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EFFICIENT COMMISSIONING OF CIS-LUNAR CONSTELLATIONS: THE ROLE OF THE LUNAR STAGING ORBIT (LSO)

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As humanity's presence around the Moon expands, so too does the need for satellite infrastructure. This includes orbiting assets for communication, position, navigation, and timing (PNT), and space situational awareness (SSA). In order to ensure global coverage of the cis-lunar space a constellation of Spacecraft must work in tandem to provide the services needed for other missions to survive. The commissioning of these constellations is troublesome due to the relatively large maneuver required to insert into a lunar orbit and the large swath of phasing and spacing necessary to achieve the full coverage. The availability of heavy lift launch vehicles is greater than ever allowing the ability to launch multiple spacecraft at once. But coordinating each spacecraft within a constellation to perform the lunar orbit insertion burn (LOI) is complicated; especially prior to the establishment of the lunar infrastructure.

The proposed solution to this problem is an orbital transfer vehicle (OTV). An OTV thrives with delivering multiple payloads to multiple destinations that would otherwise require several individual launches and mission plans. Even so, the OTV has a limited capability so clever mission design solutions are necessary to maximize the mass delivered to multiple locations and still provide value to the payload customers. This paper defines the lunar staging orbit (LSO) as a method for an OTV to efficiently commission a constellation of eight communication spacecraft into four separate, elliptical frozen lunar orbits. The LSO utilizes the instability of the Earth-Moon dynamics to perform free changes in orbit parameters in a relatively short time frame of four months. The paper defines the parameters of the LSO as well as its benefits and drawbacks.

INTRODUCTION

The demand for a comprehensive satellite infrastructure is becoming increasingly critical as industry and governments focus on the Moon. Previous studies [1,2] have shown infrastructure is essential for facilitating communication, positioning, navigation, and timing (PNT), as well as for ensuring space situational awareness (SSA) in the burgeoning cis-lunar environment. To achieve effective global coverage, a constellation of spacecraft must operate in concert [3,4], providing vital services that will support current planned architecture as well as future evolutions.

However, the commissioning of these satellite constellations poses significant challenges. The high ΔV requirements for lunar orbit insertion complicates the deployment process, while the intricate phasing and spacing needed for full operational coverage adds layers of complexity. The availability of heavy-lift launch vehicles has expanded, enabling the simultaneous launch of multiple spacecraft [5]. The coordination required for each spacecraft's lunar orbit insertion (LOI) remains a formidable task, especially in the early stages of lunar infrastructure development.

To address these challenges, this paper proposes the use of an orbital transfer vehicle (OTV) also known as an orbital maneuvering vehicle (OMV). The OTV is designed to deliver multiple payloads to various destinations, which would otherwise necessitate multiple individual launches

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and mission plans. The use of OTVs has increased substantially since 2020 [5]. Effective mission design solutions are crucial to optimize the mass delivered to different locations while ensuring value for payload customers.

Central to this discussion is the concept of the lunar staging orbit (LSO), which serves as a strategic approach for the OTV to efficiently deploy a constellation of satellites in cis-lunar space. By leveraging the dynamic instability of the Earth-Moon system, the LSO allows for significant adjustments in orbit parameters within a condensed time frame. This paper will detail the parameters of the LSO, exploring its advantages and potential drawbacks, and will highlight its role in advancing lunar satellite infrastructure.

MISSION DEFINITION AND ASSUMPTIONS

In order to prove and assess the LSO, a conceptual mission must be defined. The primary goal of this conceptual mission is to provide continuous communication relay services between Earth and the south pole of the lunar surface based on the Artemis communication architecture concepts from [6].

Modeling Assumptions

This assessment was performed using a combination of the Copernicus and FreeFlyer astrodynamics tools. The Moon Mean Earth frame was used and inertialized at the J2000 epoch (MMEJ2K). This was used primarily because it is a common frame between the two tools as well as Systems Toolkit (STK). Earth and Moon gravity were modeled using 4×4 and 8×8 zonal and tesseral terms, respectively, with the Sun modeled as a point mass. No other forces were considered; future work could include higher-fidelity dynamics. The DE430 planetary ephemeris, EGM96 geopotential file, and GRGM660PRIM selenopotential file were used for planetary motion and dynamic models. The burns were modeled impulsively.

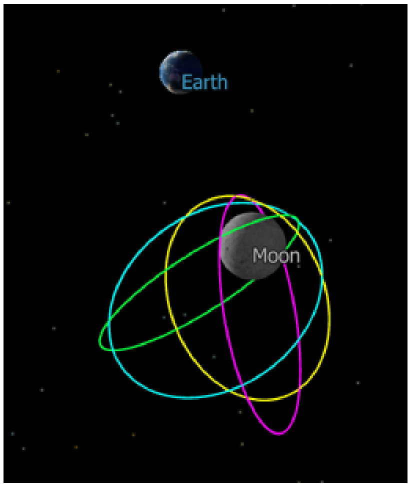

Figure 1. Visual of the elliptical frozen lunar orbit (EFLO) orbits

Target Orbit Definition

The final operating orbits for the constellation of eight spacecraft are four, separate 12-hour elliptical frozen lunar orbits (EFLOs) clocked from each other 90° in right ascension of the ascending node (Ω). Each orbits' apolune occurs over the southern hemisphere in order to increase dwell time and coverage over the south pole. The two spacecraft in each orbit are phased 180° apart from each other in true anomaly (θ). The orbits are shown in Figure 1. The specific orbital elements are defined in Table 1 where a is semi-major axis, e is eccentricity, i is inclination, ω is argument of periapsis in addition to θ and Ω defined above. These elements are derived from the EFLO analyzed in detail by [3].

Table 1. Elliptical frozen lunar orbit (EFLO) elements in the MMEJ2K frame

Orbit Name	a (km)	e (N/A)	i (deg)	Ω (deg)	ω (deg)	θ (deg)
EFLO-1	6140	0.57	57.0	48.8	90.0	$\theta, \quad \theta+180$
EFLO-2	6140	0.57	57.0	138.8	90.0	$\theta, \quad \theta+180$
EFLO-3	6140	0.57	57.0	228.8	90.0	$\theta, \quad \theta+180$
EFLO-4	6140	0.57	57.0	318.8	90.0	$\theta, \quad \theta+180$

Spacecraft Properties

For simplicity, standard payload sizes and accommodations will be used. The total stack mass of 6172 kg is comprised of two Stretched ESPA Grande rings [7] each weighing 286 kg with a payload capacity of up to 700 kg per each of the 8 available ports. The focus of the study is the LSO so the intricate details and considerations for hardware such as separation systems, center of gravity limits, payload volume in the fairing, etc. will not be considered, but could be easily added in future work.

The OTV will be assumed to have a total wet mass of 2000 kg, not including the ESPA rings, with 60% of the mass allocated as propellant and the remaining 40% being the dry mass. In this paper, the feasibility of the mission depends on the OTV having propellant remaining in addition to minimizing the ΔV-to-go for the spacecraft payloads. In order to assess propellant remaining, a simplified and conservative hydrazine system will be used with 10% of the total propellant marked as unusable to encompass performance margins, inaccessible propellant, and gauging uncertainties. The gravity constant to be used is 9.80665 m/s^2 and a specific impulse of 270 seconds.

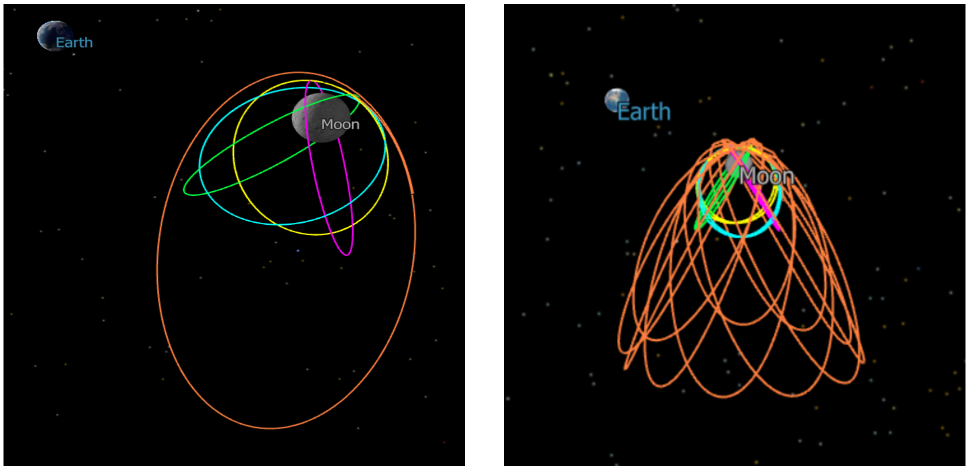

Figure 2. (Left) Visual of the initial Lunar Staging Orbit (LSO) with the EFLO orbits for reference. (Right) Visual of the propagated LSO with EFLO orbits in Body-Fixed frame.

LUNAR STAGING ORBIT DESIGN

The use of an OTV solves the issue of each spacecraft performing its own LOI but the problem of a 270° change in Ω still persists. The goal is to find an orbit to be used as a staging ground to deliver all of the spacecraft to the four separate orbits in a reasonable time frame (defined as 4 months for the purposes of this study). The final LSO will consider launch, transit, mission specific considerations, and disposal.

Lessons Learned from Sun-Synchronous Orbits

A significant portion of scientific and Earth imaging spacecraft in low Earth orbit use the Earth's oblateness to their benefit in the form of Sun-synchronous orbits (SSO). These are designed in such away to have their Ω change at the same rate that Earth orbits around the Sun so that they are synchronized with the Sun. To emphasize the point, the dynamics in these two problems cannot be directly compared but it would serve no good to ignore the lessons from decades of spaceflight in low Earth orbit.

The rate of change of Ω ($\dot{\Omega}$) can be solved for knowing the other orbit parameters [8]. The operative equation is given by:

$$\dot{\Omega} = -\frac{3}{2}J_2\left(\frac{R_E}{p}\right)^2 n * cos(i) \tag{1}$$

where J_2 is Earth's zonal harmonic coefficient, R_E is the radius of the Earth, p is the semi-latus rectum defined by $p = a(1-e)^2$, and n is the mean motion defined by $n = \sqrt{\mu/a^3}$ where μ is Earth's gravitational constant. What can be taken away from this is that $\dot{\Omega}$ is primarily a function of the a, e, and i. This provides a clue to the parameters to focus on to help calculate the desired LSO. Perhaps a future study can be performed to derive the equivalent equation(s) for the LSO but the approach herein is to perform a search of apolune radius' to achieve a 270° change of Ω within a 4 month time frame while minimizing any changes to the other orbital parameters.

Launch and Lunar Transit Design

The typical solution to delivering these spacecraft would be to have them rideshare on a launch vehicle. The payload stack mass of 6172 kg is within the capabilities of most commercial launch vehicle (CLV) providers to launch directly into a trans-lunar injection (TLI). The additional mass of the OTV would increase the total launch mass to 8172 kg. This is more limiting, but still within the capabilities of some current CLV providers.

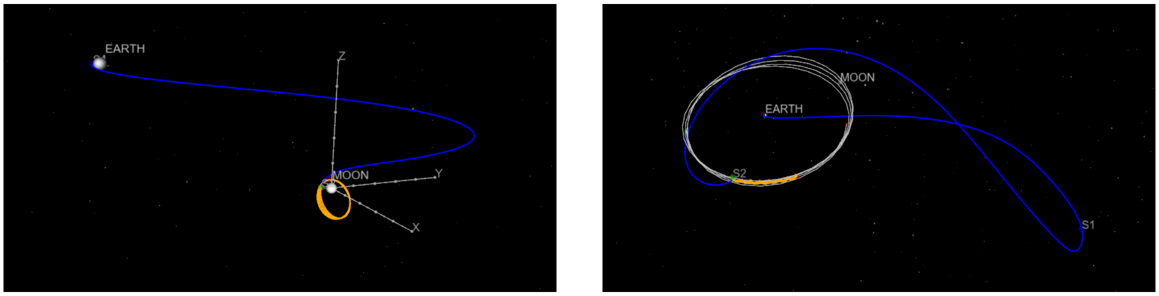

Figure 3. (Left) Direct Lunar Transfer into a 12-hour EFLO in the Earth-Moon 2-Body Rotating Frame and (Right) BLT into a 12-hour EFLO in the Sun-Earth 2-Body Rotating Frame.

For a launch rideshare scenario, the launch vehicle would propel the spacecraft up through TLI where each spacecraft is responsible for its own arrival into each of their respective orbits about the Moon. This is especially challenging for a direct lunar transfer (DLT) where there is a relatively short duration between TLI and lunar orbit insertion (LOI) to perform any meaningful trajectory changes to inject into separate orbits. Because of this unfortunate geometry, multiple launches may be necessary in order to fully commission the constellation. Not to mention that each spacecraft is individually performing the full LOI themselves of 232 or 453 m/s for a ballistic lunar transfer (BLT) and DLT respectively.

A BLT offers more flexibility to perform trajectory changes and insert into different orbits. However, a solution to insert into a swath of orbits spanning 270° of Ω is still likely out of the capability of the spacecraft when considering they each have to perform the LOI insertion independently. Figure 3 shows both the direct and ballistic lunar transfer trajectories from TLI, represented in blue, direct into one of the target orbits, represented in orange.

Lunar Staging Orbit Definition

From the SSO lessons learned, the primary orbit parameters to focus on should be a, e, and i. The other considerations from the ground rules are ΔV-to-go, mission duration, and heliocentric disposal. In order to minimize the ΔV-to-go for the payload spacecraft, the LSO should be as co-planar as possible with the target orbits. This drives an initial guess of $i = 57°$.

With i effectively fixed, that leaves a and e as the primary independent variables to maximize $\dot{\Omega}$. An alternative parameterization of a and e is periapsis and apoapsis. To simplify the orbit transfer from the LSO to the target orbit, ω and periapsis (or perilune in this case) can be fixed to match that of the target orbits. This leaves apoapsis (or apolune) as the final independent variable simplifying the trade space. The final orbit is tabulated in Table 2 and shown in Figure 2.

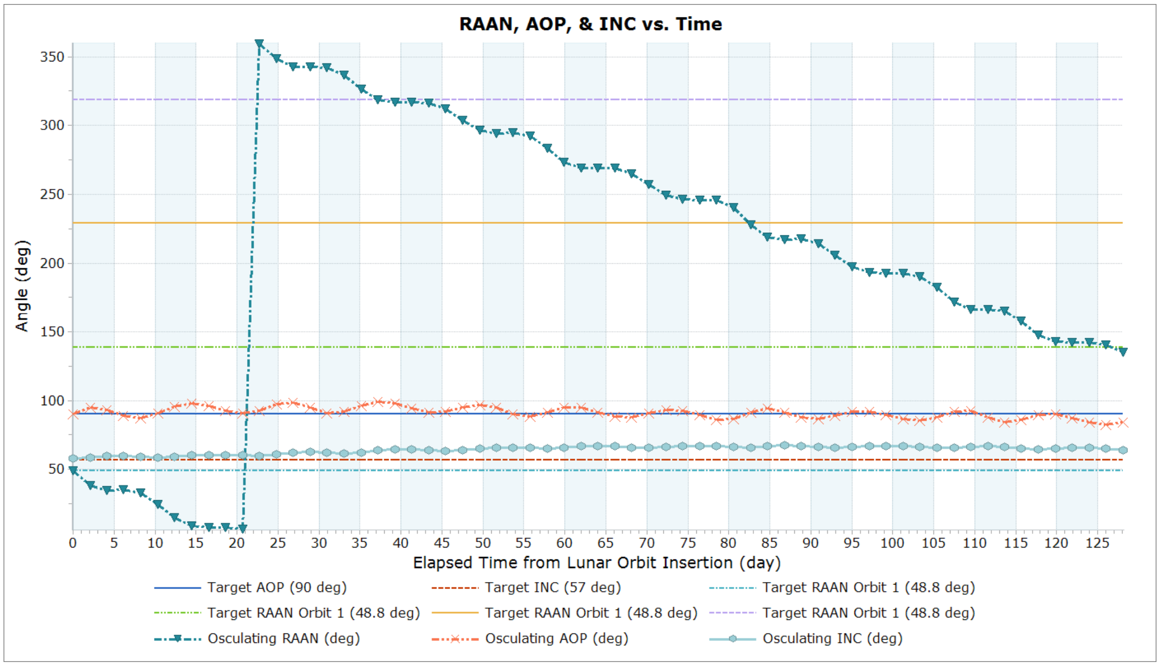

Figure 4. Trend of the osculating elements RAAN, AOP, and INC of the LSO over time without stationkeeping in the MMEJ2K frame

Figure 4 shows how the osculating orbital elements of the initial LSO trend over time without any maneuvers. Here we can see the 270° change in Ω for free within the notional 4 month time frame. The trend also shows how ω and i vary throughout the 4 month period which will be discussed in more detail in the results.

Table 2. Initial Lunar Staging Orbit (LSO) elements in the MMEJ2K inertial frame following LOI

Perilune Radius (km)	Apolune Radius (km)	i (deg)	Ω (deg)	ω (deg)
2640	29440	57.0	48.8	90.0

RESULTS

The proposed mission architecture using the Lunar Staging Orbit (LSO) and a ballistic lunar transfer (BLT) was evaluated for its ability to commission a constellation of eight spacecraft into four separate elliptical frozen lunar orbits (EFLOs) within the defined mission constraints. The results confirm that the concept is both feasible and efficient, offering significant advantages over conventional deployment methods.

RAAN Precession and Target Orbit Acquisition

The LSO design successfully exploited the natural instability of the Earth-Moon dynamics to facilitate large changes in Ω. Figure 5 varies from Figure 4 in that it includes the target orbit acquisition (TOA) maneuvers the OTV performs to help minimize the ΔV-to-go for the payload spacecraft. The operational version of the LSO achieved the required 270° Ω precession, but within approximately 6 months instead of 4. This allowed sequential deployment of payload pairs into their respective target EFLOs without the need for high ΔV plane change maneuvers.

Each deployment event was broadly timed to coincide with the desired Ω. The TOAs for a given EFLO were designed as a single burn to simultaneously target the i, Ω, ω, and perilune by varying the orbit timing and the three maneuver degrees of freedom. More performance can be gained from exploring different station keeping methodologies or more fine tuning of the TOA maneuvers.

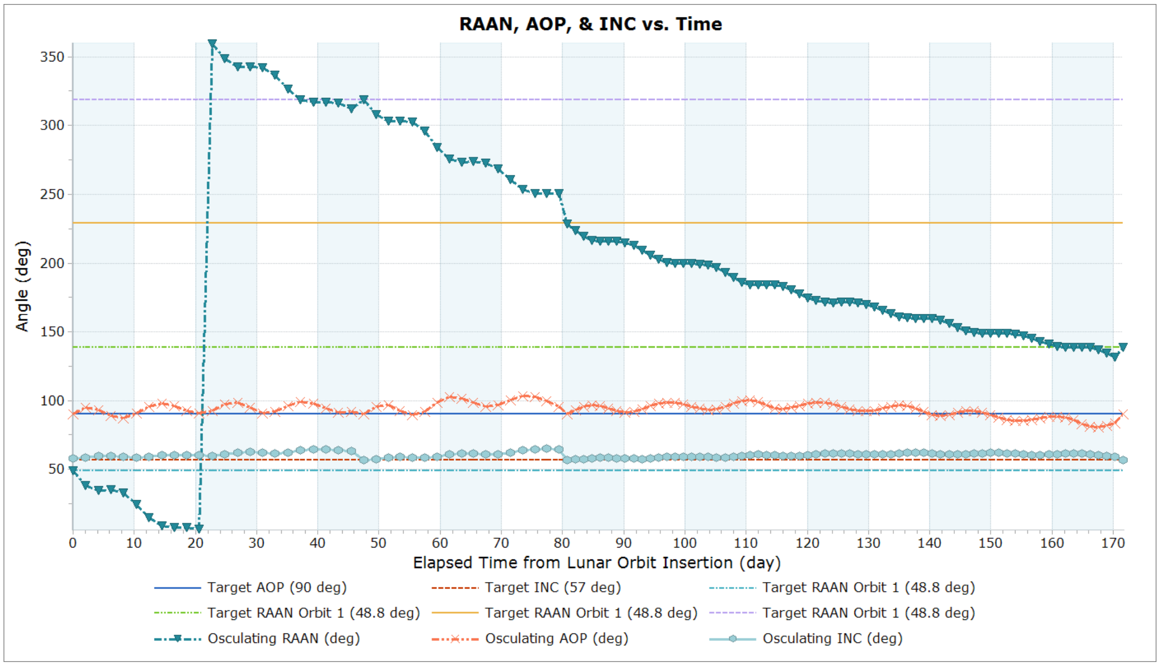

Figure 5. RAAN, AOP, and INC of the mission including Trajectory Correction Maneuvers (TCMs) in the MMEJ2K frame.

Through the design process, it was discovered that ω change was a significant contributor in ΔV. This can be seen in the ΔV optimized TOA maneuver performed for EFLO-3 (around the 80 day mark) that traded a larger change in Ω for a reduced change in ω and i. Again, pivoting to a two or even three burn maneuver scheme could result in more optimal solutions. Even so, this work proves feasibility of such a mission design that can only improve with more study.

Propellant Usage and Mission Closure

The ΔV expenditures for the OTV and payload spacecraft were tracked throughout the mission. Table 3 tabulates the sequence of events and cumulative ΔV used by the OTV for lunar orbit insertion (LOI), stationkeeping within the LSO, and individual deployment maneuvers to each EFLO required by the payload spacecraft. Total OTV propellant usage remained within capacity, with approximately 56 kg of usable propellant remaining at end-of-mission. Note that two, 700 kg spacecraft are separated at each EFLO which can be inferred by the progressively reduced propellant consumption throughout the mission.

For the payload spacecraft, the required, deterministic ΔV-to-go from the LSO to their final EFLOs was minimal, averaging approximately 120 m/s per spacecraft. This does not consider any TCMs following OTV separation or phasing. This marks a substantial reduction compared to the baseline scenario where each spacecraft performs LOI independently.

Table 3. Mission sequence of events

Event Phase	OTV Manevuer ΔV (m/s)	OTV ACS ΔV (m/s)	OTV Prop Used (kg)	Payload ΔV-to-go (m/s)
Launch to EFLO-1	75.0	30	-318	138.6
EFLO-1 to EFLO-4	65.8	15	-194	136.1
EFLO-4 to EFLO-3	151.0	15	-295	110.3
EFLO-3 to EFLO-2	89.3	30	-139	95.2
EFLO-2 to Disposal	124.7	5	-78	N/A
Total	**505.8 m/s**	**95 m/s**	**Usable Prop: 56 kg**	**Mean: 120.0**

Deployment Timeline

The deployment timeline is initialized with 0 being when LOI occurs into the LSO. The first payload deployments into EFLO-1 occurred within the first orbit following LOI. The other payload deployments into EFLO-4, EFLO-3, and EFLO-2 occur around days 47, 80, and 172 respectively. The initial LSO was sized with the intention to deploy the satellites within 4 months as was shown in the pure propagation shown in Figure 4. The operational LSO however delivered the final payloads within 5.75 months of lunar arrival. This did not meet the operational goal of commissioning the entire constellation within four months but further optimization of the TOA maneuvers could make performance available for more stringent station keeping that could maintain the orbital elements closer to the initial LSO in Table 2.

Operational Benefits

Overall, the use of the LSO enabled the deployment of all eight spacecraft using a single heavy-lift launch, centralized propulsion and deployment operations via the OTV, reducing spacecraft complexity, significant reduction in ΔV requirements for payload spacecraft, a flexible, modular commissioning approach adaptable to other constellation architectures.

These results validate the LSO as an effective strategy for commissioning cis-lunar constellations, particularly in the early stages of lunar infrastructure development when distributed propulsion capability may be limited.

CONCLUSION

This study demonstrates that the Lunar Staging Orbit (LSO) is a viable and efficient strategy for commissioning a cis-lunar satellite constellation using a single launch and orbital transfer vehicle (OTV). By capitalizing on the natural precession dynamics of a highly elliptical lunar orbit, the LSO enables large changes in right ascension of the ascending node (Ω) over a relatively short period—allowing sequential deployment of spacecraft into multiple orbital planes without requiring excessive maneuvering or multiple launches.

The notional mission architecture validated in this work successfully delivered eight spacecraft into four distinct elliptical frozen lunar orbits (EFLOs) with an average payload ΔV of only 120 m/s and approximately 5% of OTV propellant remaining above margins at end-of-mission. While the initial goal of completing deployment within four months was exceeded slightly, the six-month timeline remains operationally feasible, and future refinement of trajectory correction and station-keeping strategies is expected to close this gap.

The LSO approach reduces total mission ΔV, enables centralized deployment from a shared propulsion system, and minimizes complexity for individual spacecraft. Furthermore, the flexibility and adaptability of this architecture make it an attractive option for future cis-lunar communication, navigation, and science constellations.

Continued development of this concept, including higher-fidelity dynamic modeling, optimization of deployment sequences, and system-level integration with emerging launch and OTV capabilities, can further enhance its practicality and performance. As the cis-lunar economy matures, architectures like the LSO will be critical enablers of sustainable and scalable infrastructure.

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