(Preprint) AAS 21-697

OVERVIEW OF THE LUNAR TRANSFER TRAJECTORY OF THE CO-MANIFESTED FIRST ELEMENTS OF NASA'S GATEWAY

Melissa L. McGuire,* Steven L. McCarty,† Daniel J. Grebow‡, Thomas A. Pavlak§, Scott N. Karn**, Kaushik S. Ponnapalli††, Diane C. Davis‡‡, and Kurt J. Hack§§

This paper documents the current design reference mission planned for the first two elements of NASA's Gateway. When launched together, the Power and Propulsion Element and Habitation and Logistics Outpost comprise the Co-Manifested Vehicle (CMV).[1] The low-thrust transfer between the initial parking orbit and the final insertion into the operational Near Rectilinear Halo Orbit is described. While each specific trajectory depends on launch date, trends are identified in the dynamics and orientation of the CMV as it traverses its spiral orbit. This paper describes the interplay between various assumptions and constraints on the development of the low thrust lunar transfer.

INTRODUCTION

NASA has committed to returning to the moon, to land the first woman and the next man on its surface with a sustained human presence extending into the future. To support this effort, NASA is designing an orbital platform to be deployed in an orbit near the moon called a Near Rectilinear Halo Orbit (NRHO)[2]. This platform is known as the Gateway, and its purpose is to support crewed missions primarily to the lunar surface, to serve as a proving ground for deep space technologies, and to support human exploration beyond Earth orbit. NASA continues to study ways to reduce the cost of the human lunar landing.[3] One mission simplification is to launch the first two elements of the Gateway together on a single commercial launch vehicle (CLV). When launched together, these two elements, the Power and Propulsion Element (PPE) and the Habitation and Logistics Outpost (HALO) comprise the Co-Manifested Vehicle (CMV).[4] The CMV will take advantage of the high efficiency of the PPE's high-power Solar Electric Propulsion (SEP) to transfer a significant starting mass from an initial Earth orbit to final insertion into the NRHO.

* Chief, Mission Architecture and Analysis Branch (LSM), PPE MD Team Lead, NASA Glenn Research Center, Cleveland, OH 44135.
† PPE MD Lead Engineer, Mission Architecture and Analysis Branch (LSM), NASA Glenn Research Center, Cleveland, OH 44135.
‡ Mission Design Engineer, Mission Design and Navigation Section, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Dr., Pasadena, CA 91109
§ Mission Design Engineer, Mission Design and Navigation Section, Jet Propulsion Laboratory, California Institute of Technology, 4800 Oak Grove Dr., Pasadena, CA 91109
** Aerospace Engineer, HX-5, 21000 Brookpark Road, Cleveland, OH, 44135.
†† Aerospace Engineer, HX-5, 21000 Brookpark Road, Cleveland, OH, 44135.
‡‡‡‡ Gateway MD Team Lead, Principal Systems Engineer, a.i. solutions, Inc., Houston, TX.
§§ PPE SE&I Mission Integration, Systems Engineering and Analysis Division (LS), NASA Glenn Research Center, Cleveland, OH 44135.

This paper captures at a top level the preliminary analysis on vehicle performance, total stack mass, and trip time of the low thrust lunar transfer reference trajectory to be flown by the PPE+HALO combined vehicle. As designs mature, details such as CLV anticipated performance and the PPE main electric propulsion (EP) system performance evolve[5], and input assumptions are discussed. Where data is known but restricted due to NASA and commercial proprietary data regulations, the trajectory is discussed in terms of generalities and overview of the ephemeris (position vs mission elapsed time). These formal selections and ground rules and assumption decisions affect the final reference flight trajectory of the CMV, but the representative lunar transfer trajectory and the trades captured in this paper still hold for the physics of a low thrust lunar transfer trajectory.

TRANSFERING TO THE NRHO

Low thrust spiral transfers are one of several trajectory options to deliver mass from the Earth to the NRHO[6]. Traditional direct missions, like those of Orion and Apollo, reduce trip time at the expense of propellant through impulsive maneuvers typically provided by a high thrust, low specific impulse chemical propulsion system. An alternate transfer option recently investigated for delivery of payloads to the NRHO is the ballistic lunar transfer (BLT)[7]. While BLTs can reduce the Delta V required and subsequently propellant used by the propulsion system to insert into the NRHO, they still require performance from the inserting launch vehicle to send the payload/spacecraft toward the moon. With the decision made to combine PPE and HALO onto a single launch, no current commercially available launch vehicle can deliver their combined mass to the NRHO either by a direct transfer or via BLT. Combining the capabilities of a launch vehicle with the low thrust, high specific impulse, highly efficient performance of a SEP system enables more mass delivery for less propellant at the cost of added trip time. In this mission, the high-power SEP of the PPE is used to perform a spiral transfer from a nominally low elliptical parking orbit, taking advantage of the highly efficient SEP system on board to enable the delivery of the combined CMV to the NRHO.

Examples of these three types of transfers to the NRHO are shown in Figure 1. A direct to NRHO transfer uses an impulsive system to perform the delta V necessary to transfer to and insert into the NRHO. These transfers typically take 4-10 days to complete depending on the propulsion system and opportunity. A BLT takes on the order of 120-180 days to allow the sun's influence to reduce the delta V necessary to insert into the NRHO. The final insertion delta V can be performed with chemical or EP systems. The final transfer, and the one that the CMV will use to reach the NRHO, is a low thrust spiral. In this type of transfer, the SEP system performs nearly constant thrust maneuvers over a long period of time to raise apogee and perigee until the orbit reaches the moon; SEP also performs the final insertion maneuver into the NRHO.

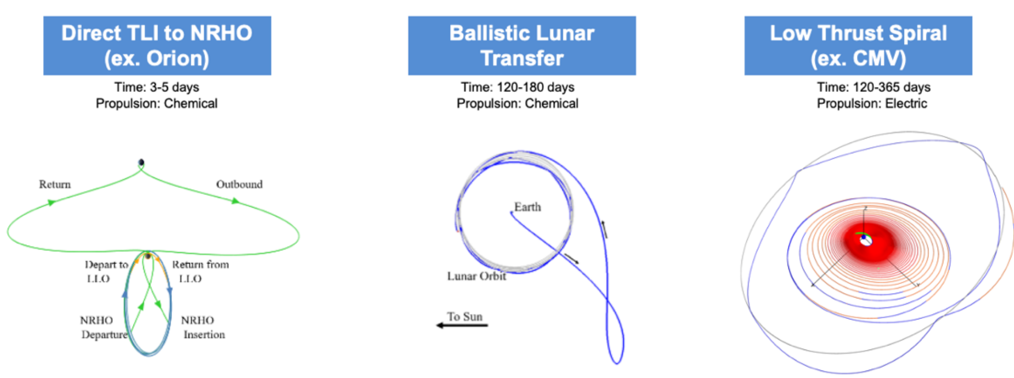

Figure 1. Mass delivered to NRHO from Falcon 9 launch.

Comparing transfer options

The CMV mass exceeds the capability of any current launch vehicle to deliver directly to the NRHO. As an illustration of mass performance of the three transfer options, the capability analysis summarized in Figure 2. Mass delivered to NRHO assuming a Falcon 9 launch. assumes publicly available Falcon 9 launch performance data[8]. The three bars represent a direct transfer to NRHO (left), a chemical BLT (center) and a low thrust spiral (right.) While the Falcon 9 can deliver mass directly to the NRHO, the use of low thrust SEP augments the capability of the Falcon 9 to deliver mass to the NRHO by 40%. For this reason, the SEP system on the PPE is key to delivering the CMV to the NRHO on a single launch.

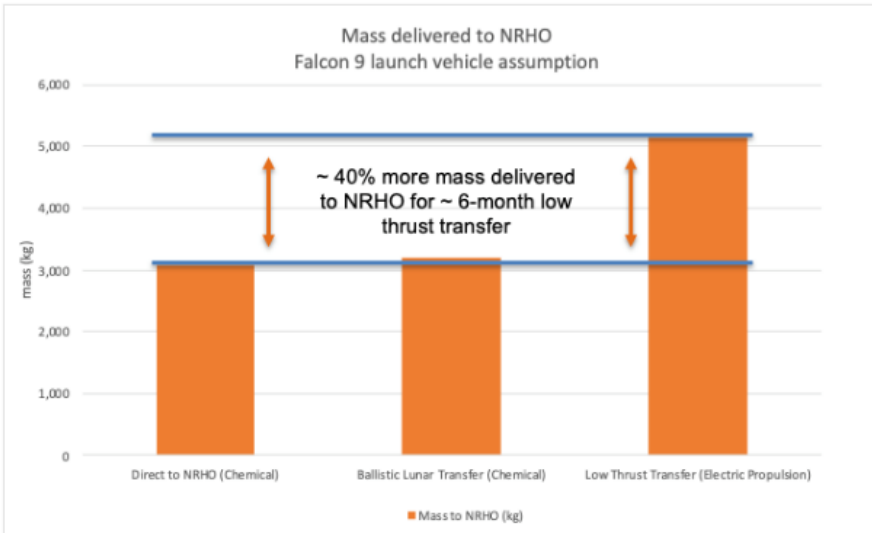

Figure 2. Mass delivered to NRHO assuming a Falcon 9 launch.

MISSION DESIGN GROUND RULES AND ASSUPTIONS

The CMV is delivered by the on-board SEP system of the PPE through a low thrust transfer to the NRHO. Ground rules and assumptions about launch vehicle and spacecraft performance have significant effects on the resulting trajectories and costs.

Launch vehicle injection orbit

In May 2021, NASA announced the selection of SpaceX as the launch provider for the CMV[9]. Launched on a Falcon Heavy, the CMV will nominally be inserted into a low elliptical orbit, with a perigee no lower than 200 km and an apogee approximately 33,900 km at an inclination of 28.5°. For the initial analysis of the transit, there is no restriction placed on the Right Ascension of the Ascending Node (RAAN) or Argument of Periapsis (AOP) for the initial orbit targeted by the launch vehicle. The actual initial values for the injection orbit in the transfer trajectory vary slightly from the nominal values due to the use of a single, standard parking orbit for the whole AOP range when generating the representative launch vehicle separation states. The injection orbit varies slightly due to changing coast duration and Earth oblateness effects in the ascent simulations.

Destination NRHO

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[10]. This NRHO is characterized by an average perilune radius of 3,366 km (~1450 km minimum altitude) over the northern hemisphere, an average apolune radius of approximately 71,000 km over the southern hemisphere, and an average period of 6.56 days.[11] In addition, the nearly stable NRHO allows long-term orbit maintenance at low cost and relatively inexpensive transfers to and from Earth and other destination orbits.[12]

Table 1. Reference NRHO characteristics.

Description	L2 southern, 9:2 lunar synodic resonance
Average Period	6.56 days
Mean Perilune Radius	3,366 km
Mean Apolune Radius	71,100 km
Initial Epoch	January 2, 2020
Ephemeris Duration	15 years

Eclipse avoidance

A requirement on the transfer trajectory is that CMV must avoid entering eclipse for greater than 90 continuous minutes. The frequency and duration of eclipses occurring during the low thrust spiral exhibit seasonal patterns and therefore depend on the launch date. The orientation of the initial parking orbit plane is also a factor. The initial conditions of the transit trajectory are specifically designed to satisfy this constraint.

Main Propulsion system assumptions

The on-board main propulsion system of the PPE is comprised of two different types of hall effect thrusters, operating in two different system modes: a high thrust mode and a high Isp mode. The high thrust mode distributes power amongst the thruster strings to maximize the total system thrust, and the high Isp mode performs a similar distribution to maximize the total system Isp. For the initial lunar transit analysis, a constant input power is assumed to be available to the SEP throughout the mission lifetime, except while in shadow, where thrust is set to zero if solar illumination is less than 100%. The division of power between thrusters is dependent upon the operating thruster configuration – the combination of thrusters and desired operating points.

All thrust arcs are designed with a 90% duty cycle to provide margin and enable time for non-thrusting activities, such as communication, tracking, or unplanned loss of thrust. The duty cycle is modeled within the trajectory design by decrementing the SEP thrust and mass flow rate to 90% of the maximum values at the chosen operating point. This model effectively implies that the SEP system is only active during 90% of the duration of any given thrust arc. This duty cycle can be thought of as a margin. As with all margins, as the trajectory is matured to the trajectory for flight, and more vehicle details are finalized, this duty cycle will be reduced.

No restrictions are placed on the net thrust direction or the rate of change of the net thrust direction. Unless designed otherwise, the thrust is free to point in any direction to satisfy the mission constraints and maximize the objective. This freedom results in a continuously time varying thrust direction over the set of thrusting arcs. Further, it is assumed that thrust is delivered through the spacecraft center of gravity and the solar arrays can be pointed to provide adequate power for any thrust direction.

LUNAR TRANSIT SUMMARY

Several trades and analyses contribute to the design of the low thrust transfer from the initial parking orbit to the final NRHO. The main requirement for the lunar transit is to avoid passing through eclipses greater than 90 minutes. The entire transfer has been specifically designed to avoid eclipses given the initial orbit and performance assumptions of the EP system. Transit selection thus involves the identification of launch dates and times that yield transfers without extended eclipses.

Mission design methods

With the announcement in 2021 of the selection of SpaceX as the launch provider, the CMV is planned for delivery to its starting orbit on a Falcon Heavy rocket. Assuming a fixed perigee, the launch vehicle's performance to an injection orbit depends heavily on that orbit's apogee. Increasing the required delivered mass to orbit results in correspondingly lower apogees achieved by the launch vehicle.

For the low thrust transfer, trip time and mass delivery depend upon the performance of the launch vehicle, the starting orbit in which the launch vehicle places the spacecraft, the on-board electric propulsion system, and the propellant capacity of the on-board tankage system. Higher mass and lower starting apogee altitudes result in longer trip times to the destination. The low thrust system trades efficiency of the low thrust system for trip time. The CMV mission design team seeks to balance the mass delivery with a desire to limit trip time as much as possible.

The trade space defining the relationship between launch vehicle performance and low thrust performance is illustrated in Figure 3. The green regions represent capabilities above the assumed launch vehicle's performance to apogee altitude. The pink regions represent low thrust transfers, at a given starting mass and apogee altitude, where the trip time exceeds one year. The purple regions represent conditions where the low thrust transfers exceed the assumed propellant capability on board the CMV. The blue lines denote times of flight to the NRHO. Thus, if a time of flight less than 365 days is desired given the selected launch vehicle, the white regions in the plots in Figure 3 represent achievable apogee-wet mass pairs. The current design reference mission, 3, is shown as the small red star on the graphic, representing a trip that takes slightly longer than 365 days to complete. This trajectory is on the launch vehicle performance curve and is well within the Xe capacity on board the vehicle.

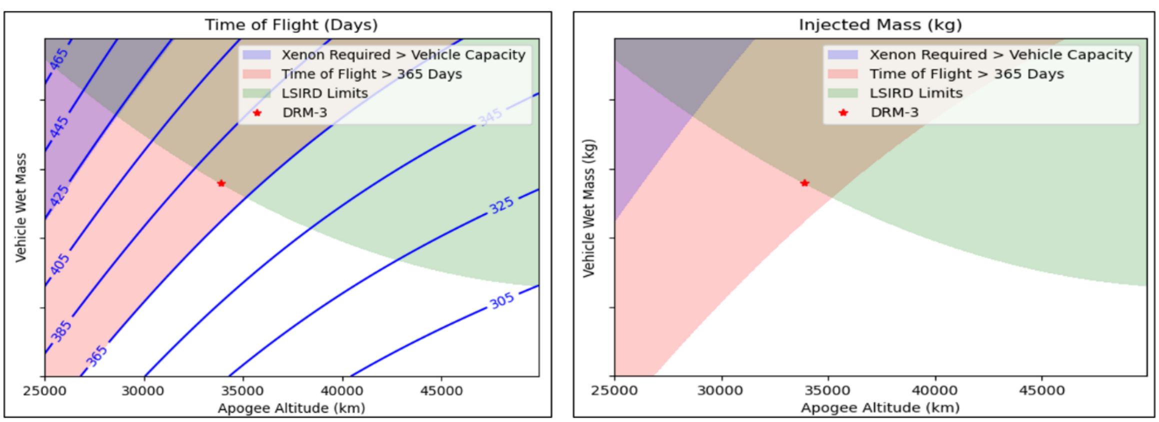

Figure 3. Time of flight (left) and injection mass (right) as a function of initial apogee altitude.

Lunar transfer

The latest CMV reference trajectory appears in the Earth-centered J2000 inertial frame in Figure 4. The Earth is located at the center of the plot axes. The distinct colors represent different thrust profiles. The initial portion of the spiral appear in red. These red arcs represent high-thrust electric propulsion mode portions of the transfer where the goal is raising the orbit as quickly as possible. The orange arcs, which are only slightly different from the red, denote high-Isp mode portions of the transfer where the primary goal is to make efficient use of the SEP thruster string performance. Eclipse periods are colored green; these are coast arcs required by the shadowing of the solar panels. Blue arcs represent zero-thrust during designed coast periods. The orbit of the Moon is shown in grey. Additionally, the phases of the transit are called out and explained in the next few paragraphs.

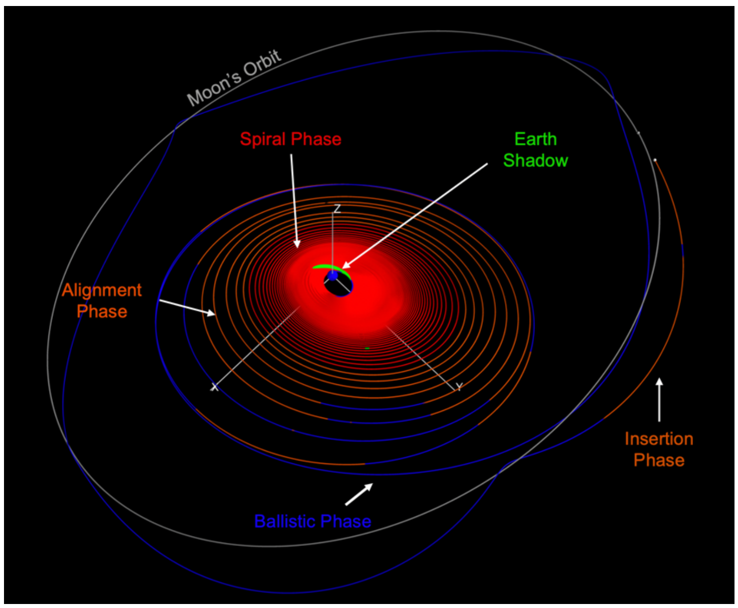

Figure 4. Lunar Transit in the Earth-Centered J2000 Frame.

The transfer is designed in four distinct subphases. Each subphase represents a portion of the spiral transfer that is different from each other in terms of operations, physics, optimization scheme, or all three. These four subphases are shown in green in each panel of the graphic in Figure 5.

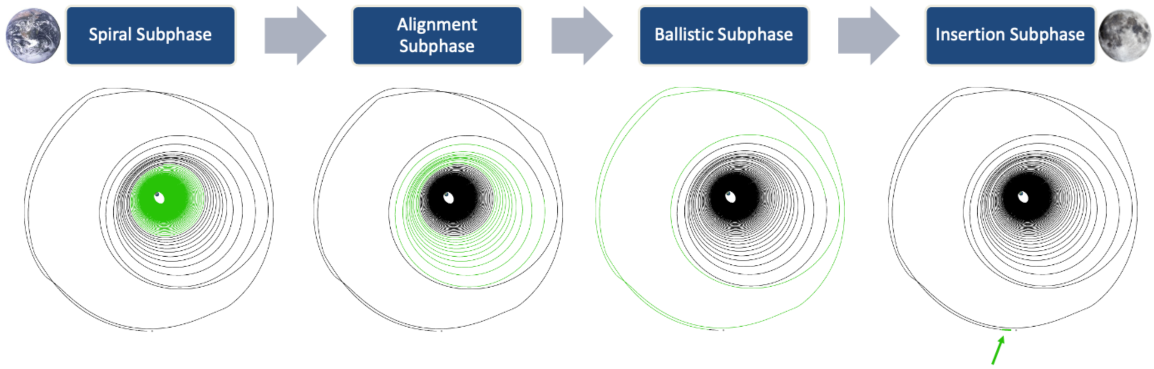

Figure 5. Lunar Transit subphase definition.

The first subphase is the spiral subphase, which encompasses the first 250 to 280 days of the nominal spiral transit. The spiral subphase corresponds to the high-thrust phase marked in red in Figure 4. Initial operations involve checkout of the on-board systems and launch vehicle dispersion makeup. Once sufficiently checked out and electric propulsion activated, this subphase consists of near-constant electric propulsion thrust arcs, except for planned operational coasts or coasts during eclipses. Operational coast periods could be planed for to exercise additional subsystem testing or other needs, but intent is to be minimized. By design, all thrusting during this subphase is in the orbital plane in the direction of the velocity vector in order to maximize the rate of addition of orbital energy and raise apogee/perigee as quickly as possible. This is the subphase that spends the most cumulative time in the earth's Van Allen Belts, and a goal of this subphase is to minimize spacecraft systems exposure time to the radiation in the belts. For the purpose of this current analysis, the end of the Spiral subphase is defined once the trajectory reaches 130,000 km semi major axis. This initial subphase is most like the traditional spirals that the current commercial electric propulsion vehicles employ to deliver payloads to Geostationary orbit (GEO).

The second subphase is the alignment subphase, which corresponds to the high-Isp phase shown in orange in Figure 4. This subphase lasts for approximation 50-90 days depending on the particular opportunity (launch date, spiral start date). This subphase includes imposed coast arcs, which may be placed in any location along the subphase. The alignment subphase is primarily where any plane change is performed.

The third subphase is the ballistic subphase, approximately 40-75 days in duration depending on the opportunity. This subphase is designed to consist primarily of coast arcs and makes use of the dynamics of the Earth, Moon and Sun to align the end of this subphase with the NRHO. While it is designed primarily as a long coast, trajectory correction maneuvers may be included. It is in this phase that the trajectory makes its closest lunar approaches over potentially several lunar flybys.

The fourth and final subphase is the injection subphase, where the trajectory has reached the location of the NRHO and the final optimal SEP thrusting completes the insertion into the NRHO. This phase lasts approximation 3 to 7 days depending on the opportunity and any operational issues that may have occurred during the transfer up to this point.

DESIGN REFERENCE MISSION

At the time of this paper, the PPE mission design team has completed the third design reference mission (DRM) for the transfer to the NRHO. These DRMs are updated as details of the performance and assumptions of the system (PPE, HALO, and launch vehicle) are solidified. The DRMs are provided to the CMV engineering subsystem teams (thermal, communications, etc.) for use in their analyses. The results of these analyses are fed back to the mission design team and affect design decisions. As the PPE, HALO and ultimately the CMV complete design reviews, the DRMs will mature to a baseline trajectory and ultimately the flight trajectory that will be transmitted to and flown by the PPE as it performs the lunar transit.

DRM3 summary

DRM3 assumes a launch in January of 2024, with a goal of transit to the moon by early 2025. This date will be changed in subsequent DRMs as development decisions and launch date decisions are finalized. A launch date change will be reflected in the upcoming DRM4. The total time of flight of DRM3 is 383 days, where 319 days are spent thrusting and 64 days are spent coasting. A given thrust arc can last up to 77 days or can be as short as a few hours in duration. The coast arcs are a combination of optimal coasts, planned coasts, and coasts during eclipse. Eclipse durations range from 13 to 62 minutes. The mission phases of DRM3 appear in Table 2, along with their durations and starting ranges from the Earth and the Moon. The range from the Earth and the Moon over time appear in Figure 6. Note the steady increase in perigee and apogee radii as the SEP engines continually adjust the trajectory. The ballistic phase at the end of the transfer results in the final close approaches to the Moon. There is currently no constraint limiting the final closest approach distance to the moon in the modeling of the transit trajectory. Analysis is ongoing to understand what constraints should be placed based on vehicle or environmental considerations.

Table 2. DRM3 Mission Phases

Event	Date	MET (days)	IPS State	Earth Range (km)	Moon Range (km)
Separation & Checkout	JAN 27, 2024	0.0	Off	6,588	406,570
Spiral Phase	FEB 03, 2024	7.0	High Thrust	6,593	388,570
Alignment Phase	OCT 29, 2024	276.8	High Isp	124,975	524,899
Ballistic Phase	DEC 31, 2024	339.5	Off	259,235	638,321
Insertion Phase	FEB 08, 2025	378.8	High Isp	389,849	47,747
NRHO Arrival	FEB 13, 2025	383.4	Off	403,543	40,472

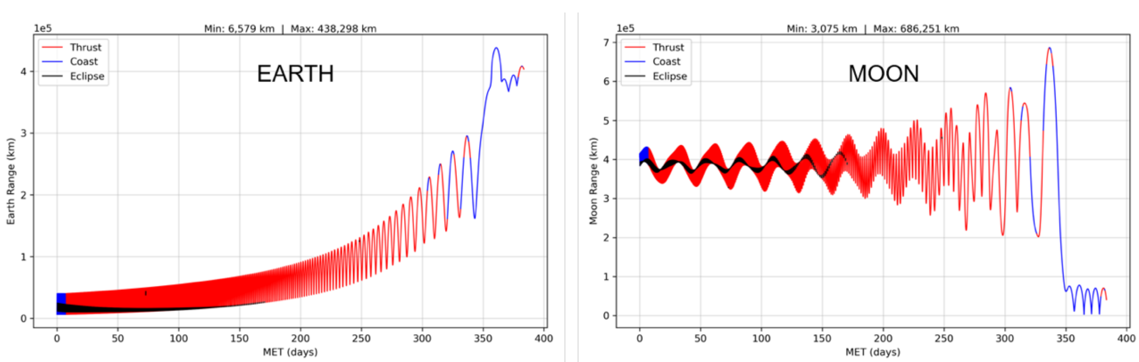

Figure 6. Range to the Earth (left) and Moon (right) during the DRM3 transfer.

CONCLUSION

The use of the highly efficient Solar Electric Propulsion system of the PPE enables the delivery of the PPE and HALO to the NRHO on a single launch. The Design Reference Missions designed for the CMV's lunar transit trajectory have been used by the CMV subsystem teams to design their systems as the combined vehicle moves from preliminary to final design reviews. The DRMs have provided the CMV team insight into environments, operations, and delivered mass capability of the combined PPE/HALO vehicle. With the PPE and HALO working toward their Critical Design Reviews, the reference missions have been used to make vehicle design decisions such as increasing Xe tank capacity, thermal system capacity, and reductions in mass.

When it flies, the CMV will represent an order of magnitude of higher power SEP and demonstrate capabilities to deliver greater payload masses to the moon than has been accomplished to date. This electric propulsion system is on the path toward the high-power electric propulsion vehicles that will enable human exploration to Mars and previously impossible robotic exploration to the outer planets.

ACKNOWLEDGMENTS

The authors wish to thank our management, both the Power and Propulsion Element and the Gateway, and leadership at our respective centers for their continued support of our efforts. Special thanks to Michael Barrett, the project manager of the PPE, Vicki Crable the PPE Systems Engineering and Integration lead and Joe Caram the lead of the Gateway Systems Engineering and Integration group for their leadership as we successfully bring the PPE and Gateway to its final home in the NRHO. We thank our families for their support and understanding of the hours we've been keeping and will continue to keep into the near future.

REFERENCES

[1] NASA press release, Feb 9, 2021. Release 21-015, https://www.nasa.gov/press-release/nasa-awards-contract-to-launch-initial-elements-for-lunar-outpost

[2] Davis, Diane, et. al. "Orbit Maintenance and Navigation of Human Spacecraft at Cislunar Near Rectilinear Halo Orbits," AAS/AIAA Space Flight Mechanics Meeting, San Antonio, TX, Feb. 2017, JSC-CN-38626.

[3] https://www.nasa.gov/feature/nasa-s-lunar-outpost-will-extend-human-presence-in-deep-space

[4] https://www.nasa.gov/press-release/nasa-awards-contract-to-launch-initial-elements-for-lunar-outpost

[5] Herman, Daniel, et. al. "Overview of the Development of the Advanced Electric Propulsion System (AEPS)," International Astronautical Congress 2017, Sept. 2017, IAC-17.C4.4.2, GRC-E-DAA-TN46546.

[6] McCarty, Steven. L, Waldy K. Sjauw, Laura M. Burke, Melissa L McGuire, "Analysis of Near Rectilinear Halo Orbit Insertion with a 40-kW Solar Electric Propulsion System," AIAA Astrodynamics Specialist Conference 2018, Snowbird, UT, GRC-E-DAA-TN59588.

[7] Parrish, Nathan L. et. al. "Ballistic Lunar Transfers to Near Rectilinear Halo Orbit: Operational Considerations," 30th AIAA/AAS Spaceflight Mechanics Meeting, Orlando FL, JSC-E-DAA-TN75856

[8] Launch Vehicle Performance Website, https://elvperf.ksc.nasa.gov/Pages/Default.aspx

[9] NASA press release, Feb 9, 2021. Release 21-015. *NASA Awards Contract to Launch Initial Elements for Lunar Outpost* https://www.nasa.gov/press-release/nasa-awards-contract-to-launch-initial-elements-for-lunar-outpost

[10] NASA White Paper: Gateway Destination Orbit Model: A Continuous 15 Year NRHO Reference Trajectory, August 20, 2019. https://ntrs.nasa.gov/api/citations/20190030294/downloads/20190030294.pdf

[11] D. E. Lee, R. J. Whitley and C. Acton, "Sample Deep Space Gateway Orbit, NAIF Planetary Data System Navigation Node," NASA Jet Proplusion Laboratory, July 2018. [Online]. Available: https://naif.jpl.nasa.gov/pub/naif/misc/MORE_PROJEC TS/DSG/.

[12] J.Willams, D.E. Lee, R.J. Whitley, K.A. Kokelmann, D.C. Davis, C.F. Berry, “Targeting Cislunar Near Rectilinear Halo Orbits for Human Space Exploration,” 2017 AAS/AIAA Space Flight Mechanics Meeting, San Antonio, TX, AAS 17-267.