Trajectory Design Considerations for Exploration Mission 1

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This study examines and presents the evolving trade studies related to the Orion Exploration Mission 1, which will be the first mission to send an uncrewed Orion vehicle to cislunar space in the fall of 2019. Details of a new scan tool are also provided and how it is used to perform the scans and post-process the results.

I. Introduction

The Orion Multi-Purpose Crew Vehicle (MPCV) Exploration Mission 1 (EM-1) was originally conceived as a lunar free-return mission, but was later changed to a Distant Retrograde Orbit (DRO) mission as a precursor to the Asteroid Redirect Mission (ARM). Whether or not ARM is eventually performed, the DRO provides an excellent opportunity to test out Orion's systems in a unique cislunar orbit. This aligns with the goal of the National Aeronautics and Space Administration (NASA) to explore the "Proving Ground" of space to prepare for missions beyond the Earth and Moon.

How the DRO is defined is important when performing repeatable analysis between many teams throughout NASA and our contractor partners. This paper describes how the DRO that is targeted for EM-1 is defined. Also, since DROs are fairly new to the realm of human spaceflight mission planning, various trajectory visualizations are included. In addition, a detailed description of the EM-1 mission, its major burns, a mission walk-through, and performance metrics from the trajectory scans are provided. To understand the required mission performance and when mission opportunities exist, a series of trajectory optimization runs was conducted using Johnson Space Center's (JSC) Copernicus spacecraft trajectory optimization tool [1]. In order for the runs to be done in a timely manner, it was necessary to employ a parallelization approach on a computing cluster with the development of a new Python-based tool. Valid mission opportunities are ones that do not exceed the usable propellant available to perform the required burns. The scan data produced by the tool data provides the propellant and Δv performance patterns for each launch period.

II. Definition and Targeting of the Distant Retrograde Orbit

A DRO is a class of stable orbits in the Circular Restricted Three Body Problem (CR3BP) (they are family f in Reference [2]). In the two-body rotating frame, Distant Retrograde Orbits move in the retrograde direction around the secondary body, appearing as large quasi-elliptical orbits. The gravitational perturbations are significant from both the primary and secondary bodies (the Earth and the Moon in this case). The term "Distant Retrograde Orbit (DRO)" was apparently first coined in Reference [3], although this class of orbits was studied and systematically classified earlier (see References [2, 4], among others). In a realistic force model, the orbits are quasi-periodic and the X-axis crossing distance oscillates over time, but still remain very stable over long periods. It is for this very reason that this type of orbit was chosen for the ARM concept back in 2013. The plan for ARM was to capture a near Earth asteroid and redirect it to a lunar DRO. The asteroid would stay in orbit for a long period of time (possibly hundreds of years) without orbit maintenance, so that humans could go visit and research it [5, 6]. The plan for ARM naturally flowed as the new lunar destination orbit for EM-1 as a precursor mission type.

The EM-1 DRO is modeled in a two-body rotating-pulsating frame [7]. This means that the X-axis is pointed along the Earth-Moon line, the Z-axis is pointed along the Moon's orbit angular momentum vector, and the Earth-Moon distance is held fixed (for this analysis, the distance is the Moon's orbit semimajor axis of 384,400 km). The center of this frame is placed at the Earth-Moon barycenter. This frame was chosen for its consistent behavior in determining

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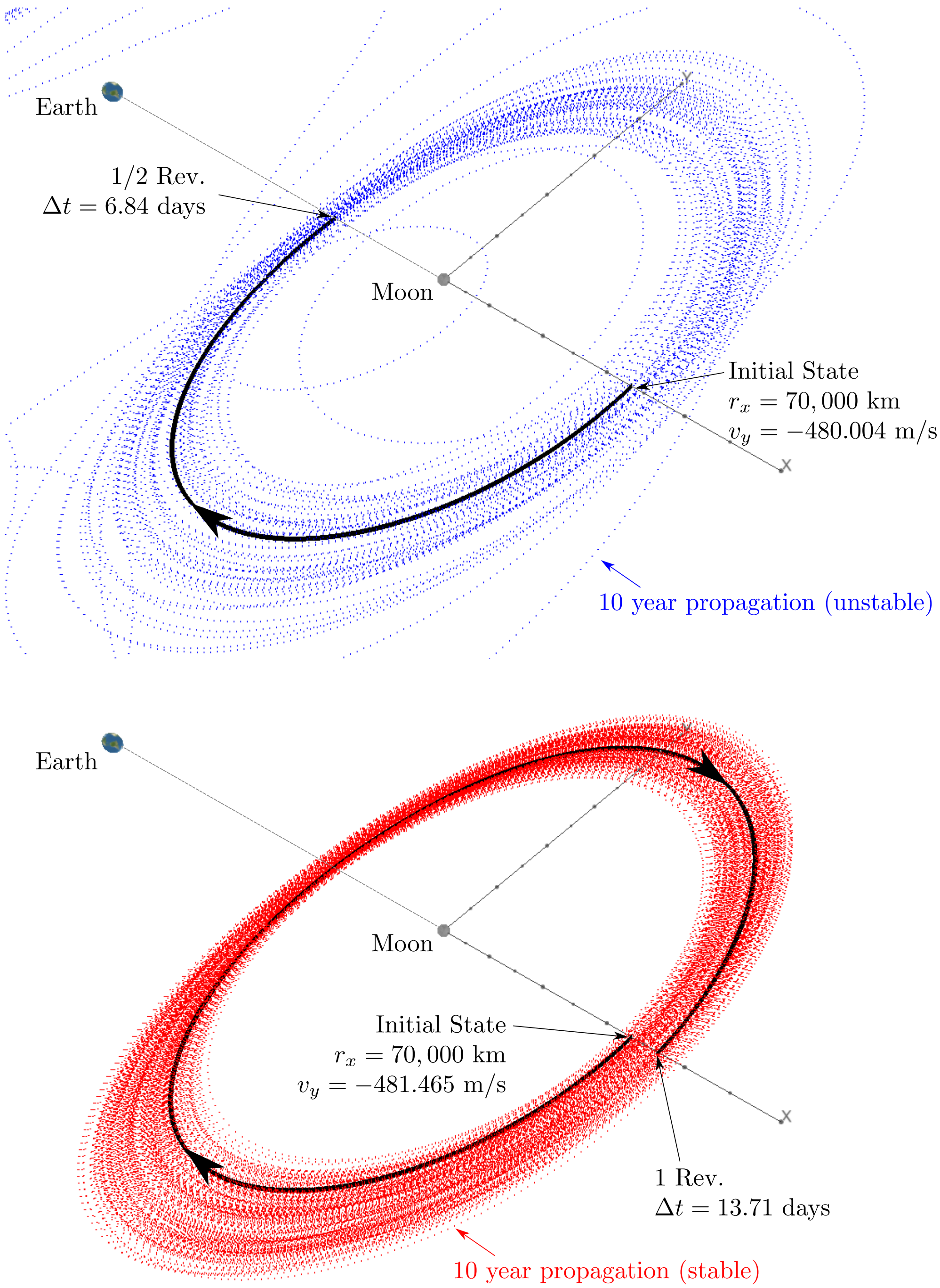

Fig. 1 DRO Targeting in a Realistic Force Model (Earth-Moon Rotating-Pulsating Frame). DROs are computed by targeting perpendicular X-axis crossings in the rotating frame. In the idealized CR3BP, one subsequent X-axis crossing is sufficient to produce a simple periodic orbit. However, when using a realistic ephemeris and force model, this is not sufficient (see top image). The bottom image shows that targeting the second X-crossing (one DRO orbit period) is sufficient to produce a 70,000 km quasi-periodic DRO that is stable for at least ten years.

a set of initial conditions to form the DRO. For EM-1, the DRO is coplanar to the Earth-Moon plane. The size of a DROs can be classified by its +X-axis crossing distance in the rotating frame (the distance measured from the center of the Moon, radially along the Earth-Moon line away from the Earth to the point where the DRO crosses the X-axis traveling in the -Y direction), which is analogous to the semi-minor axis of the quasi-ellipse. Targeting a DRO can be done in Copernicus using a forward-shooting 2D root-finding approach with a single trajectory segment (refer to Figure 1). Figure 2 shows how the DRO is defined for EM-1. The DRO is initialized at 70,000 km radius from the center of the Moon, along the +X-axis (i.e., r_x = 70,000, r_y = 0, and r_z = 0). The retrograde velocity is in the -Y direction (i.e., $v_x = 0$, $v_y < 0$, and $v_z = 0$). The initial velocity is determined by varying the value until two full revolutions is achieved. The velocity magnitude is roughly 500 m/s. The end point of the targeting problem is defined to be the point at X-Z plane crossing with no radial velocity (i.e., r_y = 0 and v_x = 0). Two periods of a DRO of this size is roughly 27 days long, so that Orion can orbit twice around the Moon in one sidereal lunar month.

III. EM-1 Mission Overview

This section provides an overview of the general Design Reference Mission (DRM) and a specific Conceptual Flight Profile (CFP) for EM-1. The term DRM is used when referring to a general class of missions. The term CFP is used when referencing a specific example of how a DRM could be performed. The EM-1 DRM involves the first launch of the Space Launch System (SLS) Block 1 rocket and the first in-space test of Orion's Service Module (SM). Orion will be uncrewed so that these new systems can be tested prior to the first crewed mission: EM-2. For additional information on the evolution of the EM-1 mission and subsequent Exploration Missions see Reference [8].

The SLS launch places its upper stage, the Interim Cryogenic Propulsion Stage (ICPS), and Orion into an insertion orbit of 22 × 975 nmi (41 × 1806 km). The ICPS performs a burn to raise the perigee to an orbit of 100 x 975 nmi (185 x 1806 km). ICPS performs the Trans-Lunar Injection (TLI) burn to target Orion's Outbound Powered Flyby (OPF) burn as it flies by the Moon. Orion performs the Distant Retrograde orbit Insertion (DRI) burn. DRI inserts Orion into a DRO that has a radius from the center of the Moon of 37,797 nmi (70,000 km). After DRO operations, the vehicle performs the Distant Retrograde orbit Departure (DRD) burn to target the lunar Return Powered Flyby (RPF) burn and 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), as well as relevant environments prior to the first crewed launch of the system.

IV. Burn Nomenclature

Through the nominal EM-1 mission there are two major burns performed by the main engine of the ICPS and four (optionally five) major burns performed by the Orion main engine, the Orbital Maneuvering System-Engine (OMS-E). The remaining Orion burns are performed using its eight Auxiliary Thrusters (Aux). A visual mission overview is shown in Fig. 3. A chronological list of the EM-1 mission burns is as follows:

- Perigee Raise Maneuver (PRM); performed by ICPS
- Trans-Lunar Injection (TLI); performed by ICPS
- Outbound Trajectory Correction (OTC-1, 2, 3, 4); performed by Orion Aux
  - Optionally an OMS-E Check-Out (OCO) burn may be performed to test out the performance of the OMS-E propulsion system as soon as reasonably possible after Orion separates from the ICPS.
- Outbound Powered Flyby (OPF); performed by Orion OMS-E
- Outbound Trajectory Correction (OTC-5, . . .); performed by Orion Aux
- Distant Retrograde orbit Insertion (DRI); performed by Orion OMS-E
- Orbit Maintenance (OM-1, . . .); performed by Orion Aux
- Distant Retrograde orbit Departure (DRD); performed by Orion OMS-E
- Return Trajectory Correction (RTC-1, n); performed by Orion Aux
- Return Powered Flyby (RPF); performed by Orion OMS-E
- Return Trajectory Correction (RTC-n+1, n+2, n+3); performed by Orion Aux

V. Conceptual Flight Profile

The following is a CFP for an EM-1 mission with a specific launch date of October 9, 2018. The current EM-1 mission is planned no earlier than December of 2019, however this CFP is still representative. The EM-1 CFP mission

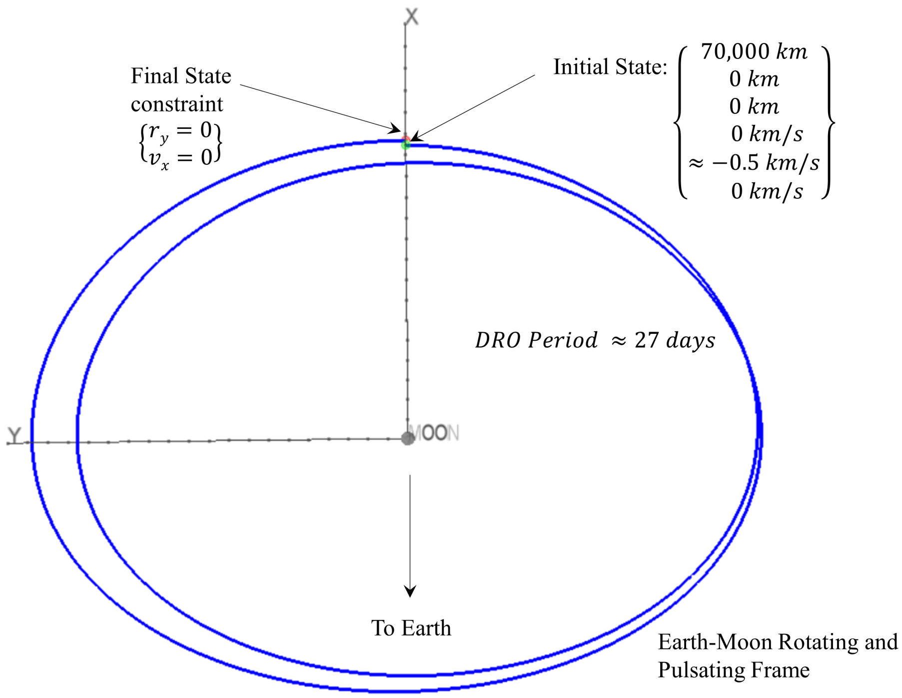

Fig. 2 EM-1 DRO Definition. When targeting a DRO, a simple forward-shooting method is used to target a state two revolutions after a perpendicular X-axis crossing. The only two control variables are the coast duration Δv and the initial y-velocity (v_y) in the rotating-pulsating frame.

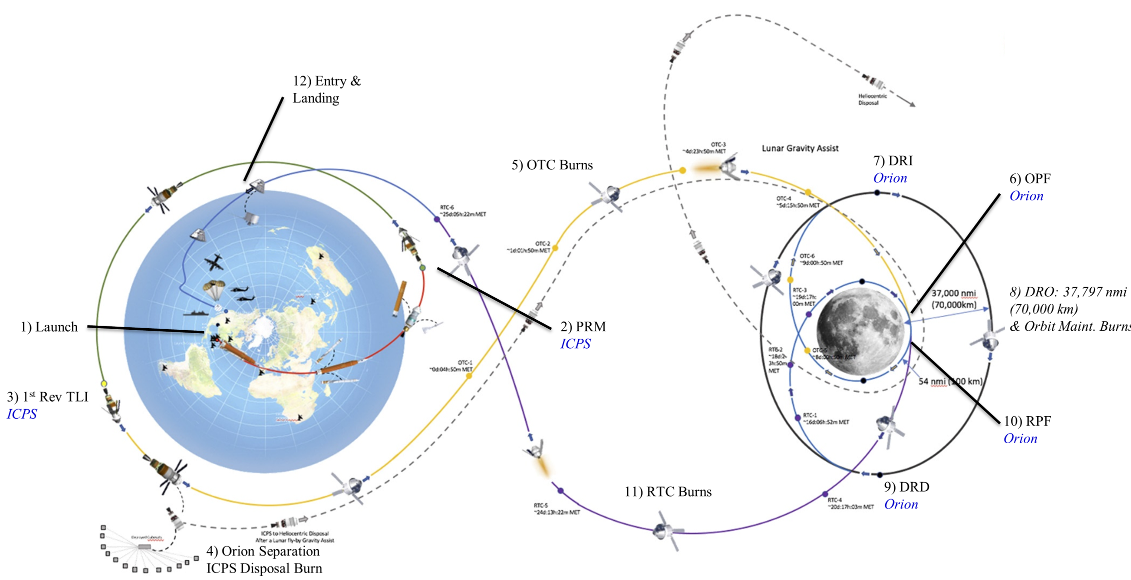

Fig. 3 EM-1 Mission Overview.

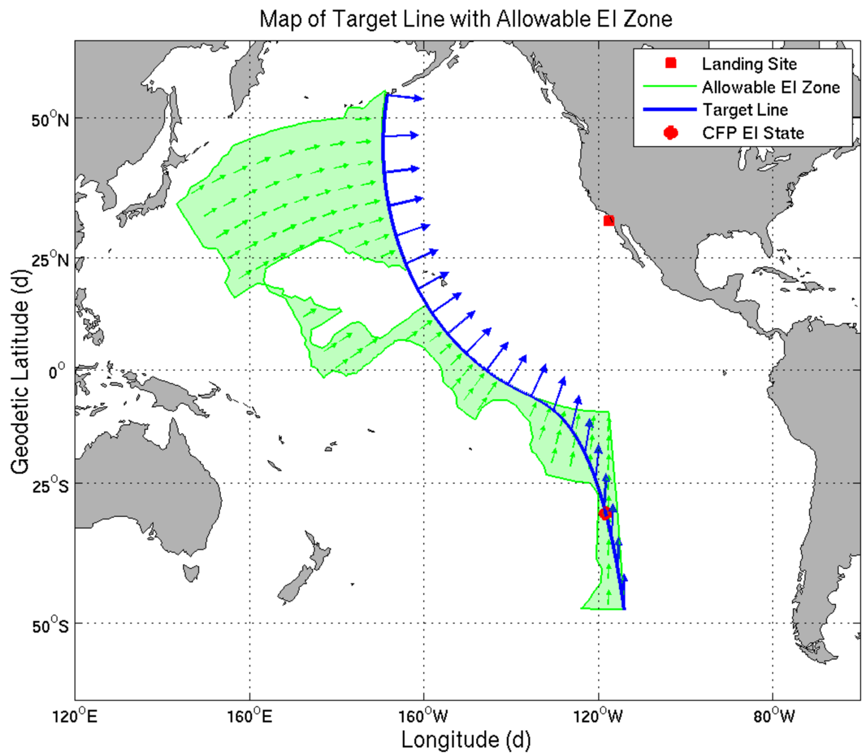

Fig. 4 Map of Target Line with Allowable EI Zone

starts at launch from Kennedy Space Center (KSC) on October 9th of 2018 at 13:02:24 TDB (13:01:15.200 UTC). This time is the opening of the launch window. The SLS inserts the ICPS and Orion into an elliptical Earth orbit with a geodetic altitude of 22 × 975 nmi (40.7 × 1806 km) and an inclination of 29.7 degrees. In addition, orbit insertion occurs at a geodetic altitude of 87 nmi (161.6 km). Orbit insertion occurs 12 seconds after Core stage Main Engine Cut-Off (MECO) and pre-SLS Core/ICPS separation. The ICPS then performs a Perigee Raise Maneuver (PRM) to increase the altitude of perigee to 100 nmi (185 km). This burn is performed at apogee.

The ICPS then performs TLI, which places Orion on a trajectory toward the flyby of the Moon and the Outbound Powered Flyby (OPF) burn. TLI targets the post-ICPS spring separation state and time. It takes about 5 days and 20 hours between TLI and the OPF burn.

ICPS/Orion spring separation occurs around ten minutes after TLI burn complete. This spring separation imparts a Delta Velocity (Δv) on both the ICPS and Orion due to the Orion's separation springs of -2.68 ft/s (0.82 m/s) on ICPS and 0.77 ft/s (0.23 m/s) on Orion. These Δvs are along the velocity vector direction. Then Orion performs a 5.6 ft/s (1.7 m/s) separation burn with the Auxiliary thrusters after the spring separation called the Aux Separation Burn (ASB). The ASB occurs 81 seconds after the spring separation to minimize Orion plume effects on ICPS.

ICPS safing, blowdown, and an eventual disposal burn using ACS thrusters starts 30 minutes after the ICPS ACS inhibit discrete, which is sent 5 seconds prior to ICPS/Orion spring separation. During the translunar coast, four Outbound Trajectory Correction (OTC) burns are reserved to cover any trajectory dispersions. In order for Orion to have an accurate navigation state post ICPS/Orion separation, OTC-1 is assumed to occur no earlier than three hours post-TLI. At the time of OTC-1 an OMS-E Check-Out (OCO) burn can optionally be performed.

The OPF burn targets the DRI burn and is performed as Orion flies by the Moon around 54 nmi (100 km). It is a Lunar Gravity Assist (LGA) powered flyby burn. The Time of Ignition (TIG) is optimally balanced for Aux downmode situation. This occurs when the OMS-E engine stops working during the burn and an automatic switch, downmode, to the backup Aux thrusters occurs. The OPF burn occurs prior to perilune and the altitude of the lunar flyby is a few kilometers below 100 km. It takes 4 days and 3 hours between the OPF and DRI burn and two OTC burns are needed to account for trajectory dispersions.

Orion performs the DRI burn to capture into the DRO. Orion then loiters in the DRO for 6 days. After this point, Orion performs the 1-burn Distant Retrograde orbit Departure (DRD) burn, which targets the Return Powered Flyby (RPF) burn on the way back to Earth. It takes about 3 days and 22 hours between the DRD and RPF. Three Return Trajectory Correction (RTC) burns are needed to account for trajectory dispersions.

RPF is performed during the flyby of the Moon around 54 nmi (100 km) altitude. It is a LGA powered flyby burn and similar to OPF, the TIG is optimally balanced for an Aux downmode situation. The RPF burn occurs after perilune. The RPF burn targets the Entry Interface (EI) target line, shown in Figure 4. The time of flight from RPF to SM EI is 5 days and 11 hours. Orion splashes down on November 3, 2018. The total mission duration is about 25 days and 10 hours. The ICPS total Δv cost is 9,359 ft/s (2,853 m/s). The total Δv cost for Orion is 2,227 ft/s (678.7 m/s).

VI. Performance Metrics

Table 1 EM-1 DRO Translational Δv Nominal Variations.

	Thrusting Δv (m/s)					
EM-1 DRO	PRM	TLI	OPF	DRI	DRD	RPF
Min	39.08	2805.18	144.35	95.77	71.46	149.60
Max	40.30	2864.32	202.30	177.79	312.03	305.71

Table 2 EM-1 DRO Translational Δv Nominal Variation Totals.

EM-1 DRO	ICPS Δv (m/s)	Orion Δv (m/s)
Min	2845.06	490.28
Max	2903.43	878.08

Multi-year trajectory scans have been performed to analyze how performance metrics vary throughout a given year. Tables 1 and 2 illustrate the variations in total Δv required for the EM-1 mission. This set of scans assume an "OMS-E optimal" Time-of-Ignition (TIG) for each of the major burns.

VII. Mission Durations

Table 3 EM-1 DRO Mission Classification.

	Class	Lower Limit (days)	Upper Limit (days)
Short	26 day	25.2	26.2
	27 day	26.2	27.2
	28 day	27.2	28.2
Long	38 day	37.2	38.2
	39 day	38.2	39.2
	40 day	39.2	40.2
	41 day	40.2	41.2
	42 day	41.2	42.2

The EM-1 mission has two classes of mission duration: short and long. The mission classification based on mission duration was derived empirically from the trajectory scan and listed in Table 3. The inclusion of the long mission duration class is necessary to provide mission opportunities throughout the year that have lighting conditions required for landing and recovery. To provide such lighting, half of the year requires short mission duration class missions and the other half long mission durations. Table 4 and Table 5 depict the time of flight between major burns. Selected missions are the subset of all the short and long mission classes that satisfy various mission requirements and desirements. For

Table 4 EM-1 DRO Mission Segment Duration Variability.

	CoreSep to PRM (mins)		PRM to TLI (hrs)		TLI to OPF (days)		OPF to DRI (days)	
	min	max	min	max	min	max	min	max
Short Class	43.90	43.90	0.49	0.70	5.11	6.38	2.90	5.27
Long Class	43.90	43.90	0.49	0.70	5.11	6.38	2.90	5.27
Selected	43.90	43.90	0.50	0.70	5.15	7.03	2.90	5.27

	DRI to DRD (days)		DRD to RPF (days)		RPF to EI (days)	
	min	max	min	max	min	max
Short Class	6.00	6.00	2.94	4.74	4.88	9.90
Long Class	13.94	22.75	1.93	11.00	3.43	11.02
Selected	2.54	18.01	2.77	11.00	4.31	9.89

Table 5 EM-1 DRO Outbound and Return Duration Variability.

	TLI to DRI (days)		DRD to EI (days)	
	min	max	min	max
Short Class	8.080	10.691	8.717	13.212
Long Class	8.080	10.691	5.365	19.169
Selected	8.080	12.123	7.215	17.089

example, the selected mission criteria could be for each mission opportunity determine which mission duration provides the minimum Orion propellant usage, avoids periods of longer than 90 minutes for Orion eclipses, and satisfies any optical navigation camera imaging requirements.

VIII. Earth-Moon Geometry

The scope of the mission duration was extended to include longer mission classes, referenced in Table 3, as a mitigation to ensure adequate lighting at landing. The Earth-Moon geometry of the nominal 26-day mission has seasonal launch and landing lighting effects. These trends are displayed in Figure 5. Each pair of points represents a valid mission that does not violate ascent propellant margin. The red represents the launch epoch and the blue is the corresponding landing epoch. The dashed lines show sunrise, and the solid lines show sunset in solar time. The dotted line defines 3 hours prior to sunset which is the preferred landing time for vehicle recovery operations. The chart demonstrates that the launch lighting requirement is met from April until September (roughly summer), whereas the land lighting requirement is met from October until February (roughly winter).

For EM-1 the SLS configuration being used requires an elliptical insertion orbit in order to maximize the effectiveness of the TLI performed at perigee by the ICPS. The launch site (KSC) is located in the northern hemisphere, thus the apogee of the insertion orbit is in the southern hemisphere. Since launch periods may only occur when the Moon's antipode is in the northern hemisphere, Orion can only encounter the Moon during half of its orbit. This, along with SLS performance constraints, results in a 10 – 11 day launch period per month. The resulting seasonal launch lighting occurs due to the Earth's orbit around the sun (see Figures 7 and 8).

The Earth rotates approximately 15° per hour, and the synodic month is approximately 29.53 days (i.e., the Moon's orbital rate is 12.19° per day). The Earth entry interface is located in the vicinity of the Moon's antipode at the flyby epoch (as shown in Figure 6). Increasing return time does not affect the landing antipode. Therefore, in order to gain 1 hour of lighting at landing, the return flyby needs to be performed 1.23 days earlier (15°/12.19°/day). See Figure 9 for the Sun-Earth-Moon geometry at the RPF Epoch.

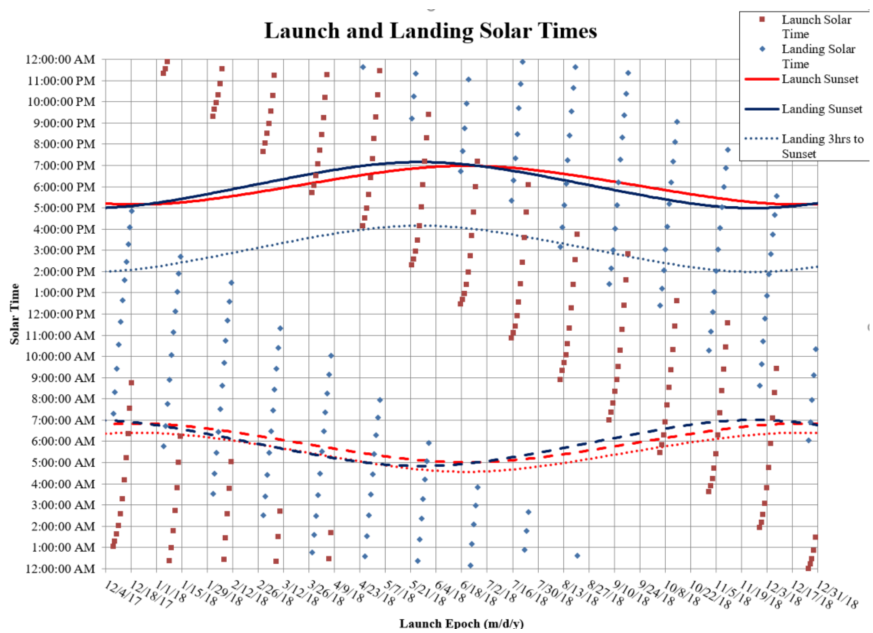

Fig. 5 Launch and Landing Solar Times for the 26-day Mission Class

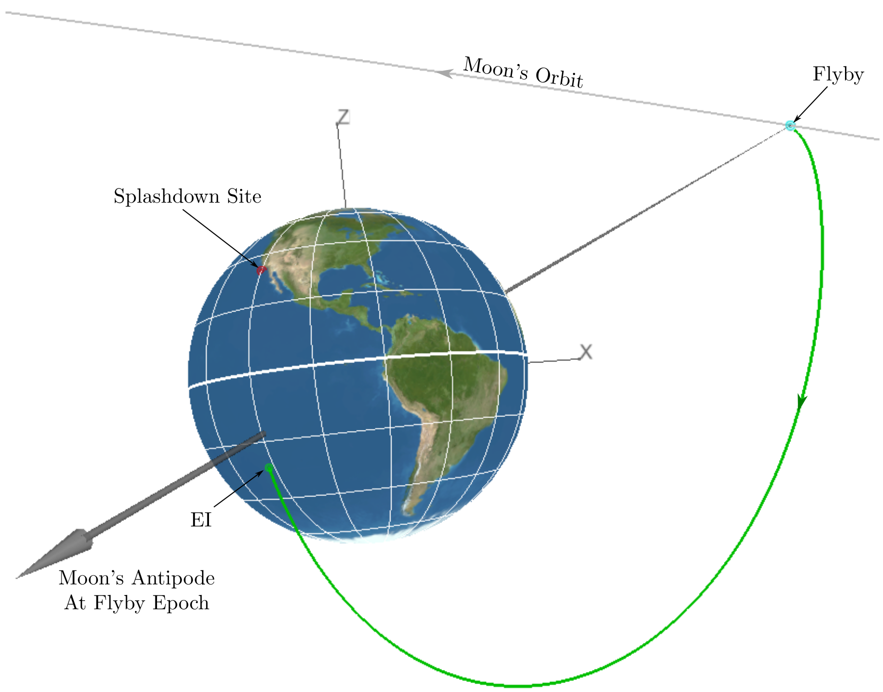

Fig. 6 Example EI Trajectory. The Orion RPF burn targets an EI state that is on the target line for a skip entry and splashdown off the coast of California.

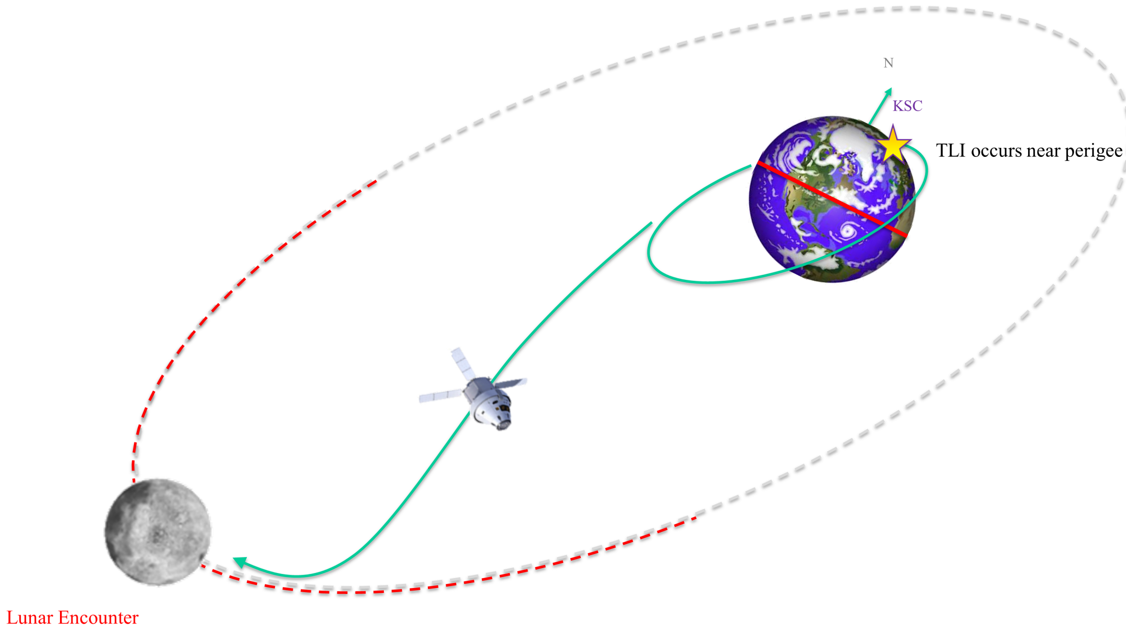

Fig. 7 Orbit Launch Restriction (Notional). Note that: (1) EM-1: SLS Block 1 and ICPS requires an elliptical orbit, (2) KSC in Northern Hemisphere sets apogee in Southern Hemisphere, and (3) Apogee fixes the Moon's Antipode, can only encounter the moon during half of its orbit.

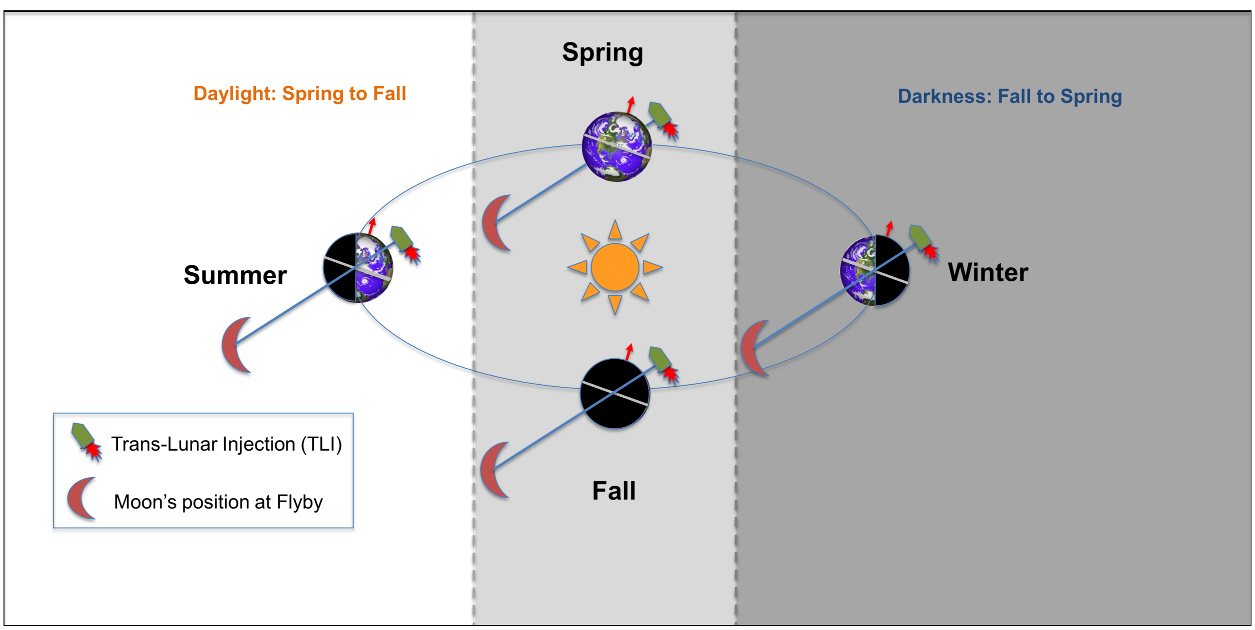

Fig. 8 Seasonal Launch Lighting (Earth-Moon-Sun Geometry).

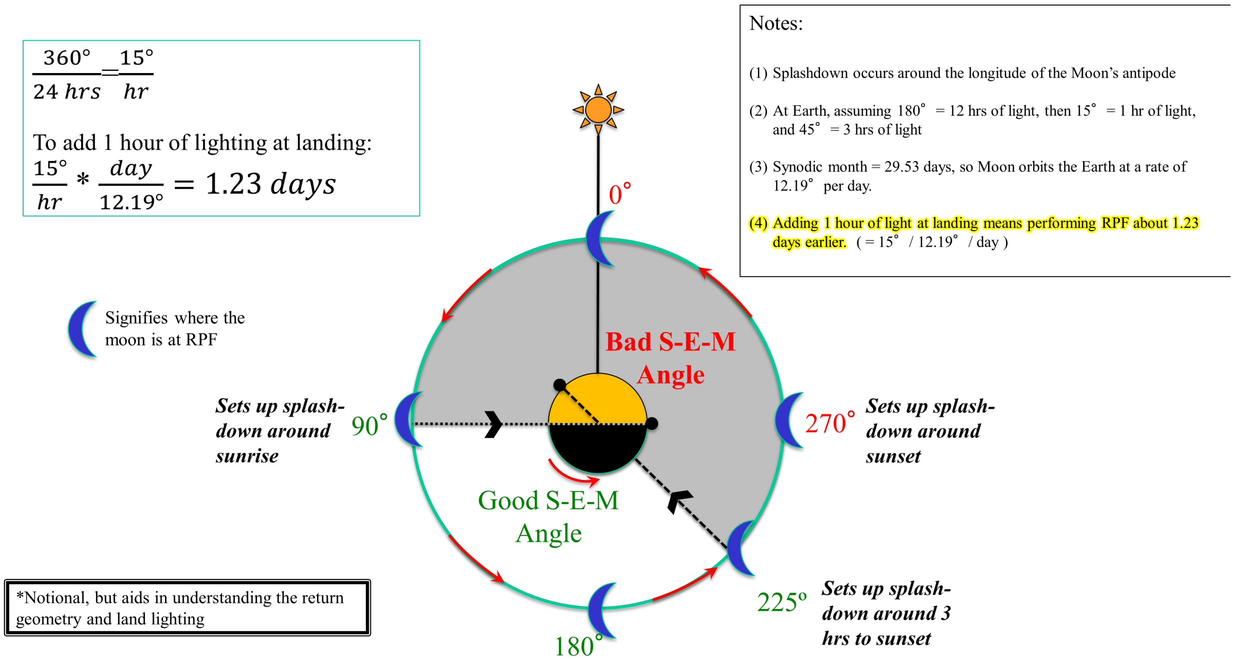

Fig. 9 Sun-Earth-Moon Geometry and the Return Powered Flyby Epoch.

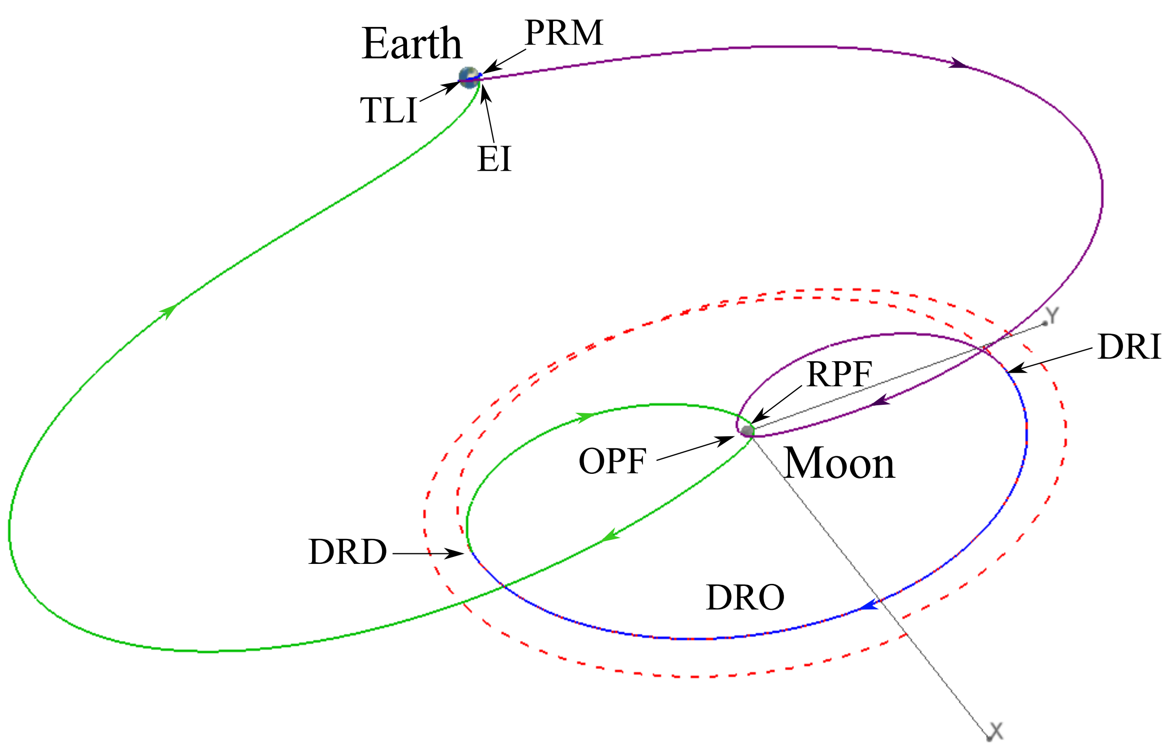

Fig. 10 EM-1 Trajectory Visualization (Earth-Moon Two-Body Rotating Pulsing Frame).

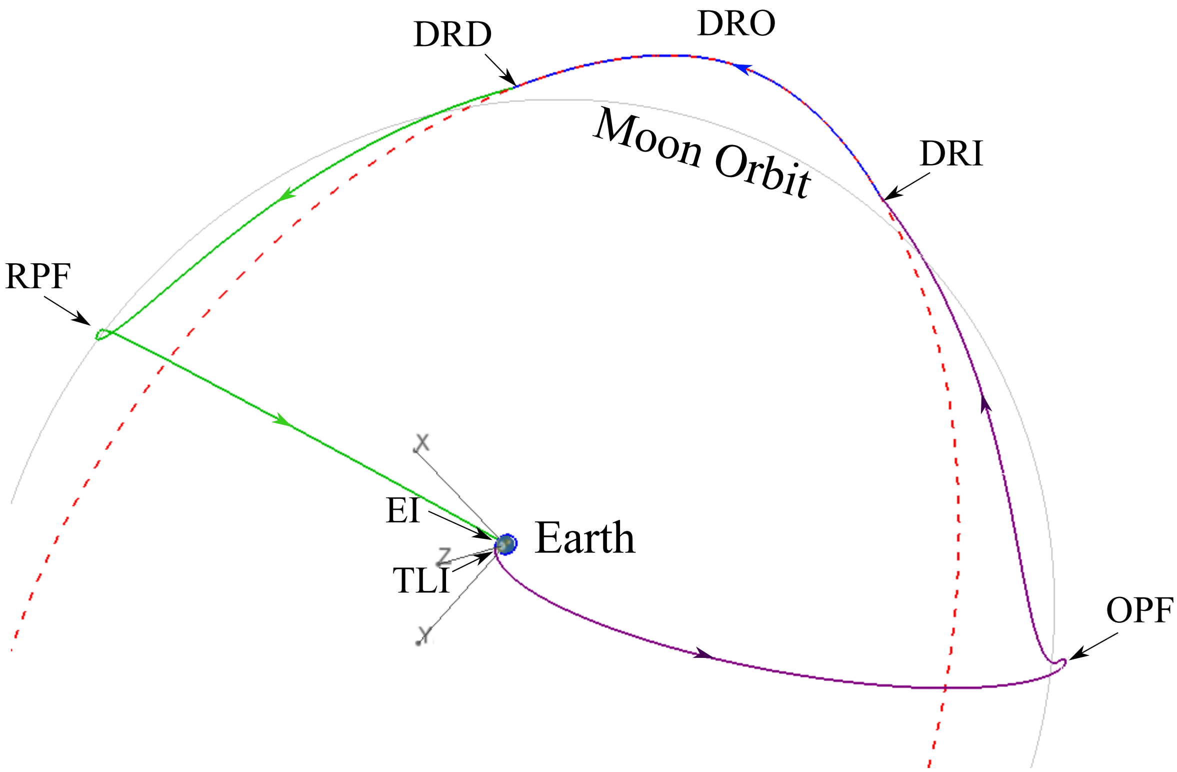

Fig. 11 EM-1 Trajectory Visualization (J2000-Earth Frame).

IX. Mission Visualizations

A Copernicus view of the 26-day mission class in an Earth-Moon Two-Body Rotating Pulsing Frame is shown in Fig. 10 with the major burns listed. Fig. 11 also shows the mission in an Earth J2000 (inertial) frame.

X. Scanning Methodology and DAMOCLES Tool Development

Evaluating mission performance incorporates requirements and constraints from a variety of vehicle subsystems. In order to characterize how the mission changes with launch epoch, trajectory scans are produced. The scan produces missions within a specified launch season, launch period, and launch window (see Fig. 12). A launch season is defined as a collection of launch periods, and in this case, over the span of a year. A launch period is a collection of daily and consecutive launch epochs possible within a month based on vehicle performance. As mentioned, the EM-1 mission has a 10-11 day launch period. A launch window is the period of time available within a given day to launch directly into a desired orbit from a particular launch site. For each nominal opportunity, there are corresponding contingency and/or alternative missions.

The first step in the scan process is to find the Δv-optimal launch date within each launch period. This corresponds to an OPF performed when the Moon is near its maximum southern declination with respect to the Earth's equator. When searching for the optimal launch date within each monthly period, a launch azimuth of 81.23° targeting an orbit inclination of 29.71° is assumed, which corresponds to the desired opening of the daily launch window.

The second step in the scan process is to determine the optimal timing for the opening of each daily launch window on subsequent and previous days throughout the launch period. Forwards and backwards solutions, at roughly 24 hour intervals, are found until the propellant requirement exceeds the usable propellant available for translational burns in either ICPS or Orion. After completing the scan database, the third step is to post-process the Copernicus output files to generate meaningful data for other subsystem teams. Copernicus can be used interactively to design and optimize spacecraft trajectories, but for very large number of cases it is not practical to do this manually. Thus, in order to efficiently produce optimized trajectory files and data products for all the various mission profiles, create and evaluate all potential trajectories to meet project deadlines, a new tool, known as the Database Automation Mission Orion

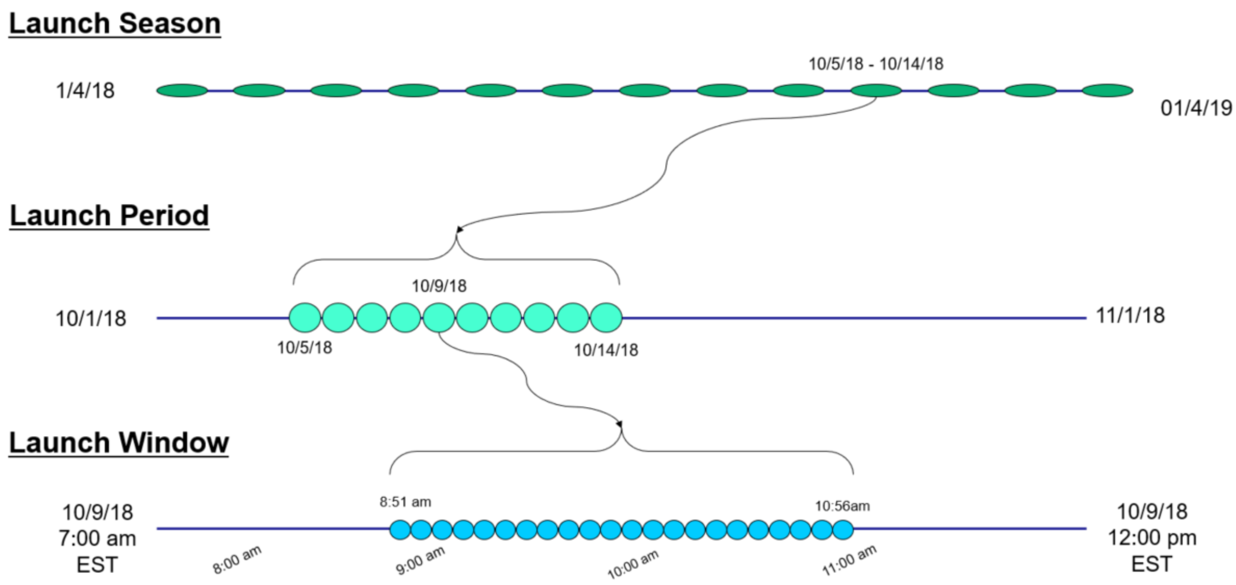

Fig. 12 Launch Terminology

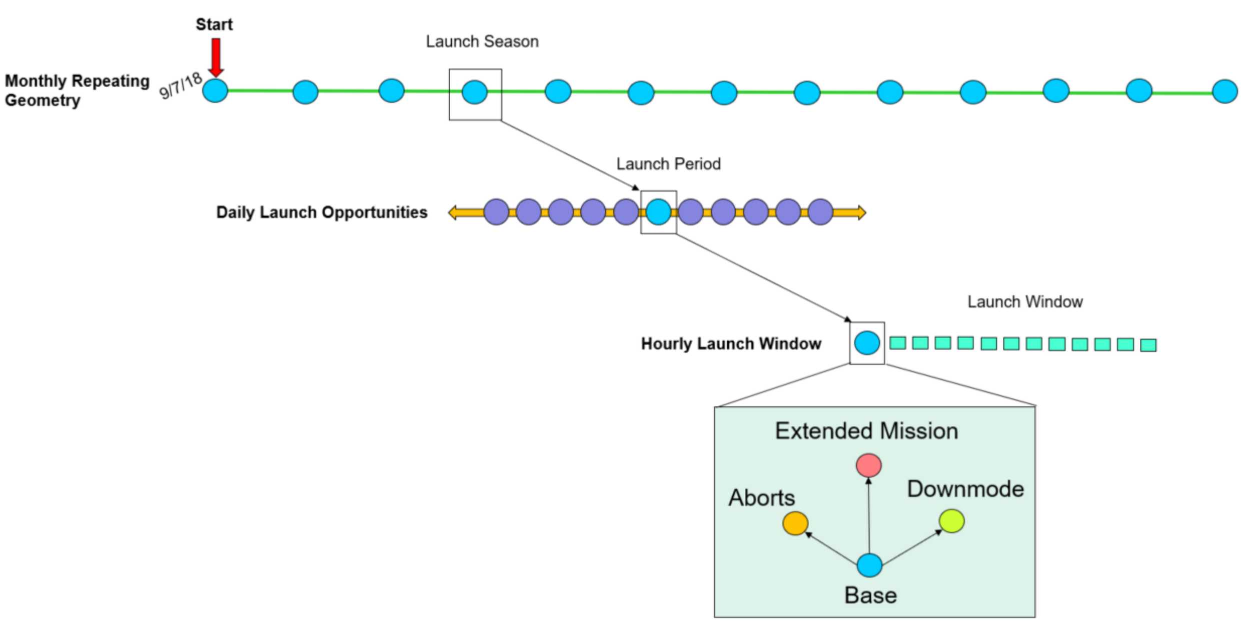

Fig. 13 Scan Methodology

Copernicus Launch Enabling Scanner (DAMOCLES), was created.

The intent of DAMOCLES for EM-1 is to produce a set of nominal and abort trajectories to support JSC's Flight Operations Directorate's need for both pre-flight and real-time data. For each nominal mission opportunity, there can also be various corresponding contingency or alternative missions. When a mission parameter is changed, this standard set of trajectories is recreated to illustrate the impact to the Orion Program. Based on these options, each scan includes between approximately 7000 and 180000 unique trajectories. Even if automated, creating, optimizing, and assessing each trajectory serially would take about 20 days (for 7000 trajectories). This run time is not sufficient for timely product deliveries. With that in mind, DAMOCLES was built to not only automate the process, but also to perform the runs in parallel whenever possible. For the 7000 trajectory case, DAMOCLES is able to create, optimize, and assess all of them within several hours (when run on a sufficiently large/powerful computer cluster).

DAMOCLES is written in Python 3.5+ and can be run on Windows and Linux. The tool manipulates Copernicus trajectory files (idecks) using a Copernicus/Python interface API with the goal of creating, converging, and extracting data from a large set of idecks. The tool has a Graphical User Interface (GUI) that allows users to intuitively configure their scan parameters. The tool can take advantage of multiple nodes/blades on computer clusters with a shared file system and is also compatible with the Torque/Maui job distribution system. The GUI starts a process to oversee and execute the required functions to accomplish the user's request. These functions are started as headless child processes (akin to daemons). The children report their status to unique files, which are then parsed by the overseer. DAMOCLES is intended to work with many mission profiles and is built so that a user can define much of its functionality. Users provide the tool with a pre-converged tuned ideck for optimization and a corresponding post-processing ideck to be used to generate trajectory time history data. They also provide a Python file that defines the trajectory logic for DAMOCLES to perform during certain tasks. Examples include checks and modifications before attempting to solve an ideck, the criteria DAMOCLES uses to assess if a solution has successful converged, what data to extract from an ideck, and how to manipulate extracted data.

DAMOCLES starts with a user-provided "seed" or base ideck that has already been converged at the initial epoch. The EM-1 mission involves the Earth-Moon system, the original "seed" ideck can serve as a good initial guess for launch epochs as long as they have similar Earth-Moon geometry. Leveraging that, DAMOCLES then creates one ideck per launch period based on the lunar sidereal month ($\approx$27.3 days) throughout the user-defined scan date range. Once those idecks are created, DAMOCLES attempts to converge them in parallel. As each case finishes, they are then used to seed the following day, the preceding day, the extended missions for the day, and the other time steps for that day's launch window. These runs all continue in parallel until the entire trajectory set has been completed or attempted (see Fig. 13). Some runs have dependencies on others (e.g. each day's launch window time step depends on the run at the opening of the window converging first), but overall, it is highly parallelized. Some trajectories are not completed as their preceding "seed" case violated propellant mass limits and over time periods of less than a few hours there's a direct relationship between incrementing the launch epoch and consuming more prop mass. The cases that are not completed for this reason are labeled as "skipped" runs. "Skipping" runs is one way to reduce the overall run time without impacting the results. By doing this, DAMOCLES eliminates over 40% of all cases during the scan.

After all the cases of a scan are completed, DAMOCLES then enters a post-processing phase. The trajectory data is generated for all the input decks from the scan. After creating the output data, DAMOCLES collects, manipulates, and concatenates it. There are no inter-case dependencies in the post-processing step, so DAMOCLES is able to generate the output data from nearly all of the trajectories in parallel. Each individual case is condensed to about 400 quantities describing various vehicle states and performance metrics. DAMOCLES outputs a single CSV file that contains trajectory data from each case. The tool also retains and makes available all of the idecks it created during the process. DAMOCLES output data feeds the Integrated Mission Analysis Tool (IMAT) which informs the Orion Program of the impact that constraints have on possible launch opportunities, various subsystems about their environments during the mission, and provides input for the Flight Operation tools used to provide updated burn targets to the Orion vehicle during flight.

Further work on DAMOCLES includes the addition of more mission options, such as various aborts at different times throughout the mission. It is expected that this addition will increase the number of unique trajectories by a factor of 100 – 1000. In order to accommodate these additions, DAMOCLES will need to be made more robust to things like hardware/software failures and changes as it pushes computer systems closer to their limits for longer run times.

XI. Conclusion

The EM-1 mission to a DRO has gone through many years of mission planning and data analysis. Those activities have helped shaped the mission into what it is today. DAMOCLES has provided the mission design team with a way to respond rapidly to many different data requests from the Orion subsystem teams (e.g., power, thermal, communications, etc.) and flight operations. For example, they use the following data to determine how their subsystem will perform on a given day:

- Local launch and landing times in relation to sunrise and sunset
- Length of eclipse periods during the in-space portion of the mission
- Earth line of sight from cislunar space
- Deep Space Network field of view looking towards cislunar space
- Variation of the downrange distance from Earth entry interface to splashdown.

Mission design trades can also be performed based on the information that the additional data shows. For example, if the landing is in darkness, but the recovery operations team desires a landing in daylight, then an analysis is performed to determine how to change the mission design to meet this desire. Also, subsystems request feasibility of alternate or contingency mission designs, such as adding an Orion main engine checkout burn or Orion completing all of its burns using only its auxiliary thrusters. These data and analyses are used to determine the final launch opportunities for EM-1.

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