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The Return of the ICPS: A Statistical Perspective on Artemis Upper Stage Disposal

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ABSTRACT

Prior to the launch of Artemis I, it was discovered that the Interim Cryogenic Propulsion Stage (ICPS) of the Space Launch System may not dispose heliocentrically as intended. This paper summarizes the results of an investigation which looked into why the ICPS may return in less than a year for certain Launch Days. Evidence is presented showing the ICPS mimics the behavior of known Three-Body Periodic Orbits for certain Sun-Earth geometries. This paper discusses the primary mechanisms causing ICPS to return, describes the statistics of these returns, and also discusses interactions between Artemis I mission design and the lunar month.

1. INTRODUCTION

Over the last decade, there has been a large increase in launch frequency worldwide (Helms 2020) and with this increase comes a larger potential for orbital debris reentering Earths atmosphere (Pardini & Anselmo 2019). From 2010 through 2022, 525 orbital stages have re-entered Earth's atmosphere uncontrolled. Orbital stages represented 80% of the total re-entered mass, averaging 92 t annually (Pardini, C. & Anselmo, L. 2024). Uncontrolled re-entry of upper stages cause undue risk (Byers et al. 2022), and it is becoming more important to understand where these stages ultimately go.

To help alleviate this, the NASA Standard 8719.14C *Process for Limiting Orbital Debris* (NASA 2021) requires that NASA Programs meet requirements for limiting orbital debris, including disposing of structures into heliocentric orbits. For lunar exploration missions, disposing of spent upper stages can be problematic given the chaotic nature of the multi-body problem and the likelihood of undesired returns to the Earth-Moon system. A study led by NASA's Engineering and Safety Center presents an evaluation of problematic lunar fly-by disposal geometries and identifies the conditions and potential impacts to NASA's Artemis missions attempting to comply with the orbital debris standard.

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Prior to the launch of Artemis I, it was discovered that the Interim Cryogenic Propulsion Stage (ICPS) may not dispose heliocentrically as intended. The Artemis I mission was designed with a high level requirement that the ICPS shall achieve a positive C3 relative to the Earth to achieve heliocentric disposal. While the compliance for this requirement was categorically met, simulations revealed specific days where the ICPS still returned into the Earth-Moon system within a matter of months, violating the intent of the requirement. Initial work was performed prior to the launch of Artemis I to quantify the number of cases that returned and identify days where the ICPS had a high likelihood of returning into the Earth-Moon system. These initial results were then used to provide cut-outs in the Artemis I launch windows as a means to prevent the Space Launch System (SLS) from launching into conditions that were not well understood.

A follow-on study was performed after the launch to further investigate why certain Sun-Earth geometries enabled the ICPS to return. Specific emphasis was applied to understand which days of Artemis Launch Periods saw returns with the goal to apply this understanding to Artemis III and future missions.

This paper summarizes the work performed prior to Artemis I launch as well as the follow on investigation. The results of this study suggest the ICPS mimics the behavior of known three-body periodic orbits with ICPS returning roughly once a lunar month. Additional results discuss Artemis I, Launch Period shifts and the impacts on both three-body periodic orbit symmetry on ICPS return.

2. BACKGROUND

The initial observation of the possibility of ICPS returning, prior to the launch of Artemis I, was particularly timely. Re-entries of debris from a SpaceX capsule in Australia (July 2022) and a Long March 5B (CZ-5B) rocket on the Phillipines (July 2022)[1] had recently become newsworthy events.

The SLS mission design team monitored possible launch opportunities for Artemis I, focusing on highlighting days where large portions of the nominal Launch Window returned. The presence of Hurricane Nicole delayed Artemis I to a day where ICPS had no returning cases, alleviating this concern. After the launch of Artemis I, additional work was performed to better understand the physics behind this phenomenon and the impact on future SLS missions.

2.1. *Overview of the Artemis I Mission*

The Artemis I mission focuses on longer-duration lunar operations for the Multi-Purpose Crew Vehicle (MPCV) (SLS-PLAN-100 2019). The primary mission objective was to demonstrate the capabilities of MPCV's heatshield with a secondary objective to maintain a Distant Retrograde Orbit (DRO). The SLS Block-1 was used for this mission and consisted of the core stage with solid rocket boosters as the first stage and the ICPS as the second, upper stage.

The core stage inserted ICPS and MPCV into a 975 x 16 nautical mile Highly Eccentric Orbit (HEO). ICPS and MPCV coast to the apogee of the HEO, where a perigee raise maneuver raised the perigee altitude to 100 nautical miles. ICPS and MPCV both coasted to perigee, where the Trans-Lunar Injection (TLI) burn was performed.

ICPS performed TLI which pointed MPCV at the Moon and provided enough Δv for both MPCV and ICPS to target separate lunar flybys with MPCV achieving insertion into the DRO and ICPS achieving heliocentric disposal. After TLI, Orion separated from ICPS and continued its mission to

[1] Aerospace Corporation Rentry Tracker: https://aerospace.org/reentries. Accessed: 2024-11-26

the Moon. ICPS performed a small disposal burn, expelling any remaining propellant and became a dead stage shortly thereafter. ICPS continued on towards the Moon and performed a lunar flyby to escape the Earth-Moon system.

The ICPS disposal trajectory is used as a constraint in the in-space mission optimization problem, where it targets a $C_3 \geq 0.1\frac{km^2}{s^2}$ 10 days after the Lunar Closest Approach (LCA) in order to exceed NASA Standard 8719.14C. Figure 1 provides an illustrative overview of the Artemis I Mission[2].

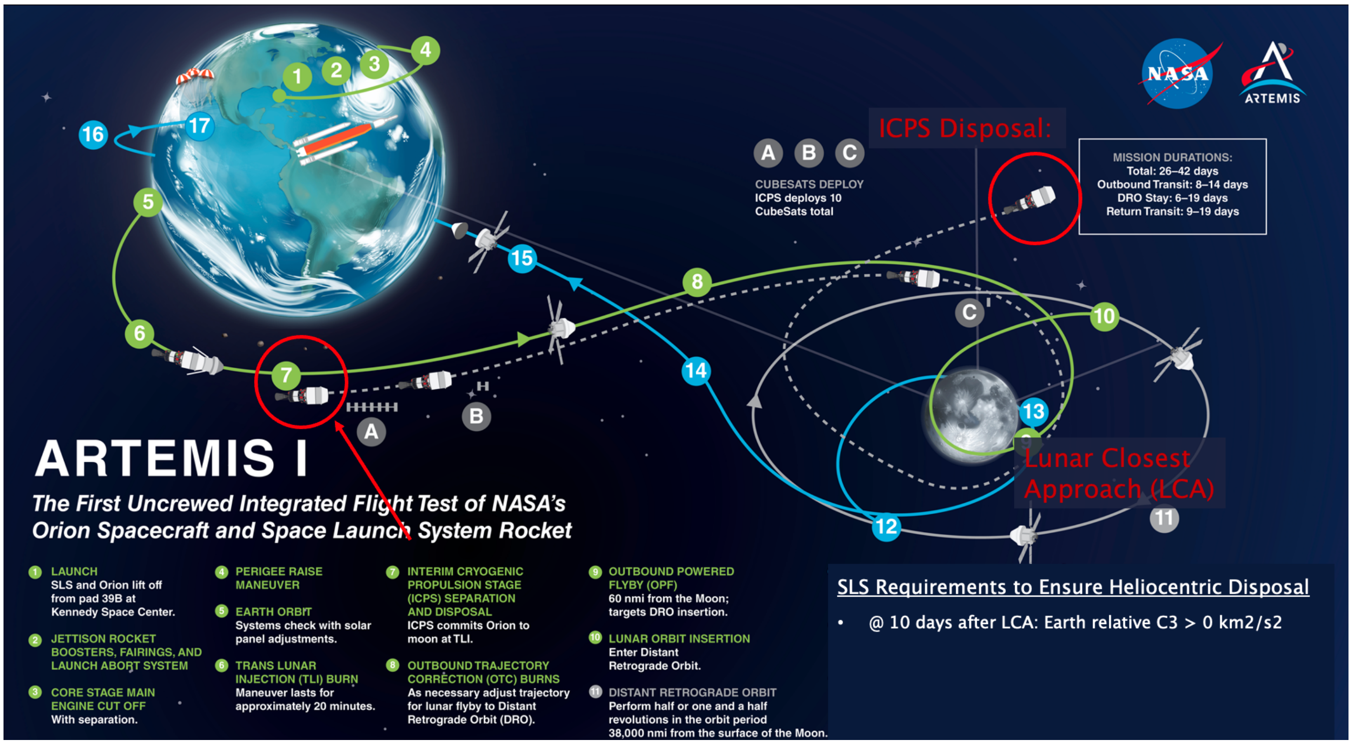


Figure 1. An Illustrative Overview of the Artemis I Mission

The Artemis I mission is simulated in two parts (Anzalone et al. 2020). First, the Program to Optimize Simulated Trajectories (POST) (Lugo et al. 2017) simulates the ascent phase of the mission to an insertion altitude of 85 nautical miles using design vehicles that capture manufacturing variations (Stein et al. 2019). The launch azimuth varies as a means to control the inclination of the insertion orbit for a fixed orbital energy. POST generates a series of insertion states dependent on azimuth to initialize the initial conditions for the second simulation performed using Copernicus.

Copernicus (Ocampo 2003) models the in-space portion of the Artemis mission and starts by interpolating over a grid of initial conditions created from the ascent trajectory to find an optimal seed for the in-space analysis. Each day in a Launch Period is optimized to find the best starting case, where Copernicus then increments each optimal point by one minute forward and backward until the mission becomes infeasible. This range of feasible minutes represents the Launch Window for a specific day.

[2] Artemis I Concept of Operations https://www.nasa.gov/image-article/artemis-i-map-2/ Accessed: 2024-08-01

This process is repeated each day for over a year to define each Launch Window and create a set of Launch Periods. A Launch Period (LP) represents a specific span of days where the Artemis mission is feasible and consists of a number of Launch Windows (LW), each on a different day. A Launch Window represents individual minutes where the mission is successful for a specific day. Launch Periods correspond with the Earth-Moon alignment whereas Launch Windows represents azimuth bounds for insertion into orbit.

Each Launch Period identifies TLI targets and Core Stage insertion targets, which are used as inputs to 6-DOF Monte Carlo analyses using MAVERIC. Upon completion of this process, the launch targets and trajectories are provided to United Launch Alliance (ULA) for analysis. ULA performs their own simulations and then provides data back to the SLS program, including the final end-of-mission state for the ICPS at LCA+10 days. This final mission state is used as the initial conditions for ICPS propagation in this investigation.

Two types of state data are provided by ULA, Nominal and Monte Carlo states. The nominal states represent the state of the ICPS at LCA+10 days for each launch minute of the Launch Window. Typically, these Launch Windows are around 120 minutes with some being shorter depending on the available azimuth range. The Monte Carlo states represent 10,000 simulated trajectories that assess various uncertainties in the Artemis I mission, including but not exclusive to uncertainties in navigation, burn duration, and the lunar flyby state. These variable time Monte Carlo are also performed at a randomly sampled time within each Launch Window.

2.2. *Overview of The Circular Restricted Three-Body Problem*

This investigation employed multiple different dynamical frameworks to model the complex dynamics of the cislunar system. The first and simplest of the frameworks as is the Circular Restricted Three-Body Problem (CR3BP). The CR3BP model uses a rotating reference frame, with the origin at the gravitational barycenter of two celestial bodies. This dynamics model assumes that the primary body (P_1) and the secondary body (P_2) orbit around their shared barycenter in circular orbits, with a constant angular rate, ω. The basis vectors of the rotating frame align the x-axis along the line connecting P_1 and P_2, and the z-axis aligned with the system angular momentum. The y-axis completes the right hand set. A third body, in this study representing ICPS, is introduced into the model with an assumed mass significantly lower than that of the celestial bodies. The third body's position in the CR3BP, $\vec{r}$, is referenced from the barycenter of the Sun-Earth system.

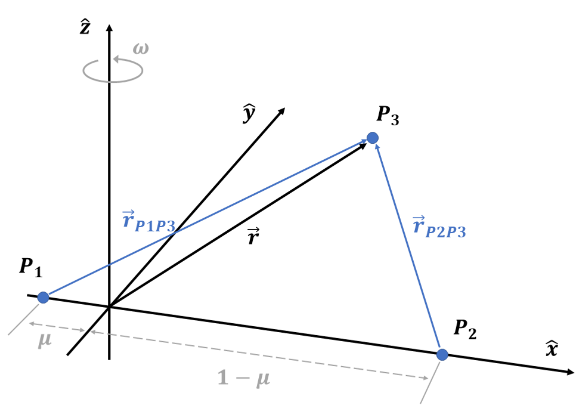


Figure 2. CR3BP Frame

Finally, the positions of the third body with respect to the primary and secondary bodies ($\vec{r}_{P1P3}$ and $\vec{r}_{P2P3}$, respectively) are required to derive the equations of motion. Note that the frame is non-dimensionalized by the mass ratio of the system, μ. The CR3BP frame is shown in Figure 2, and the equations of motion (EOMs) are shown in Equations (1), (2), and (3).

$$\ddot{x} - 2\dot{y} - x = -\frac{(1-\mu)(x+\mu)}{r_{P1P3}^3} - \frac{\mu(x-1+\mu)}{r_{P2P3}^3} \tag{1}$$

$$\ddot{y} + 2\dot{x} - y = -\frac{(1-\mu)y}{r_{P1P3}^3} - \frac{\mu y}{r_{P2P3}^3} \tag{2}$$

$$\ddot{z} = -\frac{(1-\mu)z}{r_{P1P3}^3} - \frac{\mu z}{r_{P2P3}^3} \tag{3}$$

While the CR3BP (Houin & Sood 2024) is a theoretical simplification, this framework is particularly useful for conceptualizing theoretical behavior. The CR3BP defines five equilibrium points, known as Lagrange points: L_1-L_3 are co-linear along the x-axis, while L_4 and L_5 are located 60° leading or lagging the x-axis. These equilibrium points exist where all accelerations in the frame negate each other (see Figure 2). Around these Lagrange points are various three-body periodic orbits. For example, a DRO was used during Artemis I and flown by Orion around the Moon. Gateway, a proposed space station near the Moon, will orbit on a 9:2 lunar resonance Near-Rectilinear Halo Orbit (NRHO). Other examples of three-body periodic orbits are Distant Prograde Orbits (DPOs) and the Lyapunov Orbits.

While the CR3BP provides good insight from a theoretical perspective, the simplifying assumptions of the frame don't capture the full complexity of the Sun-Earth system. Two significant sources of error between the CR3BP and ephemeris specific dynamics are perturbations from other celestial bodies, such as the Moon, and the eccentricity of the Earth's orbit.

To properly model these nonlinear inputs to the model, the Ephemeris Pulsing Rotating frame (EPPR) was also used for higher fidelity propagation (Sikes et al. 2024, 2023). The EPPR utilizes the same basis vectors ($\hat{x}$, $\hat{y}$, $\hat{z}$) and mass ratio (μ) as the CR3BP, but includes perturbations from other celestial bodies. This relies on JPL SPICE kernels for planetary ephemerides (Acton, C.H. 1996; Acton et al. 2017). Additionally, the EPPR accounts for the pulsation of the rotating frame by dynamically scaling the characteristic length of the system, enforcing a non-dimensional distance between P_1 and P_2 to be unity at all times. As such, results generated in the EPPR frame provide insights into any three-body interactions, while still maintaining complete time dependent propagation similar to inertial ephemeris integration. Therefore, the EPPR provides a more accurate representation of the physical behavior of the system and was used to predominately model the behavior of ICPS disposal. The tool was designed with computational speed, so multi-threaded parallel integration and optimization come prebuilt within the ASSET framework.

2.3. *Modeling Methods and Assumptions*

This study used the Astrodynamics Software and Science Enabling Toolkit (ASSET) for ICPS propagation and optimization. ASSET was developed by James Pezent at the University of Alabama under the NASA ROSES Grant No. 80NSSC19K1643 (University of Alabama 2019; Pezent et al. 2022, 2023).

ASSET is a free and open-source optimizer and astrodynamics tool under an Apache license. For integration, ASSET uses an adaptive step size Dormand-Prince numerical scheme by default. When performing optimization, ASSET leverages direct transcription and collocation to solve constrained problems. As part of its open-source nature, ASSET uses a custom built dual-primal interior point optimizer called the Parallel Sparse Interior-Point Optimizer (PSIOpt).

The individual trajectories in this investigation were propagated separately from ULA Nominal and dispersed Monte Carlo initial conditions based on the LCA+10 Day state data sets. For this investigation, the ULA LCA+10 Day states were propagated for 1 year by default with some additional studies exploring longer term sensitives up to 20 years. These trajectories were propagated in both the EPPR and CR3BP frames to compare actual and theoretical trajectories, with point mass gravity utilized for all planets as well as the moon. Solar Radiation Pressure (SRP) was not added to these trajectories as it did not affect the statistics over the shorter duration propagations.

A sensitivity analysis initially compared the effects of propagation duration with the impact on the statistical distribution. This analysis propagated the initial nominal state in both ASSET without SRP and Copernicus with SRP to compare the influence SRP had on the statistical distribution of the shorter duration ($\leq$ 2 year) trajectories. The purpose of this investigation is to understand the general trends of how the physics will effect the Artemis Launch Windows and are not meant to propagate the ICPS with exact precision.

Thus, precise modeling was sacrificed to increase the speed of computations due to the shear number of trajectories being propagated. This investigation was concerned with ICPS behavior at a high level statistical perspective. This analysis utilized Artemis I ICPS state data spanning Launch Periods LP22 (June 2022) through LP30 (January 2023) with a total of 1,273,011 propagated states; 12,885 nominal cases and 1,269,126 Monte Carlo cases. This dataset spans 9 Launch Periods and contains 126 separate Launch Windows.

For the purposes of this study, a *return* is defined as a case where ICPS passes through a sphere with a fixed radius of 500,000 km centered on the Earth. While this radius is strictly larger than the distance between the Earth and the Moon, it serves as a better filter since it is fixed, which clarifies some of the underlying behavior.

3. STATISTICAL ANALYSIS OF ARTEMIS I PROPAGATIONS

The initial investigation looked at the overarching statistics of the return problem by focusing on the behavior of the ICPS. The statistics were developed by propagating Artemis I LCA+10 Day nominal states for the Launch Periods 22 through 30 for a duration of 20 years using ASSET.

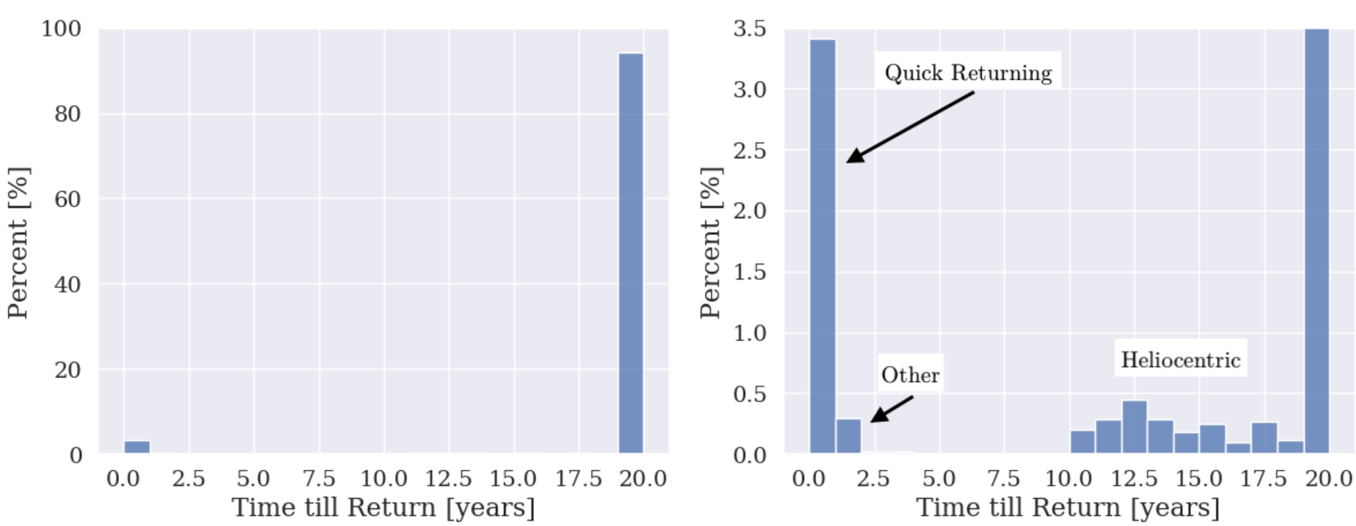
The nominal trajectories were then propagated with multiple ephemerial grid resolutions and time steps out to an excessive 250 years. The duration was then reduced while increasing resolution as a means to reduce computational run time while maintaining a constant statistical distribution. A 20 year time limit for trajectory propagation yielded little impact in the overall statistical distribution of return trajectories while maintaining a reasonable run time for a dataset of this magnitude.

Trajectories propagated in ASSET past this 20 year limit require higher-resolution ephemeral grids and smaller timesteps in order to reasonably estimate when ICPS would return, resulting in timing issues if a sufficient density was not met. This is due to ASSET utilizing spline fits of JPL SPICE kernels for ephemeris propagation instead of performing a SPICE call for every time step. Again, given that the emphasis of this investigation was to explore a large number of near-term, 1-2 year return trajectories, certain concessions were made to enable quicker trajectory propagations while preserving the overall statistics of the dataset.

It should be noted that specialized numerical algorithms (Milani & Nobili 1987; Amato et al. 2017) are necessary to accurately propagate trajectories beyond 10 to 15 years due to error propagation in the underlying numerical models. SRP would also need to be modeled as it would have a larger influence over these long durations as well (Ammar 2008). Unfortunately, good models for ICPS reflectivity and ICPS attitude past LCA+10 days do not exist. However, from a general statistical perspective, trajectories modeled to 20 years using ASSET in this manner should be representative of the problem and of sufficient fidelity to understand the statistics of the shorter duration return trajectories.

A histogram representation of the time it takes for ICPS to return into the cislunar system is provided as Figure 3. This Figure illustrates two major regions in the distribution of return times, with a large gap interior to both regions. Figure 3 consists of data from the 12,885 nominal trajectories propagated for 20 years.

The first region shows a small peak ($\approx 3.5\%$) of the trajectories that return relatively quickly when compared to the others. To the right of that region is a sizable gap of 5–10 years where no trajectories return. After the 10 year limit, a second region exists where the remainder of the trajectories return. This second region is not uniform and appears to have an interesting harmonic. But given that the numerical models used to derive these statistics are loosely applicable in this region, any conclusions made in this region would be rough estimates at best. This heliocentric region that exists past 10 years should be further investigated using the numerical modeling techniques used for longer duration trajectory propagations mentioned above, but is outside the focus of this investigation. The last bin of the histogram represents the majority of the Artemis I trajectories which return in more than 20 years.



(a) Normal Axis

(b) Truncated Axis

Figure 3. Histogram of the time to return for Artemis I ICPS disposal, LP22 through LP30

Examining individual trajectories in each of the two regions revealed three general categories of return types. The first type occurs in the far right region of the histogram where the times to return exceed 10 years. These trajectories all dispose heliocentrically and follow the intended behavior of the ICPS mission design.

The second type of trajectory return within about 1.5 years and follow the behavior of known three-body periodic orbits. These return much quicker than intended and are the primary focus of this paper. The last type of trajectory occurs with return times between 1.5 and 4.5 years. These start out following similar three-body period orbital structures, but quickly become perturbed in such a way that they orbit outside the cislunar system, eventually returning in system later than their quick returning counterparts.

3.1. *Heliocentric Disposal Trajectories*

The heliocentric disposal trajectories represent the majority ($\approx 96\%$) of the cases found in this investigation. These trajectories leave the cislunar system and propagate out around the Sun as intended. However, since ICPS disposes using a single lunar flyby, ICPS will always eventually return given that there is no secondary burn or energy change that stabilizes the orbit around the Sun. The return of ICPS becomes a matter of phasing, with returns starting around 10 years and exceeding the 20 year limit of this investigation. An example of a heliocentric disposal trajectory shown in the EPPR frame is given as Figure 4.

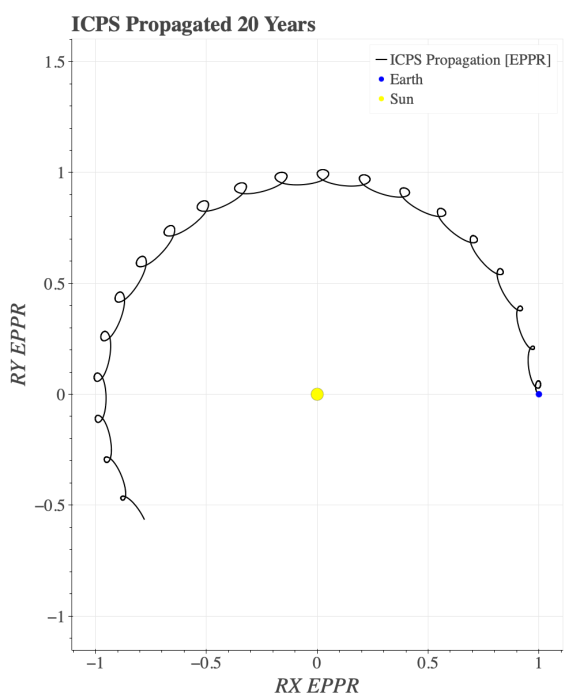
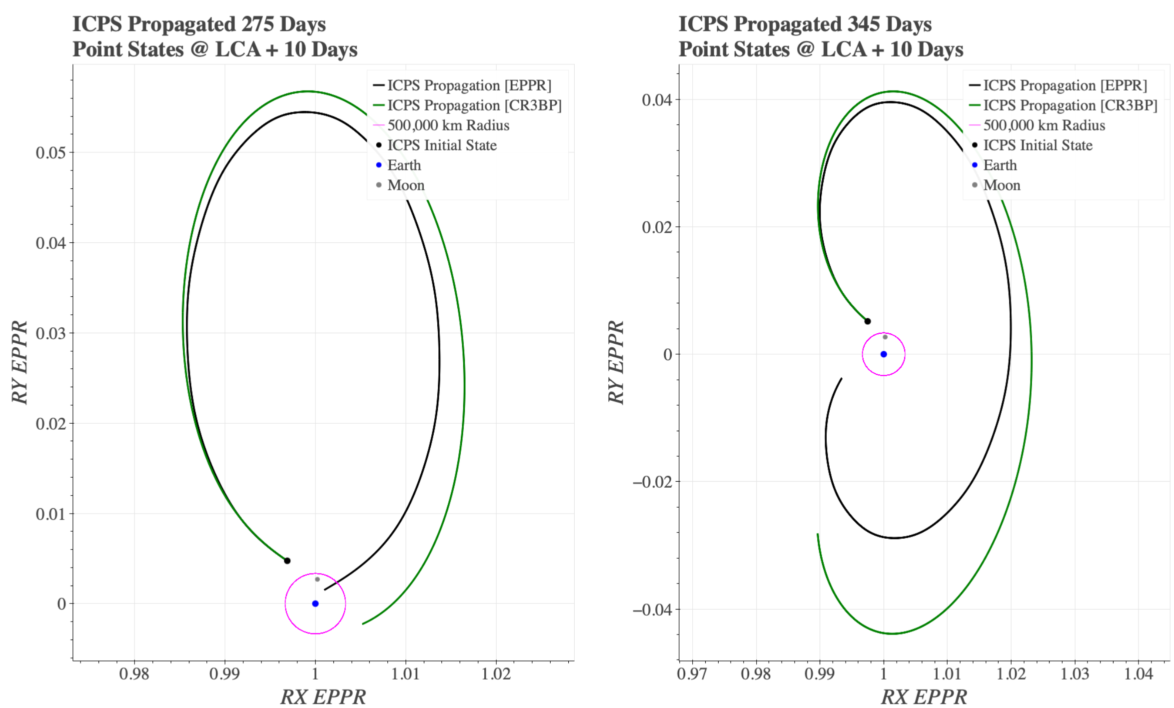


Figure 4. Example of a Heliocentric Orbit in EPPR frame: June 6th, 2022, Minute 1.

3.2. *Quick Returning Trajectories*

The second type of trajectory found in the statistical data is termed a *quick return*, one which is characterized by returning in less than 1.5 years. These trajectories exhibit behavior analogous to known three-body periodic orbits. However, due to perturbations of ephemeris dynamics, and the loss of system autonomy, these trajectories deviate from the idealized CR3BP motion. These quick return trajectories will be the primary focus of the remainder of this study.

Quick return disposal trajectories imitate the natural behavior of one of two three-body periodic orbit families; specifically Distant Prograde and Lyapunov Orbits. The term *quick return* is used instead of *three-body periodic orbit* to avoid confusion and clarify that these trajectories are not actually on a given three-body periodic orbit, but merely look like one. Examples of these trajectories are given as Figure 5 where the green line (CR3BP) shows what the orbit would become if it were a perfect three-body orbit, and the black line (EPPR) shows a more accurate depiction of the trajectory the ICPS would take given the addition of other gravitational bodies. While these trajectories shadow the six-dimensional structure of multi-body invariant manifolds found in the autonomous CR3BP (Howell et al. 1997), the motion is simulated in an ephemeris model. As such, it is to be expected that trajectories do not perfectly maintain the idealized structures present in the CR3BP. In addition, ICPS is a dead stage that does not perform targeting or orbit maintenance. Therefore, while the quick return trajectories are heavily influenced by the underlying structure of three-body orbits, dynamical chaos ensures that motion is not cleanly periodic nor predictable.



(a) Trajectory of the 38$^{\text{th}}$ Minute. (b) Trajectory of the 105$^{\text{th}}$ Minute.

Figure 5. Example of a Quick Return Orbit in EPPR frame: October 1st, 2022. Black Trajectory Propagated in EPPR. Green Trajectory Propagated in CR3BP

Typically, a quick return trajectory tends to follow one family of three-body orbits or the other due to timing of the ICPS LCA+10 Day state in the Sun-Earth frame, but there is the potential to transition between these orbit types within a single launch window. Figure 5 shows this transition during the October 1 Launch Window. Initially the trajectories early in the Launch Window follow a DPO-like behavior (Figure 5(a)), then transitions to that of Lyapunov-like orbit later (Figure 5(b)).

Figure 6 shows the same trajectory as Figure 5(b), displayed in the Sun inertial J2000 frame. In the inertial frame one can see there is little difference between Earth's orbit and that of the disposed ICPS. The disposal trajectory approximates a Hohmann transfer, with the initial burn being the lunar flyby at LCA and then ICPS enters into a heliocentric transfer orbit. However, with the absence of the second *burn*, ICPS will inevitably return and in this case, intersects Earth's orbit in multiple locations. The transfer orbit in this case is rotated as well, due to ICPS not performing the lunar flyby when Earth is at aphelion or perihelion.

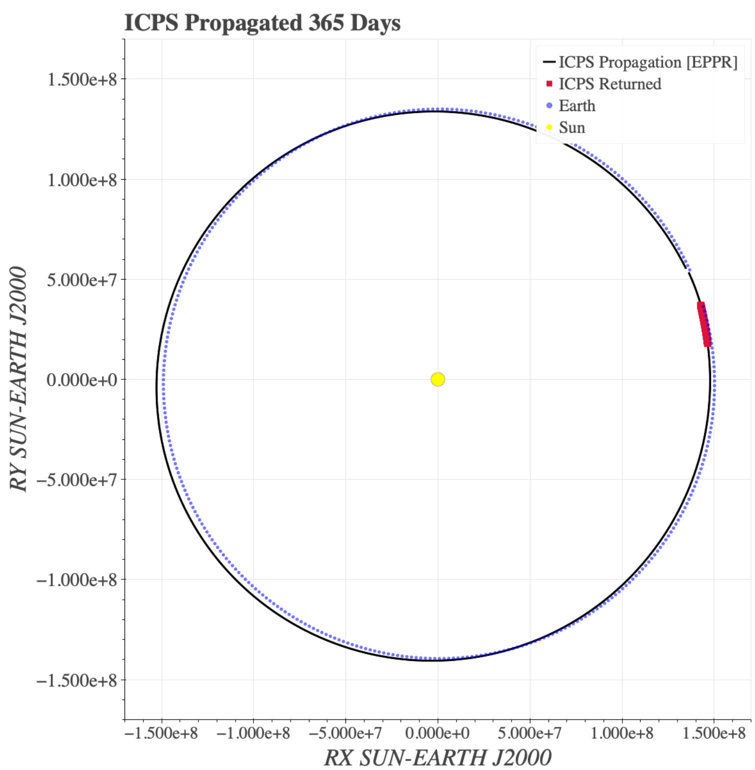


Figure 6. October 1st, 2022 Minute 105 in Earth Centered J2000.

The ICPS will always return for a mission like Artemis I. It fundamentally becomes a question of phasing between the disposal orbit and that of Earth's. While other effects such as SRP can act like a second burn and affect the phasing over longer durations, it is not controllable, especially given that the ICPS is a dead stage. This would be hard to rely on as preventative measure to ensure the ICPS never returns.

3.3. *Other Uncharacterized Trajectories*

There are a few remaining trajectories that do not fit in one of the two above categories and are a small fraction ($\approx 0.4\%$) of the trajectory dataset. These typically start as if they are mirroring a three-body periodic orbit, but become sufficiently perturbed by other gravitational bodies such that they loiter outside the cislunar system between Sun-Earth L1 and L2. Eventually, they fall back in system and are classified as a return trajectory, but typically return with a longer period, within 1.5 to 4.5 years. An example of one of these miscellaneous trajectories is shown in Figure 7.

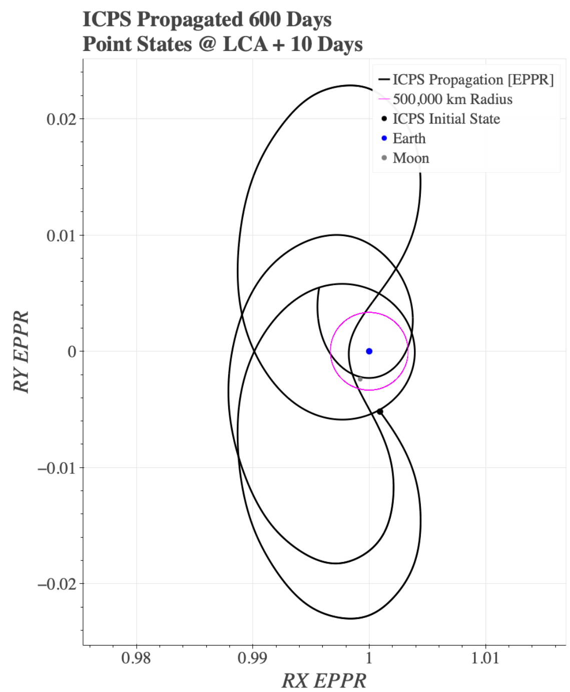


Figure 7. Example of an Uncharacterized Orbit in EPPR frame: December 13th, 2022, Minute 106.

These trajectories do not meet the intent of heliocentric disposal much like the quick returns, but are much fewer in number due to their relative instability. Also, since they tend to be oriented and initially behave similarly to the cases that return quickly, addressing the quick return trajectories should also address these remaining trajectories as well.

3.4. *Overall Summary of ICPS Return Statistics for Artemis I, LP22 through LP30*

This next section augmented the previous statistical dataset with the addition of all the Monte Carlo state propagations as well. This dataset represents the entire state data delivered for Artemis I from L22 through LP30 with a total of 1,273,011 propagated states; 12,885 nominal cases and 1,269,126 Monte Carlo cases. This dataset spans nine Launch Periods and contains 126 Launch Windows.

Table 1 shows a summary of this data for any day that had a case return in less than a year. It is grouped by Launch Period to illustrate the overall fluctuation of return frequency over the calendar year. It is of note that LP22 did not have any cases that returned nominally. This happened to coincide with the definition of that particular Launch Period, whereas the day prior to the start of LP22 did have cases that returned.

Table 1. Number of Cases that Return Summarized by Launch Period

LP	22	23	24	25	26	27	28	29	30	Total
Nominal	0	82	67	32	46	6	15	67	67	382
Monte Carlo	73	4498	5742	2618	2993	2263	3290	7135	7214	35826

Tables 2 and 3 below show this same data, but are segmented by Launch Day and Window. This allows more explicit comparisons of the return frequency on a Launch Window basis. When reviewing this data, it becomes apparent that there is a particular cadence that occurs between the days where nominal cases return.

Table 2. Number of Cases that Return Summarized by Date, LP 22-26

LP	22	23		24			25				26		
Date	6/6	7/5	7/6	8/2	8/3	8/4	8/28	9/1	9/2	9/3	9/30	10/1	10/2
Nominal	0	82	0	0	67	0	0	0	32	0	0	46	0
% Returned	0	68	0	0	56	0	0	0	27	0	0	38	0
Monte Carlo	73	4383	115	5	3795	1942	1	368	2248	1	166	2775	52
% Returned	1	44	1	0	38	19	0	4	22	0	2	28	1

Table 3. Number of Cases that Return Summarized by Date, LP 27-30

LP	27			28		29				30		
Date	10/29	10/30	10/31	11/14	11/15	12/12	12/13	12/14	12/15	1/12	1/13	1/14
Nominal	0	6	0	15	0	0	67	0	0	0	67	0
% Returned	0	10	0	22	0	0	56	0	0	0	56	0
Monte Carlo	53	2084	126	3092	198	29	6244	849	13	1019	5611	584
% Returned	1	21	1	31	2	0	62	8	0	10	56	6

First, the cadence of the cases where the nominal trajectories returned tends to repeat every lunar month, or around 27–28 days. There was one exception being between LP27 and LP28, which was closer to a half lunar month. Second, LP22 does not have any nominal cases that return, but only a few Monte Carlo cases. Finally, there are days where the Monte Carlo cases will return but the nominal cases will not. These are adjacent to days where both nominal and Monte Carlo cases return, indicating a blurring of the statistics across multiple days due to the dispersions of the states provided by the Monte Carlo data. This particular cadence of the nominal returns and the physics behind it is the subject of the discussion in the next section of this paper.

4. THE CADENCE OF ICPS RETURNS

Initial Artemis I return statistics show that there is a particular cadence that occurs between the days where a large proportion of the nominal launch window returns into the cislunar system. These days became the initial focus of the analysis prior to Artemis I launch, and during subsequent analysis, additional patterns were observed. These nominally returning days tend to repeat every lunar month with the exception between LP27 and LP28.

4.1. *ICPS May Return in Less than One Year*

It has been shown above that ICPS disposal trajectories may align with Sun-Earth three-body periodic orbital structures at certain times and imitate their natural behavior. This causes ICPS to return into the cislunar system in less than one year.

The quick return trajectories mimic either a Lyapunov Orbit or a DPO. Each orbital family has two symmetric versions, with Lyapunov families centered on either the Sun-Earth L1 or L2 Lagrange points, and DPOs exhibiting top/bottom symmetry relative to the x-axis of the CR3BP. The specific version of the orbital family the ICPS follows is dictated by Artemis I launch availability throughout the year. For Launch Periods LP22-LP27, ICPS mimics either a L1 Centered Lyapunov Orbit or the top *lobe* of a DPO. After LP27, the trajectory symmetry switched and transition to L2-centered Lyapunov Orbits and the bottom *lobe* of a DPO for Launch Periods LP28-LP30.

When comparing the orbital periods of these two orbital families one can see why the ICPS returns quickly. The idealized CR3BP system possesses one constant of integration, called the Jacobi Constant. The Jacobi Constant can be considered as a pseudo-energy term for the three-body system. As such, within a specified family of Lagrange point orbits, every Jacobi Constant within the valid range of the family is associated with a distinct orb. For the range of Jacobi constants found at the Artemis I LCA+10 Day states, the orbital period of Lyapunov orbits range between 175 and 480 days[3]. Comparatively, the orbital half-period of the corresponding DPOs range from 51 to 262 days. Half-periods are used in this comparison since ICPS would have to pass through the cislunar system in order complete a full orbit, thus half-periods should more applicably bound the times till ICPS returns. The time it takes for ICPS to return in system will deviate from these values given the influence of the other gravitational bodies and where exactly the ICPS pierces the sphere set by the return radius. But from a theoretical perspective, it can be shown that imitating these orbits would plausibly cause the ICPS to a return in roughly 2 to 16 months.

4.2. *Nominal Returns Are Contained to a Calendar Day*

When viewed relative to the Earth-Moon system, the LCA+10 Day state is not fixed. This is due to the different variations of the mission parameters over time as well as the application of various uncertainties present in the mission, especially the lunar flyby.

However, when examining the LCA+10 Day state from the Sun-Earth perspective, this state is roughly fixed relative to the Moon, any variation in state is small relative to the scale of the Sun-Earth frame. The range of operational parameters for the Artemis I mission for example, total mission δV and Orion targeting, vary by small amounts over time relative to the Sun-Earth Frame, thus appearing fixed. The Artemis I mission *fixes* the LCA+10 Day state relative to the Moon, which

[3] JPL Three-Body Periodic Orbits: https://ssd.jpl.nasa.gov/tools/periodic_orbits.html. Accessed: 2024-11-26

clocks around the Earth in a fixed orientation as a function of the lunar month. A schematic of this is presented as Figure 8.

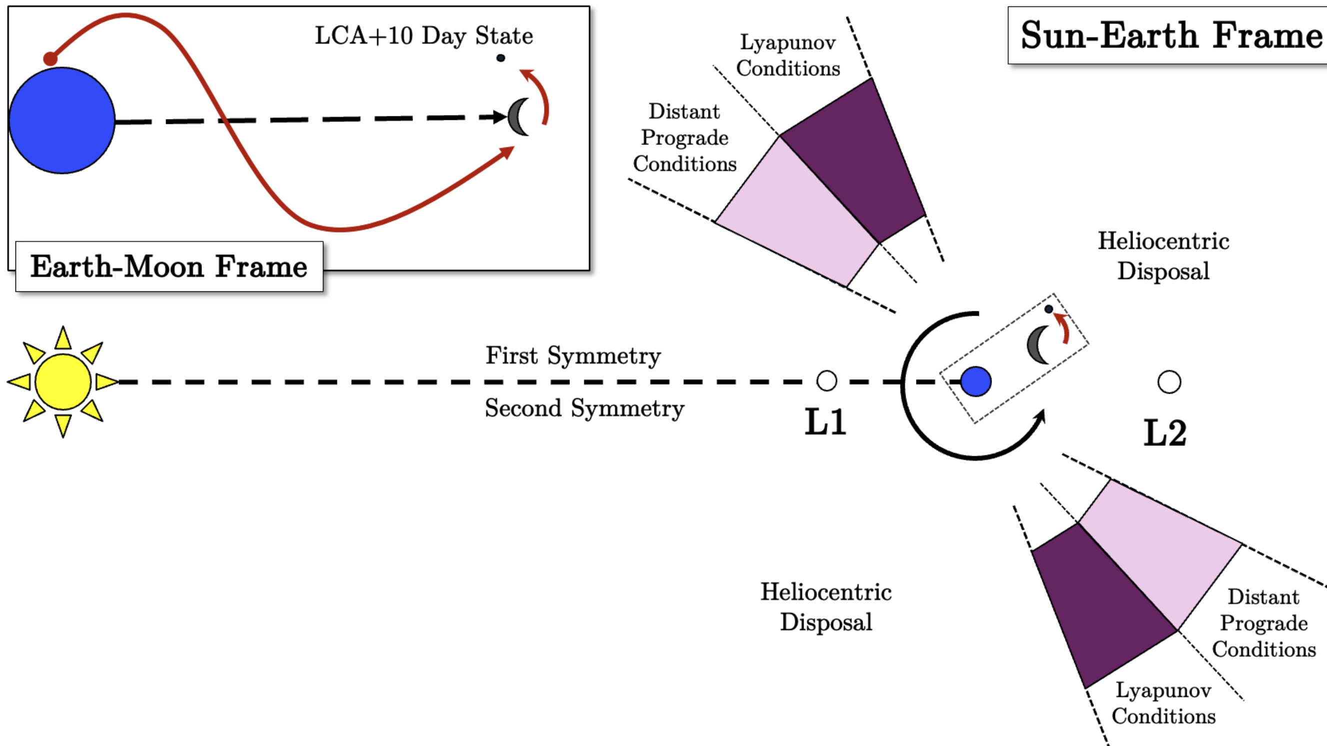


Figure 8. LCA+10 Day State Viewed in the Earth Moon Frame

The examination of many trajectories over all the Launch Period reveals distinct corridors where the ICPS returns. These corridors exist at the same angles relative to the Sun-Earth Frame such that the ICPS will start to imitate one of the three-body periodic orbits. When looking across Launch Periods, the motion of the Moon becomes the dominating factor. Similar results were discovered in an independent study conducted by the Gateway mission design team (Davis et al. 2024).

As Artemis I progresses through the Launch Period, the LCA+10 Day initial state rotates around the Earth counter-clockwise, typically starting at conditions where the ICPS will heliocentrically dispose and then rotating into this corridor where the ICPS may return. It first will rotate into conditions where the Lyapunov Orbit is imitated then aligns with the conditions of DPO imitation. Eventually, the initial states rotate out of this corridor and ICPS disposes heliocentrically again. When the LCA+10 Day state aligns with one of these corridors in the Sun-Earth system, the ICPS follows the behavior of the appropriate three-body periodic orbit. These corridors (Figure 8) serve as a type of *on-ramp* where ICPS imitates a three-body periodic orbit is perturbed. As such, this analysis demonstrates that quick-return trajectories are exclusively a function of heliocentric disposal geometry.

An example of this rotation is illustrated in Figure 9. This shows the counter-clockwise progression of the initial states throughout a subset of the Launch Period and the alignment with Lyapunov and DPO conditions. Note, only three days were chosen from this Launch Period to illustrate this phenomenon more clearly. This particular Launch Period, LP26, spans from September 20th through Oct 4th, with 17 Launch Windows.

Note, this occurs as the Moon revolves around the Earth for a Launch Period. Conversely, over the Launch Window the trajectories transition clockwise due to the relative motion of the Launch Site on Earth's with respect to the Moon's location in orbit. Thus, a Launch Window will transition from a DPO to a Lyapunov Orbit.

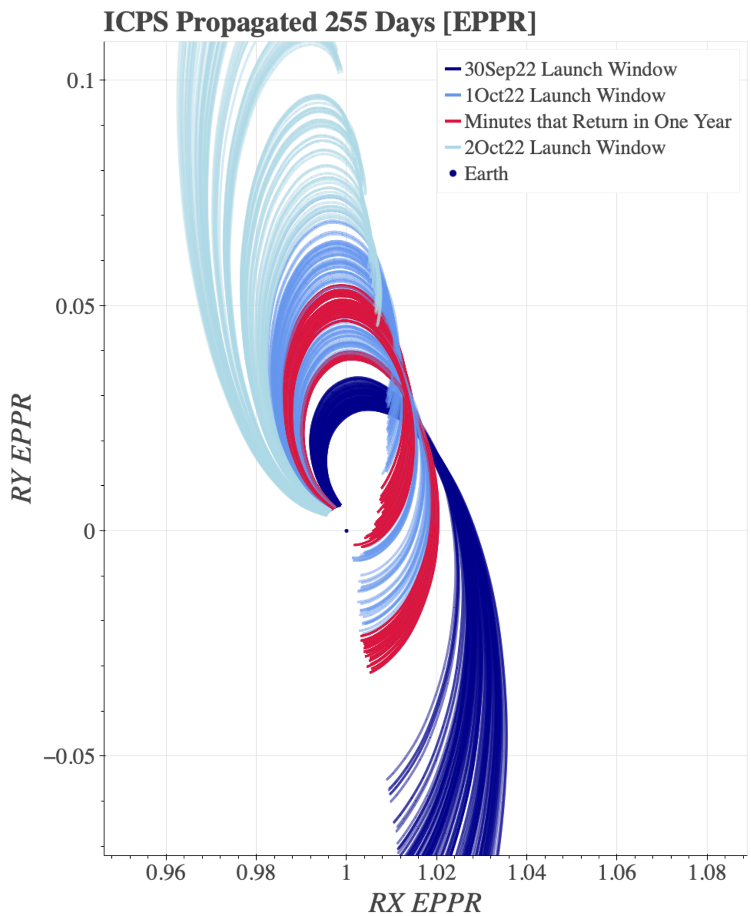


Figure 9. Launch Window Propagations of Sep 30th, Oct 1st, and Oct 2nd Launch Windows. Top Crimson Band Imitates DPOs, Bottom Band Imitates Lyapunov Orbits. Each Line Represents One Launch Minute and Transition Clockwise

When looking at any given Launch Window, the rotation of the Earth dominates. Given that each Launch Window is relatively short compared to the calendar day, roughly 2 hours or less, there is little chance of overlap between these corridors. If a given calendar day meets the conditions to mimic a Lyapunov or DPO, the Moon will revolve around the Earth enough by the next day such that it no longer meet these conditions. Thus, ICPS nominal trajectories tend to return on a single calendar day for Artemis I. October 1st is a notable exception given that it was the only Launch Window that progressed from the DPO conditions to that of a Lyapunov Orbit in a single Launch Window. But the adjacent calendar days still did not have any nominal cases that returned.

The Monte Carlo cases, on the other hand, do have days that return adjacent to the nominal day. This is mainly due to the fact that the velocity dispersions in LCA+10 Day state are sufficient to *smear* ICPS across the corridors, allowing a few cases to return as they are pointed in correct direction in the Sun-Earth Frame, but on a different calendar day. This indicates the uncertainty of the LCA+10 Day state is sufficient to dictate whether or not the ICPS returns on any given day.

This observation regarding the containment of nominal returns suggests that the calendar day can serve as a sufficient metric for estimating where ICPS disposal is most susceptible to return. This offers a means to simplify and identify where to focus analysis, since these days repeat every half or one lunar month.

The data for all the Launch Periods was combined and then propagated in the Sun-Earth EPPR Frame for a year to examine the location of these corridors. This data is shown as Figure 10 where the blue coloration simply delineates Launch Period and the crimson points show which LCA+10 Day state return within a year. Again, this data set represents a total of 1,273,011 nominal and Monte Carlo propagated states from LP22 through LP30.

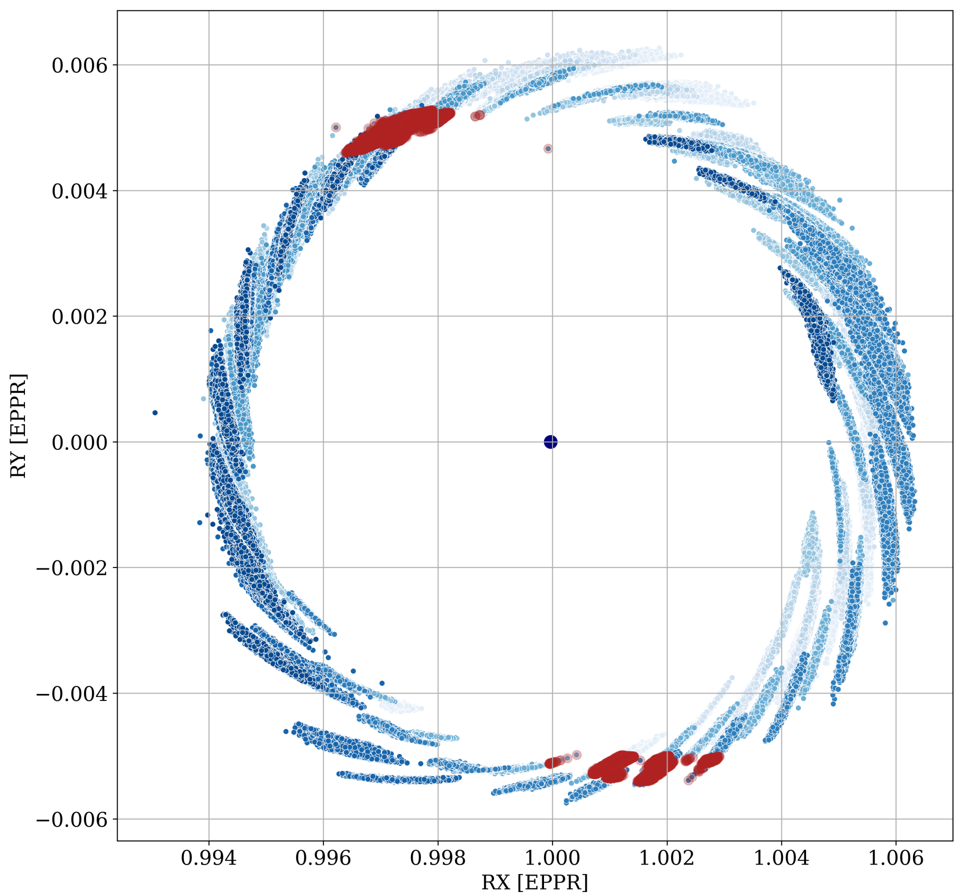


Figure 10. Location of Every Artemis I Initial State in the EPPR Frame. Blue Shading Denotes Launch Period, Red Coloration Denotes ICPS Returned.

On the Sun-Earth scale, this confirms that the orientation of all returns consistently point in the same two directions, occurring in the second and fourth quadrants. These directions align with the conditions needed to mimic Lyapunov or DPOs. The upper left corridor of Figure 10 occurs during LP22 (June) through LP27 (October), and represent the conditions needed to imitate L1-centered Lyapunov orbits or the top *lobe* of a DPO. The lower right corridor occurs during LP28 (November) through LP30 (January) and represent the conditions needed to imitate L2-centered Lyapunov orbits or the bottom *lobe* of a DPO. Trajectories that do not align with these corridors dispose heliocentrically, not returning within at least 10 years.

It is interesting to note that, while the locations of the initial conditions needed to return are contained within these two corridors, all cases inside those corridors do not necessarily return. This is mainly due to the highly non-linear nature of this region of space combined with the variation in Monte Carlo conditions of the LCA+10 Day states. Given that Monte Carlo uncertainty relative to the nominal state is sufficient for adjacent calendar days to have returns, it stands to reason that cases on days with nominal returns could also have Monte Carlo states that miss the cislunar system. This makes it very difficult to prevent ICPS from returning given the magnitude of the uncertainty of the state, which is enough to either prevent or cause ICPS to return in this region. This sensitivity has also been shown in the work by Davis et al. 2024.

4.3. *ICPS Returns once a Lunar Month*

Reviewing Tables 2 and 3 show a return cadence where the ICPS returns roughly once a month, with the noted exception between Launch Periods 27 and 28. This cadence can be attributed to the interaction between the Artemis I Launch Period and the position of the return corridors in the Sun-Earth Frame.

Figure 10 illustrates that the return corridors are fixed in the Sun-Earth Frame. The locations of these corridors can be overlayed on the location of the Artemis Launch Periods, represented via the location of the ICPS and Moon at LCA. This interaction is shown as Figure 11 where Launch Periods from LP23 through LP27 intersect the lower right corridor and Launch Periods LP28 through LP30 intersect the upper right corridor. This is different than the previous Figure 10 which shows LCA+10 Day states. Figure 11 shows the LCA position of the Moon which is more indicative of the Artemis mission and removes any uncertainties regarding the lunar flyby. Note that the Launch Period intersections have rotated due to the 10 day difference in time between the two figures.

The Artemis I lunar flyby is roughly fixed relative to the Moon and correspondingly each Launch Period and repeats using the Synodic Period. But the corridors where ICPS returns are fixed in the Sun-Earth Frame. As the Moon orbits Earth, the location of LCA sweeps out an arc in the Sun-Earth Frame for a given Launch Period. The intersection between the return corridors and the representation of the Launch Periods approximate where the days are located where the ICPS returns nominally.

These days then shift roughly two days per Launch Period. This corresponds to the difference between the Lunar Sidereal Month (27.3 days) and Synodic Month (29.5 days) since the Earth and Moon must move together for an extra 2.2 days to completely re-align with the Sun. A schematic of this motion is shown as Figure 12.

This two-day cadence is shown in Tables 2 and 3 as the number of Earth calendar days between successive nominal returning days. Note, the Table does not show a cadence of exactly two days, since both the time-of-flight for ICPS to the Moon and the distance between the Earth and Moon are not constant across all dates. Also, any given Launch Period is not exactly the same number of days in length due to Artemis I mission constraints.

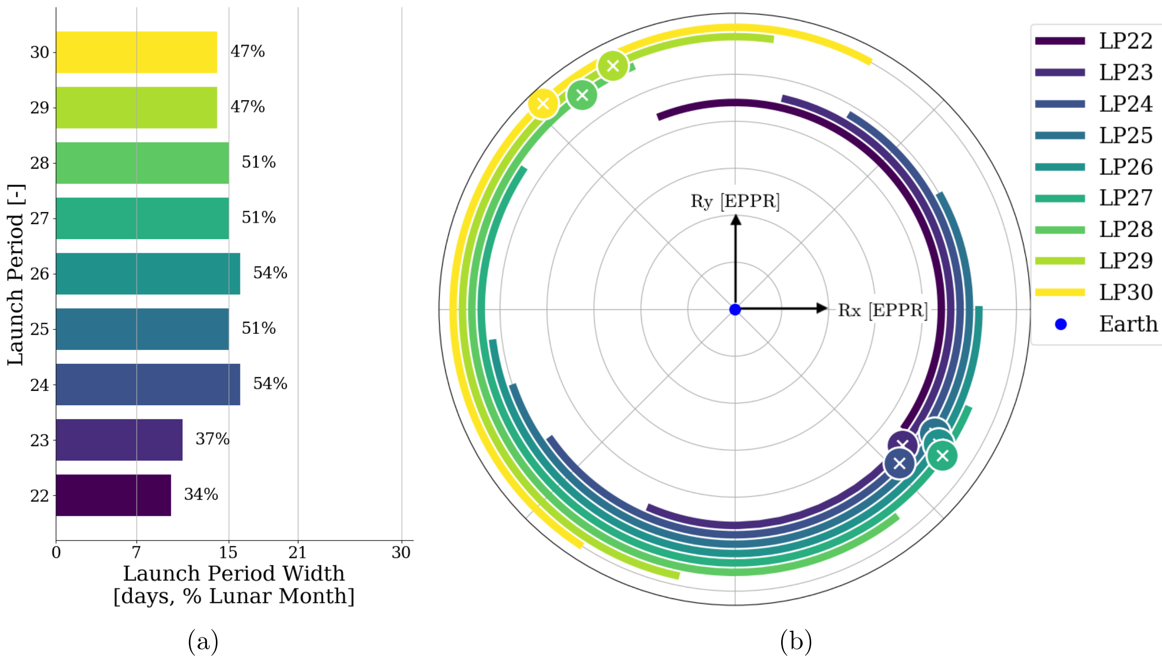


Figure 11. The Shift of the Day of Nominal Returns Denoted by LCA State in the Sun-Earth Frame with Comparisons between Launch Period Width.

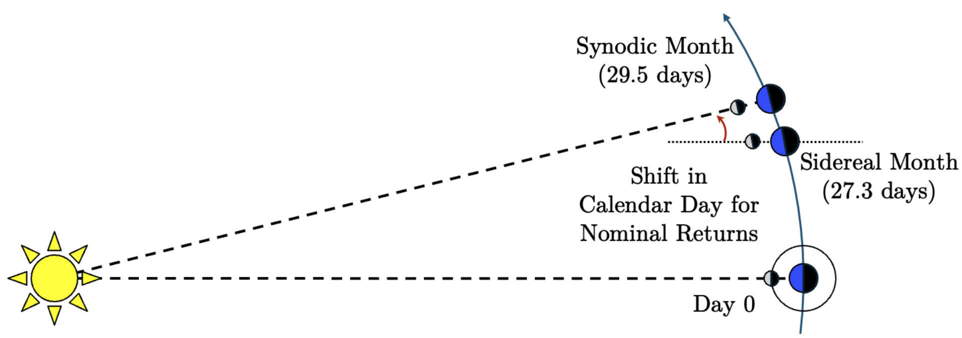


Figure 12. Comparison of Synodic and Sidereal Lunar Months.

The width of the Artemis I Launch periods is sufficient to overlap a single corridor, but not both corridors simultaneously (Figure 11). Notably, this is also why there are no nominal cases that return in LP22, it does not intersect any of the two corridors.

While there is typically only one day per Launch Period where nominal cases return for the Artemis I mission, the potential does exist for future missions where a sufficiently wide Launch Period could intersect both corridors, and thus, have multiple days of returns. Alternatively, if the Launch Period is sufficiently short, like LP22, there would be no days where ICPS would return nominally.

Between LP27 and LP28, the Launch Period traverses to a different corridor. This transition causes the relatively short 14-day period between the nominally returning days of LP27 and LP28.

This interaction should be examined for future missions like Artemis III. Since it is operationally desirable for SLS to have longer Launch Periods, the possibility might occur where the Artemis III mission creates a long enough launch period that overlaps both corridors, causing multiple days per Launch Period where ICPS returns, instead of just one.

5. PRELIMINARY ARTEMIS III COMPARISONS

Currently state data for Artemis III is scarce as analysis for this mission has just recently begun and a few Monte Carlo datasets have been simulated. Existing Artemis III data was processed using the same scripts used to explore the Artemis I data to find any cases with the same three-body periodic orbit imitation. An example case for Artemis III is provided as Figure 13. There are minor differences between the behavior of the Artemis I and Artemis III propagations, mainly attributable to the variations in the lunar flyby and the differences in the mission structure, i.e. Artemis I MPCV targeted a DRO whereas Artemis III MPCV targets an NRHO. However, the underlying physics guiding the return of the ICPS remains the same, indicating that Artemis III can possibly return via the mechanisms observed in Artemis I.

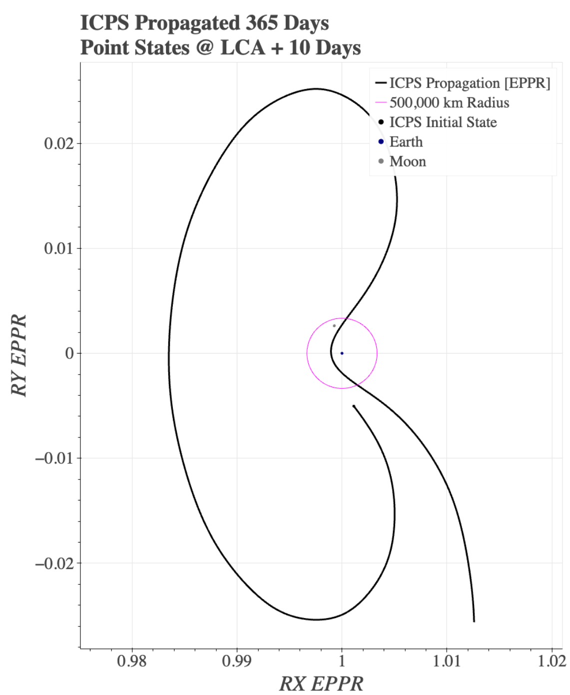


Figure 13. Initial Propagation of Artemis III Trajectories in Sun-Earth EPPR.

6. CONCLUSIONS

In conclusion, it was shown that the Artemis I ICPS will return to the Earth-Moon system under certain conditions, mainly driven by Sun-Earth geometry. These return trajectories mimic two three-body periodic orbits, namely Lyapunov and Distant Prograde Orbits. The ICPS mimics these orbits until sufficiently perturbed, resulting in a return into the cislunar system in less than a year, and are a small subset ($\approx 3.5\%$) of the overall analyzed set.

A summary of the ICPS return statistics was presented which showed the nominal return days occurred at a fixed cadence over a lunar month as well as set of nominal returns were contained in a single Launch Window. These return trajectories create corridors in the Sun-Earth Frame where they are likely to occur. These corridors provide a necessary but not sufficient condition to nominal return. The addition of Monte Carlo dispersions at the LCA+10 Day state provide sufficient perturbations which cause overlap across Launch Windows. Days with no nominal returns may return if the Monte Carlo state is sufficiently dispersed and vice versa. Also, the particular cadence of days where nominal cases return is due to a coupling between these corridors in the Sun-Earth Frame and the location of the Moon.

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