

Nuclear Electric Propulsion: Assessing the Design of Project Prometheus

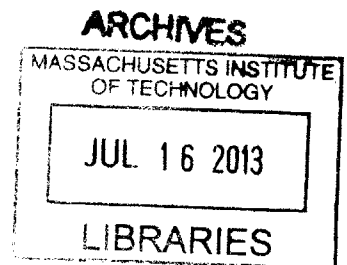
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I. ABSTRACT

The high fuel efficiency of electric propulsion makes it a viable alternative for long-distance space travel. Project Prometheus was a NASA-led project that sought to demonstrate that distant electric propulsion missions were achievable by planning an expedition to characterize the Jovian moons of Jupiter. Several requirements were estimated for the electric propulsion system to allow this mission to happen under the budget and technological constraints of the time. Although the mission was terminated early in 2005, many of the design choices during Project Prometheus have affected future long-distance electric propulsion missions. In this thesis, the design choices in response to the given constraints in the electric propulsion system are analyzed and evaluated. More specifically, the areas explored include the restriction in the amount of propellant mass, the choice of propellant used, the specific impulse and power of the thrusters, and the lifetime of the thrusters.

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II. TABLE OF CONTENTS

1	Abstract	2
2	Table of Contents	3
3	List of Figures	4
4	List of Tables	4
5	Introduction	5
5.1	History of Electric Propulsion	5
5.2	History of Project Prometheus	6
6	Theoretical Background	8
6.1	The Rocket Equation and Delta-v	8
6.2	Thrust	10
6.3	Specific Impulse	11
6.4	Power and Efficiency	12
6.5	Ion Thruster Principles and Components	12
6.5.1	Plasma Generator	13
6.5.2	Ion Accelerator	14
7	Analysis	15
7.1	Project Prometheus Requirements	15
7.2	Design Analysis	16
7.2.1	Propellant Mass	16
7.2.2	Choice of Propellant	20
7.2.3	Power and Specific Impulse	21
7.2.4	Lifetime	23
8	Conclusion	24
9	References	26

III. LIST OF FIGURES

1	Schematic of a nuclear electric propulsion spacecraft.	6
2	Schematic of JIMO mission trajectory and schedule	7
3	Schematic of the processes and components of typical ion thruster	13
4	Qualitative schematic of the sheath profile of the potential at a boundary of the plasma.	15
5	Model of the total mass of the spacecraft as a function of the operating power.	19
6	Efficiency of different propellants in an ion thruster model	21
7	Thrust of an ion thruster against the anode voltage for candidate gases krypton and xenon with a mass flow rate 1.89 mg/s	22
8	Reactor power efficiency, thruster power and efficiency as a function of specific impulse.	23

IV. LIST OF TABLES

1	Requirements and constraints for the propulsion system of Project Prometheus	16
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V. INTRODUCTION

5.1 History of Electric Propulsion

Spacecraft propulsion systems have evolved significantly since their conception. Originally, chemical fuel was used to power spacecrafts [1]. For short space missions, chemical fuel was optimal as it allowed for simpler and cheaper thruster designs [2]. However, as missions became longer, the viability of using chemical fuels decreased. The reason for this is that the range of a spacecraft mainly depends on two variables: the amount of propellant mass m_p relative to the total rocket mass m_t and the exhaust velocity of the fuel v_{exh} [3]. Longer missions would thus require more propellant mass and a higher exhaust velocity. The exhaust velocity of chemical fuels, however, is set by the chemical energy of the bonds in the fuel. Thus, the only way to increase the range of the missions was to increase the amount of chemical fuel mass m_p in spacecrafts with respect to the total mass m_t . In order to increase the amount of propellant mass, either the size of the spacecraft or the density of the propellant or the density of the propellant fuel must be increased. Increasing the size of the spacecraft would result in substantially increasing the building costs. On the other hand, increasing the density of the propellant fuel would require strong materials capable of withstanding the high energy density in the fuel [3].

Electric propulsion was brought forward as a solution to the mentioned problem [1]. In contrast to chemical fuel thruster, the power source of an electric thruster does not have to be the chemical energy of the fuel itself. Therefore, ions can be ejected from a thruster at higher exhaust velocities v_{exh} as the power of the source is not fixed. Furthermore, the efficiency of the power source may be higher than the efficiency of combustion of chemical fuel. Therefore, less propellant is needed to achieve similar velocities relative to the chemical fuel scenario. Thus, both the problem of high fuel density and the excessive bulkiness of the fuel is solved [2]. The trade-off, however, is that electric propulsion can only be used on space where friction is not present due to the low thrust force of electric propulsion. Two main electric propulsion technologies are currently used: ion and Hall thrusters[1]. The difference between the two is the direction of electromagnetic fields and thus the electromagnetic force used to accelerate the ions. The focus of this thesis will be in evaluating the design of an ion thruster electric propulsion system in a specific mission, Project

Prometheus.

5.2 History of Project Prometheus

Project Prometheus was a NASA-led project that sought to design and build a spacecraft capable of reaching the moons of Jupiter [4]. The purpose of this mission was to characterize the moons of Jupiter, while at the same time to demonstrate distant electric propulsion missions were achievable [4]. A schematic of a sample nuclear electric propulsion spacecraft can be seen in Figure 1.

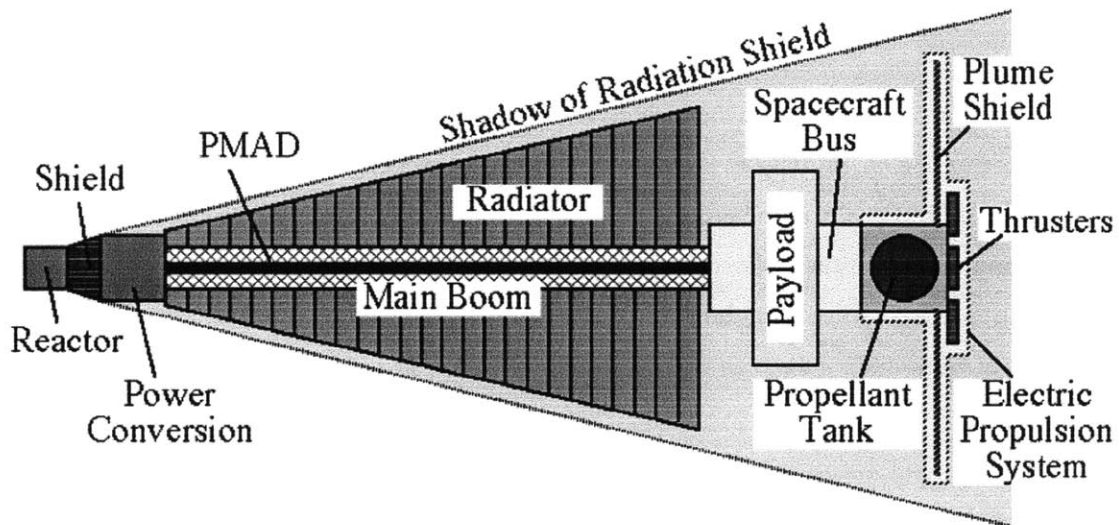


Figure 1: Schematic of a nuclear electric propulsion spacecraft. The main components that define this design are the nuclear reactor seen at the front of the spacecraft, the power conversion unit that transforms the nuclear power into electrical power, and their electric propulsion system that incorporates the thrusters, fuel tanks, and plume shields [5].

The mission faced many challenges including delivering high electric power and high specific impulse, building long-lived ion thrusters and large propellant storage systems, and picking thruster materials and components that could operate in the radiation (mostly gamma and beta) environment close to Jupiter and its moons [4] [6]. The designed spacecraft used a primary ion thruster, Herakles, for space travel and several smaller Hall thrusters for the orbiting mission around the moons [6]. A schematic of the the proposed trajectory is shown in Figure 2. The power

source for the spacecraft was a gas-cooled fission reactor with with a Brayton cycle generator [4]. Project Prometheus was cancelled in 2005, however, due to insufficient funding three years after its conception [4].

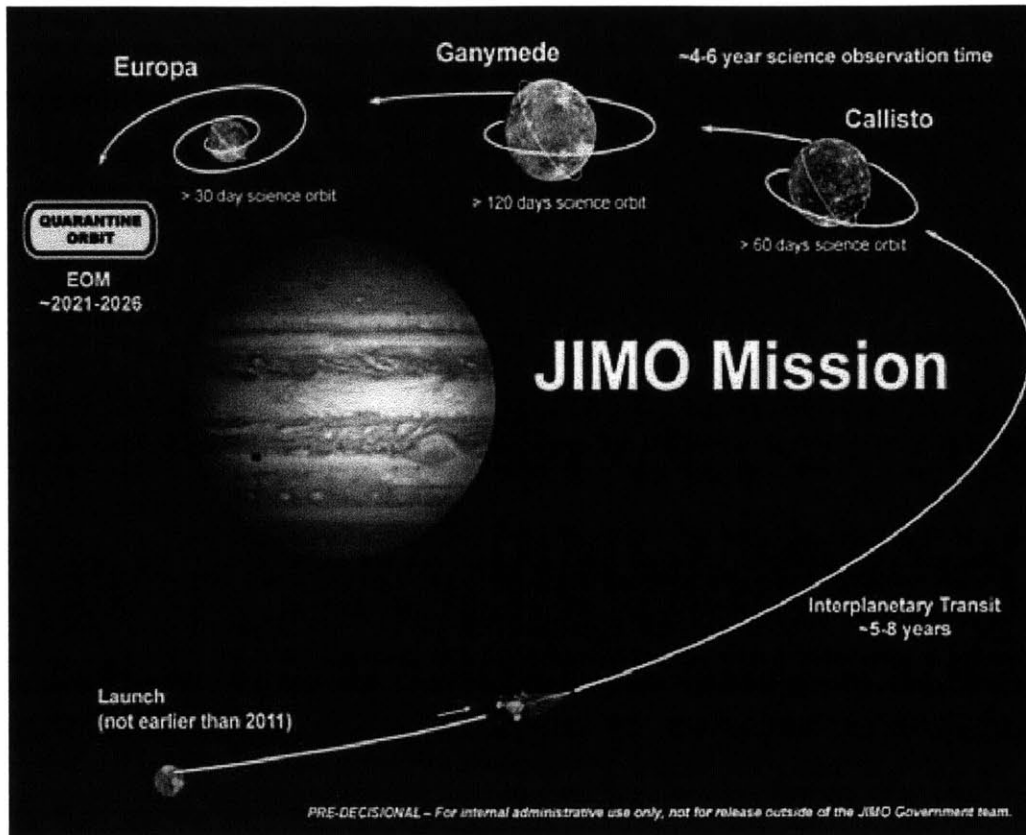


Figure 2: Schematic of JIMO mission trajectory and planning. The three Jovian moons of Jupiter were to be studied by the spacecraft [4].

Although Project Prometheus was terminated early, many important concepts may be learned from studying its design. Furthermore, the design choices made for Project Prometheus may be used as a benchmark of comparison for current electric propulsion space-travel projects.

The objective of this thesis are to:

1. Provide a a review on the relevant physics needed to understand electric propulsion
2. Analyze the different parts of electric propulsion systems with an emphasis on ion thrusters

3. Quantitatively justify required design parameters of Project Prometheus

VI. THEORETICAL BACKGROUND

6.1 The Rocket Equation and Delta-v

Rockets are propelled and change their velocity by ejecting mass through their back. By the conservation of momentum, the momentum of the ejected mass must be equal and opposite to the momentum of the rocket. Thrust is the force supplied by the engine to the spacecraft. Assuming no external forces (e.g. friction), the force (thrust) T on the rocket is thus given by:

$$T = M \frac{dv}{dt} [N] \quad (1)$$

where M [kg] is the total mass on the rocket and v [m/s] its velocity [3]. The force exerted by the propellant is given by:

$$F_{prop} = v_{exh} \frac{dm_p}{dt} \quad (2)$$

where v_{exh} is the velocity of the propellant (assumed to be constant) and m_p is the propellant mass. The total mass M in the rocket is give by the sum of the propellant mass m_p and the constant spacecraft dry mass m_d as:

$$M(t) = m_p(t) + m_d \quad (3)$$

therefore,

$$\frac{dM}{dt} = \frac{dm_p}{dt} \quad (4)$$

Newton's third law tells us that the force exerted on the rocket must be equal and opposite to the exerted by the propellant, $T = -F_{prop}$, implying that:

$$M \frac{dv}{dt} = -v_{exh} \frac{dM}{dt} \quad (5)$$

This equation can be integrated to find the speed change in a specific maneuver. Integrating

this equation from the initial velocity v_i to the final velocity v_f and from the initial mass $m_p + m_d$ to the final mass $m_d + m'_p$ (where $m'_p < m_p$) gives the following relation:

$$\int_{v_i}^{v_f} dv = -v_{exh} \int_{m_p+m_d}^{m'_p+m_d} \frac{dM}{M} \quad (6)$$

The solution to this equation gives the difference between the initial and final velocity (usually called delta-v) for a specific maneuver and is as follows

$$\Delta v = v_f - v_i = v_{exh} \ln \left(\frac{m_p + m_d}{m'_p + m_d} \right) \quad (7)$$

However, in a mission there is usually more than one maneuver in which a change in velocity must be made. For example, in JIMO, the spacecraft must modify its trajectory to orbit around three moons. In practice however, these changes in velocity must be minimized in order to minimize the amount of propellant used and reduce costs. Therefore, missions are carefully planned to take advantage of gravity in order to design routes and orbits with the least amount of velocity change.

An important property of delta-v is that it is additive. In other words, although each maneuver has its own delta-v, the final delta-v for the mission can be found by adding all the changes in velocity throughout the mission.[3]

Therefore assuming that all the propellant is used in a mission, equation 7 can be rearranged and expressed in terms of the propellant mass m_p , where Δv_t is now the sum of all the velocity changes in the mission :

$$m_p = m_d \left(\exp \left(\frac{\Delta v_t}{v_{exh}} \right) - 1 \right) \quad (8)$$

Therefore, using this form the propellant mass can be found given that the mass of the spacecraft, the exhaust velocity, and the total delta-v needed are known for a specific mission [3]. Furthermore, this equation shows the advantage that electric thruster have over chemical thrusters as shown by the v_{exh} term. For typical chemical thrusters, v_{exh} is in the order of 3 - 4 km/s whereas electric thrusters have a v_{exh} in the order of 100 km/s for gases like xenon [3] [1]. Therefore, electric thruster need much less propellant for the same type mission than a chemical

thruster. This reduction in propellant allows for smaller storage tanks reducing the building costs and the size of the spacecraft.

6.2 Thrust

The concepts covered so far apply to any general rocket system. For electric propulsion, extra consideration must be taken to the ions in the fuel and their behavior. Firstly, equation 2 may be approximated as follows:

$$T = v_{exh} \frac{dm_p}{dt} \approx \dot{m}_i v_i \quad (9)$$

where \dot{m}_i [kg] is the mass rate of the ions being expelled and v_i [m/s] their velocity. This approximation holds as long as the ratio of ions to neutrals is high and the speed of the ions is much higher than that of neutrals. In the plasma generated in ion thrusters, this approximation holds [3] [6].

From this the velocity of the ions v_i can be derived from conservation of energy:

$$v_i = \sqrt{\frac{2qV_b}{M_i}} \quad (10)$$

where V_b [V] is the net voltage that ion has been accelerated through, M_i [kg] is the ion mass, and q [C] is the ion charge.

The ion mass flow \dot{m}_i can be expressed as function of the ion current I_b :

$$\dot{m}_i = \frac{I_b M_i}{q} \quad (11)$$

Substituting the terms of 10 and 11 into 9, the thrust for a single-charged ion becomes:

$$T = \sqrt{\frac{2M_i}{e}} I_b \sqrt{V_b} [N] \quad (12)$$

Equation 12 expresses thrust as a function of easily measurable quantities in a thruster. Moreover, as expected it is seen that thrust increases with voltage, ion mass, and current, as the

momentum of the expelled ions would be higher as these variables increase [3].

6.3 Specific Impulse

Specific impulse I_{sp} is a measure of thrust efficiency defined as the ratio of the thrust to the rate of propellant consumption:

$$I_{sp} = \frac{T}{\dot{m}_p g} \quad (13)$$

where g is the acceleration due to gravity. The use of g to adjust the units of I_{sp} to seconds is a convention that has some physical meaning: if an engine's thrust could be adjusted to equal the initial weight of its propellant (measured at one standard gravity) then the specific impulse tells us how long it would last [3]. Using the expression for T in (1), I_{sp} can also be expressed as

$$I_{sp} = \frac{v_{ex}}{g} \quad (14)$$

Therefore, the rocket equation in (8) can be expressed as

$$m_p = m_d \left(\exp\left(\frac{\Delta v_i}{I_{sp} g}\right) - 1 \right) \quad (15)$$

showing that a greater specific impulse can use a lower fuel-to-spacecraft mass ratio to obtain the same speeds.

Using equations 9, 11, and 16, the specific impulse can be shown to be

$$I_{sp} = \frac{v_i \dot{m}_i}{g \dot{m}_p} = \frac{I_b M_i}{e \dot{m}_p} \quad (16)$$

Substituting equation 12 for the thrust and equation 11 for the propellant utilization efficiency into equation 16 yields an expression for the specific impulse [3]:

$$I_{sp} = \sqrt{\frac{2eV_b}{M_i g^2}} \quad (17)$$

This from of the specific impulse will be used in the next section to define the relationship

between the power and the specific impulse.

6.4 Power and Efficiency

The kinetic thrust of the a beam, also called the jet power is defined as

$$P_{jet} = \frac{1}{2} \dot{m}_p v_{exh}^2 = \frac{T^2}{2\dot{m}_p} [watts] \quad (18)$$

by substituting terms using equation 1 and 5 [3]. The rate \dot{m}_p [kg/s] is the rate at which propellant mass is accelerated and expelled from the rocket.

The total efficiency of the thruster η_T is given by the ratio of the jet power to the input power in the thruster:

$$\eta_T = \frac{P_{jet}}{P_{in}} \quad (19)$$

By dividing both sides of the equation by the thrust, rearranging the terms and substituting from equations 17, 18 and 12 the following relation between the power, the thrust, and the specific impulse may be derived:

$$\frac{P_{in}}{T} = \frac{I_{sp}g}{2\eta_T} \quad (20)$$

shows that for a given input power and total thruster efficiency, increasing the specific impulse reduces the thrust available from the electric engine. Furthermore, for a specific thrust, the specific impulse is directly proportional to the power [3].

6.5 Ion Thruster Principles and Components

Ion thrusters work by accelerating ionized particles across an electric field. Ion thrusters are composed of three main parts: the plasma generator, the ion accelerator, and the electron neutralizer. A conceptual schematic of an ion thruster and its components is showed in Figure 3. The plasma is created in the plasma generator allowing positive ions to be accelerated across the grids creating an ion beam while electrons move towards the chamber wall. The electron neutralizer

then neutralizes the ion beam as it exits the thruster avoiding charge collection on the spacecraft [3].

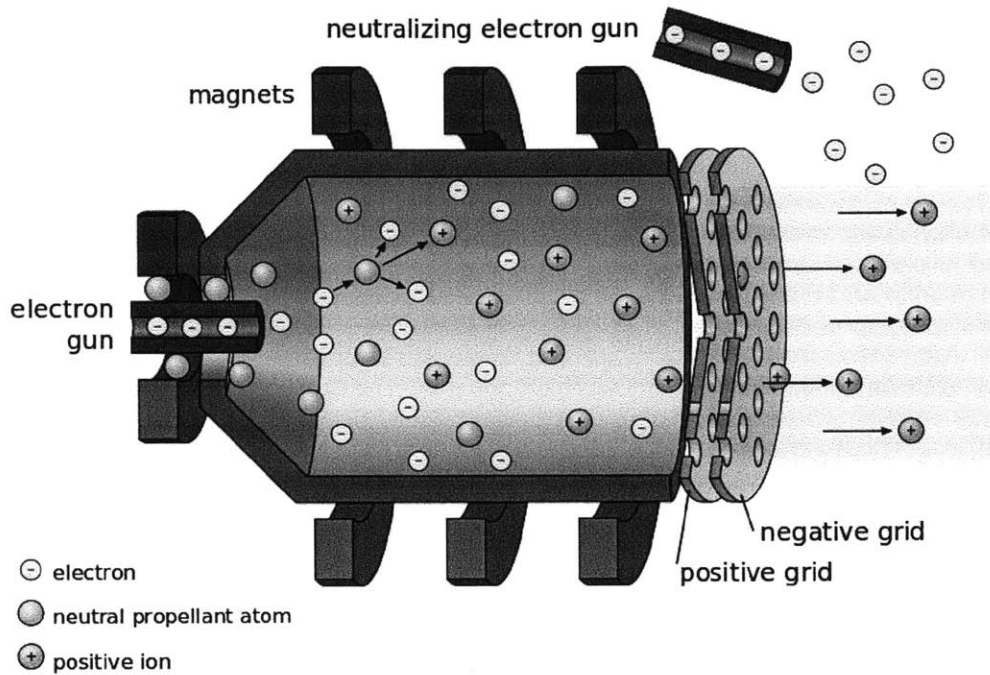


Figure 3: Schematic of the processes and components of typical ion thruster. The plasma is generated by bombarding the neutrals with electrons coming from the electron gun. The magnets bind the ions until they reach the grid in which they are accelerated across. Finally a secondary, neutralizing electron gun, neutralized discharged ions as they leave the spacecraft.

6.5.1 Plasma Generator

There are three primary mechanisms which used in plasma generators to produce the plasma: the direct current (DC) electron discharges, radio frequency (rf) discharges and microwave discharges. The main difference between the DC discharges and the other two is that the discharge hollow cathode or anode power supply are replaced by rf or microwave antenna structures, sources of microwave radiation and compatible discharge chambers to ionize the propellant gas and deliver the ions to the accelerator structure. These thrusters also utilize either applied or self-generated magnetic fields to improve the ionization efficiency of the system [3].

6.5.2 Ion Accelerator

The ion accelerator is the part of the thruster in which the ions created in the plasma generator are accelerated to then be expelled as fuel. However, accelerating ions in a plasma requires more than simply biasing the anode of a DC plasma generator. The reason for this is that plasmas remain quasi-neutral globally and create shielding sheaths under the presence of external electric fields [6] [7]. Large voltages are needed to accelerate the ions to high speeds. When the potential difference is much greater than the electron temperature in the plasma, a phenomenon called the Child-Langmuir sheath occurs in the plasma boundaries.

The Child-Langmuir sheath occurs when electrons are repelled over the majority of the sheath thickness, meaning that the electron density approaches zero approaching the sheath edge. In this case, the electron charge does not significantly affect the sheath thickness. Figure 4 shows qualitatively the sheath profile of the potential at a boundary of the plasma. Starting at the sheath potential ϕ_0 , the potential rapidly decays until it vanishes. The maximum potential of the plasma ϕ is used as the reference value for calculations that involve the plasma.

Under these limits, the ion current density J_i can be written as

$$J_i = n_i e v = n_i e \sqrt{\frac{2e}{M}(\phi_0 - \phi)} \quad (21)$$

Inserting this ion current density while neglecting the electron current density contribution into Poisson's equation give the following differential equation:

$$\frac{d^2 \phi}{dx^2} = -\frac{en_i}{\epsilon_0} = -\frac{J_i}{\epsilon_0} \sqrt{\frac{M}{2e(\phi_0 - \phi)}} \quad (22)$$

The solution to this differential equation defines the following relation between the ion current density, sheath thickness d , and the anode voltage which is called the Child-Langmuir Law [3]:

$$J_i = \frac{4\epsilon_0}{9} \sqrt{\frac{2e}{M}} \frac{V^{\frac{3}{2}}}{d^2} \quad (23)$$

The importance of this sheath is that it sets a maximum size for the grid in the ion accelerator.

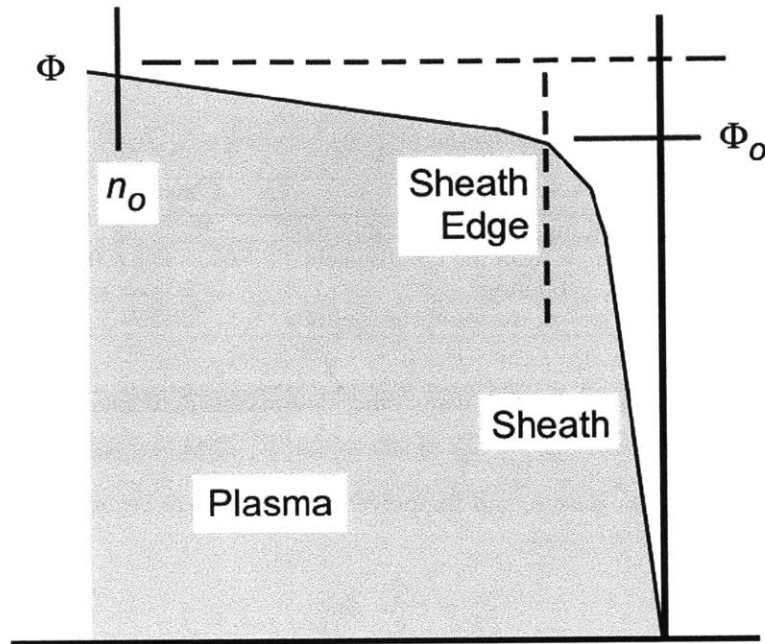


Figure 4: Qualitative schematic of the sheath profile of the potential at a boundary of the plasma. The plasma potential is highest at ϕ and rapidly starts to decline at the sheath edge where the potential is ϕ_0 .

In ion thrusters, the accelerator structure can be designed to first order using the Child-Langmuir equation where d is the gap between the accelerator electrodes [3]. If the grid size is bigger than the Langmuir-Child sheath thickness, then the electric field will not be felt by the positive ions and thus no acceleration will occur in the plasma. For typical xenon-fueled thrusters, the order of the grid spacing is between 2 mm to 5 mm with voltages around 1500 V [8].

VII. ANALYSIS

7.1 Project Prometheus Requirements

The main objective of Project Prometheus was to show that nuclear electric propulsion was a viable method for space travel by exploring the Jovian moons of Jupiter and developing a viable nuclear-fueled spacecraft [4]. Several requirements were estimated for the electric propulsion system by JPL to allow this mission to happen under the budget and technological restraint at

the time [4]. The requirements and constraints for the propulsion and spacecraft design will be exposed in this section.

Table 1: Requirements and constraints for the propulsion system of Project Prometheus

System	Constraint
Spaceship maximum total propellant mass	18,000 kg
Specific impulse from the ion thrusters	6,000 s to 8,000 s
Lifetime	> 20 years
Minimum jet power during thrust periods	130 kW

Other than the outlined requirements, many other design considerations must be taken into account for a proper spacecraft design [4]. However, other considerations such as spacecraft materials, the nuclear reactor design, and the power control unit are beyond the scope of this thesis.

7.2 Design Analysis

7.2.1 Propellant Mass

As discussed in Section 7.1, the upper bound for the propellant mass (for electric propulsion) in the Project Prometheus spacecraft is 18,000 kg. The dry mass of the spacecraft (i.e. the mass that is not propellant) was estimated to be 18,000 kg, resulting in a maximum total mass of 36,000 kg after leaving the Earth's atmosphere.

In this section, an attempt to justify this design decision will be made by making a simple estimating the required masses of the main components of the spacecraft and how they scale with the total power of the reactor.

The total mass of the spacecraft can be expressed as the sum of the mass of all other components:

$$m_t = m_{reactor} + m_{payload} + m_{radiator} + m_{thrusters} + m_{propellant} \quad (24)$$

The mass of the reactor can be expressed as a function of power making the following assumptions. The power density of the reactor p is the standard for a high-temperature gas cooled reactor, with a value of 8.4 kW/L [9]. The density of the reactor ρ_r is taken to be the density of steel of around 8 g/cm³ [9]. The reactor core of volume V is taken to be of cylindrical shape

with diameter equal to its height H . The power efficiency of the reactor η_r , this is the ratio of the power out coming out of the reactor power coming in is taken to be 38%, a standard value for a gas-cooled reactor [9]. The thermal power of reactor is then

$$P_t = pV[W] \quad (25)$$

where the volume is equal to

$$V = \frac{m_{reactor}}{\rho_r} = \frac{H^3}{4} \pi [m^3] \quad (26)$$

The mass of the reactor can be seen to be proportional to its power as

$$m_{reactor} = \frac{P \rho_r}{p} [kg] \quad (27)$$

The payload mass for the mission was estimated to be around 3000 kg [4]. This mass includes most of the instrumentation for the data collection around the moons of Jupiter.

The radiator mass depends mostly on the surface area needed to eject the waste heat from the spacecraft. The amount of heat needed to expel is approximately P_w is $(1 - \eta_r)P_t$. A study made showed that for structural stability, an aluminum radiator for space travel should have at least a thickness t of 2 cm [10]. The average temperature T [K] in the secondary system of a standard gas-cooled reactor is 650 K [9]. Using the radiative heat transfer the power radiated can be expressed as a function of the area and the average temperature (assuming the metal is at the average temperature of the coolant) [9]:

$$P_w = \sigma AT^4 \quad (28)$$

where σ [W/m^2K^4] is the Stefan-Boltzmann constant and A [m^2] is the surface area of the

radiator. The mass of the radiator is simply:

$$m_{radiator} = At\rho_{Al} = \frac{(1 - \eta_r)P_{it}\rho_{Al}}{\sigma T^4} [kg] \quad (29)$$

The thruster mass does not scale directly with the power as thrusters are able to work with a variety of different voltages. However, current ion thruster design can accept a load of 30 kW and weigh around 60 kg [4]. A conservative thruster mass to power ratio can be established as as 2 kg/kW.

The amount of propellant mass can be found using the rocket equation. The delta-V for the whole JIMO mission trajectory from Earth and changing orbits between the moons of Jupiter was estimated to be 3.6 km/s [4]. A simple calculation using equation 15 can be used to calculate the amount propellant mass required for the mission. As seen in the equation, it is the lowest specific impulse that set the maximum total propellant mass needed in the spacecraft. For example, given a dry mass of 18,000 kg, at 6,000 s of specific impulse the total propellant mass needed equals 16,300 kg. However, this calculations does not take into account fuel efficiency. Furthermore, this calculation does not take into account the fuel used by the Hall thruster system that require around 5% of the fuel for smaller maneuvers especially when orbiting around the moons [5].

Taking these consideration into account a new form of equation 15 is constructed:

$$m_p = \frac{m_d}{\eta_f} f_{Hall} \left(\exp\left(\frac{\Delta v_t}{I_{sp}g}\right) - 1 \right) \quad (30)$$

where η_f is the fuel efficiency for the propellant and f_{Hall} is the correction factor for the amount of fuel needed by the Hall thrusters. For this range of specific impulses the fuel efficiency for xenon is 0.83 and the Hall correction factor is 1.05. The upper bound for propellant mass is 18,000 kg. Thus, solving the equation for the specific impulse with these values gives the minimum specific impulse to use for this amount of fuel. From equation 20 it can be seen that the specific impulse is proportional to the power to thrust ratio. Thus, to first order, the specific impulse is proportional to the power input. The $\frac{T}{2\eta_r}$ is neglected as it is assumed to be close to unity

given that the thrust is in the order of 1 N and the efficiency is approximately 0.7 [3] [4]. This approximation is a main source of error, but greatly helps to simplify the problem and show the trade-offs between power and mass.

The mass equation 24 is then a combination of competing linear and exponential equations as shown in equation 31. At a low power, the exponential dominates, whereas at a high power the linear mass should dominate. Figure 5 shows the relationship for the total mass as a function of power.

$$m_t = \frac{P_t \rho_r}{p} + 3000 + \frac{(1 - \eta_r) P_t t \rho_{Al}}{\sigma T^4} + 2 \times 10^{-3} P_t + \frac{m_d}{\eta_f} f_{Hall} \left(\exp \left(\frac{\Delta v_t}{P_t} \right) - 1 \right) [kg] \quad (31)$$

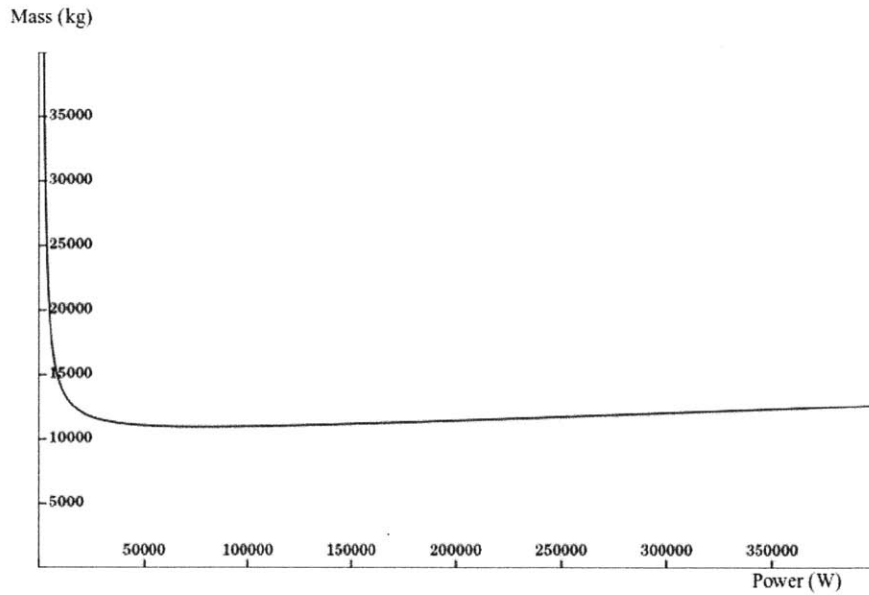


Figure 5: Model of the total mass of the spacecraft as a function of the operating power. The minima is at 77 kW with total mass of 11,000 kg

The minima is at 77 kW with total mass of 11,000 kg which is very close to the suggested total mass of 18,000 kg. The amount of propellant mass for this model is 5,800 kg, again very close to the 50% mark of the design suggested by JPL. Although this model makes many simplifying approximations, the overall behavior and trade-offs in the design can be derived. To further improve this model, more accurate parameters for the power density and coolant temperatures

of the reactors have to be made. Furthermore, the mass of the shield (both for the reactor and the external sources) must be estimated. This correction would increase the total mass of the spacecraft, making it closer to the proposed value by JPL. However, the mass of the exterior shield does not depend on the reactor power and the size of reactor shield does not vary significantly with a marginal increase in power [4].

7.2.2 Choice of Propellant

Historically, common propellant for ion thrusters include cesium, mercury, and xenon [1]. Although initially, most thrusters used cesium and mercury, the corrosive effects of cesium and the strong toxicity of mercury made these elements hard to use for long missions [3] [1]. Inert gases are neither corrosive nor explosive posing little risk to the life of the spacecraft or the personnel loading the propellant tanks [3]

Theoretically, as seen in equation 12 , xenon should offer the highest thrust of all the inert gases due to having the highest mass of the non-radioactive inert gases. However, studies have been made to show that krypton, another noble gas could be a possible substitute, by trading approximately 4% efficiency and slightly reduced thrust, for a much cheaper price . Figure 6 shows the fuel efficiency of three candidate as a function of the specific impulse.

In the region of suggested specific impulse, between the prescribed 6,650 s and 8,000 s, the efficiency of both xenon and krypton reach their peaks at 82%. This has two major implications for the fuel type used and the desired specific impulse. First, the suggested region for specific impulse does have the highest efficiency and marginal increase in specific impulse is negligible after this regime. Second, both krypton and xenon display a similar efficiency on this range, making both of them attractive candidates for long missions.

With the efficiencies of both xenon and krypton being similar, the next factor to consider is the thrust capability for each fuel. A plot from the results of a simulation for an ion thruster for both krypton and xenon is shown in Figure 7.

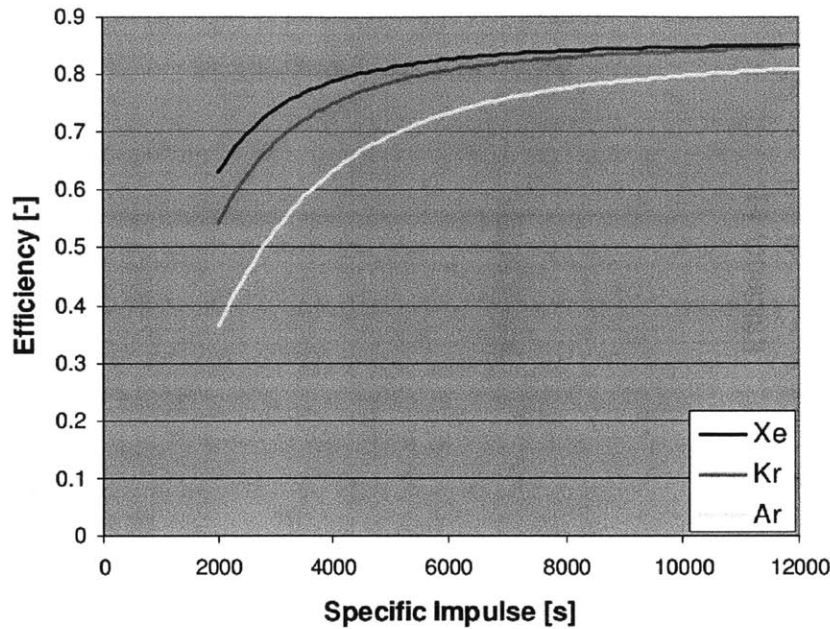


Figure 6: Efficiency of different propellants in an ion thruster model as a function of the specific impulse. The three proposed propellants are the inert gases xenon, krypton, and argon. In the Isp region of interest (6650 s - 8000 s) the efficiencies of krypton and xenon differ by less than 4%.

The results from the simulation show that the increase of thrust with anode voltage for both propellants is somewhat higher than the expected square root dependence on the anode voltage and results from the optimization towards higher specific impulses and thus anode voltages. Even though the thrust of xenon is slightly superior, the difference between the thrust of the two fuels is negligible in the high anode voltage domain. Therefore, both krypton and xenon perform similarly with their only difference being their cost and the availability and production methods. However, although these characteristics are similar, xenon has been tested much more than krypton as a propellant with more than 3 years of test hours in ion thrusters [11]. Therefore, krypton must be tested more in experimental thrusters before it can be used for potential missions, however, it is still an attractive possibility for long missions.

7.2.3 Power and Specific Impulse

The nuclear reactor on board of the spacecraft is rated to produce 200 kWe [4]. These power must be used to operate all aspects of the spacecraft such as the ion thruster, Hall thrusters, radiators,

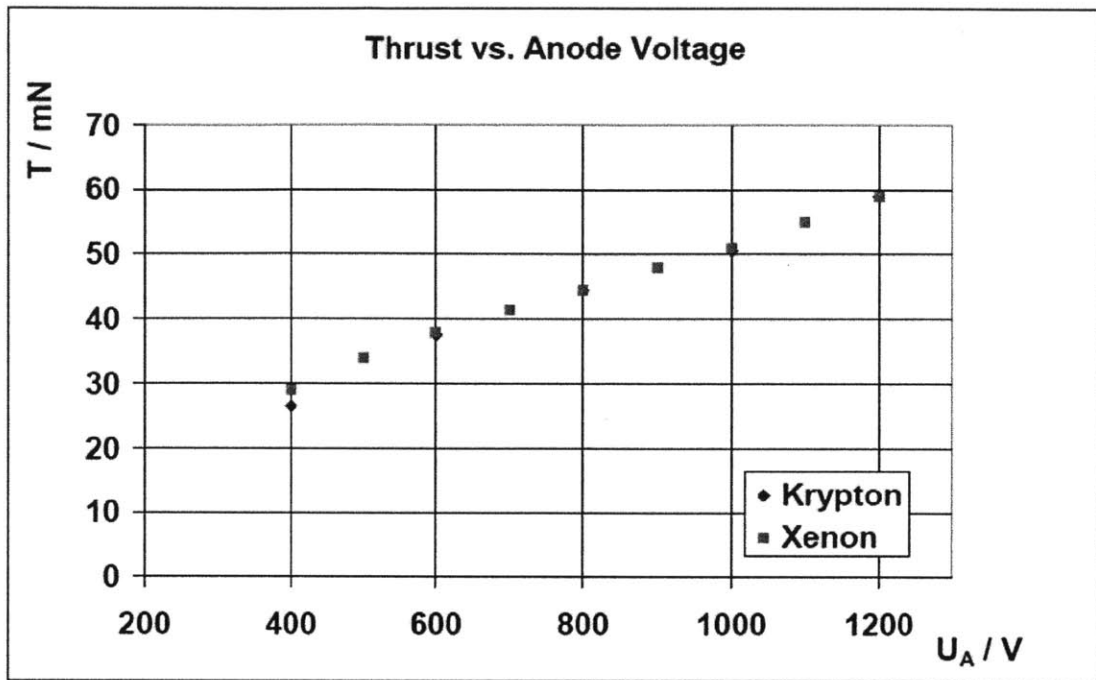


Figure 7: Thrust of an ion thruster against the anode voltage for candidate gases krypton and xenon with a mass flow rate 1.89 mg/s. The difference in thrust is very small between the two propellants for the same anode voltage.

control mechanisms, and communications [4]. The ion thrusters are expected to use at most 90% of this power in the most demanding parts of the mission [5]. Reactor power efficiency, thruster power and efficiency as a function of specific impulse is shown in Figure 8.

As seen in the figure, the thruster power increases non-trivially with specific impulse. At around 7,000 s of specific impulse, the power reaches a peak at around 30 kW of power. After this mark, the power remains relatively constant as a function of specific impulse. Therefore, in order to maximize efficiency, the specific impulse should be maximized at 8,000 s and a power of 30 kW per thruster. The problem with this implementation, however, is that (as explained in the following section) the corrosion in the ion accelerator increases with a higher specific impulse, decreasing thus the lifetime of the thruster.

With a nominal value of 7,000 s, six ion thrusters could be used for a total power of 180 kW.

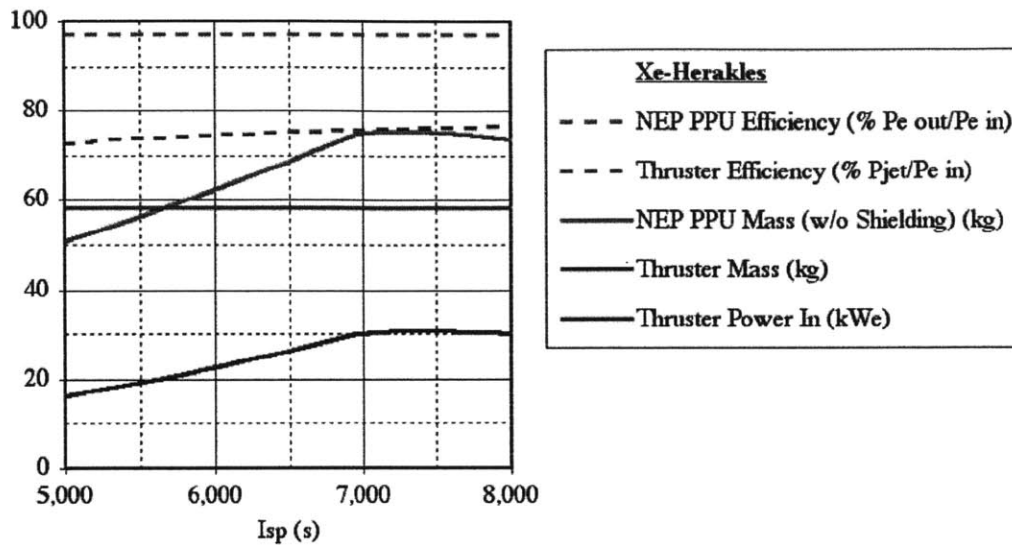


Figure 8: Reactor power efficiency, thruster power and efficiency as a function of specific impulse. The thruster power increases non-trivially with specific impulse.

However due to lifetime constraints, the specific impulse may need to be decreased to up to a minimum of 6,000 s which would imply using a power of 23 kW per thruster for a total power of 138 kW. This minimum power complies with the constraints given by the power given by the nuclear reactor and the dissipative capabilities of the spacecraft. Therefore, as long as the specific impulse is set between the recommended parameters, the power complies with the minimum (130 kW) and maximum requirements (190 kW) of the power produced.

7.2.4 Lifetime

A model for the life estimation of the ion thrusters has been made [5]. This life estimation model provides rough estimates of thruster lifetime and provides some insight into the possible failure points of ion thruster operation at different times and conditions [5]. However, the model is not a detailed physical model simulation, but rather an extrapolation of a collection of simple, independent models and fits to limited empirical data of similar devices.

The most likely ion thruster failure modes modeled are discharge cathode failure due to barium depletion, discharge cathode failure due to keeper face erosion, electron back-streaming

due to accelerator grid wall erosion, and accelerator grid failure due to pits-and-grooves erosion on the downstream surface [5]. Other important wear mechanisms have not been modeled, such as discharge cathode failure due to the formation of coatings on the emitting surface.

At the nominal operating point of a system power of 180 kW at a specific impulse of 7000 s, the life model shows that the thruster will fail at approximately 164,000 hours or roughly 18 years [5]. This lifetime is below the needed requirement of 20 years. Furthermore, no ion thruster of this power has been tested for more than 3 years continuously [5]. Possible responses to this setback are to include an emergency set of thrusters, but that would double the cost of the ion thruster propulsion system. Other solutions involve reducing the specific impulse closer to 6,000 s, thus lowering the corrosion in the accelerator grids with a trade-off of reducing the efficiency of the fuel. Therefore, further investigation must be done to find ways to extend the lifetime of ion thrusters such that they are suitable to perform long missions without reducing efficiency of adding redundant costs.

VIII. CONCLUSION

The purpose of this thesis was to review the physics and evaluate the design electric propulsion system of Project Prometheus. In order to accomplish this goal, the background physics for ion thrusters was presented, covering important topics such as thrust, specific impulse, power efficiency and ion thruster principles.

Using these principles, the design constraints for Project Prometheus as stated by JPL were presented and discussed. For each of these constraints, the possible design responses were evaluated by using the results from simulations and experiments performed in similar propulsion systems. The main design responses that were evaluated were the following: the amount of propellant mass, the choice of propellant, the specific impulse, and the lifetime of the thrusters.

The maximum amount of propellant that the spacecraft can carry is 18,000 kg. A model that estimated the dependance of the mass of each component on power was constructed. The results of this model were similar to the design parameters design by JPL at the optimal point. Several

methods, were suggested to improve this model.

The ideal propellant for the mission was xenon due to being inert. However, krypton came as a close second by having similar efficiency and thrust to xenon and being cheaper to produce. The lack of empirical testing, however, makes krypton for this mission, but could be considered for future projects with reduced budget.

The power of the thrusters was found to vary with the specific impulse. For the range between 6,000 s and 8,000 s the total power ranged between 136 kW and 180 kW. These limits complied with the power requirements by the reactor and the radiator in the spacecraft.

Some initial lifetime tests were simulated for the Herakles thruster. It was found that at 7,000 s of specific impulse, the thruster was not expected to withstand the 20 years required for the mission. Therefore, future studies must be made to see whether lowering the specific impulse (and hence sacrificing fuel efficiency) would allow the lifetime of the thrusters to increase.

Overall these results show that Project Prometheus had solid design foundations that would have allowed for a plausible space mission. However, as the project was cancelled, many of the unknowns, such as the relation between lifetime and specific impulse, or the plausibility of krypton being used as propellant for long missions will have to be explored by future projects.

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