRM) Version 3.0.¹⁷ 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). In the analyses here, we assume this Cargo transport function is performed by an electric propulsion Cargo Vehicle instead. For the EP 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} (projected¹⁷ 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

In these systems analyses, 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

There is a strong similarity in performance between the ALFA² and Ion thruster systems for an NEP Cargo Mission, as shown in Figure 17. 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 ALFA² NEP system, the optimum I_{sp} is around generally 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 ALFA², 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} .

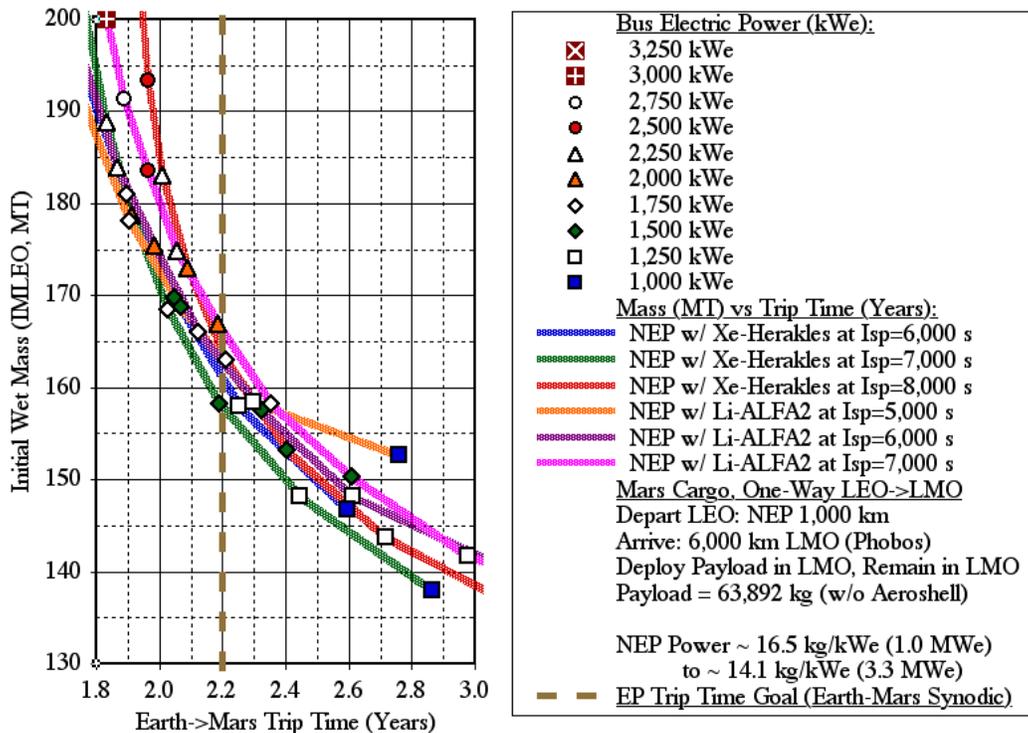


Figure 17. Variation in IMLEO and Trip Time for the NEP Mars Cargo Mission.

2. Propulsion System Complexity

Figure 18 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 Li-ALFA² system. Also shown in Figure 18 are the power levels required to achieve the target Earth-to-Mars trip time of 2.2 years. Finally, for this mission with its modest ΔV and thus propellant load, the ALFA² throughput is adequate to enable a single set of thrusters at all but the lowest I_{sp} and power levels, so the parts count curves for the three ALFA² I_{sp} cases generally fall on top of each other.

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 19 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 ALFA² 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

have 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.

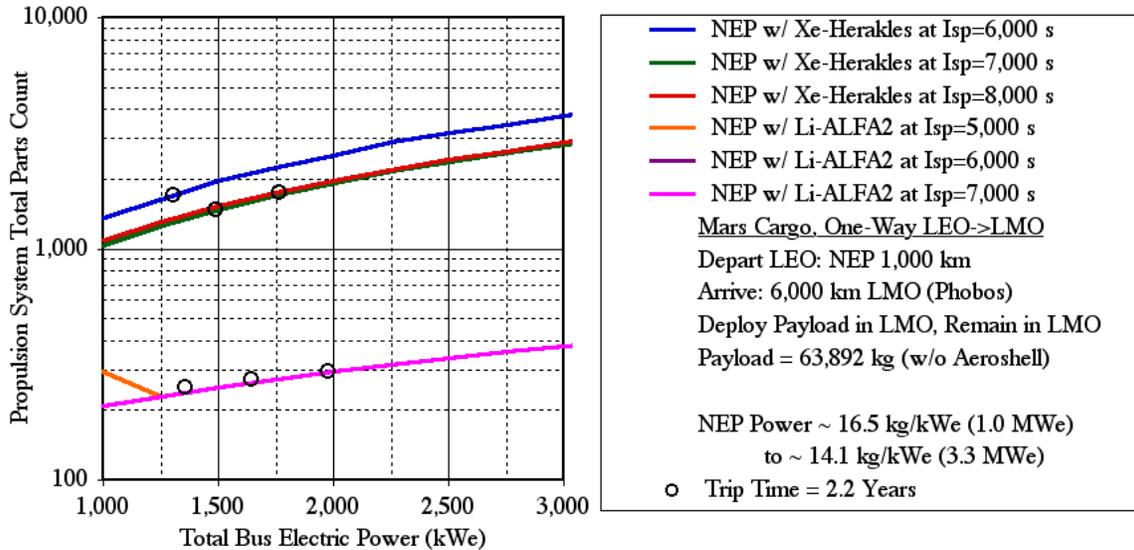


Figure 18. Electric Propulsion System Parts Count vs Total “Bus” Electric Power for the NEP Mars Cargo Mission.

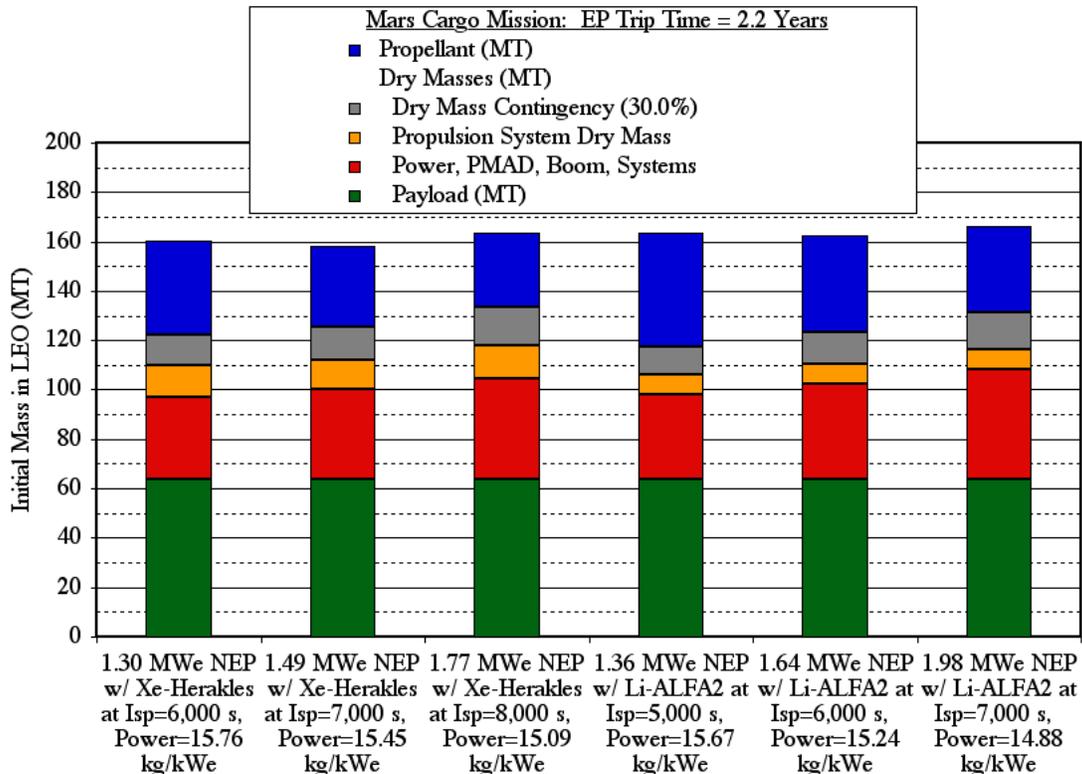


Figure 19. Mass Breakdown for NEP Mars Cargo Vehicles with a 2.2-Year Earth->Mars Trip Time. (Nuclear Thermal Propulsion Mars Cargo Vehicle one-way trip time is 0.7 years.)

E. Summary of Mission Analysis Results of ALFA² for the NEP Mars Cargo Mission

Electric propulsion in general and ALFA² high-power electric propulsion in particular holds the promise of providing significant mass savings for Cargo Missions in support of Human missions to Mars. Also, ALFA² propulsion technology improvements in throughput or efficiency are not required to match the mission performance of Ion systems for Cargo missions. Finally, as with the robotic missions, the high power-per-thruster of the ALFA² system can provide major reductions in propulsion system complexity as compared to low power-per-thruster Ion systems. The greatly reduced packaging volume and parts count of an ALFA² system dramatically simplifies the integration complexities of a 1-2 MW-class EP Cargo Vehicle.

V. SUMMARY AND CONCLUSIONS

In these analyses, based on preliminary estimates of future ALFA² technology capabilities, we found that the nominally proposed ALFA² propulsion system typically requires a modest trip time penalty (for a given IMLEO) compared to an advanced Ion (Herakles) system for the limited set of NEP outer solar system robotic missions considered. Moderate improvements in ALFA² throughput and efficiency can provide significant benefits by enabling performance comparable to Ion thrusters. For the Mars Cargo mission, the ALFA² and Ion systems have comparable mission performance even without further advancements in nominal ALFA² throughput or efficiency. Further analyses including investigation of addition mission destination scenarios and sensitivities to mission-level parameters (payload mass, science “tour” delta-V at the destination, etc.) are recommended.

For all missions examined, in response to one of the primary goals of the original NRA solicitation,¹ the inherently high power-per-thruster of the ALFA² 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. This reduction yields a corresponding dramatic decrease in the packaging volume of the thrusters and reduction in the parts count of an ALFA² propellant storage and feed system. This reduction in the ALFA² 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 with much simpler demands on the system integration and test process, and packaging into a launch vehicle payload shroud volume.

VI. ACKNOWLEDGEMENTS

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VII. REFERENCES

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