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Power and Propulsion Element (PPE) Spacecraft Reference Trajectory Document

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March 2020

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PREFACE

National Aeronautics and Space Administration's (NASA's) proposed Gateway concept would provide a crew-tended outpost in orbit near the Moon, which could be used as a staging point for exploration of the Moon and, eventually, Mars. The first piece of the Gateway to be deployed is the Power and Propulsion Element (PPE).

This document captures example reference trajectories for the PPE including a reference delivery orbit and orbit maintenance, an example cislunar orbit transfer and end-of-mission (EOM) disposal trajectory. The flexibility of electric propulsion offers, by its low thrust nature, multiple different trajectory options to transfer from one orbit to another. The trajectories captured in this document are representative examples of a low thrust transfer from the NRHO and to multiple cislunar orbits. This document provides a consistent set of data from mission design to be used in the design of the vehicle capable of flying the trajectory described.

CONTENTS

FIGURES

TABLES

1.0 INTRODUCTION

1.1 Purpose

The purpose of this document is to capture the current reference trajectories for the Power and Propulsion Element (PPE) and provide a consistent set of data from mission design to be used in the design of the spacecraft capable of flying the trajectories described. Captured in this document is the summary of the Ground Rules and Assumptions (GRA) behind the trajectory analysis for the PPE and reference trajectories developed using this GRA. The target cislunar orbit and spacecraft design assumptions represent a current baseline to use for the PPE design, but do not represent finalized mission parameters and may change as the mission design evolves.

The PPE captured in this analysis is based on the government reference configuration concept design and pre-dated the final selection of the PPE commercial partner. This document is meant to capture an application of the PPE as the propulsive element of the Gateway at the time of partner selection. The transfer trajectories are representative of what the Gateway would fly if propelled by a low thrust propulsion system. The analysis captured in this document has been used by Gateway to develop vehicle and mission requirements. Future analysis will apply the actual performance assumptions of the PPE commercial partner vehicle. The next version of this reference trajectory document will update all assumptions to be consistent with the PPE commercial partner design.

1.2 Scope

This document outlines the target Near Rectilinear Halo Orbit (NRHO) to which the Power and Propulsion Element is expected to be delivered. Also presented is a brief overview of NRHO orbit maintenance, a notional one-way orbit transfer from the reference southern NRHO to a lunar Distant Retrograde Orbit (DRO), and a longer NRHO to DRO transfer as a potential Gateway end-of-mission disposal option.

This document focuses on documenting the cislunar transfers of the Gateway using the government reference power and propulsion element assumptions, in order to capture the Xe propellant needs of the Gateway to perform two transfers during its 15-year lifetime.

Additionally, the orbit maintenance strategy and propellant estimations for the 15-year lifetime of the Gateway in the NRHO is also captured.

Any hydrazine or Xenon propellant used by the RCS or EP systems during checkout or insertion into LEO or the NRHO are ignored for the purpose of this document and left for future analysis and future reference trajectory documents to capture.

1.3 Mission Overview

In March of 2017, NASA announced its conceptual next steps for exploration of destinations beyond LEO. In order to move toward these next steps, the first phase of this exploration will involve the building of a Gateway in cislunar orbit. The Gateway is envisioned as an outpost for lunar exploration and for explorations further away from Earth and eventually to Mars.

Key to the enabling of exploration in the vicinity of the Moon, the PPE of the Gateway will be capable of transferring the entire Gateway stack between various cislunar orbits as well as capable of maintaining itself in the selected NRHO. A notional NASA graphic of the Gateway in the NRHO, can be seen in Figure 1-1. In this representative image, the notional lunar lander elements are attached at the top of the image, and an Orion piloted vehicle is shown approaching from the right to dock with the Gateway.

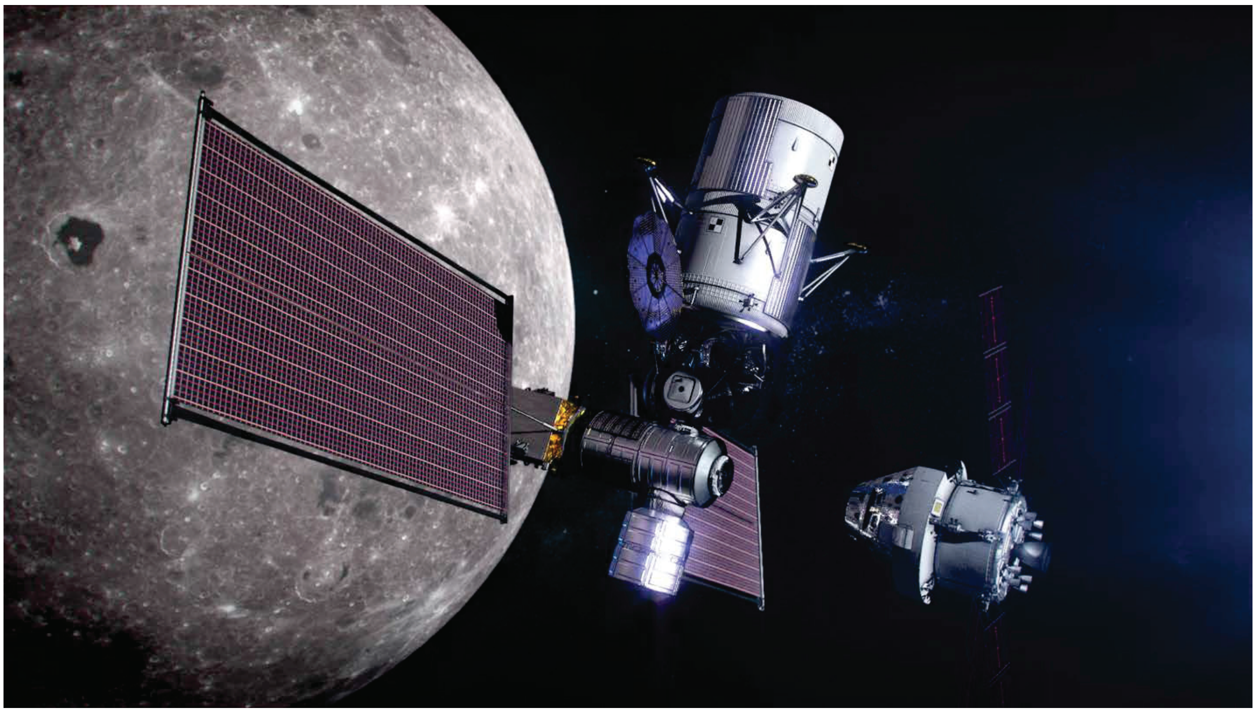

Figure 1-1—NASA Gateway Notional Illustration

2.0 MISSION DESIGN GROUND RULES AND ASSUMPTIONS

The Mission Design Team used the ground rules and assumptions (GRA) captured in this section for the development of the reference trajectories described in this document. These GRAs were taken from the government reference at the time of the analysis.

2.1 Ion Propulsion System Assumptions

The Ion Propulsion System (IPS) includes electric thrusters, power processor units (PPUs), Xenon (Xe) storage, Xe flow control hardware, and mechanical thruster-gimbals to control the direction of the thrust vector from each electric thruster. The assumptions for the operation of the IPS are listed in Table 2-1.

A nominal thruster configuration of three active and one spare thruster string (3+1) is assumed for all periods of thrusting. In all configurations, a constant available power level is assumed to be divided equally between the active thrusters.

Performance assumptions for thrust, mass flow rate and efficiency of the IPS used in the trajectory analysis, captured in Table 2-2, are the current best estimate for the predicted minimal performance of the electric propulsion strings being developed under the Advanced Electric Propulsion System (AEPS) contract with an additional 5% reduction in thrust carried as contingency. The CBE AEPS performance is an update from that in ref 1 (Herman 2017 IEPC) to reflect the Aerojet Rocketdyne hardware designs estimated performance including unit-to-unit variability and long-term, lot-to-lot variability based upon AR heritage system data. To be consistent with the PPE BAA requirements, the AEPS required throttling from 3.4 - 14 kW has been truncated to 7 - 10 kW for this trajectory analysis. The IPS performance values will be updated when more information is available from the PPE partner regarding actual expected propulsion system performance.

Additionally, the PPE Mission Design team is currently performance analysis on things such as unplanned periods of no thrusting, as well as other errors to assist the Gateway team in the development of a comprehensive low thrust trajectory design margin strategy.

Table 2-1—Ion Propulsion System Assumptions

Mission parameter	Value
Thruster String Configuration	3+1
Minimum Input Power per Thruster String	7 kW
Maximum Input Power per Thruster String	10 kW
Isp at 26.6 kW Total Power	1977 s
Duty cycle	90%

Table 2-2—Thruster String Performance Assumptions

EP String Input Power (kW)	Thrust (mN)	Mass Flow Rate (mg/s)	Specific Impulse (s)
7	338.5	20.5	1685
8	375.1	20.6	1855
9	406.7	20.8	1995
10	435.0	21.0	2111

2.2 Reaction Control System Assumptions

Each RCS thruster is modeled as a hydrazine-fueled 20N thruster with an Isp of 200s.

2.3 Power System Assumptions

This section captures the assumptions for the power available to the IPS. A constant 26.6 kW is assumed to be available to the IPS throughout the mission lifetime with the active thruster strings dividing the power equally between them. For the 3+1 configuration assumed in this analysis, each thruster string operates at 8.87 kW. These assumptions are summarized in Table 2-3. These assumptions will later be replaced with a true power curve which will depend upon the eventual spacecraft power system design. This analysis will be updated with the configuration data available from the selected PPE partner.

Table 2-3—Power Assumptions

Parameter	Value
Total Power Available	26.6 kW
Input Power per Thruster String for the 3+1 configuration	8.87 kW

2.4 Duty Cycle Assumptions

All thrust arcs are designed with a 90% duty cycle to provide margin to account for periods of no thrust due to unplanned thrust outages as well as planned spacecraft activities such as communications that require the vehicle to not be thrusting. This assumption is incorporated by setting the IPS thrust to 90% of the thrust from Table 2-2 for the thruster's operational power level.

2.5 Spacecraft Mass Assumptions

For the two transfer trajectories captured in this document, two different Gateway stack mass assumptions were used. The first transfer modeled was assumed to happen early in the life of the Gateway, when it is only part way in its configuration buildup. The second transfer modeled is assumed to be the last transfer done to deposit the Gateway into a long-term stable end of life orbit. For the end of life transfer, it is assumed that the Gateway has been fully built up and is a more massive configuration. The Gateway, constructed of its many elements, is referred to as the "stack" in several sections below. The "stack" is a term used to describe the mass that the propulsion system has to move around either through transfers or through orbit maintenance or attitude control maneuvers.

For cislunar first transfer to the DRO, the initial mass which the IPS has to deliver from the NRHO to the target orbit is assumed to be 39,000 kg. This mass was the orbit transfer mass requirement carried by the PPE at the time of designing the trajectories. This is meant to represent the mass of the Gateway early in its configuration buildup concept of operations (Conops), and includes the PPE and all propellant. This mass is the current assumption for an early configuration of the Gateway. It consists of the PPE, a mini habitat, and a logistic module. This is the mass used by the trajectory optimization at the beginning of the orbit transfer.

For the Gateway disposal trajectory analysis, it is assumed that the Gateway has been further augmented from the configuration used in the first transfer to the DRO examined in this document. At the end of its life, it is assumed that the Gateway may consist of dual habitats, additional logistic modules and other elements. For this final disposal, the initial mass at the NRHO departure is assumed to be a higher stack mass of 53,000 kg. This is the mass used by the trajectory optimization at the beginning of the orbit transfer.

2.6 Thrust Direction Assumptions

For all trajectories in this document, no restrictions are placed on the net thrust direction or the rate of change of the net thrust direction. Trajectory optimization is free to point the thrust in any direction to meet the constraints and minimize the Xe used. This translates to a time varying thrust direction over the set of thrusting segments.

3.0 REFERENCE NEAR RECTILINEAR HALO ORBIT

This section describes the reference Near Rectilinear Halo Orbit (NRHO), which is the target orbit for PPE delivery and the primary location in which Gateway is assumed to be stationed. It is also the starting orbit for all cislunar transfers described in this document.

3.1 NRHO Description

The reference NRHO is an L2 Southern NRHO with a 9:2 lunar synodic resonance, wherein there are 9 orbit revolutions for every 2 lunar months, on average. This NRHO is characterized by an average perilune radius of 3,366 km (~1450 km minimum altitude) over the northern hemisphere, apolune radius of approximately 70,000 km over the southern hemisphere, and average period of 6.56 days. In addition, the nearly-stable NRHO allows long-term orbit maintenance at low cost (see section 4.0) and relatively inexpensive transfers to and from Earth and other destination orbits (see sections 5.0 and 6.0).

The 15-year reference NRHO covers the time period from January 2, 2020 to February 11, 2035. It is continuous with a small velocity discontinuity (< 2 mm/s on average) every revolution near apolune. The 9:2 lunar synodic resonance enables the design of a reference NRHO which avoids all Earth eclipses except for two partial Earth eclipses in years 14 and 15. Eclipses by the moon are encountered regularly, but last less than 90 minutes.

The NRHO characteristics are listed in Table 3-1. The NRHO is shown graphically in Figure 3-1. See reference [1] for further discussion on the methodology for generating the reference NRHO.

Table 3-1—Reference NRHO Characteristics

NRHO Parameter	Value
Type	L2 southern, 9:2 lunar synodic resonance
Period	≈ 6.56 days
Perilune Radius	3196 – 3557 km
Apolune Radius	≈ 70,000 km
Initial Epoch	January 2, 2020
Duration	15 years

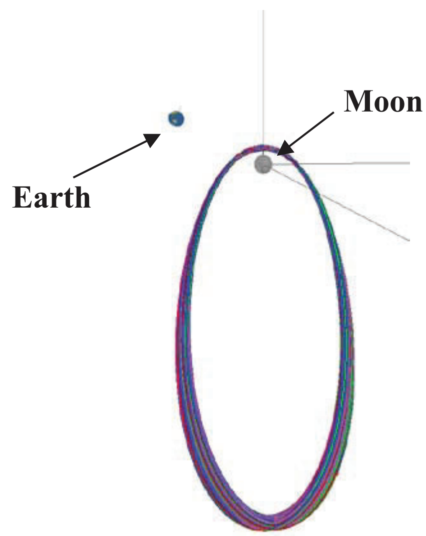

Figure 3-1—15-Year 9:2 NRHO in the Earth-Moon Rotating Frame

4.0 NRHO ORBIT MAINTENANCE

4.1 Orbit Maintenance Strategy

Analysis has been completed to determine the orbit maintenance strategy and estimate the propellant required for the 15-year lifetime of the Gateway in the NRHO. Two scenarios are investigated: uncrewed operations with a nominal 34-ton Gateway (referred to as the "stack") and crewed operations with Orion attached. For crewed operations, additional errors are considered and efforts are made to reduce the time Orion is away from its nominal tail-to-sun attitude. Either the chemical thrusters of the Reaction Control System (RCS) or passive attitude control such as wheels, can be used to slew the Gateway.

The orbit maintenance algorithm for both crewed and uncrewed operations employs a reference NRHO as a target. The spacecraft is kept near the reference by targeting the x-component of rotating velocity, v_x, in the Earth-Moon rotating frame along a receding horizon. The algorithm proceeds as follows:

1. Compute insertion error and apply to position and velocity at orbit insertion.
2. Propagate spacecraft to next apolune.
3. Varying the three components of an impulsive maneuver, target $v_x = v_{xref}$ at perilune 6.5 revolutions downstream. No errors are included in propagation during targeting.
4. If targeter fails to converge, reduce horizon to 4.5, 2.5, or 0.5 revs ahead, as needed.
5. Compute and apply navigation error to spacecraft position and velocity accordingly.
6. Compute and apply solar radiation pressure error.
7. If the targeted maneuver magnitude is greater than a minimum threshold, maneuver the spacecraft using the targeted maneuver implemented as a finite burn, perturbed by maneuver execution error. Otherwise, skip the maneuver.
8. Propagate spacecraft to next apolune. If applicable, compute and apply crewed-spacecraft perturbations in the form of impulsive maneuvers at specified intervals.
9. Return to Step 3 and repeat.

This algorithm, was adapted from the orbit maintenance algorithms successfully implemented in operations by other halo orbiters including WIND (https://wind.nasa.gov/) and ARTEMIS (https://www.nasa.gov/artemis). The approach provides robust, low-cost station keeping over long-term simulations and can be adapted to control phasing within the NRHO for eclipse avoidance. Though the maneuvers are targeted assuming impulsive maneuvers and executed using SEP finite burns, the algorithm performs well; targeting with finite burn assumptions does not significantly affect cost. Applying the minimum maneuver threshold allows very small maneuvers to be skipped without significantly affecting total orbit maintenance costs, eliminating the need to slew to maneuver orientation during these revolutions. The maneuver and target locations appear in Figure 4-1.

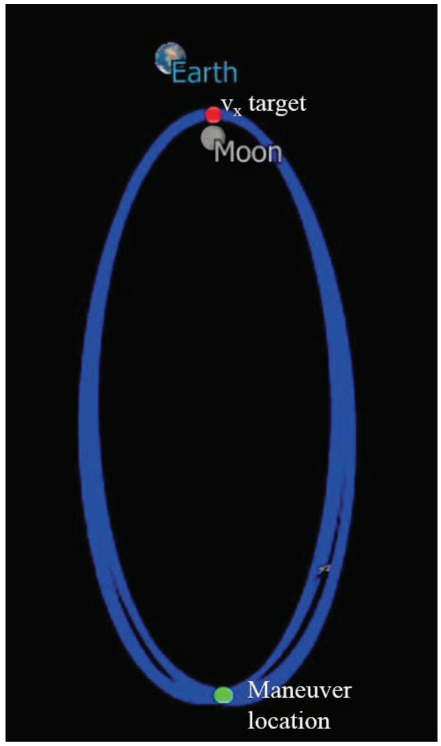

Figure 4-1—Orbit Maintenance Maneuver and Target Locations in the Earth-Moon Rotating Frame

4.2 Uncrewed Operations

For orbit maintenance cost assessment, errors are applied to the spacecraft in Monte Carlo analyses. For uncrewed operations, these errors include insertion and navigation errors in position and velocity, maneuver execution errors, perturbations from momentum desaturations, and solar radiation pressure errors. Assumptions for uncrewed operations appear in Table 4-1.

Table 4-1—Uncrewed Spacecraft Error Assumptions

Error Type	Parameter	3σ value	Frequency	Direction
Navigation and Insertion	Position	10 km	At each OMM	Random
	Velocity	10 cm/s	At each OMM	Random
Solar Radiation Pressure	Cr (Reflectivity Coefficient)	15%	At insertion	Random
	Area	30%	At each rev	Random
Desaturation perturbation	Magnitude	3 cm/s	Commanded prior to each OMM and as required to desat wheels	Random
Maneuver Execution: SEP	Fixed magnitude	1.42 mm/s	At each SEP OMM	Fixed value, Random Direction
	Direction/pointing	1°	At each SEP OMM	Random
	Proportional magnitude	1.5%	At each SEP OMM	Random

Based on the assumptions in Table 4-1, the total annual ΔV costs for orbit maintenance of the uncrewed stack remain below 5 m/s per year.

During uncrewed operations, it is assumed that passive attitude control is used to slew the spacecraft to maneuver attitude and that orbit maintenance maneuvers are executed using Xe propellant via the SEP thrusters, reducing or eliminating hydrazine usage.

4.3 Crewed Operations

For crewed operations, additional errors are assumed. When Orion and other vehicles dock to the Gateway, plume forces and docking forces are included; plume forces are also considered at undocking events. Magnitudes and directions depend on the characteristics of each vehicle and Gateway configuration at the event. During crewed operations, impulses due to carbon dioxide puffs and wastewater dumps from Orion are assumed during missions when no habitat module is managing the waste activities. Attitude slew errors are included since RCS is assumed to slew the spacecraft to orbit maintenance attitude during crewed operations, and maneuver execution errors are included for the orbit maintenance burns performed by RCS thrusters. The errors included in the crewed spacecraft Monte Carlo analyses appear in Table 4-2.

Table 4-2—Crewed Spacecraft Additional Error Assumptions

Error Type	Parameter	3σ value	Frequency	Direction
Navigation and Insertion	Position	10 km	At each OMM	Random
	Velocity	10 cm/s	At each OMM	Random
Solar Radiation Pressure	Cr	15%	At insertion	Random
	Area	30%	At each rev	Random
Desaturation perturbation	Magnitude	3 cm/s	Commanded prior to each OMM and as required to desat wheels	Random
Maneuver Execution: RCS	Direction/pointing	1°	At each RCS OMM	Random
	Proportional magnitude	1%	At each RCS OMM	Random
Docking/Undocking Perturbation	Plume impingement	Varies	At each dock and undock event	Varies
	Docking force	Varies	At each dock event	Varies
CO_2 puff perturbation*	Magnitude	20 kgm/s (fixed value)	10 min	{-0.5, -0.866, 0.0} body-fixed
Wastewater dump perturbation*	Magnitude	132 kgm/s (fixed value)	6 hours	{-0.5, 0.0, -0.866} body-fixed

*CO_2 and Wastewater venting modeled as fixed-magnitude perturbations

Based on the assumptions in Table 4-2, the total annual ΔV costs for orbit maintenance of the crewed stack remains below 5 m/s per year.

All analysis on orbit maintenance and attitude control made assumptions as to the passive attitude control hardware available on the PPE. These analyses will be revisited once the capabilities of the partner PPE are known.

Further details on the orbit maintenance algorithm and costs for both uncrewed and crewed operations appear in reference [2].

4.4 Orbit Maintenance Propulsion Allocation

It is assumed that the PPE is capable of performing orbit maintenance for the Gateway stack during its entire 15-year lifetime. Both the RCS thrusters and the SEP thrusters are capable of executing orbit maintenance maneuvers. For uncrewed operations, the SEP thrusters are assumed to execute orbit maintenance maneuvers. Figure 4-2 summarizes the current estimates for Xe mass to perform orbit maintenance of the Gateway over time as the stack is built up from the PPE to a notional full stack over the course of 15 years. In this figure, LM are the Logistic Modules, and the EM calls out the forecasted Exploration Mission launches. These stack configurations are notional and subject to change as the Gateway is further defined.

Orbit maintenance analysis suggests that the total Xe propellant required for Phase 1 (i.e., the first 5 years) is about 21 kg. Extending the analysis to 15 total years gives a total Xe usage of about 150 kg. A margin of 15% is carried on this preliminary analysis, leading to a total of about 172 kg of Xe for 15 years of orbit maintenance.

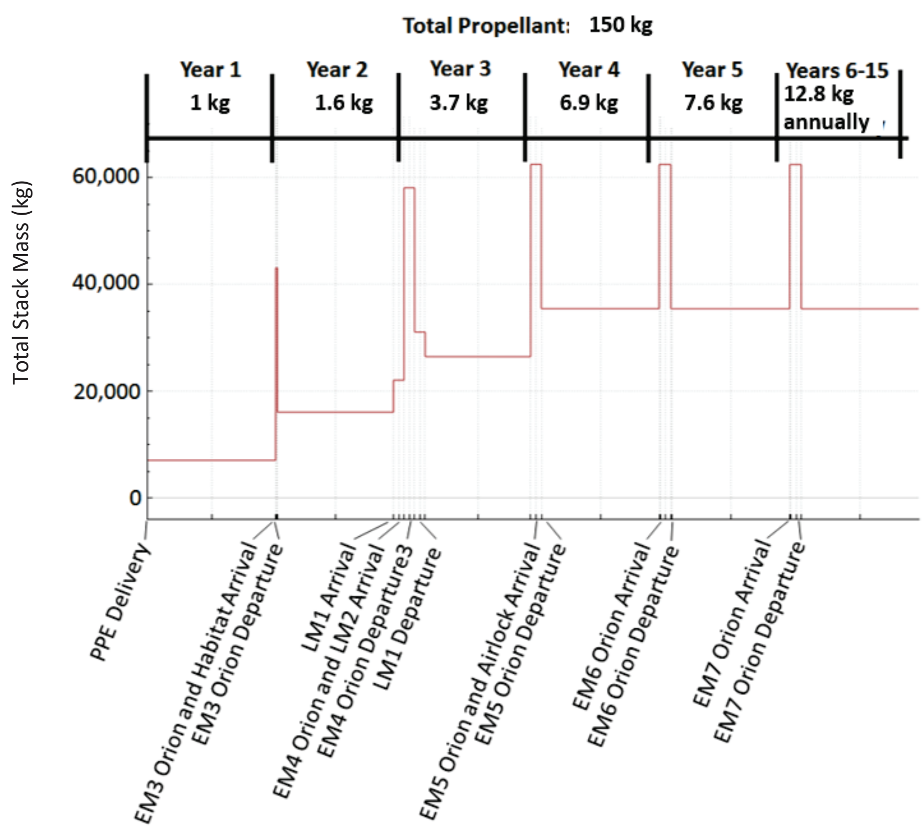

Figure 4-2— Orbit Maintenance Xe propellant usage over Gateway 15-year lifetime

During crewed operations, it is assumed that hydrazine is used to slew the spacecraft and perform orbit maintenance burns to complete the slew-burn-slew sequence within the 3-hour timespan that Orion is allowed for excursions from tail-to-sun attitude. The hydrazine cost per mission varies widely depending on the length of the Orion stay, the configuration of the Gateway components, and whether carbon dioxide and wastewater venting occurs through Orion's unbalanced vents or if both are handled non-propulsively through a Gateway module. For example, for a minimal Gateway configuration with Orion venting during a 10-day stay, the mission requires 30-40 kg of hydrazine. The same hydrazine mass will control a full Gateway configuration for a 30-day Orion stay if Orion is not venting. Figure 4-3 summarizes the current estimates for hydrazine mass to perform orbit maintenance and attitude control during crewed operations and attitude control during uncrewed operations, assuming venting via Orion only occurs during the first crewed mission.

Current analysis suggests that a 15-year lifetime budget for hydrazine, assuming venting of carbon dioxide and wastewater through Orion occurs only during the first crewed mission, ranges from 650-820 kg. If wastewater venting occurs through unbalanced Orion vents throughout the Gateway lifetime, the hydrazine budget ranges from 1120 – 1320 kg.

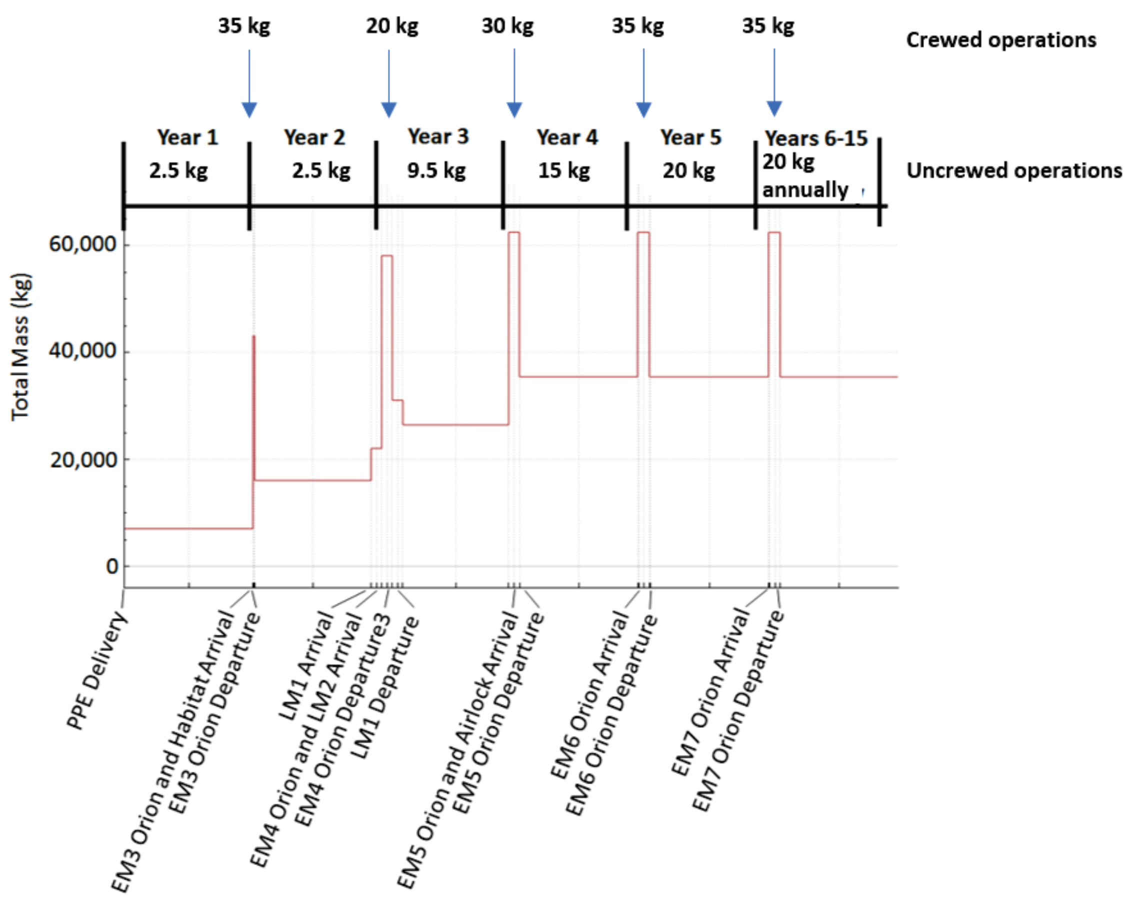

Figure 4-3— Orbit Maintenance hydrazine propellant usage over Gateway 15-year lifetime

5.0 REPRESENTATIVE CISLUNAR ORBIT TRANSFER: NRHO TO DRO

The DRO exists because of three body gravitational effects between the Earth and Moon. The following section details a representative transfer trajectory from the reference NRHO to a Distant Retrograde Orbit (DRO). This transfer departs from the L2 Southern 9:2 NRHO described in section 3.1 and uses SEP to transfer to the final destination.

The assumed target DRO has a radius of approximately 70,000 km and is stable on the order of >100 years. The initial 9:2 NRHO and target DRO are shown in Figure 5-1. This representative SEP transfer assumes a total Gateway stack mass of 39,000 kg at departure from the L2 Southern 9:2 NRHO (Section 3.1) and optimized the final mass arriving in the DRO.

The one-way cislunar transfer to the DRO takes a total of 161.2 days, with 33.0 days of thrust and 128.2 days of coast. This trajectory is shown in Figure 5-7**.**. A total of 159.7 kg of Xe is required to perform the 79.6 m/s total mission ΔV using 3 thruster strings operating at 26.6 kW. A return transfer back to the NRHO is assumed to require similar propellant mass, ΔV and trip time.

Though assumed to be representative, the presented trajectory is a single point case. Transfers with differing epochs and planetary alignment may result in better or worse performance. In addition, the values shown for ΔV and Xe mass are ideal and do not include margin for any dispersions, off-nominal scenarios, etc.

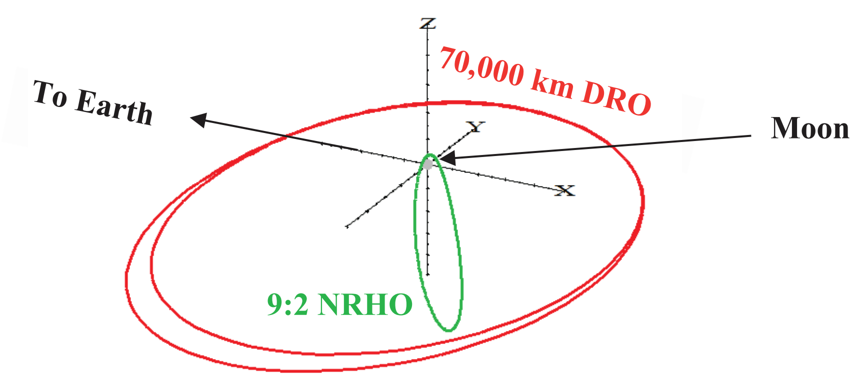

Figure 5-1—Initial 9:2 NRHO and Target 70,000 km DRO in the Earth-Moon Rotating Frame

5.1 NRHO to DRO Transfer Summary

Table 5-1 summarizes the total trip time, thruster time, and coast time for the representative NRHO to DRO transfer. All values presented are for a one-way transfer from the initial NRHO to the target orbit. The return trip to the NRHO is assumed to require a similar amount of time and propellant mass to perform. Therefore, to estimate values for a round trip mission, simply double the propellant mass and the trip time.

Table 5-1—One Way Transfer Summary

One-Way Transfer	**NRHO to DRO**
Total Transit Time (days)	161.2
Total SEP Thrusting Time (days)	33.0
Total Coast Time (days)	128.2
Maximum SEP Burn Arc (days)	10.9
Number of SEP Burn Arcs	7
Total SEP ΔV (m/s)	79.6
Total Unmargined Xe (kg)	159.7

5.2 Reference Trajectory Events

Table 5-2 lists the thrust and coast segments throughout the transfer from the NRHO to the DRO. Any close approaches to the moon with altitudes < 1,500 km are also listed.

Table 5-2—Relevant Values at Major Mission Events

Mission Event	MET (days)	Total Mass (kg)	Solar Range (AU)	Earth Range (km)	Moon Range (km)
Departure	0.0	39,000.0	0.990	381,533	67,734
Coast 1	0.0	39,000.0	0.990	381,533	67,734
Thrust 1	1.8	39,000.0	0.991	375,451	35,910
Coast 2	3.0	38,994.1	0.992	378,228	29,618
Thrust 2	15.6	38,994.1	0.997	407,299	30,310
Coast 3	16.5	38,989.8	0.997	409,447	21,767
Thrust 3	53.1	38,989.7	1.011	758,139	998,128
Coast 4	58.7	38,962.2	1.012	795,156	1,161,192
Thrust 4	64.7	38,962.2	1.012	789,838	1,032,648
Coast 5	64.7	38,962.2	1.012	789,825	1,032,548
Lunar Flyby (564 km)	82.5	38,962.2	1.011	357,864	2,302
Thrust 5	84.5	38,962.2	1.010	425,295	100,280
Coast 6	94.9	38,911.7	1.014	373,973	315,866
Thrust 6	114.1	38,911.7	1.015	291,788	151,224
Coast 7	118.0	38,893.1	1.016	407,727	223,648
Thrust 7	152.0	38,893.1	1.016	401,654	111,747
Coast 8	163.0	38,840.3	1.014	300,761	69,473
Arrival	167.9	38,840.3	1.010	425,863	90,869

5.3 Eclipse Avoidance

It is assumed that the PPE Flight System must provide power during eclipse events for up to 1.5 hours duration. From a mission design perspective, any continuous eclipse event (partial, full or a combination) with a duration greater than 90 minutes is considered a violating eclipse event. The majority of representative NRHO to DRO transfers studied at various departure epochs encounter at least one eclipse event during the transfer. Approximately 60% of transfers have an eclipse event lasting longer than 90 minutes.

Preliminary analysis has shown that transfer trajectories which naturally avoid eclipses occur during an 11-day departure window approximately every 60 days. For the 40% of transfers with eclipses lasting longer than 1.5 hours, thrust arcs can be adjusted (e.g. extended or delayed) to reduce the amount of time spent in eclipse to satisfy the requirement.

The representative NRHO to DRO cislunar transfer detailed in this section avoids eclipses longer than 1.5 hours in duration.

5.4 Reference Trajectory Plots

The data in this section are plotted as a function of Mission Elapsed Time (MET), starting from the representative NRHO departure date of March 5, 2025. The red lines represent thrust periods while the blue lines represent coast periods. All displayed coast periods are optimal given the constraints of the mission design. Aside from the duty cycle of 90%, which effectively embeds a 10% coast margin into the trajectory, no additional explicit coast periods have been inserted for missed thrust or other margin considerations.

Figure 5-2, Figure 5-3 and Figure 5-4 show the range to the center of Earth, Sun and Moon throughout the trajectory, respectively. The maximum Earth range is 800,000 km and the maximum solar range is 1.019 AU, and the maximum and minimum lunar range is 116,000 km and 2300 km respectively.

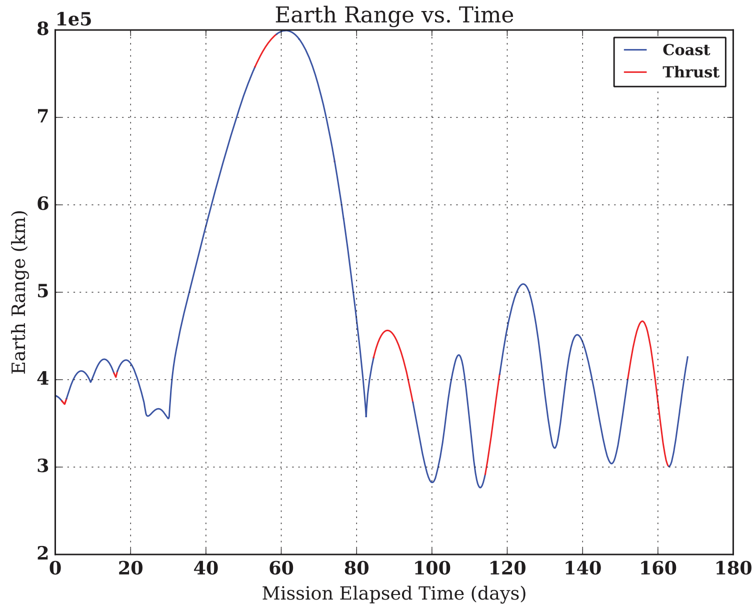

Figure 5-2—Earth Range (km) vs. MET

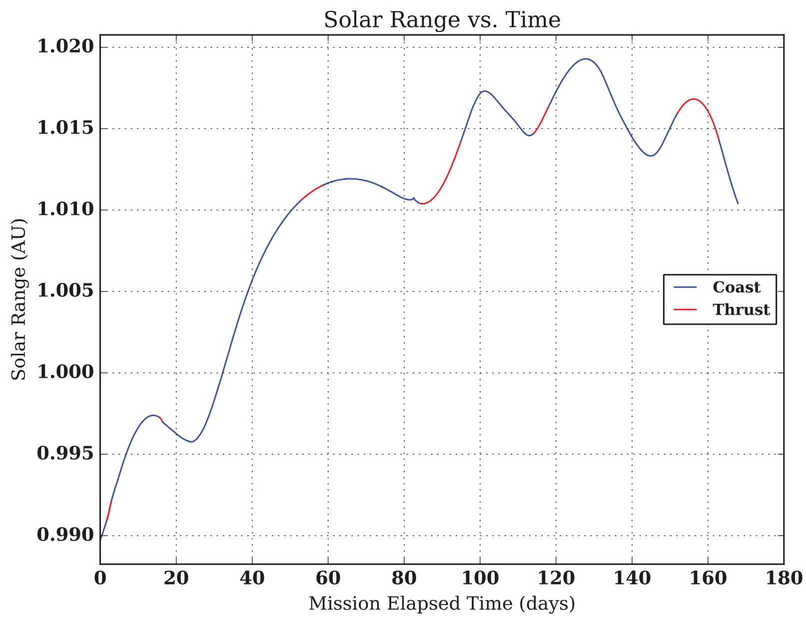

Figure 5-3—Solar Range (AU) vs. MET

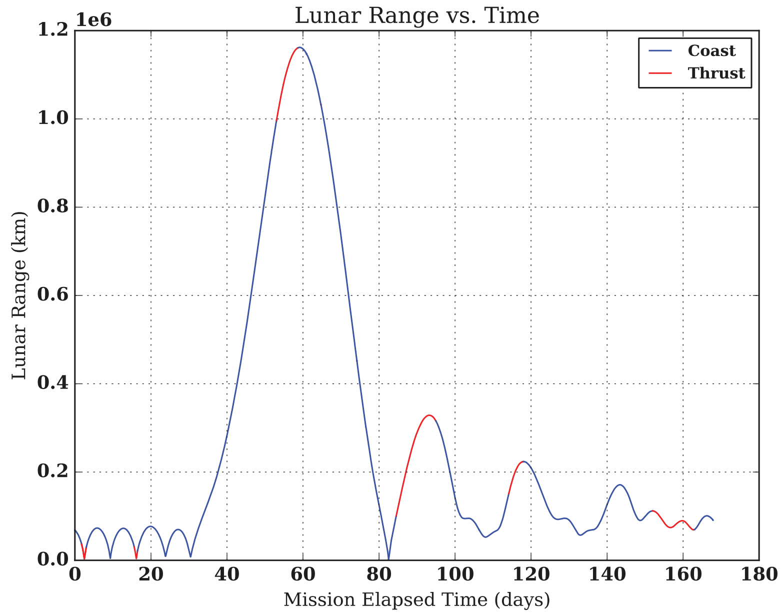

Figure 5-4—Lunar Range (km) vs. MET

Figure 5-5 plots the three (x, y, z) components of the inertial thrust direction unit vector in the J2000 Ecliptic frame. The x-direction is plotted in red, the y-direction is plotted in blue, and the z-direction is plotted in green. Gaps in the data correspond to periods of coast.

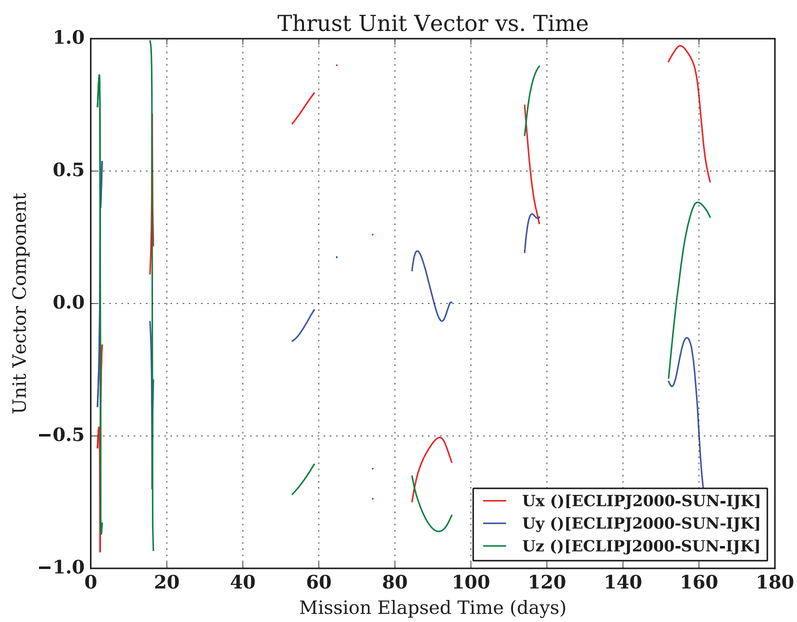

Figure 5-5—Thrust Unit Vector vs. MET

Figure 5-6 shows the evolution of the stack mass throughout the mission. All changes in mass correspond to Xe used during SEP thrusting.

Figure 5-7 shows the reference trajectory in the Earth-Moon rotating frame. The Earth is located at the center of the plot. The major events of NRHO departure and DRO arrival are annotated in the graphic.

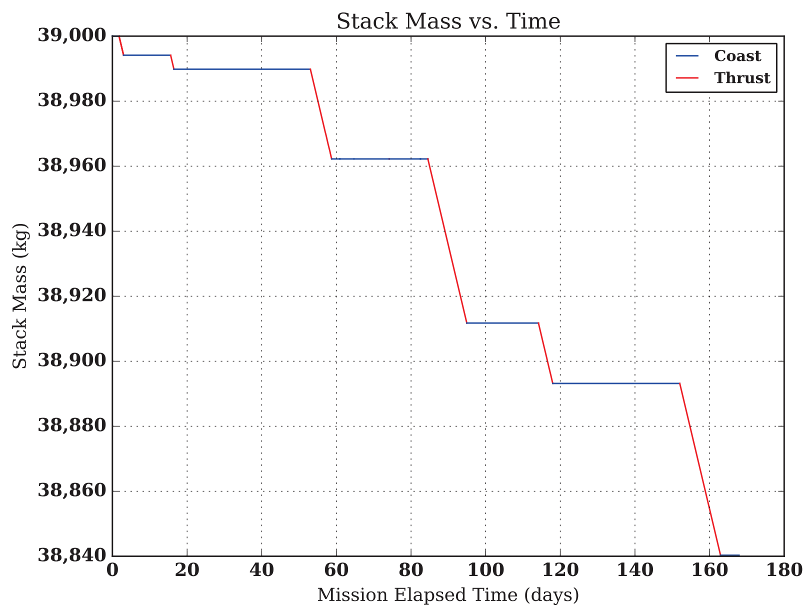

Figure 5-6—Stack Mass (kg) vs. MET

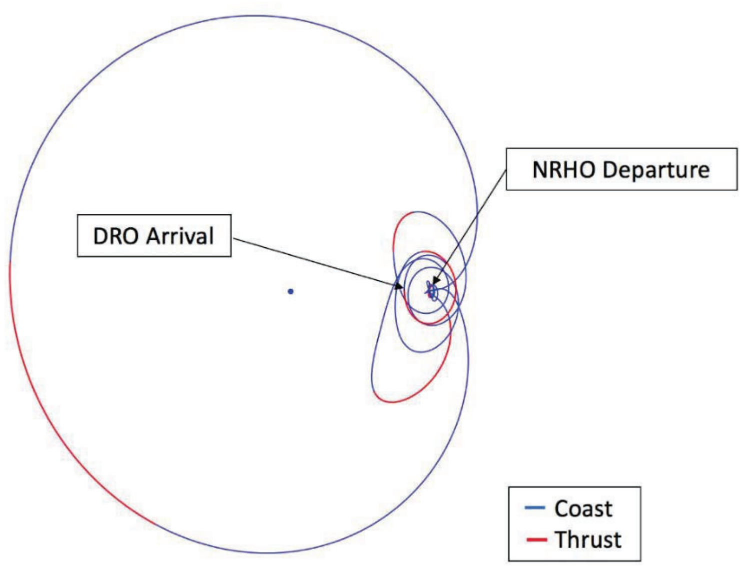

Figure 5-7—Cislunar Transfer Trajectory in Earth-Moon Rotating Frame

6.0 REPRESENTATIVE NRHO TO DRO DISPOSAL TRAJECTORY

A separate representative NRHO to DRO disposal trajectory was created for the transfer of Gateway to the DRO at its end-of-life. Since this transfer occurs at the end of Gateway's lifetime, a longer transfer time does not impact crewed operations and can be extended to reduce the required ΔV. Several relevant parameters for the disposal orbit transfer trajectory are identified and captured in Table 6-1. The total effective ΔV required to perform the cislunar transfer is derived from the Isp of the SEP system and the Xe mass expended.

Table 6-1—Summary of NRHO to DRO Disposal Trajectory Parameters

Item	Value	Notes
Initial Mass	53,000 kg	Starting mass at departure from the NRHO
Ideal Xenon Propellant Mass	85 kg	Xe required to perform DRO disposal transfer
Effective ΔV	31 m/s	ΔV required to perform DRO disposal transfer

6.1 Trajectory Events

Table 6-2 lists the thrust and coast segments throughout the transfer from the NRHO to the DRO. This transfer does not include any close approaches to the moon with altitudes less then 1,500 km.

Table 6-2—Relevant Values at Major Mission Events

Mission Event	MET (days)	Total Mass (kg)	Solar Range (AU)	Earth Range (km)	Moon Range (km)
Departure	0.0	53,000.0	0.981	418,742	71,375
Coast 1	0.0	53,000.0	0.981	418,742	71,375
Thrust 1	9.5	53,000.0	0.983	371,775	12,134
Coast 2	10.1	52,997.3	0.983	374,194	25,340
Thrust 2	15.9	52,997.3	0.986	378,339	16,633
Coast 3	16.5	52,994.5	0.986	379,283	22,130
Thrust 3	75.9	52,994.5	0.990	714,637	1,054,124
Coast 4	77.4	52,987.4	0.990	728,050	1,099,364
Thrust 4	248.7	52,987.1	1.011	433,981	111,432
Coast 5	260.2	52,931.7	1.006	332,194	91,371
Thrust 5	321.8	52,931.5	0.988	465,789	74,359
Coast 6	323.6	52,923.0	0.987	461,051	76,852
Thrust 6	377.1	52,922.7	0.984	460,072	77,946
Coast 7	378.6	52,915.4	0.984	467,414	75,711
Arrival	382.3	52,915.4	0.983	388,629	90,911

6.2 Trajectory Plots

The data in this section are plotted as a function of Mission Elapsed Time (MET). In each plot, periods of thrusting are indicated with red line segments while periods of coast are indicated in blue. All displayed coast periods are optimal given the constraints of the mission design. No additional coast periods have been inserted for missed thrust or other margin considerations.

Figure 6-1, Figure 6-2 and Figure 6-3 show the range to the center of Earth, Sun and Moon throughout the trajectory, respectively. The maximum Earth range is 760,000 km and the maximum solar range is 1.019 AU, and the maximum and minimum lunar range is 116,000 km and 3300 km respectively.

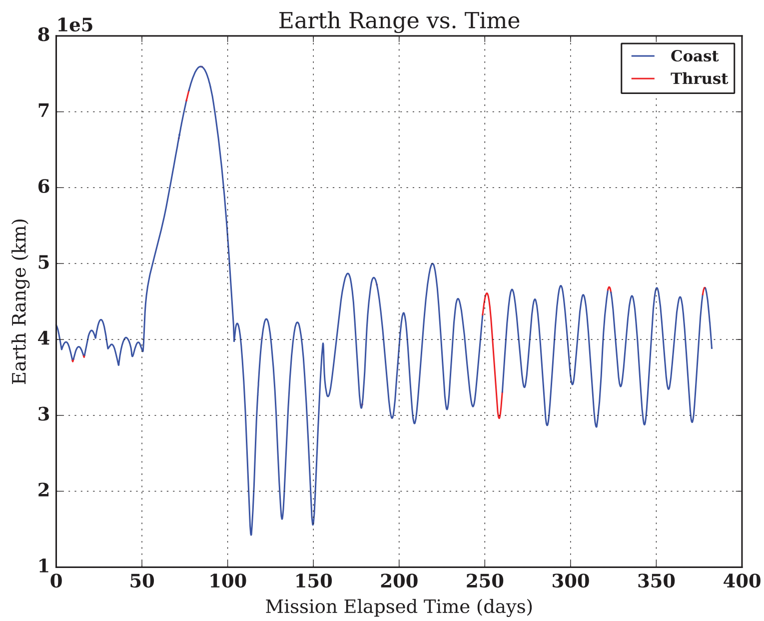

Figure 6-1—Earth Range (km) vs. MET

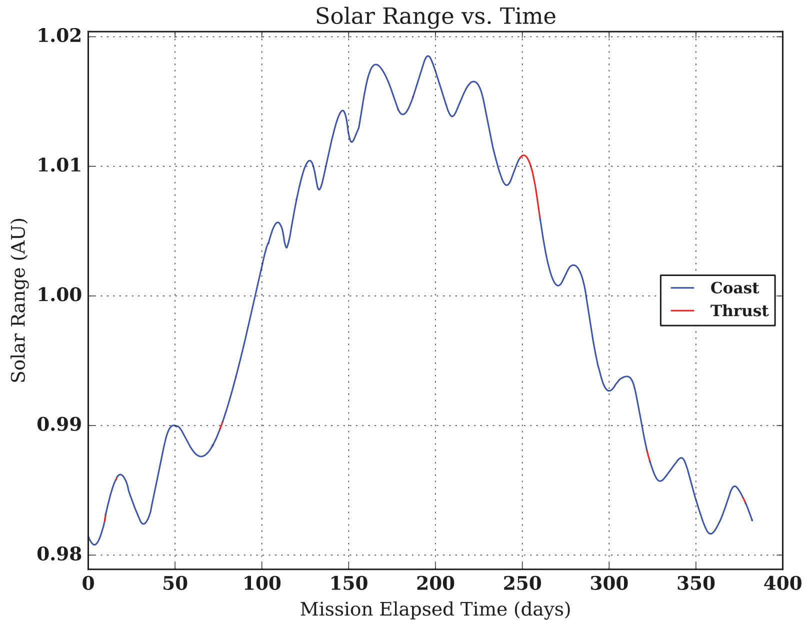

Figure 6-2—Solar Range (AU) vs. MET

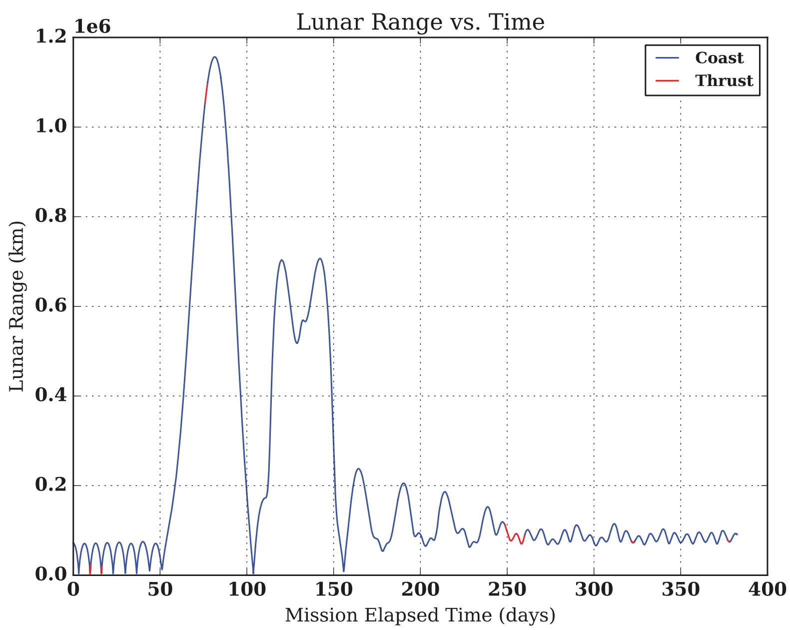

Figure 6-3—Lunar Range (km) vs. MET

Figure 6-4 plots the three (x, y, z) components of the inertial thrust direction unit vector in the J2000 Ecliptic frame. The x-direction is plotted in red, the y-direction is plotted in blue, and the z-direction is plotted in green. Gaps in the data correspond to periods of coast.

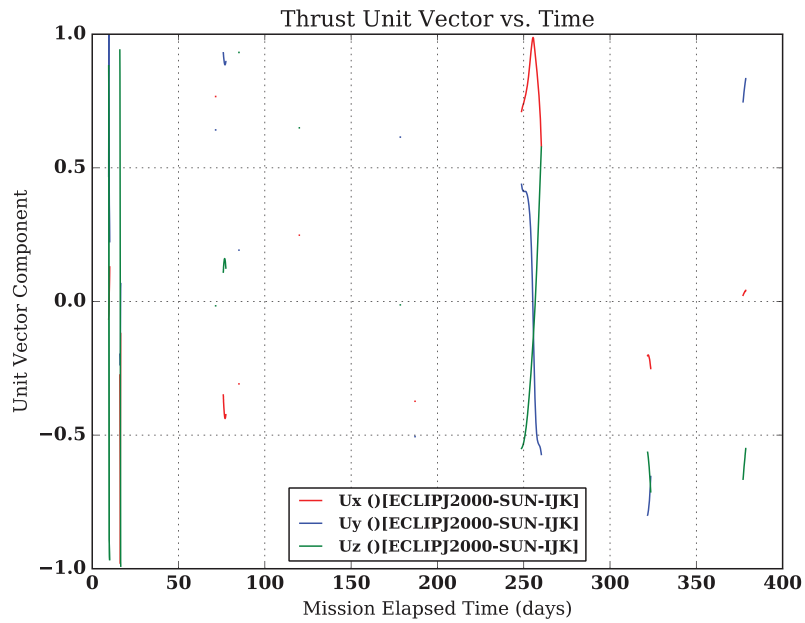

Figure 6-4—Thrust Unit Vector vs. MET

Figure 6-5 shows the evolution of the stack mass throughout the mission. All changes in mass correspond to Xenon (Xe) used during SEP thrusting.

Figure 6-6 shows the reference trajectory in the Earth-Moon rotating frame. The Earth is located at the center of the plot. The major events of NRHO departure and DRO arrival are annotated in the graphic.

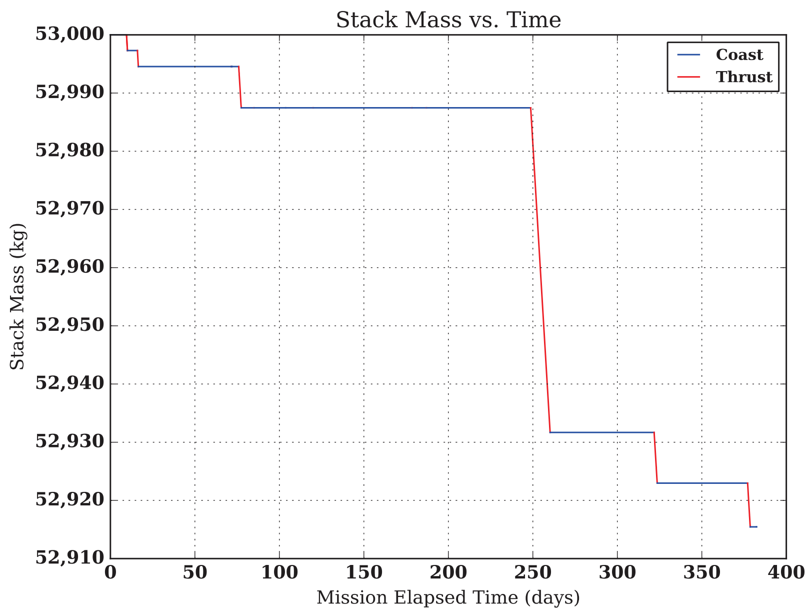

Figure 6-5—Stack Mass (kg) vs. MET

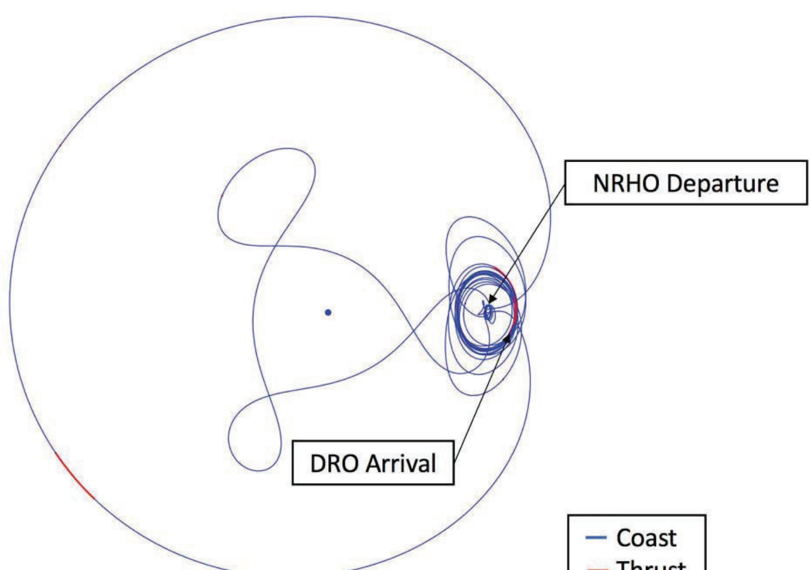

Figure 6-6—Disposal Transfer Trajectory in Earth-Moon Rotating Frame

APPENDIX A. ACRONYMS AND ABBREVIATIONS

AU	Astronomical Unit
BOM	Beginning of Mission
CMG	Control Moment Gyroscope
Cr	Reflectivity Coefficient
DRO	Distant Retrograde Orbit
EP	Electric Propulsion
EOL	End of Life
EOM	End of Mission
GRA	Ground Rules and Assumptions
IPS	Ion Propulsion System
Isp	Specific Impulse
LM	Logistics Module
MET	Mission Elapsed Time
NRHO	Near Rectilinear Halo Orbit
PPE	Power and Propulsion Element
PPU	Power Processing Unit
RCS	Reaction Control System
SEP	Solar Electric Propulsion
Xe	Xenon

APPENDIX B. DEFINITIONS

J2000 Inertial Frame: A non-rotating SPICE frame, defined by the Earth mean equator, dynamical equinox of the J2000 epoch (January 1, 2000 at 12:00:00 TDB).

J2000 Ecliptic Frame: A non-rotating SPICE frame, using ecliptic coordinates based on the J2000 frame

APPENDIX C. ADDITIONAL FILES

C.1 Scope

A number of supplementary files are included with this document. These include the raw data used to generate the included plots and ephemeris files for the reference trajectories.

C.2 Reference Trajectory Excel File(s)

The trajectory data (ranges, mass, power, etc.) versus mission date used to produce the plots for the cislunar transfers in this document are available in separate excel files. The files can be used to examine the data in more detail and perform secondary calculations. The files also include thrust magnitude, mass flow rate, and thrust unit vector direction throughout the mission.

Trajectory	Excel File Name
Representative NRHO to DRO Orbit Transfer Data	PPE_NRHOtoDRO_data _Doc0079RevD_Jun2019.xlsx
Reference NRHO to DRO Disposal Transfer Data	PPE_NRHOtoDRO_Disposal_data _Doc0079RevD_Jun2019.xlsx

C.3 SPK File(s)

The trajectory ephemeris data for the reference NRHO, representative orbit transfer, and representative disposal trajectory available in the form of Spacecraft Kernel (SPK) files. Details on how to read and use the SPK file format can be found at JPL's Planetary Data System Navigation Node site:

- http://naif.jpl.nasa.gov/naif/
- https://naif.jpl.nasa.gov/pub/naif/toolkit_docs/C/req/spk.html

Trajectory	SPK File Name
Reference NRHO	PPE_Reference_NRHO_traj _Doc0079Rev2_Mar2018.bsp
Reference NRHO to DRO Orbit Transfer Data	PPE_NRHOtoDRO_traj _Doc0079RevD_Jun2019.bsp
Reference NRHO to DRO Disposal Transfer Data	PPE_NRHOtoDRO_Disposal_data _Doc0079RevD_Jun2019.bsp

APPENDIX D. DOCUMENT HISTORY LOG

Status (Preliminary/ Baseline/ Revision/ Canceled)	Document Revision	Effective Date	Description
Baseline	0	12/01/2017	Initial Release of reference trajectory
Revision	A	12/15/2017	Incorporate inputs/edits from internal PPE review. Edits from D. Davis at JSC. Update Table 3.4 to better capture masses based on feedback from the study contractors. Inputs from LMB on thrust vector, and theta. Add in Isp/Thrust to the tables of the transfers. Add in section 7.0 to summarize Xe and hydrazine needs for the PPE.
Revision	B	05/09/2018	3/1/18 - Added additional detail for reference NRHO. Updated orbit maintenance section. Updated cislunar transfer plots with latest trajectories including updated assumptions. Sent for inspection and sign off. Additional CM editorial changes, including removed “GRC-” from prefix of document number and header/footer changes per new CM directives.
			Corrected for BAA release to change “DSG” and “Deep Space Gateway” to “Gateway” throughout the document. Update cover page to indicate available to public.
Revision	C	08/21/2018	Remove reference to PPE SEP Performance Document and include reference to D. Herman 2017 paper on AEPS thruster performance modeling.
Revision	D	02/21/2020	Update reference trajectory to include 10 kW maximum power to IPS. Details were added for eclipse avoidance and for a separate reference disposal trajectory, adding in more detail on the orbit maintenance strategy section 4.0 to reflect the current modeling.

APPENDIX E. SIGNATURE PAGE

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REFERENCES

[1] J. Williams, D.E. Lee, R.J. Whitley, K.A. Bokelmann, D.C. Davis, and C.F. Berry, "Targeting Cislunar Near Rectilinear Halo Orbits for Human Space Exploration," 27th AAS/AIAA Space Flight Mechanics Meeting, Feb. 2017. (AAS 17-267)

[2] D.C. Davis, S.A. Bhatt, K.C. Howell, J. Jang, R.L. Whitley, F.D. Clark, D. Guzzetti, E.M. Zimovan, and G.H. Barton, "Orbit Maintenance and Navigation of Human Spacecraft in Cislunar Near Rectilinear Halo Orbits," 27th AAS/AIAA Spaceflight Mechanics Meeting, Feb. 2017.

[3] M. McGuire, L. Burke, S. McCarty, K. Hack, R. Whitley, D. Davis, C. Ocampo, "Low Thrust Cis-Lunar Transfers Using a 40 kW-Class Solar Electric Propulsion Spacecraft," AAS/AIAA Astrodynamics Specialist Conference, Aug. 2017. (AAS 17-583)