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ARTEMIS I TRAJECTORY DESIGN AND OPTIMIZATION

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This paper presents the overall trajectory design and optimization process for NASA's Artemis I mission to send an uncrewed Orion vehicle to a lunar Distant Retrograde Orbit (DRO). The on-orbit trajectory begins at the Space Launch System (SLS) core separation and ends at the Orion service module Earth Entry Interface (EI) point. The details of the trajectory optimization process are presented, including design of nominal and extended mission options, launch windows, and abort options. Novel design techniques are also discussed to account for contingencies, such as using auxiliary thrusters to protect against main engine failure and applying trajectory shaping to mitigate or reduce eclipse durations.

INTRODUCTION

Artemis I is the mission formerly known as Exploration Mission 1.[1,2] It will be the first integrated test of NASA's Orion spacecraft and the Space Launch System (SLS) rocket.[3] The Artemis program is the foundation for sustainable human lunar exploration and beyond. The key characteristic of the Artemis mission design is to support extensible capabilities through an architecture utilizing the SLS and Orion vehicles for an array of missions. SLS and Orion are used as the crew transport for missions into cislunar space, including a rendezvous with a reusable habitation element (called Gateway), and a Human Landing System (HLS) to return astronauts to the surface of the Moon.[4] To achieve a successful lunar landing for Artemis III, Artemis I has key mission priorities that demonstrate NASA's capability to extend human presence on the Moon, including:

- demonstrate Orion's heat shield at lunar return re-entry conditions,
- validate required system performance that is mandatory to support crewed missions,
- demonstrate SLS ascent and launch vehicle operations,
- operate systems in flight environment,
- demonstrate Orion deep space environmental performance, communications, propulsion, and navigation systems,
- demonstrate Exploration Ground Systems (EGS) and day of launch operations,

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- demonstrate flight operations management and execution, and
- retrieve the Orion Crew Module (CM)

For Artemis I, an uncrewed Orion will fly a round-trip mission to a lunar Distant Retrograde Orbit (DRO) on a trajectory that includes two lunar flyby maneuvers, providing an excellent opportunity to test out Orion's systems in the cislunar regime. This paper describes how the Artemis I reference trajectories were designed and optimized while adhering to subsystem constraints, and integrating ascent, orbit, and entry phases of flight.

GENERAL ASSUMPTIONS

Mission Design and Optimization

The Artemis I mission is optimized with NASA's Copernicus spacecraft trajectory design and optimization application[5] using the SNOPT optimization method.[6] Copernicus (shown in Figure 1) makes use of a multiple-shooting approach, and the mission is designed using numerous coast and burn trajectory segments that are numerically integrated both backwards and forwards in time. In addition, a number of customized Copernicus plugins are employed to model specific aspects of the mission.[7] Figure 2 displays a Directed Acyclic Graph (DAG) of all the mission components (segments and plugins) as defined in Copernicus to optimize this mission. Segments 1–25 represent the SLS/ICPS part of the mission, while segments 26–52 represent the nominal Orion mission, segments 53-54 represent the DRO, and segments 55–70 represent the Auxiliary Down Mode (XDM) segments.

Vehicle Assumptions

The Orion spacecraft serves as the primary crew vehicle for NASA's Exploration Systems Development (ESD) missions. It is capable of conducting in-space operations in conjunction with payloads delivered by the SLS. It consists of four main elements: the CM providing a habitable pressurized volume to support crew members and cargo, the Service Module (SM) consisting of the European Service Module (ESM) and the Crew Module Adapter (CMA) responsible for propulsion, heat rejection and power generation, the Spacecraft Adapter (SA) as a structural connection to the launch vehicle, and the Launch Abort System (LAS) providing abort capability to transport the CM away from the launch vehicle during ascent. For this trajectory design and optimization, only the CM and SM of Orion are included.

Orion is attached to the Interim Cryogenic Propulsion Stage (ICPS), the upper stage of SLS. The ICPS performs all propulsive maneuvers until the ICPS/Orion spring separation post-TLI, after which the Orion SM provides propulsive support the rest of the mission. The Orion SM consists of three engine types: the main engine Orbital Maneuvering System Engine (OMSe), secondary Auxiliary (AUX) thrusters, and the Reaction Control System (RCS). Copernicus only models the first two. The OMSe has a thrust of about 26,000 N, while the eight AUX thrusters combined have a thrust of about 3,400 N (and a slightly lower Isp).

Environment Simulation

In Copernicus, all mission phases are integrated explicitly using the DDEABM integration method (a variable-step size variable-order Adams-Bashforth-Moulton implementation) with a 10^{-12} error tolerance.[8] The JPL DE 421 ephemeris[9] was used for the celestial body ephemeris of the Earth,

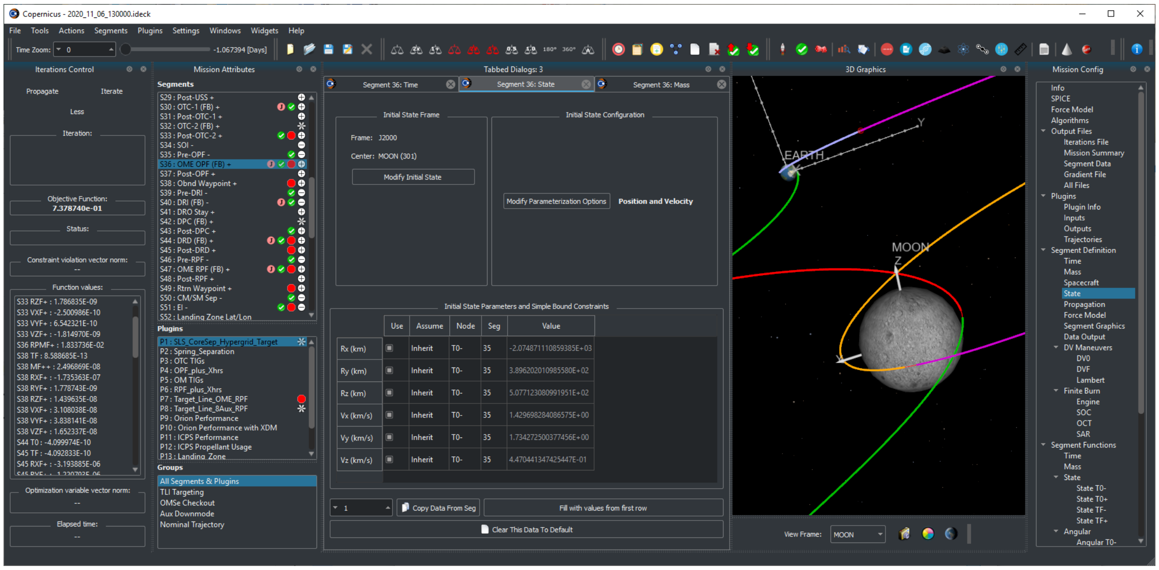


Figure 1: Screenshot of Copernicus. Copernicus is used to design the Artemis I trajectory, run the performance scans, and will also be used during the mission to provide real-time optimization support.

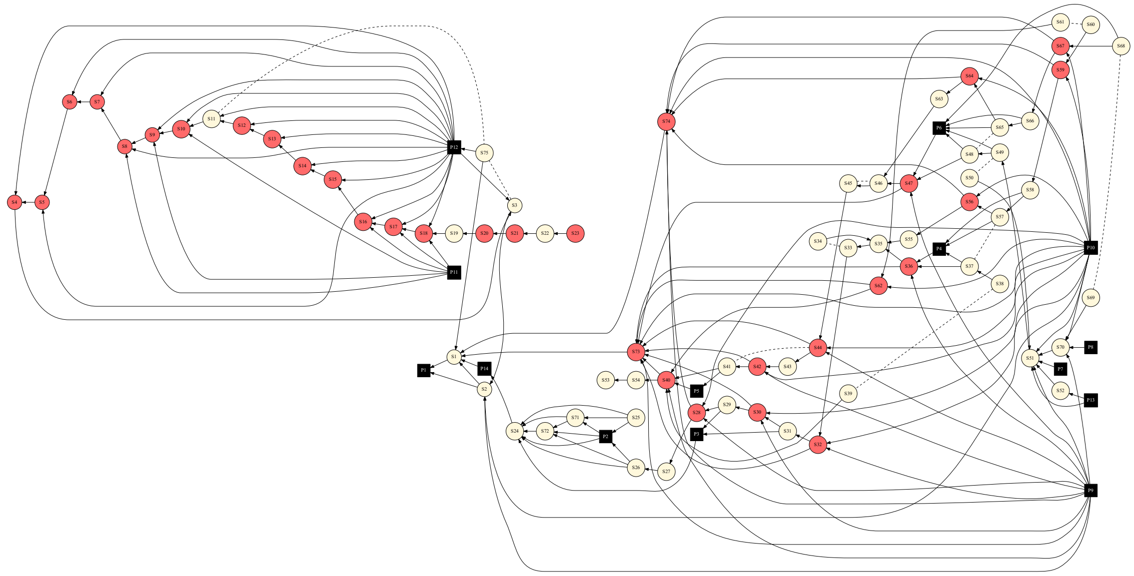


Figure 2: Copernicus Dependency Diagram. This shows the dependency information of the segments and plugins in the mission. Circles are segments (red circles include finite burn maneuvers) and squares are plugins. Arrows represent dependencies and dashed lines represent constraints.

Moon, and Sun. The Earth departure phases use an 8×8 Gravity Recovery And Climate Experiment (GRACE) GGM02C Earth gravity model.[10] The Moon-centered phases use the Gravity Recovery and Interior Laboratory (GRAIL) GRGM660PRIM gravity model.[11] A 50×50 model is used for the close lunar flybys, a 4×4 model is used in the DRO, and an 8×8 model is used in other cases. Third bodies are treated as point masses. The high-fidelity body fixed frames for the Earth and Moon are used (i.e., IAU_76_80 from SOFA[12] and MOON_PA from SPICE, respectively) for gravity models and the computation of geodetic parameters.

Destination Orbit

The destination orbit for Artemis I is a Moon-centered DRO. The details of this orbit were described in an earlier paper,[2] and an example is shown in Figure 3. The DRO is computed along with the nominal mission as part of the same optimization problem. The initial state of the DRO is propagated backwards in time to serve as the target for the DRO Insertion (DRI) burn. Note that the nominal DRO is planar (i.e., in the Earth-Moon plane), but during the eclipse mitigation process, it may be inclined (see subsequent eclipse section for details on this process).

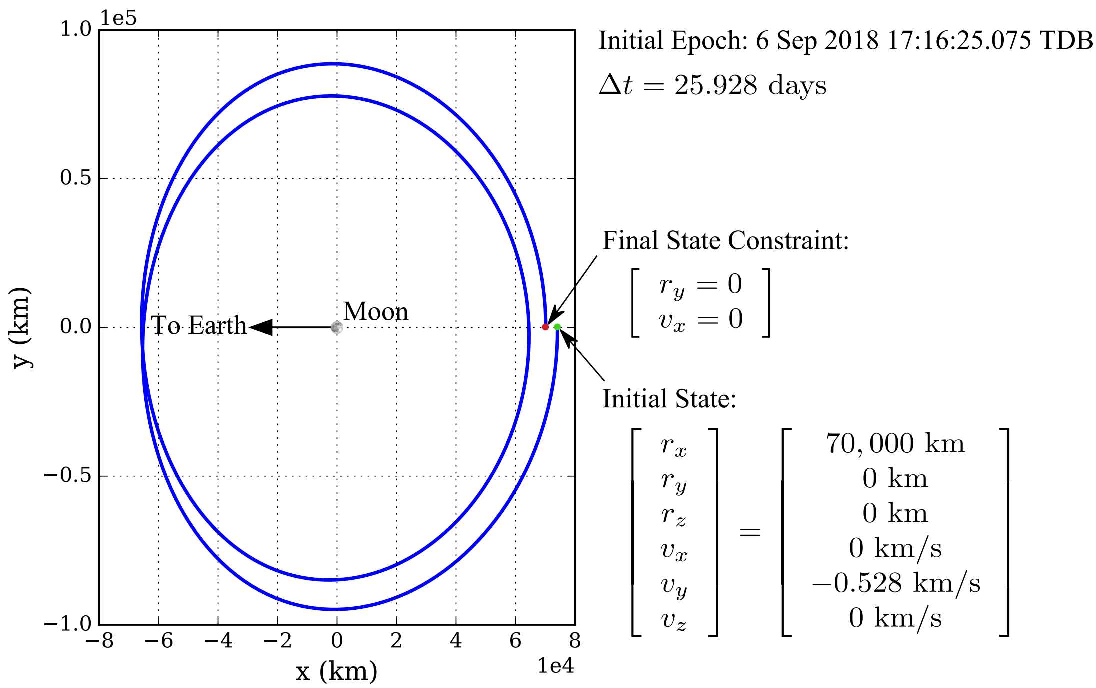


Figure 3: Representative DRO Example (Earth-Moon Rotating-Pulsating Frame).[2] When targeting a DRO, a simple forward-shooting method is used to target a state two revolutions after a perpendicular X-axis crossing. The only two control variables are the coast duration Δt and the initial y-velocity (v_y) in the rotating-pulsating frame.

NOMINAL MISSION OVERVIEW

Figure 4 shows a visualization of a nominal Artemis I trajectory in Copernicus. The general round-trip DRO trajectory design is largely based on earlier studies conducted for the Asteroid

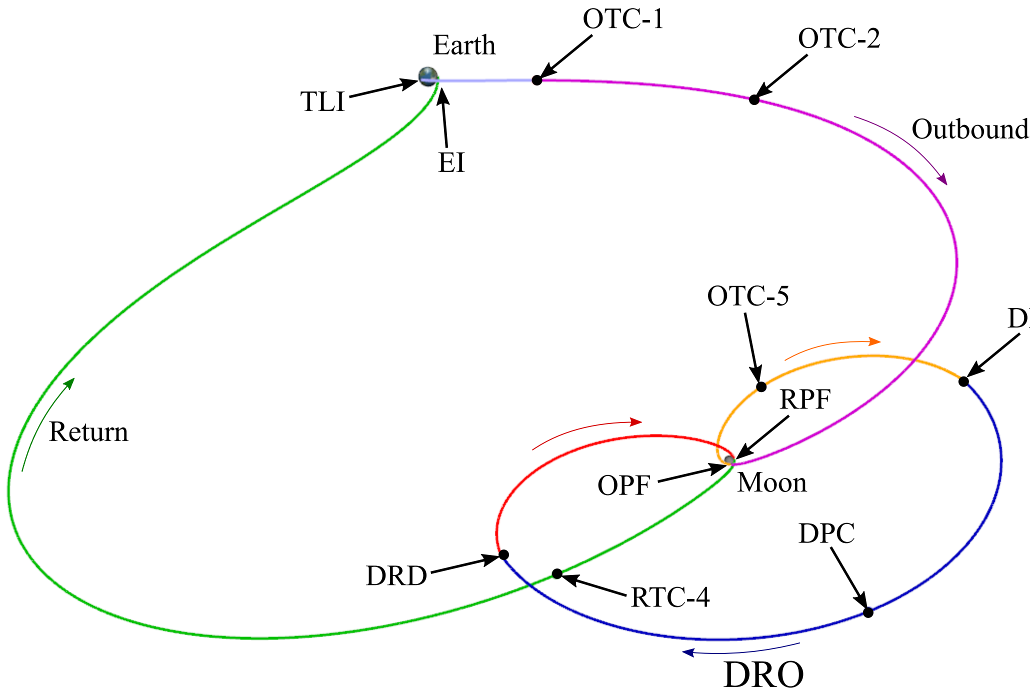


Figure 4: Example Artemis-I Mission (Earth-Moon Two-Body Rotating Pulsating Frame). This shows an example nominal mission case for a November 6, 2020 launch.

Redirect Crewed Mission (ARCM).[13] The trajectory can be broken into three phases of flight: Ascent, In-Space, and Entry. The Copernicus trajectory optimization software specializes in on-orbit trajectory design, and therefore Copernicus plugins are used to simulate the ascent and entry phases of flight which are discussed in the subsequent sections.

The trajectory begins at SLS Core Separation which places the ICPS/Orion stack into a 16 $\times$ 975 nmi orbit (30 x 1806 km). The ICPS performs a Perigee Raise Maneuver (PRM) to raise the perigee to 100 nmi (185 km). Then ICPS performs a Trans Lunar Injection (TLI) burn to target the Outbound Powered Flyby (OPF) burn, which Orion performs as it flies by the Moon. About 10 minutes after TLI, an impulsive spring separation plugin models the separation between the ICPS and Orion vehicles. The ICPS targets the Orion Target Interface Point (TIP) 40 seconds after the spring separation. After another 41 seconds, Orion executes the Upper Stage Separation (USS) burn to increase the separation distance with the ICPS. At 6 hours after TLI, Orion performs a 30 second Outbound Trajectory Correction (OTC)-1 burn to test the OMSe engine prior to its first major burn behind the Moon with no communication to ground support – this is also called the OMSe Checkout (OCO) burn component. Since ICPS performs a heliocentric disposal via lunar flyby, the OTC-1 burn also reduces the ICPS and secondary payload recontact risk which is needed since they are also headed towards the Moon. An optional OTC-2 burn can be used to provide the optimizer with additional control to target OPF throughout a launch window. At the lunar flyby, Orion performs its first major burn, OPF, followed by its second major burn, DRI, which inserts Orion into the DRO. While in the DRO, an optional DRO Plane Change (DPC) burn may be used to incline the DRO in order to avoid vehicle eclipse duration violations. After DRO operations, the Orion performs the DRO Departure (DRD) burn to target the lunar Return Powered Flyby (RPF) burn onto the Earth return trajectory. The return trajectory targets a high speed atmospheric entry on the order of 36,000 ft/s (11 km/s), suitable for demonstrating the performance and effectiveness of the Orion Thermal Protection System (TPS) heat shield, as well as relevant environments prior to the first crewed launch of the system.

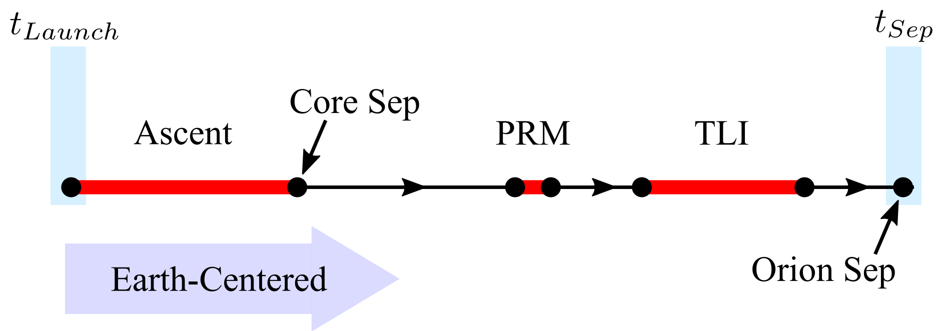
ASCENT TO SEPARATION

The first part of the mission (see Figure 5a) consists of the period from the SLS launch to the separation of the Orion vehicle from the ICPS. This phase is initially included in the optimized trajectory in Copernicus, using a plugin that queries a database of SLS launch trajectories[14] generated at MSFC by the Program to Optimize Simulated Trajectories (POST) tool.[15] This provides Copernicus a close approximation of the ascent phase that can be used during optimization.[16] The SLS database consists of a set of data grids of core separation state vectors computed at monotonically increasing launch azimuth values. Each point in the grid represents a single POST run over an incremented launch azimuth sweep. There is one grid per month which uses the mean monthly trajectory in order to account for seasonal wind variation. When setting up the mission for optimization, the Copernicus plugin reads the launch epoch from the trajectory, and based on the calendar month, will use the corresponding grid. During optimization, the CoreSep state is interpolated from the grid using the launch azimuth as the independent variable.

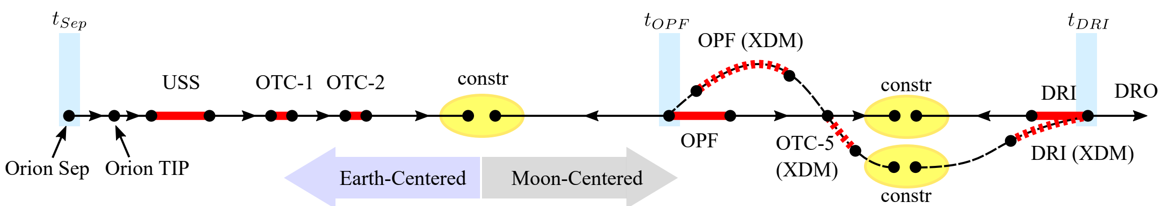
All launch azimuths target a 16 $\times$ 975 nmi insertion orbit at Core Separation. Approximately 45 minutes after CoreSep, the vehicle stack has coasted near apogee where ICPS performs the PRM to raise the Low Earth Orbit (LEO) perigee altitude to achieve a stable orbit condition. The TLI burn begins prior to the next perigee passage and occurs at a true anomaly between $-40°$ and $-95°$ in order to intercept the Moon throughout the launch period each month. The objective function to be minimized for this problem is the combined ICPS and Orion total Δv. It should also be noted that these targets are generated for a nominal short-class mission without the OTC-1 and OTC-2 maneuvers, or any Orion trajectory mitigations.

This entire mission phase, as seen in Figure 5a is considered an estimate since the ICPS phase of flight is not officially modeled in Copernicus. After the initial optimization of the entire mission (see Figure 6), feasible missions from this phase* are reprocessed by the launch vehicle provider to create a high fidelity simulation. The Orion trajectory team incorporates the high fidelity TLI targets *back* into Copernicus through the use of a TLI targets plugin. Therefore, the Pre-ICPS/Orion Spring Separation (see "Orion Sep" in Figure 5a) is then used as the starting point for the on-orbit mission optimization. After these high fidelity targets are implemented, everything from launch to this point is no longer directly modeled. The one optimization variable for this part of the mission then becomes the launch epoch, which uniquely determines the corresponding Orion Sep state by interpolation of the database. Note that, while the TLI targets plugin does support interpolation, for production runs, interpolation is not necessary since all points in the database lie on the integer UTC minute. These correspond to the points computed in the launch window scan (see subsequent sections).

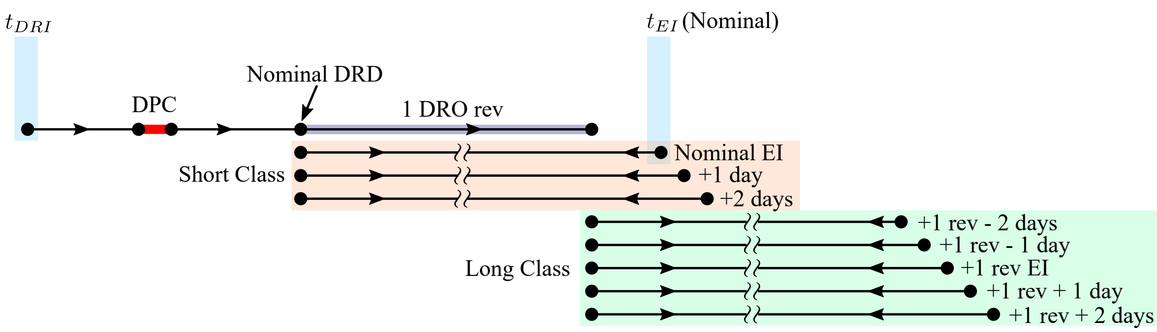
*Points that exceed the available vehicle performance (e.g., if the azimuth and inclination bounds exceed SLS range safety limits) are excluded.



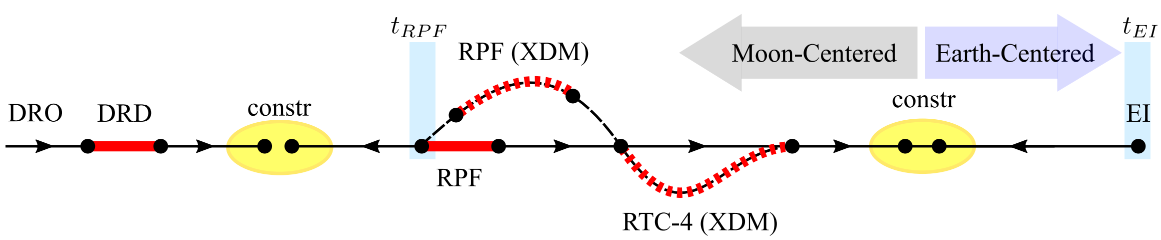
(a) Launch to Orion Spring Separation. Ascent is performed by the SLS and the Perigee Raise Maneuver (PRM) and Trans Lunar Injection (TLI) are performed by the ICPS.



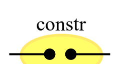
(b) Orion Spring Separation to DRO Insertion. The outbound part of the mission includes two major burns: OPF and DRI. The Auxiliary Down Mode (XDM) branch is constrained to arrive at the post-DRI epoch at the same time, state and mass as the nominal OMSe branch.



(c) DRO Stay Time and Mission Duration for Different Mission Classes. Extended missions can be used to achieve desired landing lighting conditions not otherwise possible with the nominal mission of approximately 26 days.



(d) DRO Departure to Earth EI. The return part of the mission includes two major burns: DRD and RPF. The Auxiliary Down Mode (XDM) branch is constrained to arrive back on the nominal trajectory at the conclusion of RTC-4.

Figure 5: Mission Timelines. In these charts, 

indicates continuity constraints in the multiple shooting transcription. States are propagated forward and backward from the epochs (the direction of propagation is indicated by the arrows). The red lines are burns and the black lines are coasts.

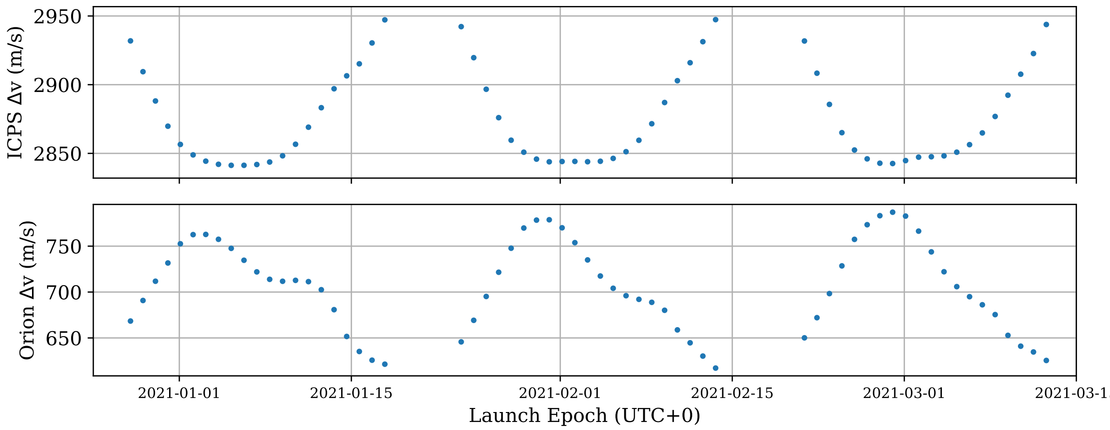


Figure 6: ICPS and Orion Vehicle Performance for Launch Targets Generation Scan. For the initial scan, the entire mission is optimized from ascent to EI which is used to generate the launch targets. This plot shows three example launch periods at the beginning of 2021.

ORION ON-ORBIT TRAJECTORY

The Orion in-space portion of the trajectory begins at the ICPS/Orion Spring Separation ("Orion Sep" in Figure 5a). For optimized trajectories incorporating the TLI targets plugin, this state and time are entirely functions of the launch epoch. Figures 5b-5d show the detailed timelines of the on-orbit portion of the Artemis I mission, from Orion Spring Separation to Entry Interface (EI). In these diagrams, the arrows indicate the direction of propagation of the mission phases in the Copernicus multiple-shooting transcription. Segments numerically integrated forward from one time (e.g., t_{OPF}) and backwards from a later time (e.g., t_{DRI}) have intermediate state, time, and mass constraints imposed to achieve a continuous mission when converged. Once the ascent to Orion separation mission phase is eliminated using the TLI database, the objective function is the total Orion Δv required, which is to be minimized.

The thrust directions for all burns are modeled by the spherical angles α (right ascension) and β (declination), as functions of time:[17]

$$\alpha(t) = \alpha_0 + \dot{\alpha}_0(t - t_0) \tag{1}$$

$$\beta(t) = \beta_0 + \dot{\beta}_0(t - t_0) \tag{2}$$

Where t_0 is the burn start time, and the direction of the thrust vector $\hat{\mathbf{u}}$ at time t is given by:

$$\hat{\mathbf{u}}(t) = [\cos\alpha(t)\cos\beta(t)]\hat{\mathbf{e}}_1 + [\sin\alpha(t)\cos\beta(t)]\hat{\mathbf{e}}_2 + [\sin\beta(t)]\hat{\mathbf{e}}_3 \tag{3}$$

Where the basis vectors $[\hat{\mathbf{e}}_1, \hat{\mathbf{e}}_2, \hat{\mathbf{e}}_3]$ can be:

- the `IJK` frame ($\hat{\mathbf{e}}_1 = \mathbf{i}$, $\hat{\mathbf{e}}_2 = \mathbf{j}$, $\hat{\mathbf{e}}_3 = \mathbf{k}$)
- the Copernicus `VUW` controls frame ($\hat{\mathbf{e}}_1 = \mathbf{v}/\|\mathbf{v}\|$, $\hat{\mathbf{e}}_3 = \mathbf{h}/\|\mathbf{h}\|$, $\hat{\mathbf{e}}_2 = \hat{\mathbf{e}}_3 \times \hat{\mathbf{e}}_1$).
- the Copernicus `VNC` controls frame ($\hat{\mathbf{e}}_1 = \mathbf{v}/\|\mathbf{v}\|$, $\hat{\mathbf{e}}_2 = \mathbf{h}/\|\mathbf{h}\|$, $\hat{\mathbf{e}}_3 = \hat{\mathbf{e}}_1 \times \hat{\mathbf{e}}_2$)

Note that these frames can be defined in either an inertial or rotating reference frame. In some cases (e.g. OPF and RPF), burns are parameterized in a "VUW" Earth-Moon rotating-of-date frame* so

*An "of-date" frame in Copernicus is an inertial frame where the rotating frame is fixed at an epoch that when stepping through epochs in the scan, the previous case seeds a better initial guess for the next case. The significant burns in the mission are:

- Upper Stage Separation (USS) : Used to reduce near-field recontact risk. A fixed $\Delta v = 5.5 \text{ ft/s} \ (1.6764 \text{ m/s})$ burn along the velocity vector.
- Outbound Trajectory Correction (OTC)-1 : Includes the OCO burn component, a 30 sec burn, optimized α and β, parameterized as `J2000-VNC`.
- Outbound Trajectory Correction (OTC)-2 : A small AUX burn (used nominally during the launch window as an extra control in order to allow the OPF epoch and DRO state to be fixed), optimized α and β, parameterized as `J2000-IJK`.
- Outbound Powered Flyby (OPF) : A major burn with optimized Δt, optimized α and β, parameterized in an Earth-Moon Two-body rotating "of date" `VUW` frame. This burn is mostly in the anti-velocity direction.
- DRO Insertion (DRI) : A major burn with optimized Δt, optimized α and β, parameterized in an Earth-Moon Two-body rotating "of date" `IJK` frame.
- DRO Plane Change (DPC) : An optional burn used to incline the DRO, with optimized Δt, optimized α and β, parameterized in an Earth-Moon Two-body rotating "of date" `IJK` frame.
- DRO Departure (DRD) : A major burn with optimized Δt, optimized α and β, parameterized in an Earth-Moon Two-body rotating "of date" `IJK` frame.
- Return Powered Flyby (RPF) : A major burn with optimized Δt, optimized α and β, parameterized in an Earth-Moon Two-body rotating "of date" `VUW` frame. This burn is mostly in the velocity direction.

In general, the major optimization variables are:

- The finite burn control law parameters α_0, β_0, (and optionally $\dot{\alpha_0}$, and $\dot{\beta_0}$) for each of the optimized maneuvers.
- The burn time Δt for each of the optimized maneuvers.
- The various intermediate flight times.
- The launch epoch.
- The OPF and RPF flyby epochs.
- The OPF and RPF flyby state parameters (periapsis radius, eccentricity, inclination, ascending node, argument of periapsis, and true anomaly). At the start of the burn, the flyby periapsis radius is bounded to be no less than 100 km above the lunar surface. The true anomaly is bounded to be within $\pm 90°$.
- The EI epoch.
- The EI longitude, latitude, velocity, azimuth, and flight path angle. The EI state is numerically integrated backwards and constrained to provide state and time continuity with the forward-integrated return trajectory (this allows for specifying the EI altitude directly rather than being imposed as a nonlinear constraint). The geodetic altitude is specified to be 121.92 km.

The main problem constraints are:

- The various time and state continuity conditions along the trajectory (see Figure 9).
- There is a minimum 5.15 day outbound (launch to OPF) flight time constraint in order to maximize flight time and ensure consistency for the secondary payloads on board ICPS.
- The EI longitude, azimuth, and flight path angle are constrained to be on the EI target line (see Figure 7). The nominal and XDM return phases are specified to have the same EI state and time.

- Auxiliary Down Mode (XDM) constraints as explained in the next section.

It is important to note that the EI state is parameterized in an Earth-fixed reference frame, which will produce locally-optimal solutions roughly every day for the EI epoch. The problem setup (i.e., propagating backwards in time from EI) makes it straightforward to vary the EI epoch in the scan by one day increments and quickly re-converge the solution.

AUXILIARY DOWN MODE

While the nominal Artemis I mission utilizes the OMSe throughout, the OPF and RPF phases each include a parallel trajectory option in which Orion performs an Auxiliary Down Mode (XDM), and continues the duration of the region using these secondary thrusters, deviating from the nominal OMSe trajectory. These parallel trajectories are a part of the optimization problem in order to protect against a main engine failure. During the Shuttle program, trajectory flight controllers assessed thruster downmode contingencies in the event of an OMSe failure, in an attempt to minimize the penalty that resulted from changing the number of thrusters from two nominally planned OMSe, to one.[18] Similarly, the Artemis I trajectory uses this strategy if a contingency requires switching from the main OMSe to the secondary auxiliary thrusters. These engine configurations have a tremendous impact on performance due to different thrust and Isp capabilities if the Time of Ignition (TIG) of the auxiliary thrusters occurs at the OMSe optimal time. This results in a performance penalty that is beyond what is available in Orion's propellant margin. Therefore, an optimization strategy assuming both the OMSe and AUX produces a balanced ignition time for the outbound and return flyby burns, thus reducing the downmode penalty in the event of a main engine failure. This comes at a cost compared to an OMSe-optimal TIG, however, pre-planning a potential downmode protects Orion's propellant budget.

Figure 5b and Figure 5d represent the XDM timelines for OPF and RPF, respectively. For the OPF XDM, the OPF AUX burn starts 10 seconds after the OPF OMSe TIG (assuming a failed OMSe startup). The burn time, the burn α and β angles, and the burn $\dot{\alpha}$ angular rate are optimized for the AUX engines and target the position vector on the nominal trajectory at 18 hours after the OPF OMSe TIG. The 18 hour waypoint was chosen to allow Orion to be in a Tail-to-Sun attitude for 15 hours for thermal considerations, followed by a 2 hour optical navigation observation of either the Earth or Moon. At this outbound waypoint, OTC-5 is an AUX burn with an optimal burn time and burn α and β angles that targets the DRI, which is backwards propagated from the nominal DRO. DRI uses the recovered OMSe and uses an optimized burn time, and burn α and β angles to target the nominal DRO insertion epoch, state, and mass. The total propellant mass used by the XDM OPF through DRI segments is constrained to be equal to the propellant mass used by the nominal OPF through DRI segment burns.

The RPF XDM sequence is initially similar to the OPF XDM sequence. The RPF AUX burn starts 10 seconds after a failed RPF OMSe startup. Burn time, burn α and β angles, and burn $\dot{\alpha}$ angular rate are optimized to target the nominal position vector 18 hours after the RPF OMSe TIG. At this return waypoint, RTC-4 uses an optimal burn time and burn α and β angles to target the nominal EI epoch, mass, and state, which is backwards propagated to an area close to the lunar sphere of influence. The total propellant used by the XDM RPF through RTC-4 segments is constrained to be equal to the propellant mass used by the nominal RPF segment burn.

EXTENDED MISSION CLASSES

To increase launch availability throughout the year and still maintain desirable launch and landing lighting conditions, the optimal nominal mission can be adjusted to produce a set of extended mission options.[2] These missions vary according to the total mission duration, as shown in Figure 5c. Short class missions consist of the nominal mission, nominal $+1$ and $+2$ days of extra total mission duration. Long class missions add an additional DRO stay time period, and ± 1 and ± 2 days of extra mission duration. The initial guess for these classes are generated by incrementing the EI epoch from the nominal case by the appropriate time offset (both the DRD and EI epochs are allowed to vary). In addition, the long class extended missions can ideally have up to two generally local optimal solutions, and therefore, both of these are checked and the mission with the lowest total Orion Δv is selected. This methodology of using DRO stay time to gain back landing lighting was originally seen as an option, if desired, by the Artemis Program. However, in the years of Artemis I design iterations, these extended missions are now considered a part of the nominal trajectory. Within a given launch period, the mission class will only be allowed to switch once to limit operational complexity.

EARTH ENTRY INTERFACE

The terminal mission constraint for the optimization problem is the EI target line, an analytical curve that approximates the allowable landing zone for a skip entry off the coast of San Diego.[19,20] The target line has both horizontal and vertical components, and defines constraints on longitude, azimuth, and flight path angle as functions of geodetic latitude and velocity. The horizontal target line is shown in Figure 7. The EI target line constraints are implemented as a Copernicus plugin.

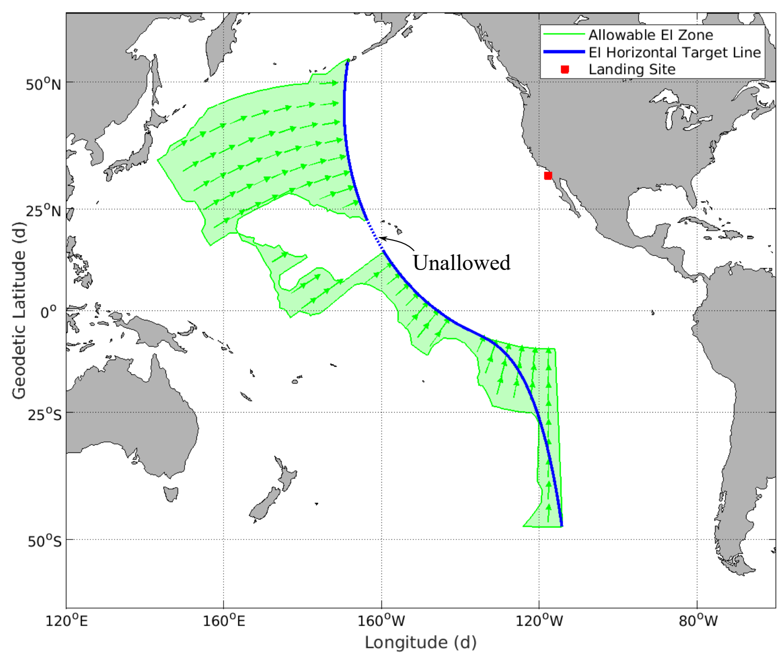


Figure 7: EI Target Line with Allowable Zone. The blue target line approximates the green region in the optimization problem. The unallowed part of the curve is excluded due to the SM disposal requirements near the Hawaiian islands.

ADDITIONAL CONSTRAINTS AND MITIGATIONS

Some desired constraints are not included in the nominal mission optimization because they are discontinuous, or otherwise make convergence difficult. For these constraints, the strategy is to exclude them from the nominal mission scan (see Figures 9a-9c) but then to check them after the fact. If none of the constraints are violated, then the mission is accepted. If any are violated, a mitigation process is introduced. The constraints of this type are:

- Maximum and minimum bounds for distance traveled between EI and splashdown site
- Sunrise and sunset time constraints for landing
- Maximum eclipse durations throughout the trajectory
- Non-viable latitude locations for EI

Specific criteria are established to define the bounds for violations in each mission. The following subsections provide the current limits for these bounds and a brief description of how these mitigation strategies are implemented.

EI Downrange

The EI downrange limit, or entry range, is defined as the distance traveled by the Orion spacecraft during descent from EI to the landing site. The entry downrange constraint[21] is defined to be:

$$2498 \text{ nmi} \leq |\vec{R}_{targ}|\theta_{range} \leq 4000 \text{ nmi} \tag{4}$$

where $|\vec{R}_{targ}|$ is the landing site position vector magnitude, and θ_{range} is the range angle between EI and the landing site in radians. Note that this constraint is always activated when a mission is examined for mitigation. This safeguard prevents any other mitigation algorithm from causing this relatively simple constraint to be violated.

Landing Lighting

Landing lighting pertains to the time to and from sunset and sunrise of splashdown, respectively. The duration of Orion's descent from EI to the landing site is estimated to be 20 minutes. With this duration, the EI epoch, the latitude/longitude of the landing site, and the time from sunrise and time to sunset can be calculated using a Copernicus lighting plugin. The default buffers required for these times are 1.0 hour after sunrise and 0.5 hour prior to sunset. Depending on whether the violation is closer to sunrise or sunset, either constraint will be activated. This restriction is necessary to allow enough sunlight to perform recovery operations of the vehicle post-splashdown. The landing lighting constraint would reduce mission availability by almost 50% if the long class mission durations were not employed.

Eclipse Mitigation Algorithm

The Orion spacecraft is limited to eclipse durations of no longer than 90 minutes.[22,23] This restriction was originally put in place due to the susceptibility of the solar cells to debond from Orion's solar arrays when in the extremely cold environment of an eclipse. This restriction is currently in place also due to power and battery capacity concerns. The eclipse constraint, on its own, reduces mission availability by a minimum of 18%.[23,24] To attempt resolving these issues and bring these missions back into the fold, a mitigation algorithm was developed to reshape the trajectory and bring violating eclipses under the duration threshold.

For the eclipse mitigation analysis, the trajectory is divided into six regions. These regions will often identify what type of trajectory shaping strategy is necessary or indicate whether mitigation is even required. The regions are color coded in Figure 4 and also described in Table 1.

Table 1: Eclipse Mitigation Regions

Region Number	Label	Boundary	
1	Pre-OTC-1	Core Sep	$\rightarrow$ OTC-1
2	Pre-OPF	OTC-1	$\rightarrow$ OPF
3	Post-OPF (outbound)	OPF	$\rightarrow$ DRI
4	DRO	DRI	$\rightarrow$ DRD
5	Pre-RPF	DRD	$\rightarrow$ RPF
6	Post-RPF (return)	RPF	$\rightarrow$ EI

The algorithm must also account for the XDM mission phases, where Regions 3 (outbound) and 6 (return) include the parallel trajectories that deviate from the nominal trajectory. Consequently, each path requires an individual eclipse analysis. Using the trajectory from each region, if an eclipse is found, the data provided to the eclipse mitigation algorithm includes the eclipse start time, duration, and eclipse type (Earth, Moon, or multi-body). Using this information, along with a Copernicus plugin that dynamically calculates the percentage of sunlight experienced by Orion, any violations of the eclipse constraint throughout the mission are located, and the proper eclipse mitigation method to apply is determined. There are five possible trajectory shaping methods that can be utilized:

1. Inclining the DRO
   (a) Optimizes v_z of DRO state and leverages Δv from OTC-1 and OPF burns to incline the entire DRO
   (b) Most common method of eclipse mitigation
2. Activating the DRO Plane Change (DPC)
   (a) Activates DPC burn three days after DRI to incline from the Earth-Moon plane (see Figure 5c)
3. Combining Inclination Methods
   (a) Employs methods 1 and 2 to apply two layers of inclination to the DRO
4. Switching OTC-1 Burn Direction
   (a) Reverses the burn direction of OTC-1 from inward to outward (toward or away from the Earth-Moon line respectively) by modifying the α and β burn direction angle values
5. Utilizing DRD and RPF Burns
   (a) Optimizes DRD and RPF burn durations

The eclipse mitigation algorithm will choose the appropriate strategy from this list and apply the necessary mitigation to the trajectory. The trajectory is then reoptimized to obtain a minimum Δv solution where the mitigated eclipse duration is equal to the allowable limit. When the solution has converged, the trajectory will then be checked to see if another eclipse violation exists in the mission. If so, a separate mitigation will be performed for each eclipse violation, one at a time, in mission chronological order using this process until all violations are mitigated.

EI Latitude Mitigation

It can be seen in Figure 7 that there is a section of the EI target line between 14° and 21.9° latitude, in which the manifold does not have an acceptable solution. This is due to the SM disposal requirements near the Hawaiian islands. Thus there is a discontinuity in the EI constraint. This is accounted for by splitting the latitude violation boundary into thirds. If the EI state is on the boundary of one of these regions, a latitude constraint is activated to constrain it to the upper or lower regions of the target line. If the EI state is in the unallowable region, two separate optimizations are performed (one with the EI latitude constrained to the upper target line region and the other being constrained to the lower target line region). The converged solution with the lowest Δv is then selected.

LAUNCH WINDOW

To generate launch window trajectories, the "optimal point" is found first, which is the optimal launch azimuth point per day that corresponds to the minimum ICPS plus Orion total Δv. An appropriate mission class and length is selected based on Orion constraints and any additional constraints are added in order to mitigate violations. The final optimized trajectory is used as the seed to compute the launch window. The optimization problem is then altered in the following ways:

- The launch epoch is fixed.
- The secondary payload outbound transit time with a minimum bound of 5.15 day from launch to OPF is removed.
- The OPF epoch t_{OPF} is fixed to ensure a variable launch azimuth mission design approach is achieved. This helps to ensure more consistent outbound transit times.
- OTC-2 is enabled as an optimized burn
- The DRO insertion state and time (t_{DRI}) is locked down. Operationally this helps the mission planning for the FOD team to ensure the same DRO is achieved during the entire window.

Each day consists of a 1-minute launch window sweep of up to $\pm$ 4 hours before and after the optimal point (see Figure 9c). Note that, since the launch window scan is done using the TLI database (beginning at the Orion Sep point), only the Orion Δv is being minimized in the objective function. Since the epochs that are infeasible for SLS/ICPS have already been excluded from the database, only Orion needs to be considered. The launch scan continues (forward and backwards) until any of the following conditions are violated:

- The problem does not converge (i.e., the mission is infeasible)
- Orion exhausts its available propellant for major burns
- The launch to OPF duration is < 4 days

An example launch window in shown in Figure 8.

SCAN METHODOLOGY

A continuation method is used to perform epoch scans of the basic mission, mission classes, and computation of the launch window. This method is implemented in the JSC tool DAMOCLES,[2] which is a wrapper to Copernicus for performing large scans in parallel in an HPC environment. There are three kinds of scans: an epoch scan (see Figure 9a), a mission class scan (see Figure 9b), and a launch window scan (see Figure 9c).

During the scan, the lunar geometry is used in various ways to update the initial guess to provide robust convergence. It was mentioned above how parameterizing states and burns in the Earth-Moon rotating frame helps with this. Additionally, for epoch steps (see Figure 9a), rather than using integer days, a better time step is computed for each epoch using a simple lunar ephemeris[25] so that the Earth-Moon geometry is similar. The average size of this step is about 24.84 hours. Another

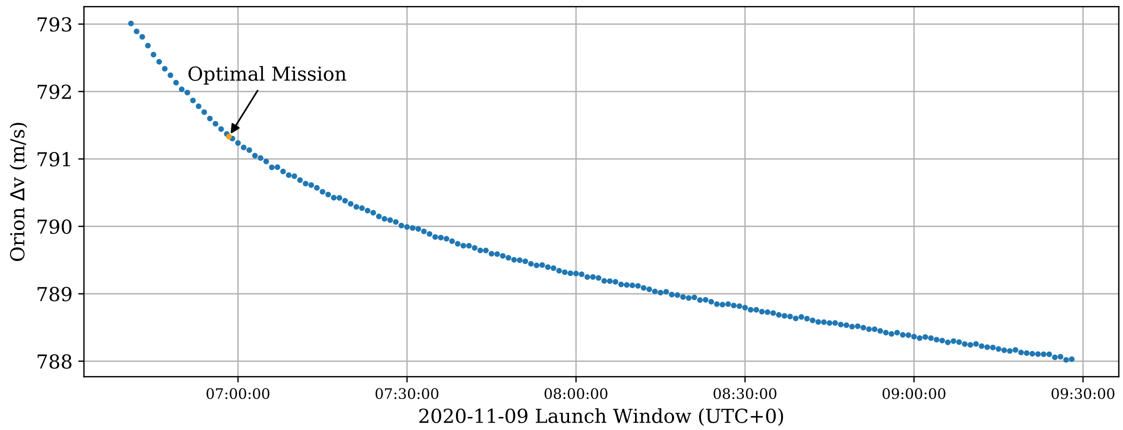


Figure 8: Example Launch Window Scan. The launch window scan proceeds in one minute steps backwards and forwards from the optimal mission to the start and end of the window.

strategy is to update the initial guess for the DRO at each epoch using the following empirically-determined approximation:

$$2T_p = 26.426 \text{ days} \tag{5}$$

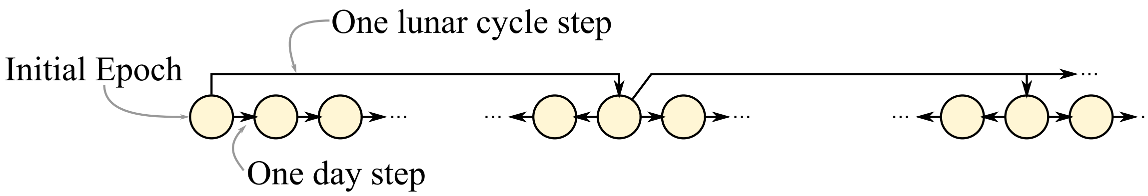
$$v_y = 1.17228407314 \times 10^{-6} r - 0.952752324405 \text{ km/s} \tag{6}$$

where $2T_p$ is the period of two revolutions in the DRO, v_y is the v velocity of the DRO initial state in the rotating frame (see Figure 3), and r is the Earth-Moon distance at the selected epoch.

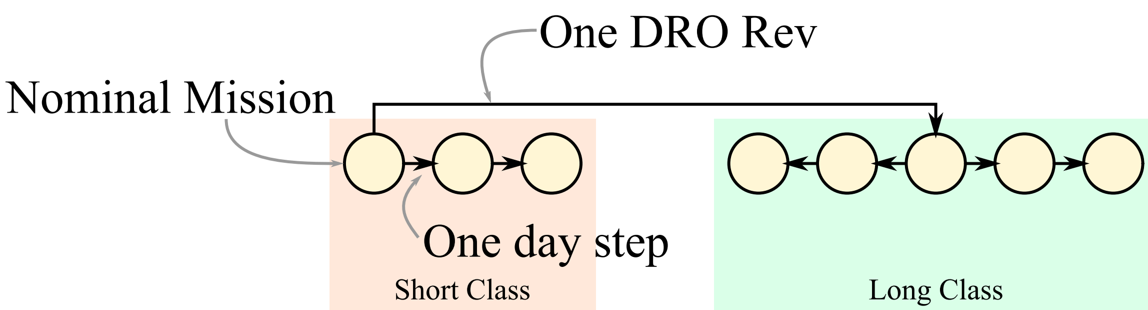
OFF NOMINAL MISSIONS

For every nominal mission trajectory, the Artemis I mission design team will create a set of off-nominal missions (each of which requires the solution of a new optimization problem spawned off from the nominal mission). These off-nominal missions provide pre-built solutions to conceivable single-point failures. They are designed such that the mission accomplishes objectives in order of priority. The off-nominal trajectories can be broken into three categories depending on the goal of the off-nominal trajectory: alternate missions, returns to nominal trajectory, and aborts:

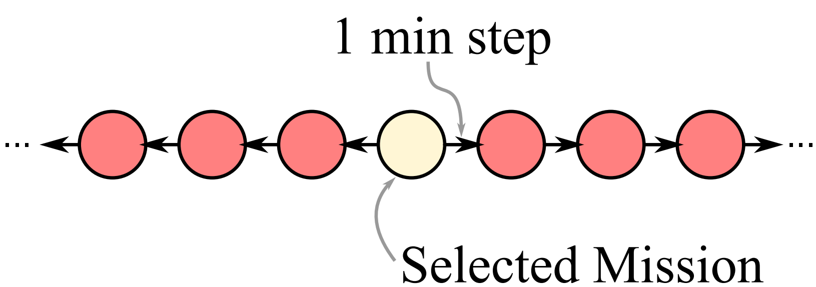
- An alternate mission is a response to a failure that occurs prior to the Orion-ICPS separation. In this scenario, the baseline mission is infeasible and is replaced with a scenario within vehicle constraints. For example, a partial TLI burn may force Orion to target a highly elliptical Earth orbit instead of a lunar flyby into the DRO. This alternate mission will maximize the Orion re-entry speed, but may not meet other objectives, such as testing optical-navigation capability.
- A return to the nominal trajectory is a response to a relatively small failure, such as a missed or partial Orion burn. In this scenario, a burn did not perform as expected due to a failure or operational constraint. A recovery burn is used to return Orion to the baseline Artemis I trajectory and ensures all intended mission objectives are met. For example, if the DRD burn is missed, a recovery burn will be performed later to target the RPF burn.
- An abort trajectory is a response to a significant failure. It is used in the event the vehicle cannot complete the nominal mission and must abort to ensure vehicle recovery. For example,



(a) Nominal Mission Scan Logic. An epoch sweep is completed from an initial epoch to a final epoch by stepping (in parallel) by about one day time increments (due to the Earth-Moon geometry) and by one lunar cycle time increments, until the full epoch range is completed from beginning to end.



(b) Short and Long Class Missions. For each day, starting with the nominal short class mission, the others are then generated by adding time to the total mission duration. Eight possible missions are generated per day.



(c) Launch Window Scan. For the selected mission class on a given day, the launch window is then generated in one minute steps (truncated to the nearest integer UTC minute). The scan proceeds forwards and backwards in time until the entire window is generated.

Figure 9: Scan Methodology. A continuation method is used for all scans. The arrows indicate that one case is used as the initial guess for the next case. Parallelization is an important component of the scan methodology that ensures results can be generated in a reasonable amount of time. The final results from the scan are the launch window solutions (as well as any off nominal missions spawned from them).

in the event of a propellant leak, an abort may be performed to take advantage of the current propellant reserves by placing the vehicle on a trajectory to a viable EI.

Each of these off-nominal cases can be further divided into sub-cases with specific parameters. For example, an abort can be a direct return abort or include a lunar flyby. The abort TIG can occur at a variety of times, with different engine types, and with a variable number of subsequent abort burns. Due to the large amount of cases, sub-cases and parameters, a Mission Design Matrix (MDM) was formulated to track the different trajectories that need to be created, verified, and delivered. An estimated 10,000 off-nominal trajectories will be generated for each nominal launch day.

INTERACTION WITH FLIGHT OPERATIONS

The design for the Artemis I trajectory is an iterative process with the JSC Flight Operations Directorate (FOD), specifically with the Flight Dynamics Officers (FDOs). For the success of the operational trajectory, the FDOs and Orion engineering trajectory design team work closely to alter the design in order to meet operational requirements. In addition, various trajectory products, including optimized Copernicus mission files and plugins, ephemeris files, and summary report data are transferred to the FDOs to support flight rule development and pre-flight and real-time operational needs. Successful convergence of the Copernicus trajectory is critical, since FDO tools rely on Copernicus mission files to create an initial trajectory guess that is used for vehicle burn plan generation and as an input to the FOD ground targeting software, which ultimately generates the data loaded directly onto the vehicle.

In addition to the Copernicus trajectory, the data is also post-processed to generate summary reports to provide the FOD with crucial information needed for real time operations. A report for each launch time contains a timeline of mission events (e.g., burns, ground sight loss of signal, and separations), a summary of eclipse times and durations, and Orion's view of ground stations and Tracking and Data Relay Satellite System (TDRSS). They also contain various plots of trajectory parameters such as velocity and altitude, as well as relative motion plots of Orion and the ICPS after separation to check for possible recontact. For each launch day in a scan, a summary report is generated with the aggregate eclipse, vehicle performance, Deep Space Network (DSN) coverage, and lighting with respect to important mission events. Another report summarizing the eclipses is generated for an entire launch window. All reports are automatically generated at the end of the scan process.

CONCLUSION

Trajectory design is a complicated merging of vehicle capabilities, programmatic constraints and desirements, risk trades, mission objectives, and physics. The creation of a flight-ready reference trajectory relied heavily on the integrated efforts of the SLS, Orion, EGS programs, ESD, and FOD. The Artemis I mission design has evolved over the years, but even when the performance based reference trajectory for the optimal launch per day was solidified, many permutations needed to be analyzed and adjustments made to the trajectory. Maximizing mission availability, which is a key asset to have when launching a system for the first time, drove these permutations. The alterations are also necessary to create a large enough pre-flight product suite for operations to provide initial guesses for vehicle targets and to account for off-nominal situations. This impelled the JSC team to develop novel trajectory design techniques and the tools Copernicus and DAMOCLES to emulate what whole organizations did in past human spaceflight programs. Even though Artemis I is uncrewed, the trajectory design process that has been created will be used as the groundwork for all future human spaceflight missions.

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NOTATION

ARCM	Asteroid Redirect Crewed Mission
AUX	Auxiliary
CM	Crew Module
CMA	Crew Module Adapter
DAG	Directed Acyclic Graph
DPC	DRO Plane Change
DRD	DRO Departure
DRI	DRO Insertion
DRO	Distant Retrograde Orbit
DSN	Deep Space Network
EGS	Exploration Ground Systems
EI	Entry Interface
ESD	Exploration Systems Development
ESM	European Service Module
FDO	Flight Dynamics Officer
FOD	Flight Operations Directorate
GRACE	Gravity Recovery And Climate Experiment
GRAIL	Gravity Recovery and Interior Laboratory
HLS	Human Landing System
HPC	High Performance Computing
ICPS	Interim Cryogenic Propulsion Stage
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
LAS	Launch Abort System
LEO	Low Earth Orbit
MDM	Mission Design Matrix

MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
OCO	OMSe Checkout
OMSe	Orbital Maneuvering System Engine
OPF	Outbound Powered Flyby
OTC	Outbound Trajectory Correction
POST	Program to Optimize Simulated Trajectories
PRM	Perigee Raise Maneuver
RCS	Reaction Control System
RPF	Return Powered Flyby
RTC	Return Trajectory Correction
SA	Spacecraft Adapter
SLS	Space Launch System
SM	Service Module
SNOPT	Sparse Nonlinear OPTimizer
SOFA	Standards of Fundamental Astronomy
SPICE	Spacecraft, Planet, Instruments, C-matrix and Events
TDRSS	Tracking and Data Relay Satellite System
TIG	Time of Ignition
TIP	Target Interface Point
TLI	Trans Lunar Injection
TPS	Thermal Protection System
USS	Upper Stage Separation
UTC	Coordinated Universal Time
XDM	Auxiliary Down Mode

REFERENCES

[1] J. P. Gutkowski, T. F. Dawn, and R. M. Jedrey, "Evolution of Orion Mission Design for Exploration Mission 1 and 2," AAS Guidance, Navigation and Control Conference, Feb. 2016. AAS 16-111.

[2] T. F. Dawn, J. P. Gutkowski, A. L. Batcha, J. Williams, and S. M. Pedrotty, "Trajectory Design Considerations for Exploration Mission 1," AIAA/AAS Space Flight Mechanics Meeting, Jan. 2017. AIAA 2018-0968.

[3] K. Hambleton, "Around the Moon with NASA's First Launch of SLS with Orion," Feb. 2020. https://www.nasa.gov/feature/around-the-moon-with-nasa-s-first-launch-of-sls-with-orion.

[4] N. F. Merancy, M. Sarafin, and J. Gruber, "Human Lunar Mission Design: Then and Now," 70th International Astronautical Congress (IAC), Oct. 2019. IAC-19-B3.1.9.

[5] J. Williams, A. H. Kamath, R. A. Eckman, G. L. Condon, R. Mathur, and D. C. Davis, "Copernicus 5.0: Latest Advances in JSC's Spacecraft Trajectory Optimization and Design System," AAS/AIAA Astrodynamics Specialist Conference, Aug. 2019. AAS 19-719.

[6] P. E. Gill, W. Murray, and M. A. Saunders, "SNOPT: An SQP Algorithm For Large-Scale Constrained Optimization," *SIAM Journal on Optimization*, Vol. 12, 1997, pp. 979–1006.

[7] J. Williams, "A New Architecture for Extending the Capabilities of the Copernicus Trajectory Optimization Program," *Advances in the Astronautical Sciences: Astrodynamics 2015*, Vol. 156, 2016. AAS 15-606.

[8] L. F. Shampine and H. A. Watts, "DEPAC – Design of a User Oriented Package of ODE Solvers," Tech. Rep. SAND-79-2374, Sandia National Labs, Sept. 1980.

[9] W. M. Folkner, J. G. Williams, and D. H. Boggs, "The Planetary and Lunar Ephemeris: DE 421," Memorandum IOM 343R-08-003, Jet Propulsion Laboratory, California Institute of Technology, March 2008.

[10] B. Tapley, J. Ries, S. Bettadpur, D. Chambers, M. Cheng, F. Condi, B. Gunter, Z. Kang, P. Nagel, R. Pastor, T. Pekker, S. Poole, and F. Wang, "GGM02: An Improved Earth Gravity Field Model from GRACE," *Journal of Geodesy*, Vol. 79, 2005.

[11] F. G. Lemoine, S. Goossens, T. J. Sabaka, J. B. Nicholas, E. Mazarico, D. D. Rowlands, B. D. Loomis, D. S. Chinn, D. S. Caprette, G. A. Neumann, D. E. Smith, and M. T. Zuber, "High-Degree Gravity Models from GRAIL Primary Mission Data," *Journal of Geophysical Research: Planets*, Vol. 118, No. 8, 2013, pp. 1676–1698.

[12] IAU SOFA Board, "IAU SOFA Software Collection," http://www.iausofa.org.

[13] J. Williams and G. L. Condon, "Contingency Trajectory Planning for the Asteroid Redirect Crewed Mission," SpaceOps 2014 Conference, May 2014. AIAA 2014-1697.

[14] J. Williams and A. Batcha, "EMPT Artemis 1 FRAC0 Delivery Memo," Flight Dynamics Team Technical Brief, NASA Johnson Space Center, Oct. 2019. EG-TARGO-19-05.

[15] R. A. Lugo, J. D. Shidner, R. W. Powell, S. M. Marsh, J. A. Hoffman, D. K. Litton, and T. L. Schmitt, "Launch Vehicle Ascent Trajectory Simulation Using the Program to Optimize Simulated Trajectories II (POST2)," *AAS/AIAA Space Flight Mechanics Meeting*, Feb. 2017. AAS 17-274.

[16] R. Whitley, J. Gutkowski, S. Craig, T. Dawn, J. Williams, C. Ocampo, D. L. William B. Stein, R. Lugo, and M. Qu, "Combining Simulation Tools for End-to-End Trajectory Optimization," AAS/AIAA Astrodynamics Specialist Conference, Aug. 2015. AAS 15-662.

[17] C. Ocampo, "Finite Burn Maneuver Modeling for a Generalized Spacecraft Trajectory Design and Optimization System," *Annals of the New York Academy of Science*, Vol. 1017, May 2004, pp. 210–233.

[18] W. H. Tracy, "An Assessment of Thruster Downmode for the Lunar Flyby Maneuvers for EM-1," Internal NASA JSC Memorandum, Apr. 2014.

[19] J. Rea, "Exploration Mission Entry Interface Target Line," Tech. Rep. FltDyn-CEV-15-025, rev. EM1.1.0, Aeroscience and Flight Mechanics Division (EG5), NASA Johnson Space Center, December 2017.

[20] J. Rea, "Orion Exploration Mission Entry Interface Target Line," AIAA/AAS Space Flight Mechanics Meeting, Feb. 2016. AAS 16-485.

[21] J. Rea, "Parameters for Entry Interface Dispersion Requirement Definition," Tech. Rep. FltDyn-CEV-14-33, rev. 5, Aeroscience and Flight Mechanics Division (EG5), NASA Johnson Space Center, June 2018.

[22] N. Merancy, J. Dithmer, J. Hojnicki, D. Dannemiller, K. Pohlkamp, D. Lantz, K. Jennings, and D. Crook, "ESM Cold Case Non-compliance Impacts," National Aeronautics and Space Administration, February 2016. Presented at the March, 2016 Orion Engineering Review Board, Houston, TX.

[23] S. Smallwood and D. Lee, "Artemis I Eclipse Mitigation Strategies," Tech. Rep. EG-TARGO-19-8, Aeroscience and Flight Mechanics Division (EG5), NASA Johnson Space Center, February 2020.

[24] "Cross Program End-to-End Performance Ground Rules and Assumptions for EM-1," Tech. Rep. ESD 10042, National Aeronautics and Space Administration, July 2018.

[25] D. G. Simpson, "An Alternative Lunar Ephemeris Model for On-Board Flight Software Use," *Proceedings of the 1999 NASA/GSFC Flight Mechanics Symposium*, May 1999, pp. 175–184.