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Electric Propulsion System Modeling for the Proposed Prometheus 1 Mission

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Abstract

The proposed Prometheus 1 spacecraft would utilize nuclear electric propulsion to propel the spacecraft to its ultimate destination where it would perform its primary mission. As part of the Prometheus 1 Phase A studies, system models were developed for each of the spacecraft subsystems that were integrated into one overarching system model. The Electric Propulsion System (EPS) model was developed using data from the Prometheus 1 electric propulsion technology development efforts. This EPS model was then used to provide both performance and mass information to the Prometheus 1 system model for total system trades. Development of the EPS model is described, detailing both the performance calculations as well as its evolution over the course of Phase A through three technical baselines. Model outputs are also presented, detailing the performance of the model and its direct relationship to the Prometheus 1 technology development efforts. These EP system model outputs are also analyzed chronologically showing the response of the model development to the four technical baselines during Prometheus 1 Phase A.

Nomenclature

e	Unit Charge, 1.602×10^{-19} C
E_b	Discharge Losses (W_e/A or eV/ion)
f	Doubles to Singles Current Ratio
f_{ab}	Ratio of Accelerator Grid Current to Beam Current
$f(\omega)$	Coefficient of PP&C Beam Module Mass Equation (kg/$kW_e^{0.87}$)
g_0	Acceleration due to Earth's gravity, 9.80665 m/s^2
I_{SP}	Specific Impulse (seconds)
J_b	Beam Current (A)
J_d	Discharge Current (A)
m_{Beam}	PP&C Beam Module Mass (kg)
$m_{Discharge}$	PP&C Discharge Module Mass (kg)
m_i	Xenon atomic mass, 2.180×10^{-25} kg
$\dot{m}_T$	Total Mass Flow Rate (kg/s)
η	Total Thruster Efficiency
η_α	Doubles to Singles Correction Factor
η_ε	Electrical Efficiency
η_θ	Beam flatness parameter, the ratio of the average to peak current density

η_u	Propellant Utilization Efficiency
P_l	Sum of the Miscellaneous Power Losses (W_e)
P	Total Power Into the Thruster (W_e)
$P_{PP\&C}$	Total Power Into the Power Processing & Control Unit (kW_e)
R	R-ratio, ratio of beam plus coupling voltage to beam plus absolute value of accelerator voltage
T	Thrust (N)
V_b	Beam Voltage (V)

The Prometheus Project, started in 2002 as the Nuclear Systems Initiative, was tasked with developing nuclear electric propulsion (NEP) capability for the National Aeronautics and Space Administration (NASA) civilian space missions[1]. NASA conducted studies from mid-2002 through early 2003 to assess what was then called a Jupiter Icy Moon Tour mission[2]. Early in 2003, Congress appropriated funding to begin studying a Jupiter Icy Moons Orbiter (JIMO) mission[1,3], which was originally slated as the first mission for this NEP system. Technology tasks were also started as part of the Prometheus Project to develop the reactor[1], power conversion[1,4,5], and electric propulsion systems[1,6].

One task created as part of the Prometheus Project was the System Modeling and Analysis (SM&A) task. The goal of this task was to develop a system model for the Prometheus 1 spacecraft[7]. This model, containing eight modules representing different subsystems of the spacecraft, the launch vehicle, and mission design, was then used to perform linked analyses of the mission and systems[8]. The Electric Propulsion System (EPS) model, as part of the SM&A task, was required to accept inputs, such as power level and specific impulse, and provide an EPS designed for the particular inputs. The EPS model would then output parameters needed by the Prometheus system model and other models, such as masses and efficiencies, so that the design could be iterated upon and a converged mission and system solution obtained.

II. Electric Propulsion System Baseline Designs

The Prometheus Project had three Technical Baselines that required EPS model development for the JIMO mission. Technical Baseline 2 (TB2) was defined in late 2003 and early 2004, Technical Baseline 2.5 (TB2.5) was defined in the Fall of 2004, and Prometheus Baseline 1 (PB1) was defined in late 2004 and early 2005. TB2 and TB2.5 were the results of the government-only Phase A studies, whereas PB1 incorporated the team from Northrop Grumman Space Technologies (NGST) after the award of the Phase A/B contract. An updated model, based on PB1, is also being developed, which includes updated ion thruster performance and lifetime modeling

Two EPS designs were carried for TB2 because two ion thrusters and two power conversion systems were still under development. One design paired Brayton power conversion with the High Power Electric Propulsion[9] (HiPEP) ion thruster and the other paired thermoelectric power conversion with the Nuclear Electric Xenon Ion System[10] (NEXIS) ion thruster. Each carried individual power and I_{SP} set points as well as voltage input type to the PP&C system (AC for Brayton and DC for thermoelectric). TB2 also included a 4.5 kW_e Hall thruster system to provide acceleration augmentation during the moon tour at Jupiter. The TB2 nominal parameters for each configuration are presented in Table 1.

Table 1. Summary of Key Design Parameters for Each of the EPS Baselines

	TB2		TB2.5		PB1
	Brayton/HiPEP	TE/NEXIS	Brayton/HiPEP	Brayton/NEXIS	Brayton/Herakles
Ion Thruster System					
Total Power Into the EP System	95 kW_e	75 kW_e	130 kW_e	130 kW_e	180 kW_e
Specific impulse	6500 seconds	5500 seconds	6000 seconds	6000 seconds	7000 seconds
Voltage Type into PP&C System	AC	DC	AC	AC	AC
Ion Thruster Type	HiPEP	NEXIS	HiPEP	NEXIS	Herakles
Number of Operational Ion Thrusters	4	6	6	6	6
Number of Spare Ion Thrusters	2	2	2	2	2
Tank Capacity	14000 kg	14000 kg	18000 kg	18000 kg	12000 kg
Hall thruster systems					
Total Power to the Hall PPUs	24 kW_e	25 kW_e	0 kW_e	0 kW_e	120 kW_e
Hall Specific impulse	2059 seconds	2059 seconds			2000 seconds
Number of Operational Hall Thrusters	5	5	0	0	6
Number of Spare Hall thrusters	1	1	0	0	0
Total Number of Reaction Control Hall Thrusters	0	0	12	12	12

While Brayton power conversion was selected for TB2.5 and all following baselines, TB2.5 still included two EPS designs because the separate HiPEP and NEXIS thruster developments were still continuing. TB2.5 redirected the function of the 4.5 kW_e Hall thruster system to perform attitude and reaction control functions, requiring a larger quantity of thrusters. Each of the TB2.5 EPS designs used the same defining parameters as shown in Table 1.

PB1 introduced the first unified design of the EPS. The HiPEP and NEXIS developments were combined and one ion thruster design was developed named Herakles[11]. This baseline also incorporated the team from NGST to help develop the baseline design. The EPS design for PB1 included a total of 26 electrostatic thrusters, 8 ion, 6 high power Hall thrusters, and 12 attitude/reaction control Hall thrusters[12]. The defining parameters for the PB1 EPS are shown in Table 1.

III. Prometheus Electric Propulsion System Modeling

The Prometheus EPS Model is required to estimate the mass of the EPS, performance information (efficiencies and quantities required of the different components), and in some cases dimensional information. Each of the EPS models estimates the mass, critical dimensions, and performance of the thruster systems as well as its associated support equipment (PP&C system, propellant management system, and xenon tank). Depending on the EPS configuration for each of the Technical Baselines, the model also calculates the mass and performance of up to two Hall thruster systems, one for acceleration augmentation and the other for attitude and reaction control. Some of the models also include a small xenon cold gas reaction control system.

The models were designed to operate from a simple set of inputs, with the other pertinent parameters assumed based on the baseline designs. The ion thruster system was sized by providing input power, specific impulse, and total propellant throughput required, and based on calculations performed in the ion thruster model, the required quantity of thrusters was determined and used to calculate the mass of the system. The acceleration augmentation Hall thruster system was sized in much the same way as the ion system in the PB1 model, however for the TB2 model, the input power was determined by specifying the quantity of ion thrusters turned off to operate the Hall system, not the Hall system input power directly.

A. Ion Thrusters

Two ion thruster performance models were implemented over the course of the model development. The first, a purely physics-based model, was used during the early stages of development of the ion thruster system hardware (HiPEP and NEXIS). The second, incorporates ion thruster testing experience and was implemented as the ion thruster hardware design was finalized (Herakles). Each of these models requires power input to the thruster and required specific impulse in order to calculate the thruster operating point and performance. The thruster mass models were tied directly to mass estimates provided through the technology development efforts, whereas the performance estimates were somewhat unrelated to the technology development until the Herakles thruster design was developed.

1. Physics-based Ion Thruster Model

The purely physics based model is adapted from work by Wilbur[13] and Brophy[14] with the Current Best Estimate (CBE) of performance constants provided by the ion thruster technology development team (see Table 2) and updated periodically as the thruster development progressed. The equations, as presented in simplest form, require inputs of beam current (J_b) and beam voltage (V_b), and calculate the thruster performance with knowledge of the performance constants. These equations were solved algebraically to require thruster input power and specific impulse as inputs for the physics-based model.

Table 2. Performance Constants for the Physics-based Ion Thruster Model

Parameter	TB2 & TB2.5	PB1
η_u	0.87	0.86
E_b	170 W/A	210 W/A
f	0.05	0.053
η_θ	0.98	0.98
P_l	30 W	0 W
R	0.9	0.86
f_{ab}	0.005	0.008

Total thruster input power is calculated using the beam current and voltage as well as knowledge of the power losses of the thruster.

$$P = J_b V_b + J_b E_b + J_b f_{ab} \, {V_b}/{R} + P_l \tag{1}$$

The electrical efficiency of the thruster is the ratio of beam power to total power (Eq. 1).

$$\eta_\varepsilon = \frac{J_b V_b}{P} \tag{2}$$

The doubles-to-singles correction factor is a function of the ratio of doubly charged to singly charged ions thruster constant.

$$\eta_\alpha = \frac{1 + f\,0.5^{0.5}}{1 + f} \tag{3}$$

The input beam current and propellant utilization allow calculation of the thruster total mass flow rate.

$$\dot{m}_T = \frac{m_i J_b}{e \eta_u} \tag{4}$$

The thrust produced by each ion thruster is a function of both beam current and voltage.

$$T = \left(\frac{2m_i}{e}\right)^{1/2} \eta_\alpha \eta_\theta J_b V_b^{1/2} \tag{5}$$

Specific impulse is calculated with knowledge of the thrust produced and total mass flow rate.

$$I_{sp} = \frac{T}{g_0 \dot{m}_T} \tag{6}$$

Finally, the total thruster efficiency is calculated as a function of the above calculated and specified efficiencies.

$$\eta = \eta_\varepsilon \eta_u \eta_\alpha{}^2\, \eta_\theta^2 \tag{7}$$

Because sufficient modeling of the ion thruster life-limiting mechanisms had not been completed when this performance model was in use, a simple algorithm was used to determine the number of thrusters required at a given operating condition. First, the current density was limited to 2.2 mA/cm^2, which determined the minimum quantity of operating thrusters required at any operating point. (The current density is defined as the beam current (J_b) divided by the active beam area of the thruster. The active beam area is the area of the ion thruster grids that contain orifices.) Second, the total propellant throughput per thruster was limited to 1.5 kg/cm^2 of active beam area. This throughput limit determined the quantity of thrusters required to process the total amount of propellant.

2. Physics and Test-based Ion Thruster Model

The most recent iteration of the ion thruster performance model includes some performance relationships based on test data from the NSTAR[15] and NEXT[16] thrusters as well as other precursor and lab hardware. This model uses many of the relationships presented in the last section as well as some of the

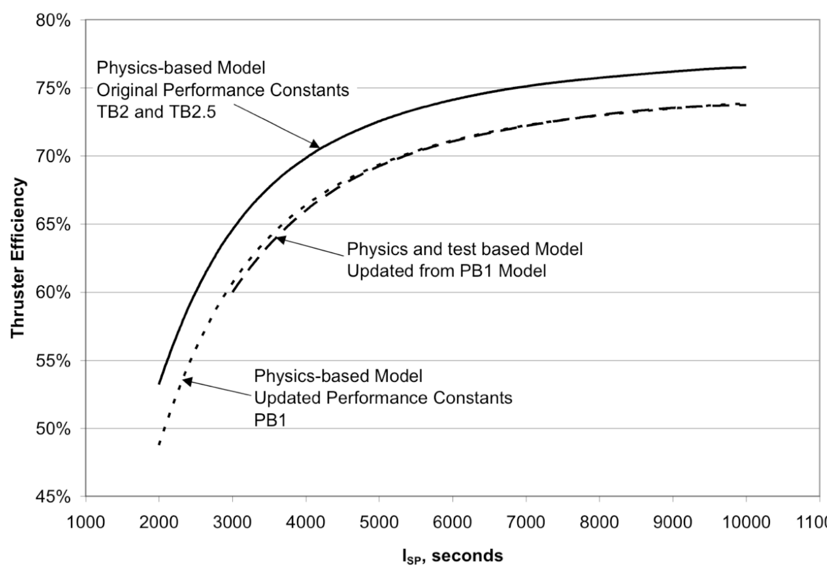


Figure 1. Performance Model Comparison

same performance constants, but incorporates knowledge gained from ion thruster test experience. This inclusion of testing experience results in slightly lower thruster efficiency at the lower range of I_{SP}. A comparison of these models is presented in Figure 1.

3. Ion Thruster Life Model

The current version of the life estimation model was implemented to replace a more crude method of estimating the lifetime of ion thrusters. The earlier methodology involved placing a current density limit on the thrusters at which current density it was assumed that the thruster could survive long enough to process 1.5 kg of xenon per square centimeter of grid area. The result was a conservative estimate of how long an ion thruster might survive, but it failed to provide any insight into what mechanism would cause the eventual failure, nor did it provide a sense of the effect of changes to the system. The new life estimation model was developed as a tool for systems engineers to provide what the old model lacked.

The life estimation model provides rough estimates of thruster lifetime for systems engineering purposes and provides some insight into the ramifications of ion thruster operation at different conditions on thruster lifetime. However, it is not a detailed physical model. It is a collection of simple, independent fits to more detailed models and fits to empirical data. Four of the 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 hole wall erosion, and accelerator grid failure due to pits-and-grooves erosion on the downstream surface. Other wear mechanisms have not been modeled, such as discharge cathode failure due to the formation of coatings on the emitting surface. The results the life model produces are obtained quickly with conservatism built-in and the interface is meant to be usable by non-experts. A separate life modeling and testing effort is developing detailed physics-based models that accurately predict lifetime and are validated by extensive experimental data[17].

At the nominal operating point of a system power of 180 kW at a specific impulse of 7000 seconds, the life model shows that the thruster will fail at approximately 164 khrs. The life model is also used to calculate the quantity of thrusters needed for the mission. To determine the quantity of thrusters needed, the ion thruster model iterates the life model through the applicable operating points and chooses the minimum number of thrusters that are able to achieve the required lifetime.

4. Ion Thruster Mass Model

The ion thruster mass model was simply a result of the ion thruster technology development. The TB2 and TB2.5 mass models were based on the individual HiPEP and NEXIS ion thruster CBE mass, while the PB1 models used the CBE of the Herakles ion thruster mass. The Herakles design incorporated technologies developed for both HiPEP and NEXIS as well as lessons learned, and was not merely a down select to one thruster as the mass estimate may imply[11]. Ion thruster masses used in the system model for each technical baseline are tabulated in Table 3.

Table 3. Ion Thruster Masses for Each Baseline Design

	TB2	TB2.5	PB1
HiPEP ion thruster	46.5 kg	46.5 kg	
NEXIS ion thruster	28.7 kg	28.7 kg	
Herakles ion thruster			49.5 kg

B. Acceleration Augmentation Hall Thruster System

An additional electric propulsion system model module was developed to allow the evaluation of augmenting the thrust provided by the ion thrusters to meet minimum acceleration requirements. Initially, this system was based on a 4.5 kW_e Hall thruster system, but after TB2 the need was recognized for a higher power, newly designed Hall thruster system. This auxiliary system was based on Hall thruster technology that has demonstrated specific impulses in the range of 1400 – 2800 seconds at power levels up to 50 kilowatts[18]. Thruster performance was predicted based on an empirical relationship between thruster efficiency and specific impulse based on a range of experimental data[18,19,20,21,22]. The empirical curve fit had the following form:

$$\eta = 0.024\, I_{sp}^{0.4} \tag{8}$$

Thruster mass was estimated based on the thruster power and thruster size. A second empirical relationship between specific impulse and discharge voltage based on the experimental data shown in Figure 2 was used to determine the discharge voltage corresponding to the specific impulse of interest. This empirical relationship was:

$$V = 0.013\, I_{sp}^{1.35} \tag{9}$$

The discharge current was then determined based on the desired thruster power and discharge voltage assuming thruster power could be estimated as the product of discharge voltage and discharge current. This assumption neglected the electrical power that is required to operate electromagnets, however, based on the prior experimental data this power is on the order of 1-2% of the total input power. As a result this error was considered acceptable. The thruster size was then determined based on propellant density requirements and the relationship between propellant mass flow rate and discharge current[23]. The result of these assumptions was a curve of Hall thruster specific mass as a function of specific impulse. This relationship, shown in Figure 3, was an attempt to capture the mass penalty associated with increased thruster size corresponding to low specific impulse operation resulting from a scaling methodology based on maintaining constant current and mass flow density. The state-of-the-art 10.1 kilogram 4.5 kW_e BPT-4000 Hall thruster specific mass[24] compared favorably with the model of thruster specific mass versus specific impulse used for this investigation. The acceleration augmentation Hall thruster masses at each baseline design are shown in Table 4.

Table 4. Acceleration Augmentation Hall Thruster Parameters for Each Baseline Design

	TB2	TB2.5	PB1
Input Power	4.5 kW_e		20.0 kW_e
Hall Thruster Mass	12.3 kg		41.2 kg

Because this system was used to augment the acceleration of the ion thruster system, joint operation of ion and Hall thrusters was assumed. This secondary system, composed of both ion and Hall thrusters, provided sufficient acceleration to maintain stable orbits at Jupiter and its moons. Therefore, calculation of the performance of the combined system was required. The combined performance was calculated by summing the thrust and mass flow rate of the two systems and then solving for specific impulse and efficiency.

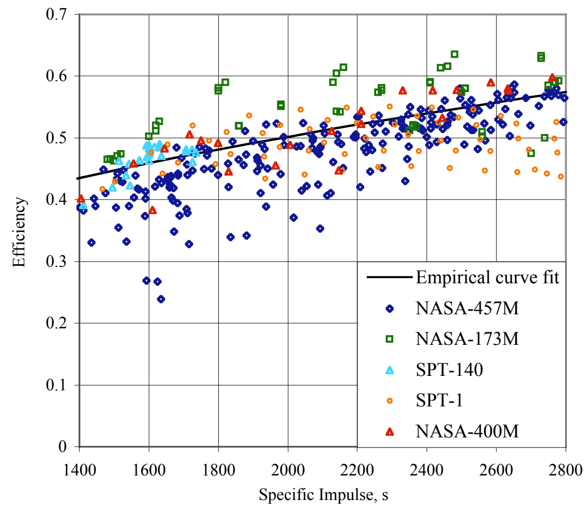


Figure 2. Hall thruster performance data and empirical fit

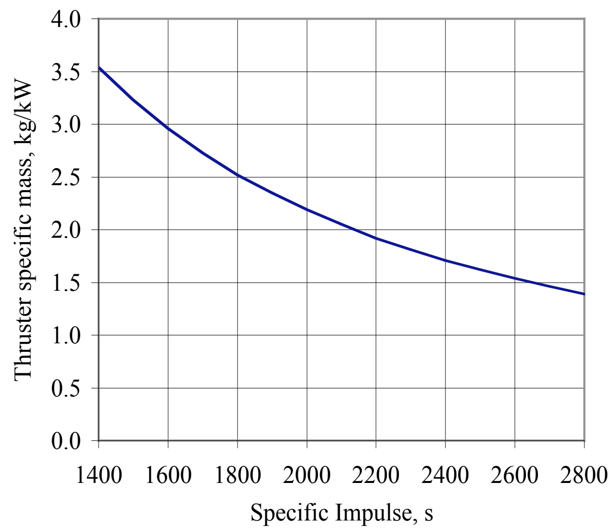


Figure 3. Hall thruster specific mass

C. Attitude and Reaction Control Hall Thruster System

TB2.5 and PB1 utilized a Hall thruster system to perform attitude and reaction control functions for the spacecraft. The TB2.5 model utilized a reconfigured 4.5 kW_e Hall thruster system to perform these functions. Specifically, the thruster quantity was increased to 12 thrusters cross-strapped to six PP&C units. The PB1 model introduced a smaller Hall thruster system, also composed of 12

Table 5. Attitude and Reaction Control System Parameters for Each Baseline Design

	TB2	TB2.5	PB1
Quantity of Thrusters		12	12
Thruster Input Power		4.5 kW_e	0.7 kW_e
Hall Thruster System Mass		274 kg	83 kg

thrusters cross-strapped to 6 PP&C units, with 700 W_e thrusters. The attitude and reaction control Hall thruster system parameters at each technical baseline are presented in Table 5.

D. Power Processing and Control System

The Power Processing and Control (PP&C) System model is a direct representation of the PP&C technology development[25]. Initially, two PP&C systems were designed to accept AC and DC input from the Brayton and thermoelectric power conversion systems, respectively. The DC PP&C system was scaled from the NEXT power-processing unit[26] while the AC PP&C system's discharge and ancillary supplies (cathode heaters and keepers, accelerator supply, controller, etc.) were also scaled from the NEXT power-processing unit and mated with a newly designed transformer-rectifier module to provide main beam power. PP&C system efficiencies were estimated based on the current state of the technology development for each baseline and for each input voltage type and assumed constant over the range of input power levels (see Table 6). PP&C unit dimensions were also estimated to allow calculation of radiation shielding mass. Because calculation of radiation shielding mass was done externally to the EPS model and the PP&C unit dimensions have no impact on the EPS design, neither the PP&C unit dimensions nor the radiation shielding mass are discussed herein.

Table 6. Ion PP&C System Parameters for Each Baseline Design

	TB2	TB2.5	PB1
AC PP&C Unit			
PP&C Unit Input Power	23.8 kW_e	32.5 kW_e	30.0 kW_e
Efficiency	96.5%	96.5%	95.0%
AC Frequency	1000 Hz	1000 Hz	2200 Hz
$f(\omega)$	2.92	2.92	1.47
PP&C Unit Mass	57.9 kg	72.4 kg	50.3 kg
DC PP&C Unit			
PP&C Unit Input Power	12.5 kW_e		
Efficiency	94.0%		
PP&C Unit Mass	37.6 kg		

The PP&C system mass model evolved with each technical baseline as the technology development progressed. The DC PP&C model implemented for TB2 combined component module masses from the NEXT power-processing unit. These modules were either scaled up in mass, or extra components were added to accommodate higher power levels. The AC PP&C model evolved with each baseline design as the PP&C system technology was developed. Each AC PP&C model utilized a simple power function (Eq. 8) to size the beam module transformers with different coefficients that were dependent on the frequency of the input power supplied.

$$m_{Beam} = f(\omega) P_{PP\&C}^{\ 0.87} \tag{12}$$

The AC frequency coefficients ($f(\omega)$) are presented in Table 4. The masses of the other PP&C system components were modeled differently for each of the technical baselines. For TB2 and TB2.5 the discharge and ancillary power supplies were modeled as a constant 12 kg mass. The discharge power supply was modeled as

$$m_{Discharge} = 0.105 J_d \tag{13}$$

for PB1 with a constant 17 kg carried for the ancillary supply mass. The PP&C unit masses at each of the technical baselines are also presented in Table 6.

The Hall thruster PP&C units were modeled in much the same way as the ion thruster PP&C units. The TB2 and TB2.5 Hall systems, although performing different functions, were 4.5 kW_e Hall thruster systems. This Hall system was estimated to have a PP&C system design mass of 6 kg in the direct-drive configuration of TB2 and 12.6 kg for TB2.5, both with AC input. The PB1 model used the transformer mass estimate (Eq. 12) and to it added 9 kg of mass for ancillary supplies for the acceleration augmentation Hall system and 4 kg for the attitude and reaction control system. System parameters for the Hall thruster PP&C systems at each baseline design are presented in Table 7.

Table 7. Hall PP&C System Parameters for Each Baseline Design

	TB2		TB2.5	PB1
	Brayton/HiPEP	TE/NEXIS	HiPEP or NEXIS	Herakles
Acceleration Augmentation				
PP&C Unit Input Power	4.7 kW_e	4.7 kW_e		20.0 kW_e
Efficiency	96.0%	96.0%		94.0%
PP&C Unit Mass	6.0 kg	4.2 kg		28.9 kg
Attitude & Reacton Control				
PP&C Unit Input Power			4.7 kW_e	0.7 kW_e
Efficiency			96.5%	94.0%
PP&C Unit Mass			12.6 kg	5.1 kg

E. Propellant Management System

The Propellant Management System mass model was comprised of three parts; the isolation and pressure regulation module, the xenon flow control assembly, and the xenon recovery system. The isolation and pressure regulation module isolates the tank from the downstream components and regulates the pressure of the xenon from the tank to a lower pressure useable by the xenon flow control assemblies. The xenon flow control assemblies receive the regulated xenon from the isolation and pressure regulation module and splits and controls the flow to the two cathodes and the main discharge of the ion thrusters. The xenon recovery system allows xenon to be extracted from the tank once the tank pressure reaches a pressure too low for normal propellant management system operation. One isolation and pressure regulation module and one xenon recovery system is needed, as well as one xenon flow controller per thruster. This model was based on the current design of the propellant management system at the time of the technical baselines. The propellant management system masses at each of the technical baselines are presented in Table 8. The changes in masses between technical baselines were due to updates to the CBE mass as the propellant management system design evolved.

Table 8. Propellant Management System Masses for Each Baseline Design

	TB2		TB2.5	PB1
	Brayton/HiPEP	TE/NEXIS	HiPEP or NEXIS	Herakles
Shared Hardware				
Isolation and Pressure Regulation Module	10.2 kg	10.2 kg	10.2 kg	16.4 kg
Xenon Recovery System	23.1 kg	23.1 kg	23.1 kg	23.0 kg
Ion Thruster Hardware				
Xenon Flow Control Assembly	3.9 kg	3.9 kg	3.8 kg	5.6 kg
Tubing, fittings, structure, etc.	10.3 kg	10.3 kg	10.3 kg	6.3 kg
Quantity of Ion Thrusters	6	8	6	8
Acceleration Augmentation Hall Hardware				
Xenon Flow Control Assembly	0.7 kg	0.7 kg		2.7 kg
Tubing, fittings, structure, etc.	0.0 kg	0.0 kg		1.7 kg
Quantity of Large Hall Thrusters	6	6		6
ACS Hall Thruster Hardware				
Xenon Flow Control Assembly			0.7 kg	2.1 kg
Tubing, fittings, structure, etc.			0.2 kg	0.4 kg
Quantity of Large Hall Thrusters			12	12
Total	123.0 kg	151.4 kg	129.3 kg	192.0 kg

F. Xenon Tank

Two different xenon tank models were implemented in the EPS model. The first was based on point designs at the time of each technical baseline, with any capacity deviation completed by scaling the tank mass linearly with propellant capacity mass. The second xenon tank model is a higher fidelity model based on previous tank design experience and the known properties of xenon at the required state. Both models assumed that the xenon was stored as a supercritical fluid. TB2 and TB2.5 utilized the simple linearly scaled tank, while PB1 used the higher fidelity tank model. The xenon tank masses are presented in Table 9.

Table 9. Xenon Tank Masses for Each Baseline Design

	TB2	TB2.5	PB1
Tank Capacity	14000 kg	18000 kg	12000 kg
Tank Mass	361 kg	476 kg	352 kg

G. Xenon Reaction Control System

A small xenon cold gas thruster system was included in TB2 and TB2.5 to cancel any perturbations due to the launch vehicle separation before power was available for the EPS. This system was designed using currently available commercial components. The I_{SP} of this system was assumed to be 29 seconds, and a constant mass of 5.1 kg was included in the EPS model for this system.

IV. Electric Propulsion System Model Performance

The performance of the EPS models developed for each of the technical baselines provides insight into both the baseline designs and the evolution of the EPS model. The four models, TB2, TB2.5, P1, and updated P1, were run at input power levels between 50 kW_e and 300 kW_e at a constant I_{SP} of 7000 seconds (note that only the Brayton/HiPEP configuration was used for the TB2 model data). The total EPS mass output from each model, as well as marks denoting the three technical baselines can be seen in Figure 4. Mass differences between models are primarily due to EPS configuration changes (i.e. addition of Hall thruster systems). The discrete steps in the data occur when the EPS model increments the quantity of thrusters, either ion or Hall, to process the total amount of power input to the EPS.

The mass breakdown of each EPS model output provides further understanding of the performance of the models. Figure 5 shows plots of the mass breakdown of each EPS model. Also detailed are the discrete jumps in the data where the quantity of ion thruster changes. Examination of the data labeled "Hall Mass," also reveals discrete jumps where the quantity of Hall thrusters changes. The Hall thruster system mass for the TB2 model (shown in

Figure 5a) is the result of setting the input power of the Hall thrusters as the power gained by shutting down one ion thruster at all input power levels, whereas the Hall thruster system mass for the two PB1 models (shown in Figure 5c and d) is the result of approximately maintaining the ratio of Hall system input power to total EPS input power.

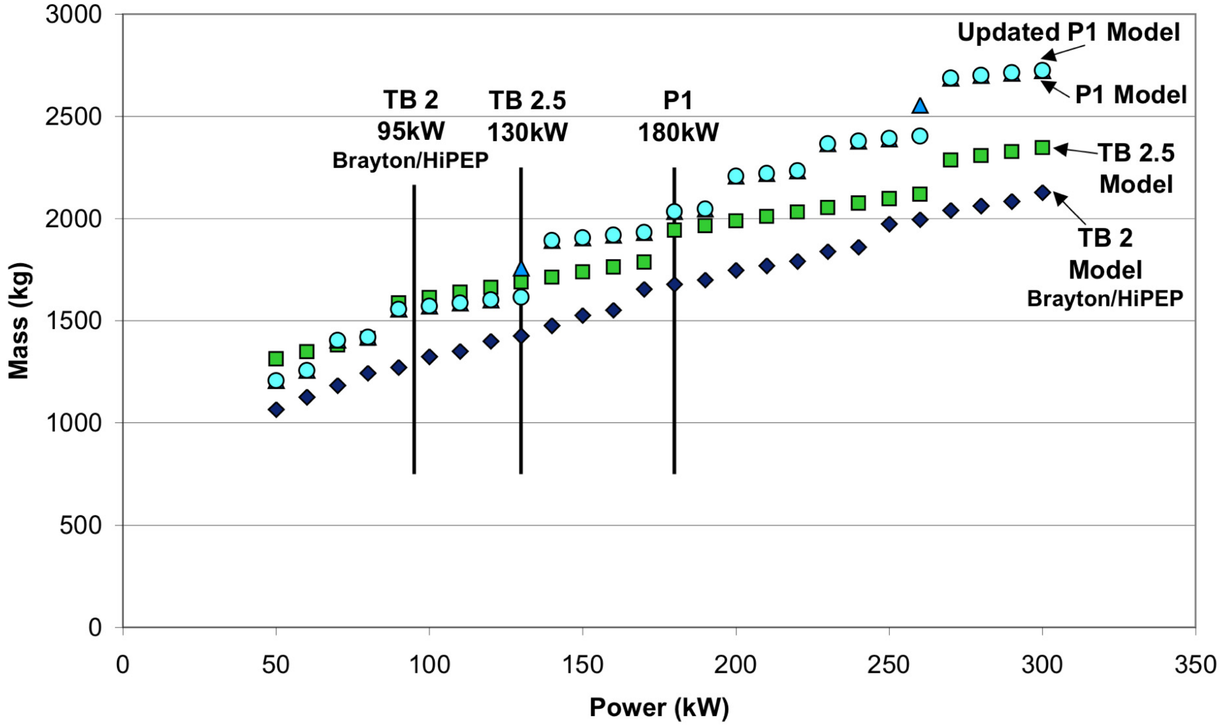
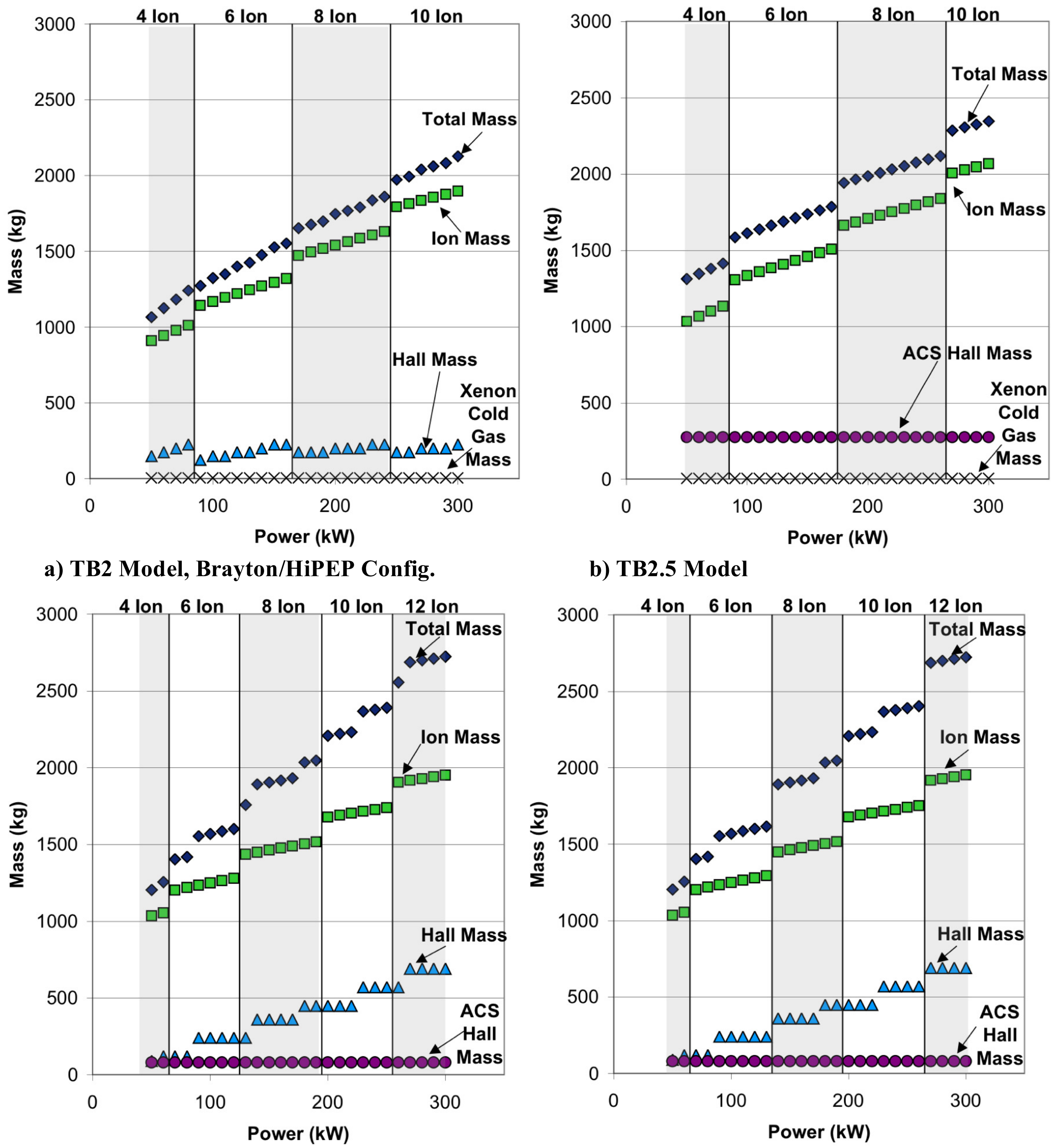


Figure 4. Mass Performance of the EPS Model Iterations

V. Future Plans for the Electric Propulsion System Model

Future modifications to the EPS model fall into two categories; modifications based on technology development and capability enhancements. Modifications based on technology development include implementation of any design changes throughout the development of the EPS hardware, modifying the performance assumptions as performance tests of the EPS hardware are completed, and updating life model assumptions to reflect the wear mechanisms examined through long duration hardware tests. Many performance and life tests are planned as part of the Prometheus Project, which may result in design modifications of EPS hardware. These design modifications must be reflected in the EPS model in order to accurately represent the EPS in trades performed with the Prometheus system model. Capability enhancements that may be implemented mainly add flexibility to the EPS model and make it a more general tool, not necessarily tailored to the proposed JIMO mission. These capability enhancements could include several thruster modules representing different technologies that can be switched out easily depending on the mission requirements, a second tank and propellant management system model for cryogenic storage of propellant, or a higher fidelity approach to tracking uncertainties in the performance of the thrusters over the length of the mission.



a) TB2 Model, Brayton/HiPEP Config.

b) TB2.5 Model

c) PB1 Model

d) Updated PB1 Model

Figure 5. Mass Performance Breakdown of Each EPS Model

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13. ABSTRACT *(Maximum 200 words)*

The proposed Prometheus 1 spacecraft would utilize nuclear electric propulsion to propel the spacecraft to its ultimate destination where it would perform its primary mission. As part of the Prometheus 1 Phase A studies, system models were developed for each of the spacecraft subsystems that were integrated into one overarching system model. The Electric Propulsion System (EPS) model was developed using data from the Prometheus 1 electric propulsion technology development efforts. This EPS model was then used to provide both performance and mass information to the Prometheus 1 system model for total system trades. Development of the EPS model is described, detailing both the performance calculations as well as its evolution over the course of Phase A through three technical baselines. Model outputs are also presented, detailing the performance of the model and its direct relationship to the Prometheus 1 technology development efforts. These EP system model outputs are also analyzed chronologically showing the response of the model development to the four technical baselines during Prometheus 1 Phase A.

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