SPACE LAUNCH SYSTEM DEPARTURE TRAJECTORY ANALYSIS FOR CISLUNAR AND DEEP-SPACE EXPLORATION

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The Space Launch System will insert Orion into different orbits for Artemis I and Artemis II. The Artemis program has considerations beyond the immediate mission of inserting Orion into its desired trajectory. Primarily, following separation from Orion, the Interim Cryogenic Propulsion Stage must be safely disposed, and another is that secondary payloads will be deployed only after Orion separation to ensure safety of the primary mission. The first consideration (ICPS disposal) constrains the latter (secondary payload trajectories). In this paper, we give an overview of the constraints and opportunities provided by Artemis missions for secondary payloads within the Earth-Moon system and beyond.

INTRODUCTION

For Artemis I, the Space Launch System (SLS) will insert Orion into a Distant Retrograde Orbit (DRO) (see Figure 1). Following the injection of Orion into the Outbound Transit Orbit (see Figure 1), the SLS Interim Cryogenic Propulsion Stage (ICPS) will perform a disposal maneuver to target a heliocentric orbit.

The purpose of the disposal maneuver will be to target a lunar flyby to add energy to enable escape from the Earth-Moon system. There are 13 secondary payloads manifested on SLS, and these will be ejected at regular intervals called Bus Stops (see Figure 2). There will be navigational dispersions on the ICPS disposal trajectory state vector as well as additional delta-V imparted to the secondary payloads by the deployer mechanism. Secondary payloads that wish to remain in the Earth-Moon system must possess a propulsive capability. Additionally secondaries can use propulsive capability to target a different lunar flyby than the one ICPS is targeting.

Artemis II will be a crewed spaceflight mission that inserts into a free return circumlunar trajectory for crew safety (see Figure 3). Artemis II will also perform a heliocentric disposal of the ICPS, but alternate scenarios, including Earth disposal, are under discussion. Additionally, details of secondary payload opportunities are also less defined at the time of this paper. The number of payloads, size and mass of payload allowed, deployment times and other details have yet to be completely defined. For the purposes of this paper, we will assume the nominal heliocentric disposal of the ICPS but also consider the implications of Earth disposal. ICPS will

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separate from Orion in a high elliptical orbit farther from the Moon than Artemis I, which is another change that must be considered.

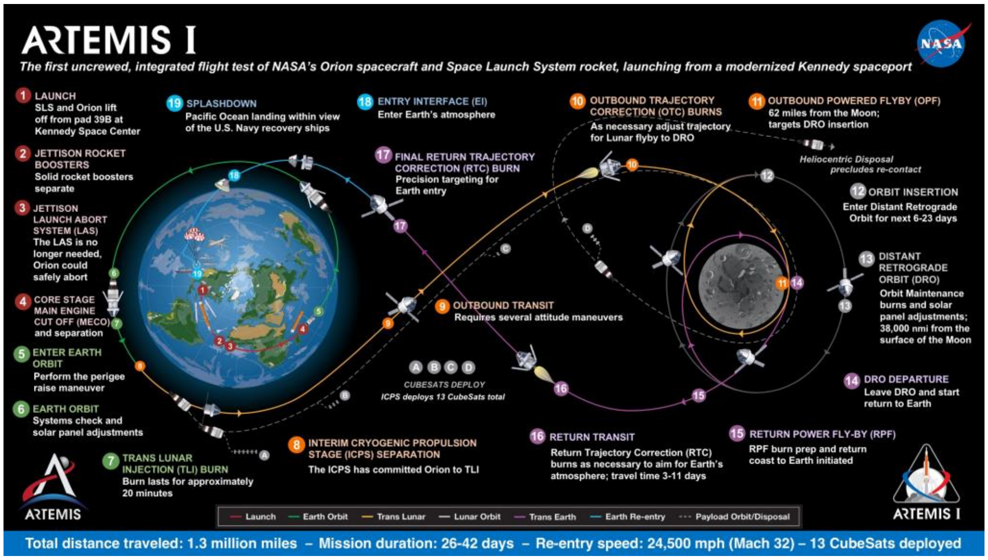

Figure 1. Artemis I Trajectory

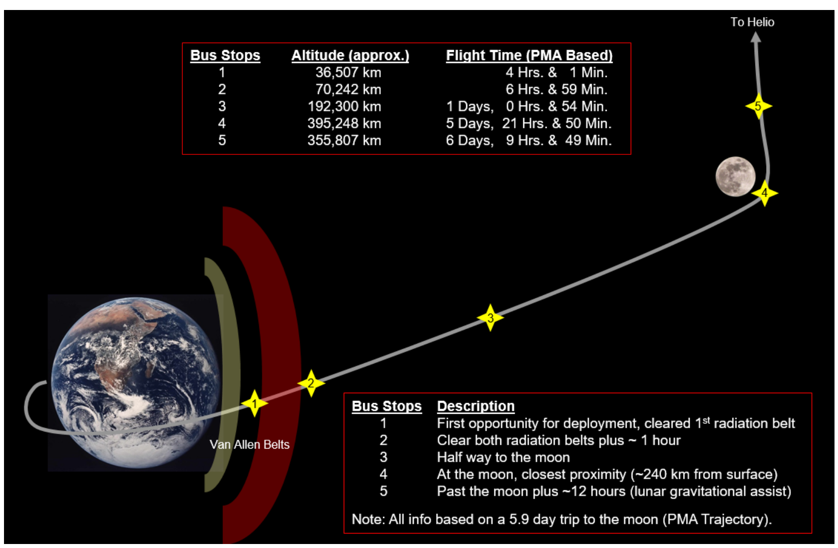

Figure 2. Artemis I Secondary Payload Bus Stops

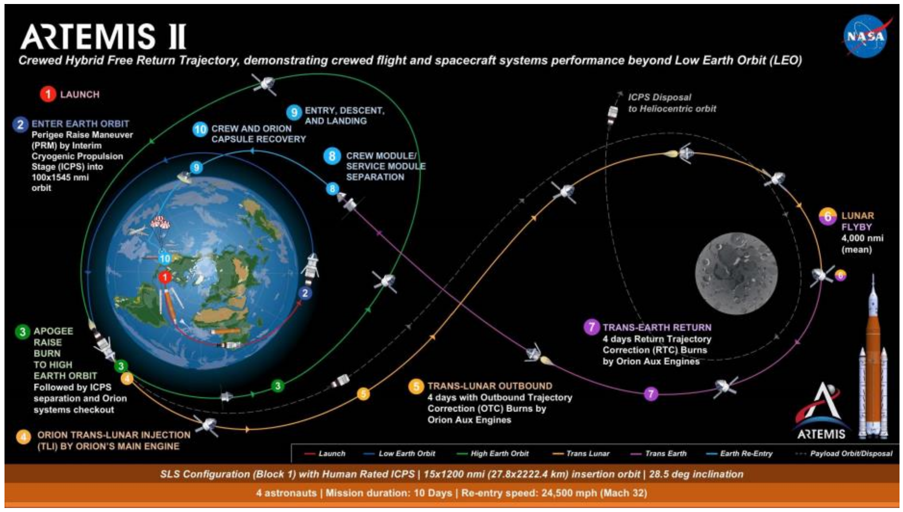

Figure 3. Artemis II Trajectory

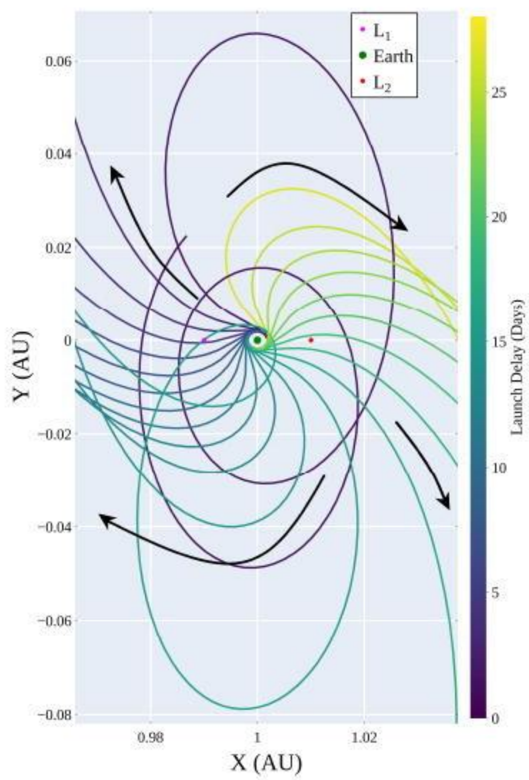

Figure 4. Departure Trajectories as a Function of Delay from Launch Date for One Lunar Month as Viewed in the Sun-Earth Rotating Frame Centered about the Earth

ARTEMIS I

A study to support the Artemis I secondary payload Near Earth Asteroid (NEA) Scout demonstrated that the range of possible heliocentric departures (leading vs. trailing) is a strong function of launch date. Figure 4 depicts various departure trajectories from the Earth-Moon system at different departure angles when viewed in the Sun-Earth rotating frame. Figure 4 essentially captures the monthly variation caused by launch dates for Artemis I in which the secondary payloads follow cost-effective trajectories that enable them to either depart or stay within the Earth-Moon system without additional propellant requirements. One of the conclusions of the NEA Scout study is that departures along and opposed to Earth's velocity vector will be more efficient at achieving rendezvous orbits with near-Earth asteroids. Here, we determine if this result is more generally applicable to other rendezous orbits such as Mars and Venus as well as other potential missions.

Although not all SLS data for Artemis I can be released to the public, the SLS program did release an initial state vector for Artemis I to the secondary payloads. This trajectory appears in Table 1. For Artemis I, SLS carries the Orion spacecraft into orbit, then performs a Trans Lunar Injection (TLI) burn as depicted in Figure 1. After the TLI, Orion separates from the SLS Interim Cryogenic Propulsion Stage (ICPS) and the ICPS performs a small disposal burn. The purpose of the disposal burn is to target a Lunar Gravity Assist (LGA) that sends the ICPS into a heliocentric orbit.

Table 1: Post Disposal Nominal State Vectors in J2000

Parameter	Value
Julian Date (UTC)	2458399.1507421
Semi Major Axis (km)	206959.1154
Eccentricity	0.966351726
Inclination (deg)	28.26180851
RAAN (deg)	35.37723043
Argument of Perigee (deg)	41.23445507
True Anomaly (deg)	118.0730252

Although the trajectory in Table 1 is for an October 7 2018 launch date, it still is representative of later launch dates and can be used to draw general conclusions about the Artemis I insertion state. First, Artemis I generally will try to insert into lunar orbit when the moon is near perigee, which occurs monthly, thus the Artemis I launch window recurs on a monthly basis. Second, the energy of the orbit transfering Orion from Low Earth Orbit (LEO) to the moon will generally have a simliar energy for the nominal midpoint of the launch window from month to month, since lunar perigee changes very slowly due to solar and other perturbations. Third, since the launch window is correlated on a monthly basis, in a given year's worth of launch windows the inertial direction of Artemis I will appear to rotate with respect to the Earth-Sun line as depicted in Figure 4.

After the flyby of the moon to achieve the LGA, the ICPS will nominally have an excess hyperbolic velocity (Vinf) of 0. 382 km/sec and other parameters as shown in Table 2. Thus any secondary payload on Artemis I that does not have a propulsive capability will achieve this state, plus a small Delta-V from the cubesat deployer that is on the order of ~1.5 m/s. The dispersion of the velocities imparted by the cubesat deployer including any uncertainty in the direction of that Delta-V is beyond the scope of this paper, but could easily be included. However the nominal post-LGA ICPS state vector in Table 2 is still generally representative of a state achieved by any secondary lacking propulsive capability after the lunar flyby.

Table 2: Post LGA Nominal State Vectors in Earth-Centered J2000

Parameter	Value
Julian Date (UTC)	2458411.1507421
Outbound Vinf (km/sec)	0.3381687400
Right Ascension of Vinf	106.2250834
Declination of Vinf (deg)	26.72177403
RAAN (deg)	29.18910466
Radius of Perigee (km)	293593.7486
True Anomaly (deg)	76.552880537

Having presented two state vectors of secondary payloads, one prior to the LGA and one after, we can now address three options that secondary payloads have on Artemis I to achieve missions of interest, and also address how realistic some of those options can be for the 6U Cubesats carried by Artemis I.

There are three main options for the secondaries, two of which assume a propulsive capability by the secondary:

1. To accept the post-LGA state provided by ICPS (ignoring variations from the Delta-V imparted by the deployer mechanism). This is the only option available for those secondaries without propulsive capability.

2. The secondary can perform a manuever prior to or during the LGA to either (a) insert into orbit around the moon, (b) fly a more distantly from the moon (thus either lowering the C3 of the escape trajectory or actually staying in the Earth-Moon system), or (c) fly closer to the moon to increase the energy imparted by the LGA (thus increasing C3).

3. Accept the LGA as is, and begin manueuvering after the LGA. This might be the option selected if the ICPS disposal trajectory is acceptable as a starting point for the secondary, or if constraints on the secondary do not allow for an earlier manuever prior to the LGA.

For each of the above cases, various missions are feasible even given the mass and volume constraints on the 6U cubesats carried by Artemis I. There are secondary payloads currently on the manifest in each category above, which will be discussed and then other potential mission options that could be exercised for the Artemis I mission as an example for future potential secondary payloads.

In the first category of a spacecraft having no propulsive capability is the Biosentinel mission. Biosentinel's only trajectory requirement is to escape from the Earth's protective magnetic field to assess the effects of deep space radiation on biological samples. Thus this is not a particularly interesting case, since any ICPS trajectory for any launch date will escape to heliocentric orbit (if the ICPS disposal burn is successful). However, if Biosentinal (or a hypothetical other mission) discriminated between Earth-trailing or Earth-leading heliocentric orbits then launch date would matter.

Numerous secondary payloads manifested on Artemis I are planning to insert into lunar orbit, which is only logical since Artemis I is a lunar mission. One example in this category is Lunar Flashlight. Lunar Flashlight plans to use a green propellant propulsion system to insert into lunar orbit. The Lunar Flashlight propulsion system will provide 290 m/sec of delta-V. That amount of delta-V combined with repeated flybys of the moon will be enough to insert Lunar Flashlight into a lunar orbit. Lunar Flashlight is one example of a secondary payload inserted into lunar orbit using it's own propulsive capability.

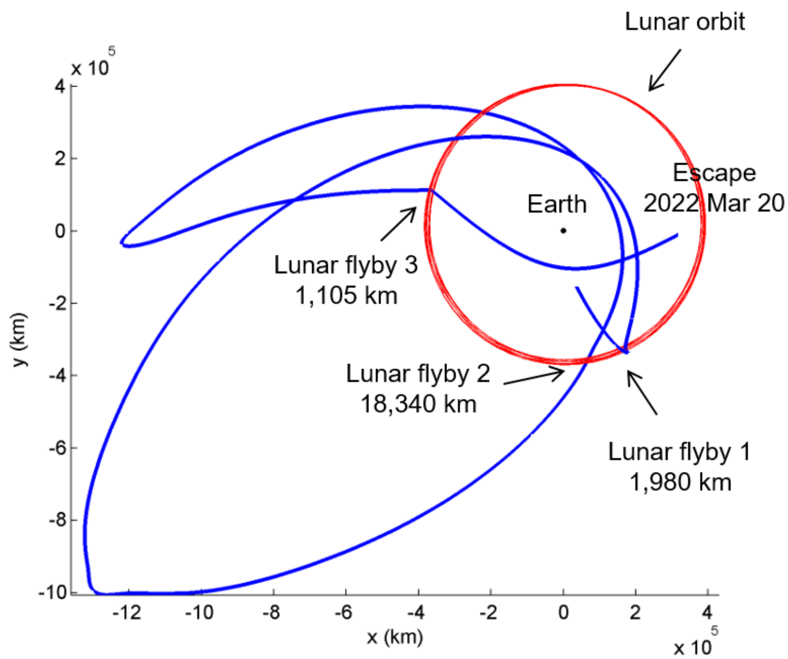

Figure 5 - Lunar escape trajectory (Earth-centered inertial frame)

The third category is when a secondary payload can provide propulsive capability after the LGA. This propulsive capability can be used to leave the Earth-Moon system to targets in heliocentric space. In this category is the Near Earth Asteroid Scout, or NEA Scout. NEA Scout will use a solar sail deployed after the first LGA to target subsequent LGAs in order to escape the Earth-Moon system in the correct direction and with enough energy to rendezvous with a Near Earth Asteroid within 2 years. Although uncertainty in the Artemis I launch window means that the NEA Scout mission must frequently update what asteroid it will target (due to orbital phasing issues), the July 2020 Design Reference Mission (DRM) is depicted in Figure 5.

Figure 5 illustrates that escape from the Earth-Moon system can be accomplished with a series of LGAs, thus targeting destinations in heliocentric space. Other destinations such as Mars or Venus might also be attractive for future small sat secondary payloads.

A quick assessment of the feasibility of a secondary payload mission to Mars from an Artemis I orbit can be done. Following the approach of NEA Scout of using a solar sail, a departure v-inf of up to 1.5 km/sec is possible which is a significant portion of the Vinf necessary to reach Mars. With a NEA Scout sized sail, it would be possible to do a fly by of Mars in approximately 4.5 years, assuming that the full 1.5 km/sec of departure Vinf is available. This is not a good time of flight for a mission to Mars, but it does show that it's possible.

ARTEMIS II

At the time of the writing of this paper, Artemis II secondary payload opportunities are less well defined than for Artemis I. However the most likely scenario is still that the Artemis II ICPS will be disposed heliocentrically, although Earth disposal is a possibility. Earth disposal is a possibility for Artemis II due to differences in the orbits of Artemis I and II. However, since at time of publication heliocentric disposal is still the baseline, we will assume a heliocentric disposal for Artemis II.

Thus Artemis II secondary payloads will face many of the same choices as they do for Artemis I. Artemis II will feature 12U cubesats in place of the 6U cubesats that will fly on Artemis I. Therefore, Artemis II secondary payloads should be able to achieve greater propulsive capabilities and thus have access to a wider range of missions.

As shown in Figure 4, the time of year at which Artemis I launches greatly affects the characteristics of the departure orbit for the heliocentric disposal. Specifically the time of year affects the direction of the departure orbit. This will also be true for Artemis II.

A few example cases can show what is possible with the greater volume and mass allowed for secondaries by Artemis II. First consider a secondary payload similar to Lunar Flashlight that can now carry more propellant. If we assume that for a 24 kg "Lunar Flashlight-like" cubesat that a total of 10 kg can be propellant, then the Delta-V possible increases to 1980 m/sec. With that amount of Delta-V combined with a Vinf of departure possible from LGAs, the upgraded Lunar Flashlight propulsive capability could allow a flyby of Mars or Venus, though not a rendezvous.

The higher volume allowed by the 12U cubesat could also allow a larger solar sail for a "NEA Scout-like" mission. In this case, time of flight to the asteroids of interest could be greatly reduced. For instance, for a flight to asteroid 2019GF, the time of flight is reduced from 563 days to 413 days, a significant improvement.

Thus mission design for Artemis II secondary payloads does appear promising if the greater volume and mass allowed by the secondary payloads is used to increase the propulsive capability of the secondary payloads. It should be easier for the secondary payloads on Artemis II to achieve mission objectives if the heliocentric ICPS disposal is followed for Artemis II.

CONCLUSIONS

For secondary payloads manifested on Artemis I, having a propulsive capability greatly enhances the probability of mission success for lunar orbiters and deep space secondaries. However for deep space secondary payloads the time of year that Artemis I is launched will have a major effect on the departure Vinf, thus constraining some launch times to be better than others.

For secondary payloads manifested on Artemis II, assuming that Artemis II uses heliocentric disposal for the ICPS, the secondaries will face similar mission design choices to those faced by the secondaries on Artemis I. Secondary payloads on Artemis II will be able to include more propulsive capability if desired due to the higher mass and volume allowed by 12U cubesats. This increase in propulsive capability will allow greater ability to achieve mission objectives and also increase probability of mission success.

The opportunity to include secondary payloads on Artemis creates opportunities for a variety of missions. These missions must include the constraints imposed by Artemis, including launch window and trajectory constraints that are driven by the primary Artemis mission requirements. In this paper we have presented some strategies to optimize the mission opportunities and mission success of secondary payloads on the Artemis missions.

REFERENCES

K.F. Robinson and G. Norris, "Space Launch System: Deep-Space Delivery for Smallsats" AIAA Space and Astronautics Forum and Exposition, September 2017, Orlando, FL.

J. B. Pezent, R. Sood, and A. F. Heaton, "Contingency Target Analysis, Trajectory Design, and Analysis for NASA's NEA Scout Solar Sail Mission" Advances in Space Research, accepted for future publication, available online February 17, 2020.

Ricoo, Tony, et al, "BioSentinel: DNA Damage-and-Repair Experiment Beyond Low Earth Orbit", NSTRS 20190001657.

Andrews, Dawn, _ Lightsey, E.G, "Design of a Green Monopropellant Propulsion System for the Lunar Flashlight Mission", Special Georgia Tech Project AE 8000 Report, Dec, 2019.

Cohen, B., "Lunar Flashlight: Illuminating the Moon's South Pole", GSFC Planetary Cubesat/Small Sat Symposium, June 2012, Greenbelt, MD.