

SYNCHRONIZED LUNAR POLE IMPACT & PLUME SAMPLE RETURN TRAJECTORY DESIGN

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The presented trajectory design enables two maneuverable spacecraft launched onto the same trans-lunar injection trajectory to coordinate a steep impact of a lunar pole and subsequent sample return of the ejecta plume to Earth. To demonstrate this concept, the impactor is assumed to use the LCROSS mission's trajectory and spacecraft architecture, thus the permanently-shadowed Cabeus crater on the lunar south pole is assumed as the impact site. The sample-return spacecraft is assumed to be a CubeSat that requires a complimentary trajectory design that avoids lunar impact after passing through the ejecta plume to enable sample-return to Earth via atmospheric entry.

INTRODUCTION

This paper presents a trajectory design inspired by the Lunar Crater Observation and Sensing Satellite (LCROSS) mission¹ that enables a lunar sample return (to Earth) from one of the Moon's polar regions that contain water-ice in permanently shadowed craters.

The architecture assumes an ejecta plume is created by a steep (greater than 60°) and fast (at least 2.5 km/s) impact of a lunar pole by a centaur upper stage that is guided (i.e., "shepherded") by a spacecraft, as required and flown by the LCROSS mission. The plume sample collection is assumed to occur by a CubeSat that would be deployed from the shepherding spacecraft shortly after trans-lunar injection (TLI) and must fly through the ejecta plume at a low altitude (100 km or lower) and return a sample to Earth. Unlike the LCROSS Shepherding spacecraft that flew through the centaur upper stage's plume and impacted the Moon directly thereafter, the presented CubeSat's trajectory must collect a sample and avoid a lunar impact to allow its return to Earth.

The above requirements can be achieved if the perilune location of the CubeSat's orbit is above the impact plume site and low enough to collect the sample shortly after the impact occurs (i.e., minutes). This orbital geometry is demonstrated via the presented trajectory designs, performed within the Systems Tool Kit (STK) Astrogator module, which utilized an N-body force model and high-order Runge-Kutta numerical integrator. For comparison, other solutions that utilize a single maneuverable spacecraft with a deployable impact mass are presented as well.

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TRAJECTORY DESIGN

The presented trajectory utilizes the LCROSS trajectory for the centaur-created impact plume at the lunar south pole, thus the starting orbit for all spacecraft is that of the Lunar Reconnaissance Orbiter (LRO) spacecraft, i.e., a post-TLI transfer trajectory launched out of the Cape Canaveral Air Force Station (CCAFS) (Fig. 1, A). Several hours after the TLI, LCROSS would separate from the primary payload, LRO (Fig. 1, B) and shortly thereafter the Shepherding spacecraft would deploy a propelled CubeSat (Fig. 1, B').

This follow-up mission to LCROSS would then require the guided centaur upper stage to perform a 6 m/s (of ΔV) maneuver (Fig. 1, C) to target a south pole lunar flyby (Fig. 1, D) while the sample-collecting and sample-return CubeSat would perform its first maneuver, requiring 78 m/s of ΔV (Fig. 1, C') to target a lunar north pole flyby (Fig. 1, D'). These lunar flybys, both unpowered, yield two opposing-motion, high-energy, highly inclined planes with respect to the lunar orbit plane. Thus both spacecraft are able to perform steep (85°) and fast (~ 2.5 km/s) approaches of the Moon from opposite directions (Fig. 1, E and E'). This unique geometry enables a lunar surface sample to be returned to Earth without capturing into an orbit around the Moon.

Other views of the presented trajectory are shown in Fig. 2, with the plume creation and collection phase shown in Fig. 3.

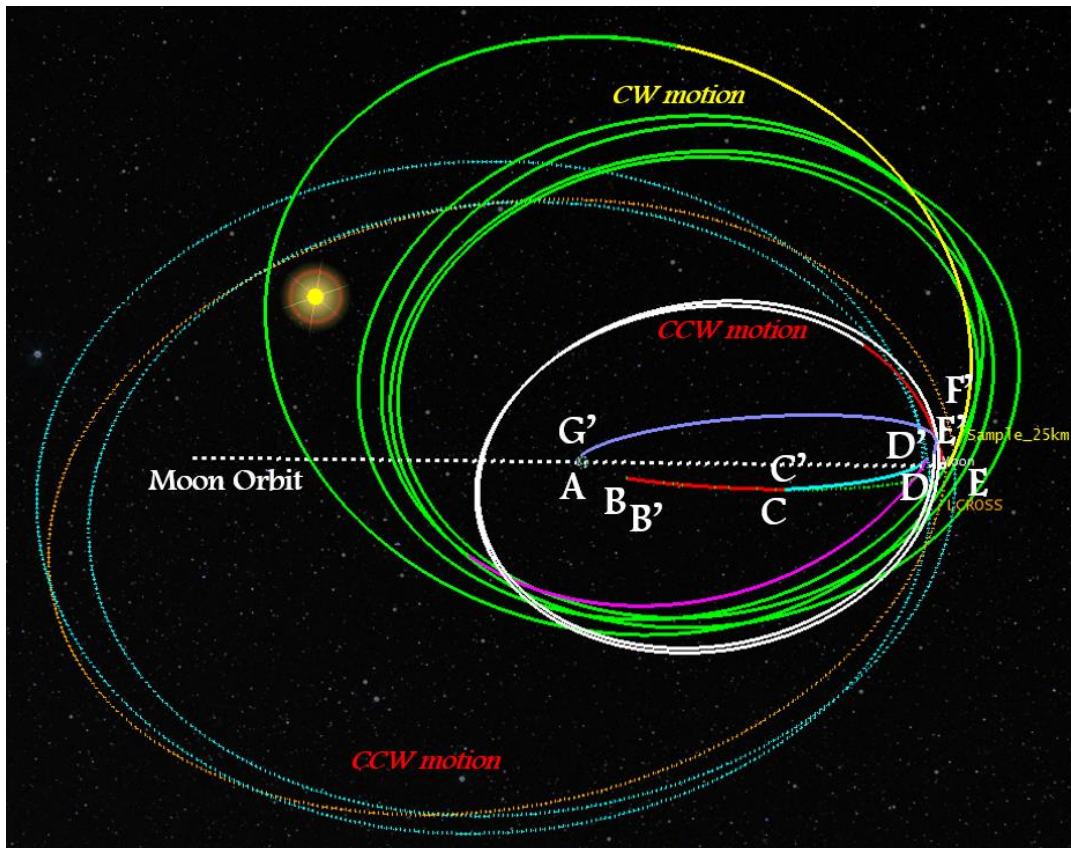


Figure 1. Centaur upper stage orbit (dotted) and sample-return spacecraft orbit (solid) shown in Earth inertial frame, with view edge-on the lunar orbit plane.

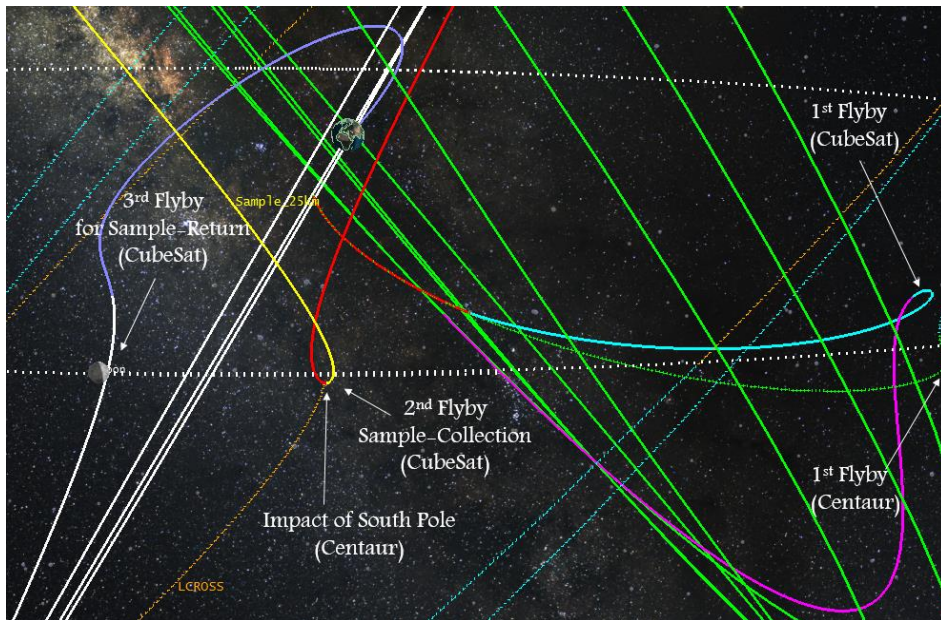
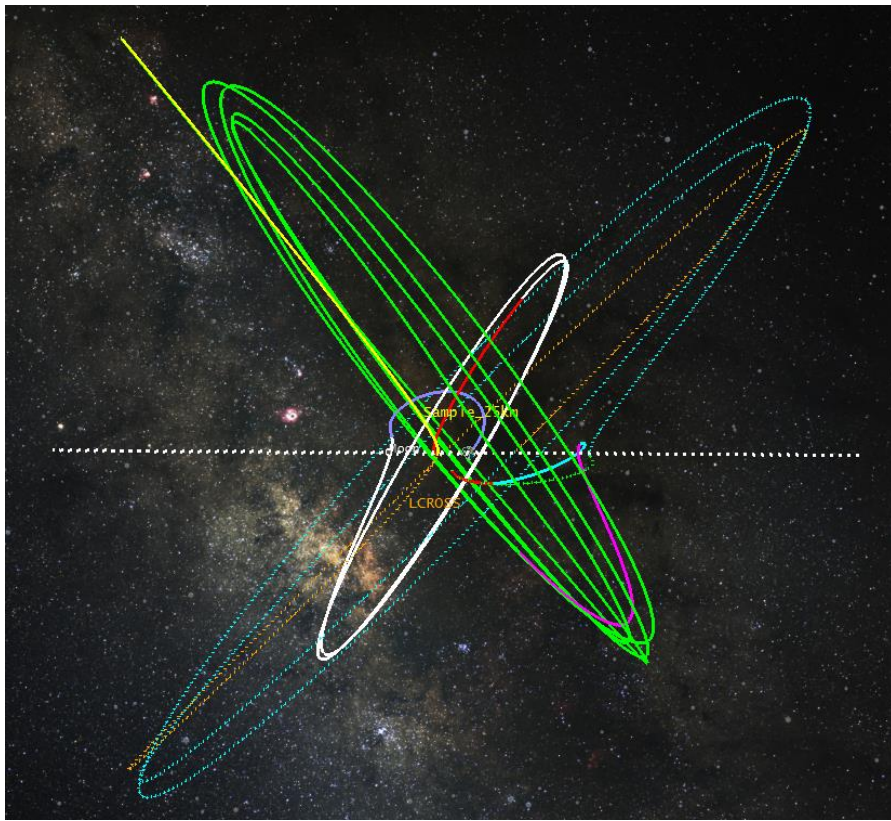


Figure 2. Centaur upper stage orbit (dotted) and sample-return spacecraft orbit (solid) shown in Earth inertial frame, with view edge-on to the lunar orbit plane (top; rotated 90 deg from the view in Fig. 1) and angled to the lunar orbit plane, focused on the four lunar encounters (bottom).

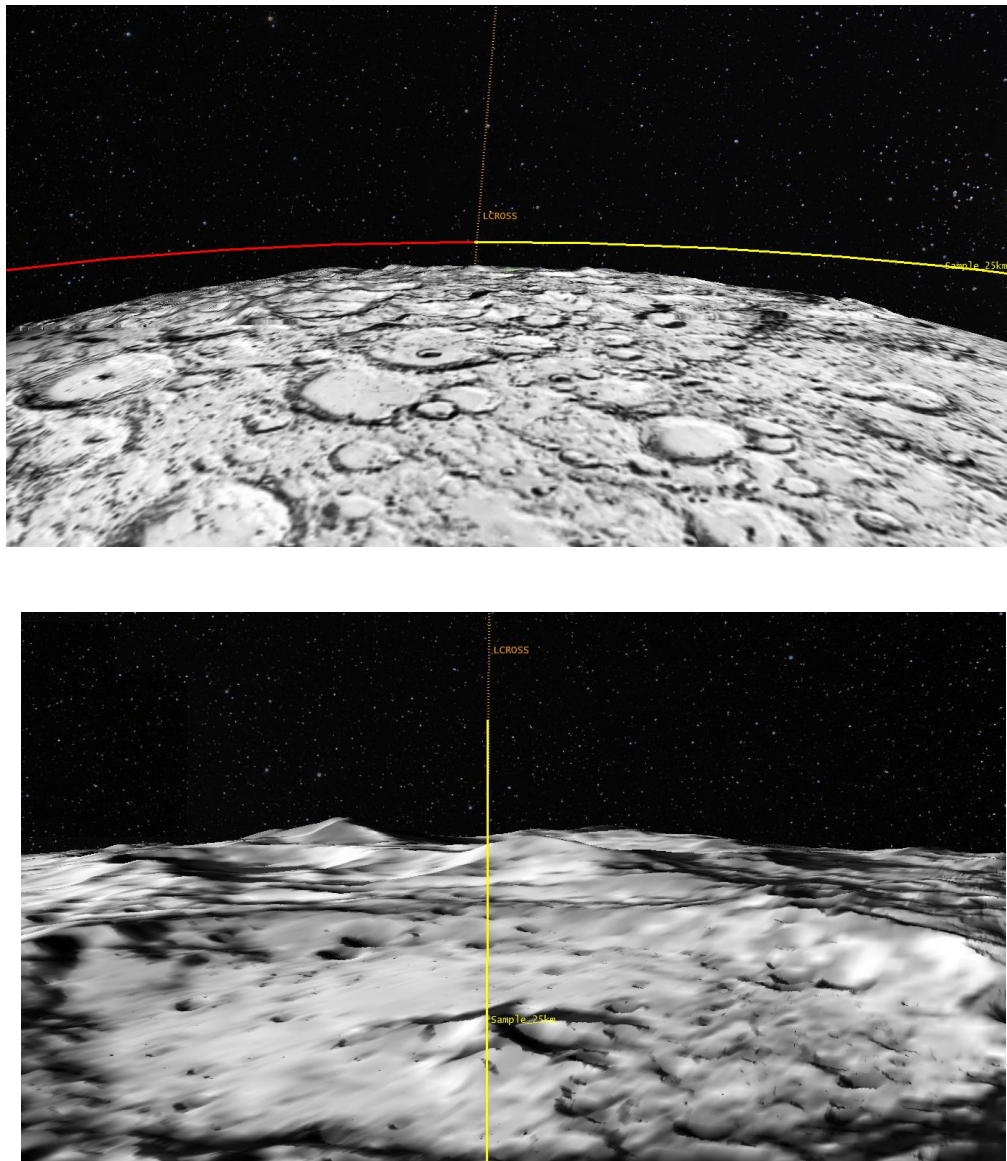


Figure 3. Views of the impact and sample-return trajectory focused at lunar south pole (Cabeus crater), shown in the Moon inertial frame (XY view on top, YZ view on bottom).

Characteristics of the centaur-created impact plume were modeled using in-house software developed at NASA Ames Research Center and previously used for the LCROSS mission. Details of the plume altitude and curtain radius (plotted against the time after the centaur impact) can be found in Figs. 4 and 5. The ejecta cloud was modeled as an expanding conical form, with the curtain being formed by the ballistic flights of the ejecta; thus the curtain is low and has a relatively small radius at the start and as it expands the curtain height increases as well. Aside from a steep impact maximizing plume height (especially helpful if the sample-return spacecraft's flight path doesn't lie on that of the centaur's), it can be seen that a steep and fast impact are desired for maximizing crater diameter, crater depth, and total mass ejected (Fig. 6).

As the perilune and fly-through altitudes match in the presented solution, the minimum fly-through altitude is not constrained by the trajectory but rather likely by navigation constraints and the mass of the ejecta at very low altitudes. For instance, a very low perilune and fly-through altitude of 3 km was chosen for this solution initially, but such an altitude would pose navigational challenges and too much ejected mass would be present for the 150-second sample-collection timing delay assumed to capture ejecta shortly after their peaks. Thus, a more practical fly-through altitude of 25 km was chosen for this solution (per Fig. 4). This means the corresponding perilune altitude was also 25 km; for safety reasons, the minimum perilune altitude for all solutions presented in this paper is constrained to at least 10 km.

The total ΔV required by the sample-return spacecraft was calculated as 309 m/s with a 145-day total duration, while the centaur upper stage required 27 m/s of ΔV and 112 days of duration.

For the sample-return spacecraft, this total ΔV of 309 m/s includes a 78 m/s maneuver to target the first lunar flyby (Fig. 1, D'), a 157 m/s maneuver calculated to target the second lunar encounter for the sample-collection (Fig. 1, G'), and a 67 m/s maneuver to target the third lunar flyby (Fig. 1, F') to return to Earth (Fig. 1, G') at 10.98 km/s entry speed. All lunar flybys are unpowered.

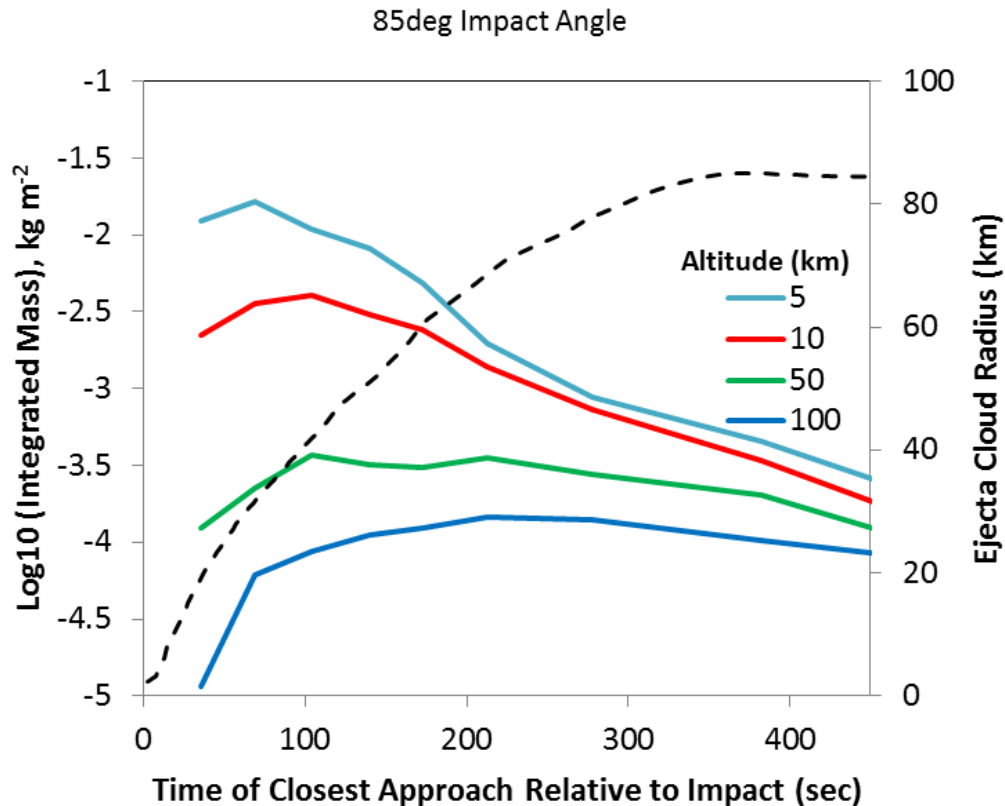


Figure 4. Ejecta Mass per Surface Area vs. fly-through delay timing and altitude by the CubeSat sample collection spacecraft (colored curves that correspond to Y-axis on left). 2500 kg impactor mass and 85° impact angle assumed. Ejecta cloud radius shown as dotted black curve corresponding to Y-axis on right.

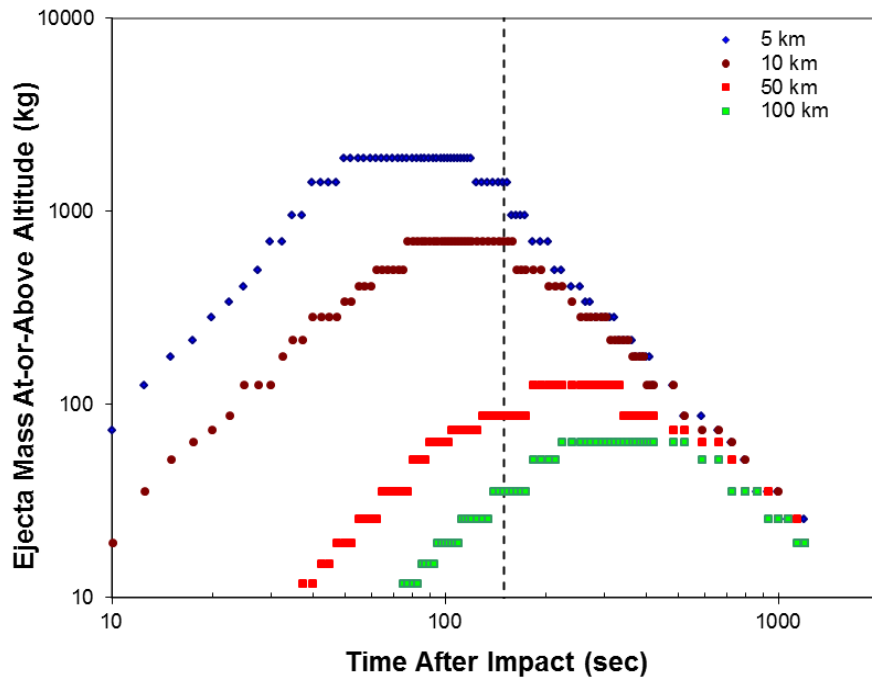


Figure 5. Ejecta Mass at or above fly-through altitude for various sample-collection timing delay by the CubeSat sample collection spacecraft. 2500 kg impactor mass at 85 degree impact angle and 2.5 km/s speed.

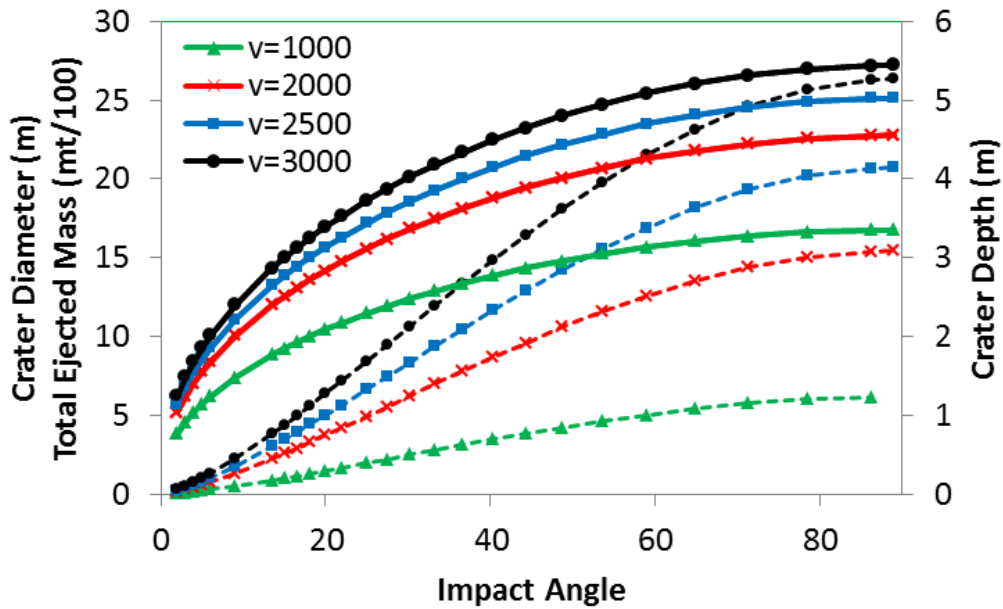


Figure 6. Impact crater depth and ejected mass vs impact angle. Dashed curves correspond to ejected mass axis values while the solid curve corresponds to both crater depth and crater diameter axes values. 2500 kg impactor mass assumed.

Lower Energy LCROSS Solution

Since the perilune altitude need not equal the fly-through altitude (i.e., perilune occurs either shortly before or after the minimum plume fly-through altitude is reached) for the given assumptions, a solution with a fly-through altitude on the order of 100 km is shown in Fig. 7. Such a plume fly-through altitude was shown to yield significant science value per Figs. 4 and 5. After constraining the perilune altitude to 10 km, the corresponding fly-through altitude was calculated as 73 km, thus a sample-collection timing delay of 400 seconds was assumed (per Figs. 4 and 5).

This solution in Fig. 7 required 126 m/s of total ΔV for the sample–return spacecraft. This total ΔV requirement includes 67 m/s to target the first lunar flyby, 18 m/s to target the second flyby, and 41 m/s for the third flyby. It is of note that the third lunar flyby, used to direct the sample toward Earth for reentry, occurs on the opposite side of Earth with respect to where the impact occurred (in the inertial frame), about 17 days after the sample is collected. Thus the total duration is 129 days from launch until the sample enters the Earth’s atmosphere for reentry. All lunar flybys are unpowered.

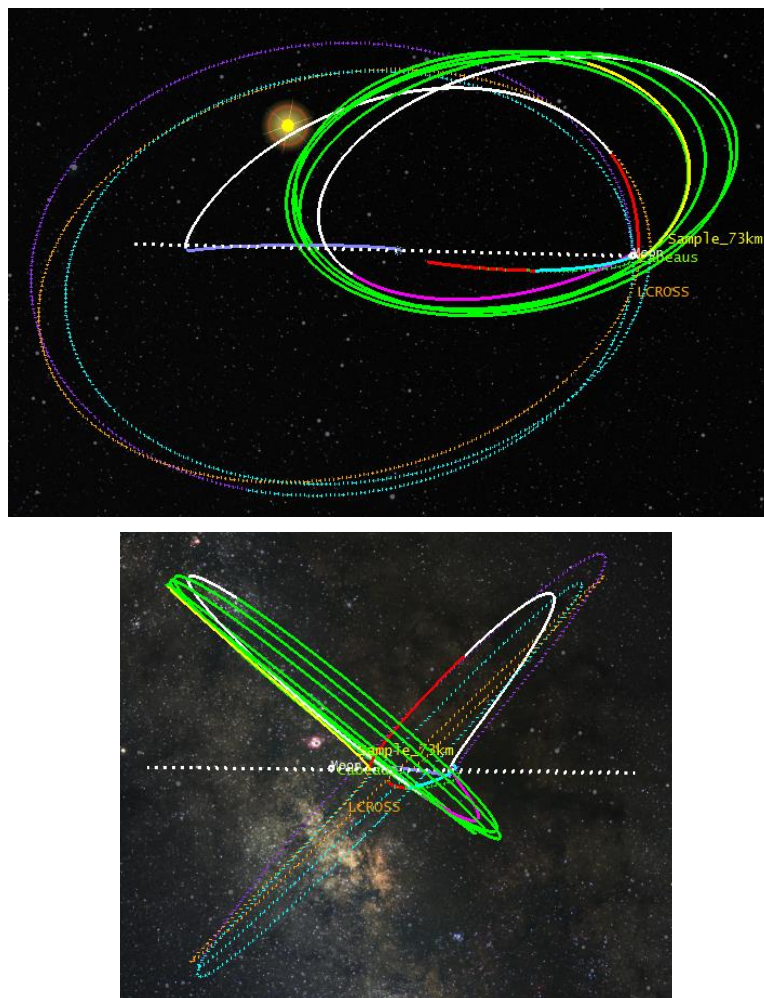


Figure 7. Solution requiring a total of 126 m/s of ΔV for the sample-return spacecraft, with a 73 km fly-through altitude (10 km perilune altitude). Impact solution uses LCROSS trajectory. Two Earth-centered, Earth inertial views (XY on top, YZ on bottom).

BackFlip Solution

Instead of using the LCROSS transfer for the centaur spacecraft, the BackFlip transfer designed by Uphoff² is implemented. The actual reference trajectory used for this impact launches from Vandenberg Air Force Base and yields an impact near the lunar north pole. Specifically, a permanent shadowed region of Peary crater is targeted for this centaur impact which occurs at 2.5 km/s (i.e., the same speed as the LCROSS centaur impact). Despite the impact angle (64°) being lower than that of the LCROSS mission (85°), it is still above the required 60 degrees and the resulting ejecta plume is comparable to that created by the LCROSS centaur (Figs. 8 and 9).

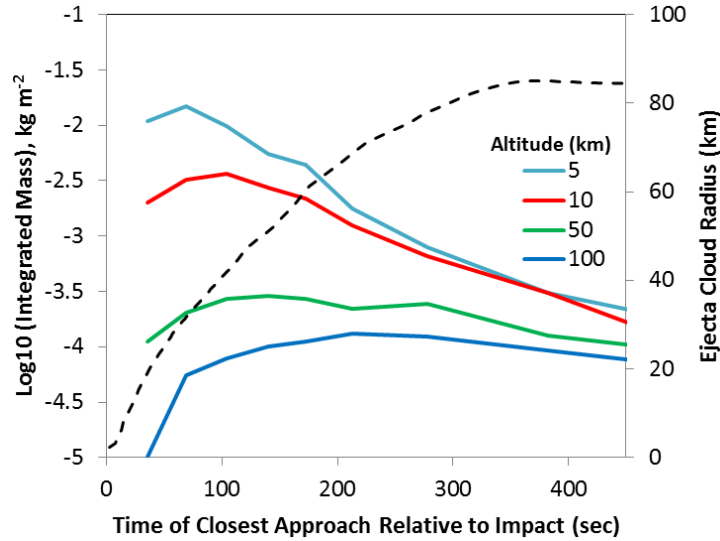


Figure 8. Ejecta Mass per Surface Area vs. fly-through delay timing and altitude by the CubeSat sample collection spacecraft (colored curves that correspond to Y-axis on left). Ejecta cloud radius shown as dotted black curve corresponding to Y-axis on right.

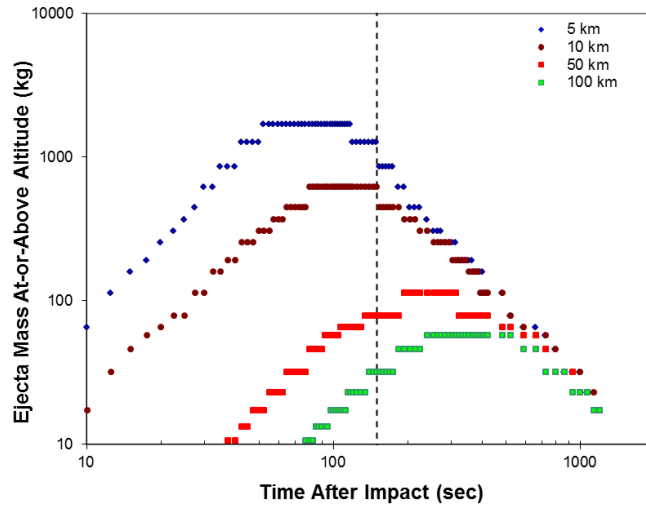


Figure 9. Total Ejecta Mass at or above fly-through altitude for various sample-collection timing delays by the CubeSat. 2500 kg impactor mass at 64° impact angle and 2.5 km/s speed.

This resulting solution is shown in Fig. 10, with both the BackFlip centaur trajectory (dotted) and CubeSat sample-collection spacecraft trajectory (solid) shown in Earth-centered, Earth inertial frames. The ΔV needed to target the first lunar flyby (all lunar flybys are unpowered) was calculated as 67 m/s, for the second flyby it was 9 m/s, and for the third and final lunar flyby, the ΔV cost was 25 m/s. Thus the total ΔV required for this solution was 101 m/s, with a total duration of 48 days (i.e., the lowest ΔV and duration solution presented in this section). The fly-through altitude of the CubeSat is 65 km which corresponds to a 10 km perilune altitude; coupled with the 64° impact angle at 2.5 km/s. Although a 150-seconds vertical line for the timing delay is shown (in Fig. 9), this solution's delay was increased to 400 seconds to account for the higher fly-through altitude.

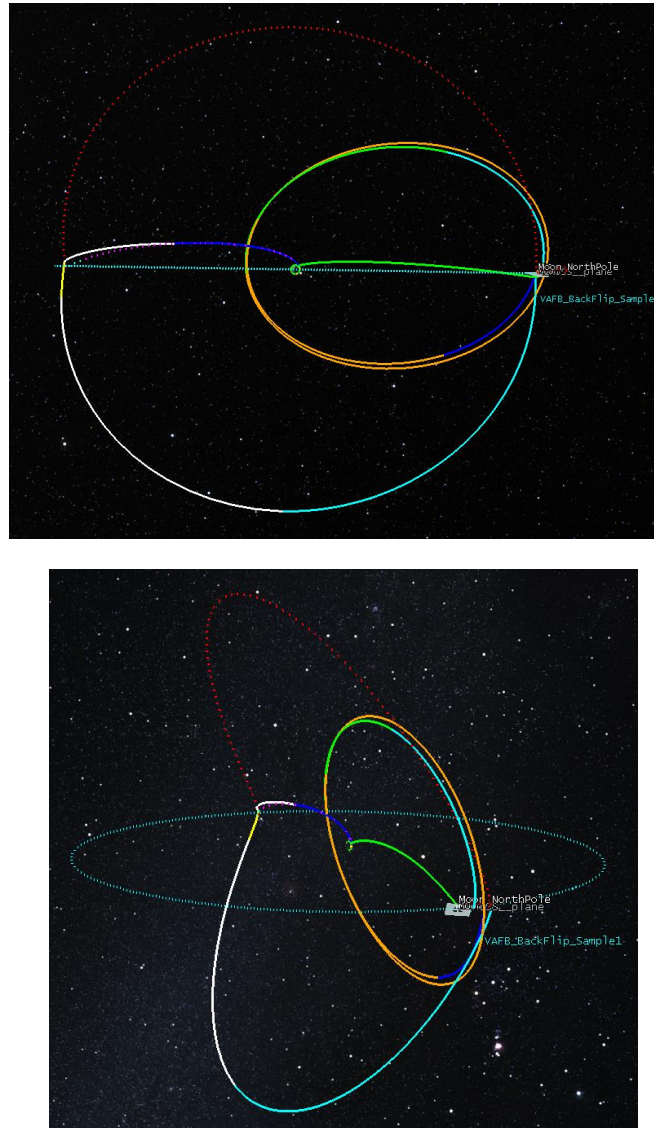


Figure 10. Solution requiring a total of 101 m/s of ΔV for the sample-return spacecraft, with a 65 km fly-through altitude (10 km perilune altitude). Impact solution uses BackFlip trajectory. Two Earth-centered, Earth inertial views (XY view on top, and angled view on bottom).

Intermediate Solution

Presented in Fig. 11 is a solution considered intermediate, in that the impact angle yielded of 74° is between that of the BackFlip (64°) and LCROSS (85°) solutions previously presented. This centaur impact occurs 31 days after launch at 2.5 km/s and is centered on Cabeus crater (i.e., the impact target assumed for the LCROSS mission as flown), which is located about 5 degrees from the lunar south pole. The TLI used for this solution assumes a launch out of CCAFS.

The CubeSat performs a plume fly-through at 25 km altitude (which corresponds to a 17.9 km perilune altitude) at a speed of 2.47 km/s. This fly-through occurs 200 seconds after the impact. The total ΔV requirement for the sample-return CubeSat was calculated as 204 m/s, which includes 75 m/s of ΔV to target first lunar flyby, 55 m/s of ΔV to target the second lunar flyby, and 74 m/s of ΔV to target the third lunar flyby. All lunar flybys are unpowered.

This solution is also considered intermediate since its total mission duration (for the CubeSat) of 62 days is also between that of the BackFlip (48 days) and LCROSS (112 days) solutions previously presented.

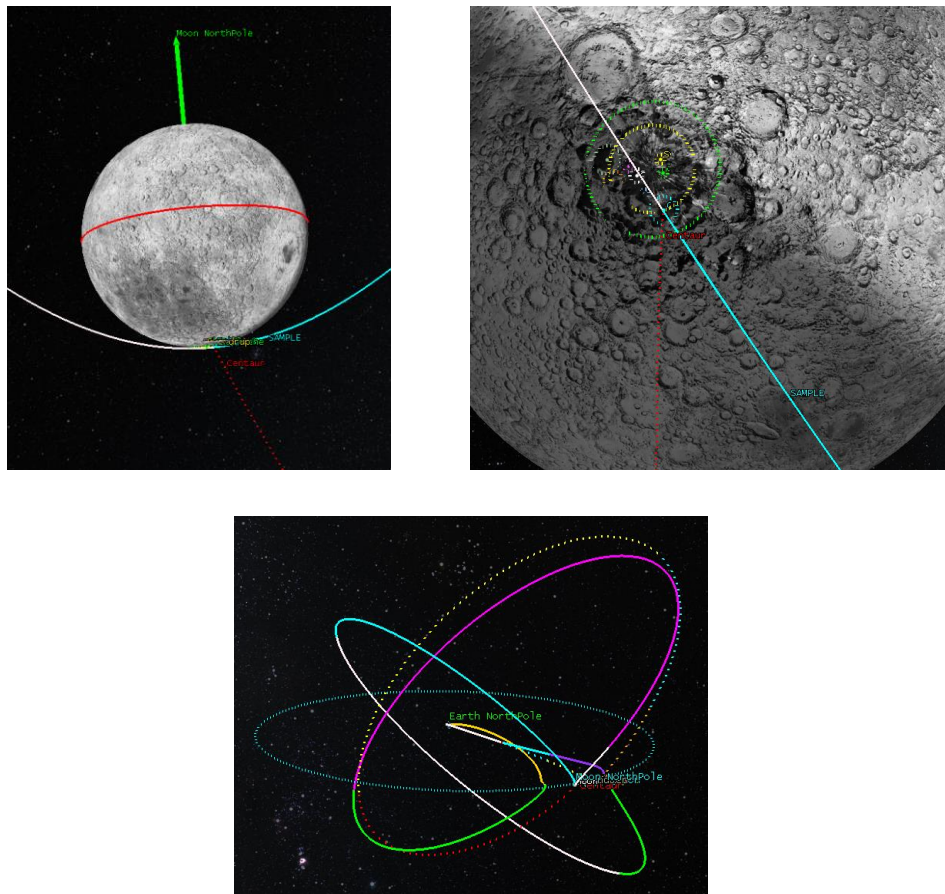


Figure 11. Intermediate Solution requiring a total of 62 days from launch to sample-return to Earth. Assumes a 25 km plume fly-through altitude (17.9 km perilune altitude) by the CubeSat. Trajectory shown in two Moon inertial views moments before impact by the centaur (top-left and top-right). Transfer trajectory shown in Earth inertial frame (bottom).

Single Maneuverable Spacecraft Solutions

A single maneuverable spacecraft architecture was considered for comparison purposes. In this scenario, one maneuverable small spacecraft is equipped with a relatively small, spherical mass (i.e., a 20 kg tungsten ball) to serve as the impactor that creates the sampled plume. The small sample-collection spacecraft (i.e., “mothership”) would need to perform a survival maneuver after deploying the tungsten mass on an impact trajectory; thus the mothership would be on the (essentially) same flight path as the deployed tungsten mass. This geometry constrains the impact location to near the limb of the approach asymptote for a low-altitude sample collection.

Assuming transfer on a TLI (launched from CCAFS), this type of impact can be targeted at either lunar pole (Fig. 12); however, the sampling altitude is well above the required 100 km maximum. Furthermore, the post sample collection trajectory may escape depending on the flyby conditions and apogee location with respect to the rotating Sun-Earth reference frame. For the assumed example, a north pole impact yielded a direct Earth-escape while a south pole impact yielded an orbit still captured at Earth, albeit weakly (i.e., solar gravity helped to maintain an Earth-captured post sample-collection orbit).

While both north and south pole impact trajectories are shown in Fig. 12, a complete solution for the south pole impact case is shown in Fig. 13. In this south pole case, the impact angle yielded was about 30° while the mothership’s fly-through altitude was 438 km (via a 10 km per-lune altitude) with a plume fly-through speed of 2.34 km/s. The ΔV requirement to return the sample for this case includes: 77 m/s to target impact conditions for the tungsten mass, a 5 m/s maneuver performed one day after deploying the tungsten mass to avoid an impact (i.e., raise per-lune to 10 km) after flying through the impact plume, and 79 m/s performed at the first apogee of 2.35 million km altitude to target an Earth reentry without the use of a lunar flyby. This solution required a total ΔV of 161 m/s and total duration from launch to sample return (11.06 km/s reentry speed at 121 km Earth altitude) of 255 days.

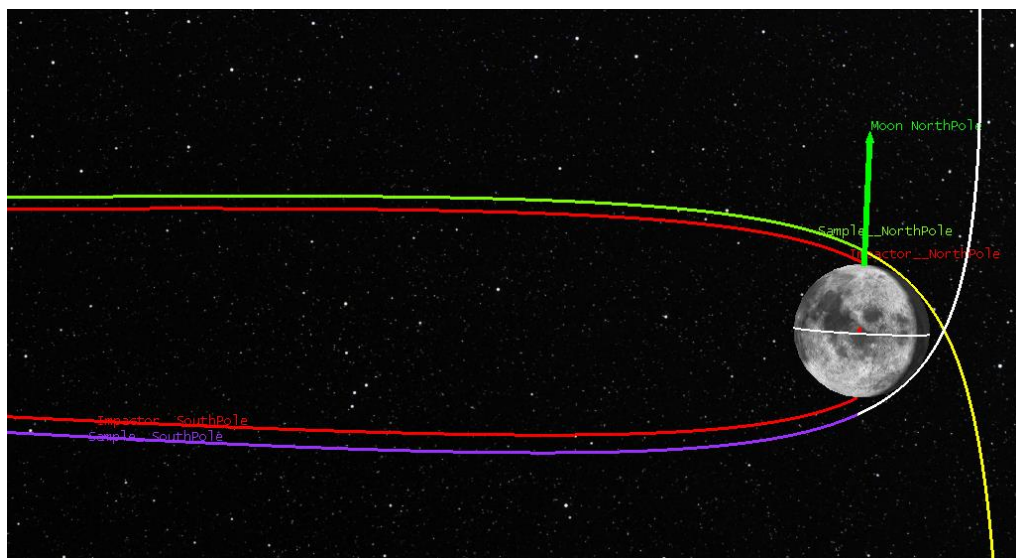


Figure 12. Both north and south pole impact solutions shown from a standard TLI, given a single maneuverable spacecraft equipped with a 20 kg tungsten mass to be deployed for the impact. 3-day transfer duration from TLI to lunar close-approach. Trajectories shown in the Moon inertial frame.



Figure 13. Direct TLI CCAFS South pole impact case shown through sample-return to Earth. Solution shown in the Earth inertial frames (normal to the lunar orbit plane on the left, and edge-on the lunar orbit plane on the right). Highest apogee altitude is 2.35 million km.

The relatively high 438 km fly-through altitude, however, is not favorable for plume collection; thus an attempt was made to lower this altitude to near 100 km. Such conditions are possible when the impact angle is about 15 degrees, thus updated plume ejecta plots are shown in Figs. 14 and 15 where this significant drop in ejecta mass can be seen (as compared to Figs. 4, 5, 8, and 9). The 438 km fly-through altitude can be reduced for this given TLI at the expense of the impact location moving away from the south pole. Specifically, for a 15° impact angle, the fly-through altitude of the sample-collection spacecraft was calculated as 109 km (with a plume fly-through speed of 2.54 km/s) which corresponded to an impact location of 73° north latitude (Fig. 16). This trajectory naturally escapes Earth orbit after collecting the plume sample and thus the total ΔV requirement for this solution is about 80 m/s less than that of the 438 km solution presented in Fig. 13). If a south pole impact site was chosen instead (as in Fig. 12), then this impact location would move farther from the pole, to 67 degrees (not shown).

If instead the TLI from the Vandenberg Air Force BackFlip case was used (from Fig. 10), the tungsten mass can be deployed closer to the pole with a 15° impact angle (i.e., within 5 degrees of the south pole in this case). However, the solution shown in Fig. 17 assumes an impact in Shackleton crater on the south pole (89.9° south latitude), which corresponds to an impact angle of 17.9° at a speed of 2.5 km/s. In this solution, the sample is collected at 152 km above the impact location at 2.56 km/s fly-through speed, with a timing delay of about 10 minutes. Once again, the trajectory naturally escapes Earth orbit after sample-collection. In such cases, the sample is assumed to be analyzed by the sample-collection spacecraft's instruments while on its escape trajectory to heliocentric space, making sure to relay the data back to Earth while communication is possible. The heliocentric orbit can serve as a disposal orbit when the mission is complete. The total ΔV requirement for this solution was calculated as 77 m/s (68 m/s to target the impact site for the tungsten mass and 9 m/s for the survival maneuver that yields sample-collection).

The total duration required for these direct TLI, direct Earth-escape solutions (i.e., Figs. 16 and 17) includes 3 days for the transfer from TLI to the tungsten impact and the time needed for the sample-collection spacecraft to relay the science data back to Earth.

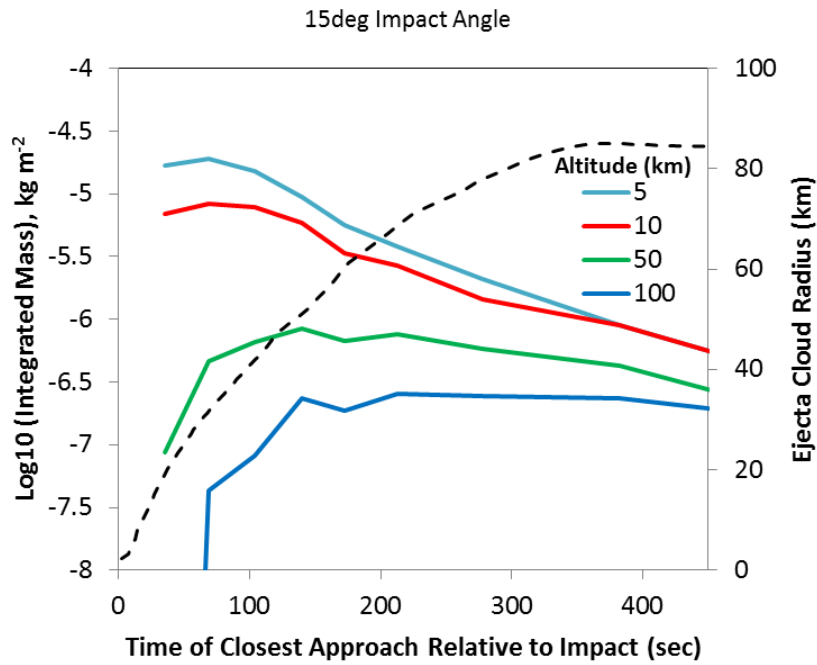


Figure 14. Ejecta Mass per Surface Area vs. fly-through delay timing and altitude by the sample collection spacecraft (colored curves correspond to Y-axis on left). Ejecta cloud radius shown as dotted black curve corresponding to Y-axis on right. Impactor assumed as 20 kg tungsten mass, with 15 degree impact angle at 2.5 km/s speed.

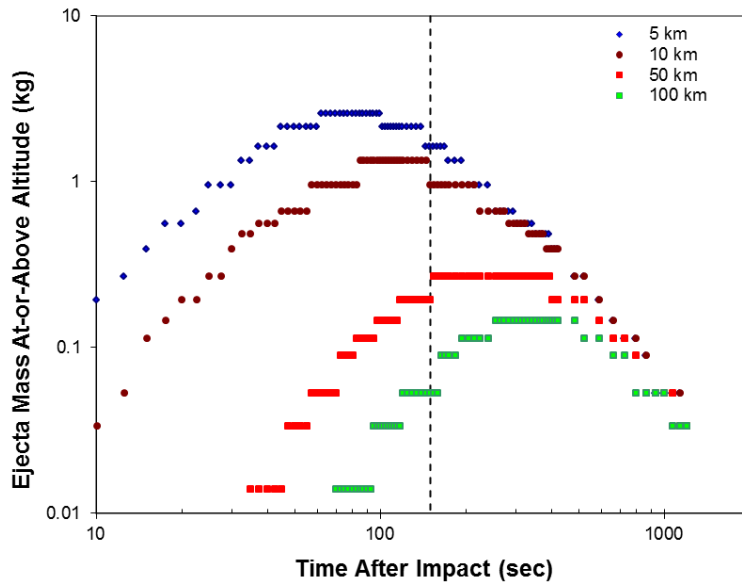


Figure 15. Ejecta Mass at or above fly-through altitude for various sample-collection timing delay by the sample collection spacecraft. 20 kg tungsten impactor mass at 15 degree impact angle and 2.5 km/s speed assumed.

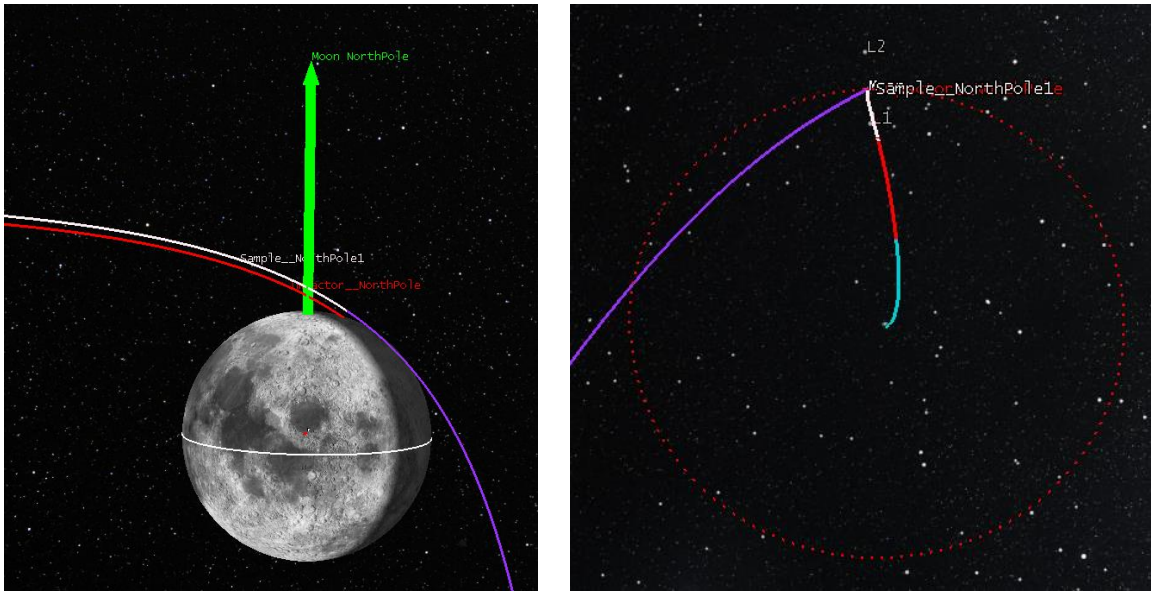


Figure 16. Direct TLI CCAFS case: 73 deg north latitude impact at 15° impact angle and 2.5 km/s impact speed. Mothership collects sample at 110 km altitude after peak ejecta mass plume. Moon inertial view shown moments before the impact and subsequent sample collection (left) and Earth inertial view shows the natural escape from Earth orbit after the sample collection (right).

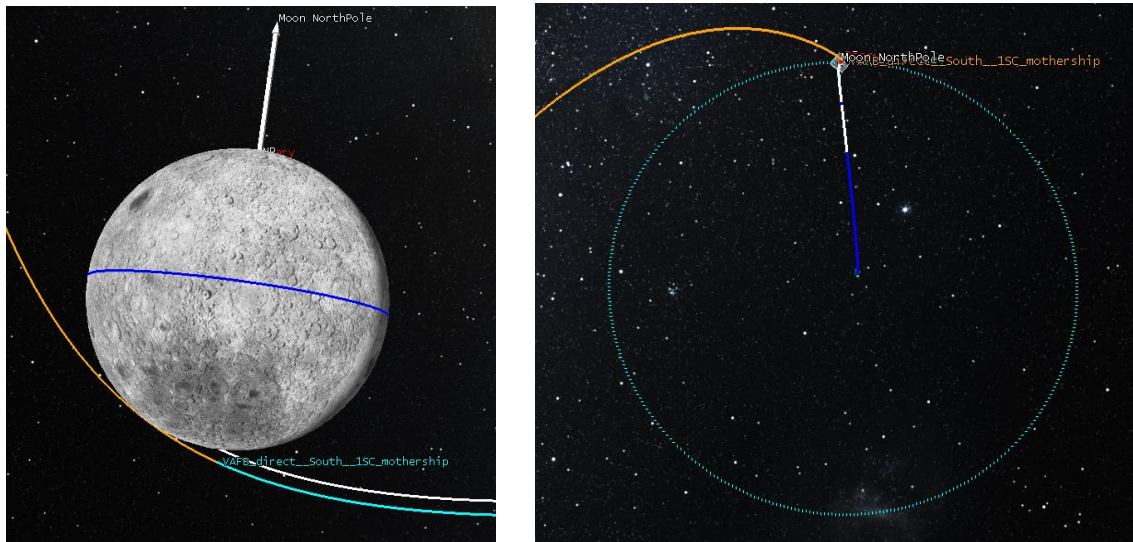


Figure 17. Vandenberg Air Force Base Direct TLI case: 90 degree south latitude impact at 17.9 degree impact angle and 2.5 km/s impact speed. Mothership collects sample at 152 km altitude after peak ejecta mass plume. Moon inertial view shown moments before the impact and subsequent sample collection (left) and Earth inertial view shows the natural escape from Earth orbit after the sample collection (right).

The final solution presented in this section utilizes the LCROSS TLI and the general transfer of the sample-return spacecraft (i.e., CubeSat) from the solution presented in Fig. 7. The purpose of this solution is to demonstrate that an impact location near the pole is possible from the same steep approach previously utilized in the two-spacecraft scenario presented earlier.

In this presented solution (Fig. 17), the impact angle was constrained to 15° so that the sample-collection could occur near 100 km. This resulting impact was located (centered) at 85° south latitude with an associated impact speed of 2.5 km/s impact speed. The sample collection was timed to occur 7 minutes after the moment of impact at a minimum altitude of 114 km above the impact location (via a 10 km perilune altitude constraint) at 2.45 km/s fly-through speed.

The total ΔV requirement for this spacecraft was calculated as 171 m/s, which includes: 67 m/s to target the first lunar flyby, 18 m/s to target the second lunar flyby and impact trajectory, and 11 m/s to avoid the impact (this survival maneuver is performed 12 hours before the impact to yield a reasonable error ellipse for the impact site; specific mission needs will drive this requirement), and 75 m/s to target the third lunar flyby to return the sample via Earth atmospheric reentry at 10.99 km/s. All lunar flybys are unpowered. The total mission duration was calculated as 131 days.

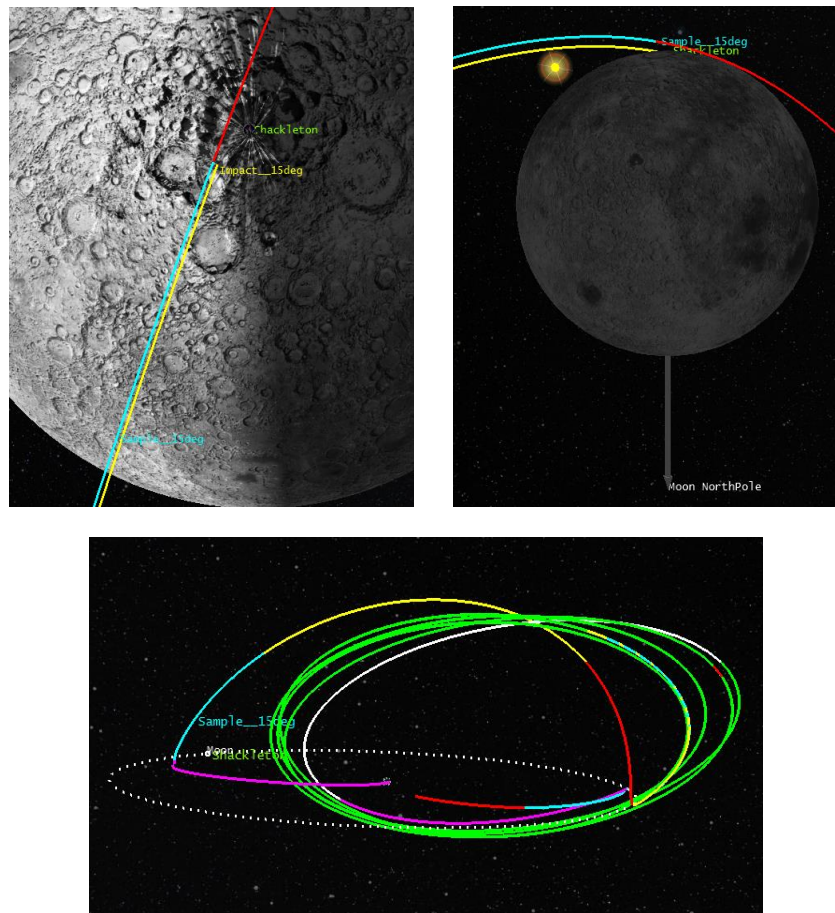


Figure 18. LCROSS style solution with one maneuverable spacecraft that deploys a tungsten mass to create an impact plume at 2.5 km/s speed and 15 degree angle. Plume is sampled at 114 km altitude. Moon inertial views at moment of impact (top-left) and moment of sample collection before impact (top-right). Transfer solution shown in Earth-inertial frame (bottom).

APPLICATIONS

Although the presented orbits never enter orbit around the Moon, the same steep lunar approaches can be used to place the perilune of an elliptical lunar orbit above the north or south pole. Such a lunar orbit could be used to fly-through the presented impact plume but at lower velocities and for spacecraft wishing to conduct science measurements at low altitudes near either lunar pole (e.g., hydrogen mapping of these shadowed craters via a neutron spectrometer^{3,4}).

To demonstrate this design alteration, a primary payload is assumed to be on a TLI (Fig. 19, A) headed to a low inclination lunar orbit. One day after TLI, the secondary CubeSat performs a maneuver (75 m/s of ΔV ; Fig. 19, B) to target a 6,700 km lunar flyby (Fig. 19, C). This flyby occurs over the lunar north pole so as to significantly change the CubeSat's orbit inclination to that of near-polar.

After a perigee maneuver of 32 m/s (Fig. 19, D) for lunar phasing, the Moon is revisited one lunar day (28 Earth days) after the north pole (unpowered) flyby. The steep lunar approach from the north enables the perilune to be placed near the south pole above Shackleton Crater (Fig. 19, images on bottom). The lunar orbit insertion (LOI) is performed at 50 km perilune altitude to capture into lunar orbit (Fig. 19, E). The instantaneous ΔV for the LOI is 175 m/s, which yields a relatively stable orbit that will subsequently have its energy lowered. It is of note that sufficient thrust is needed to ensure capture at the Moon and to target the first lunar flyby.

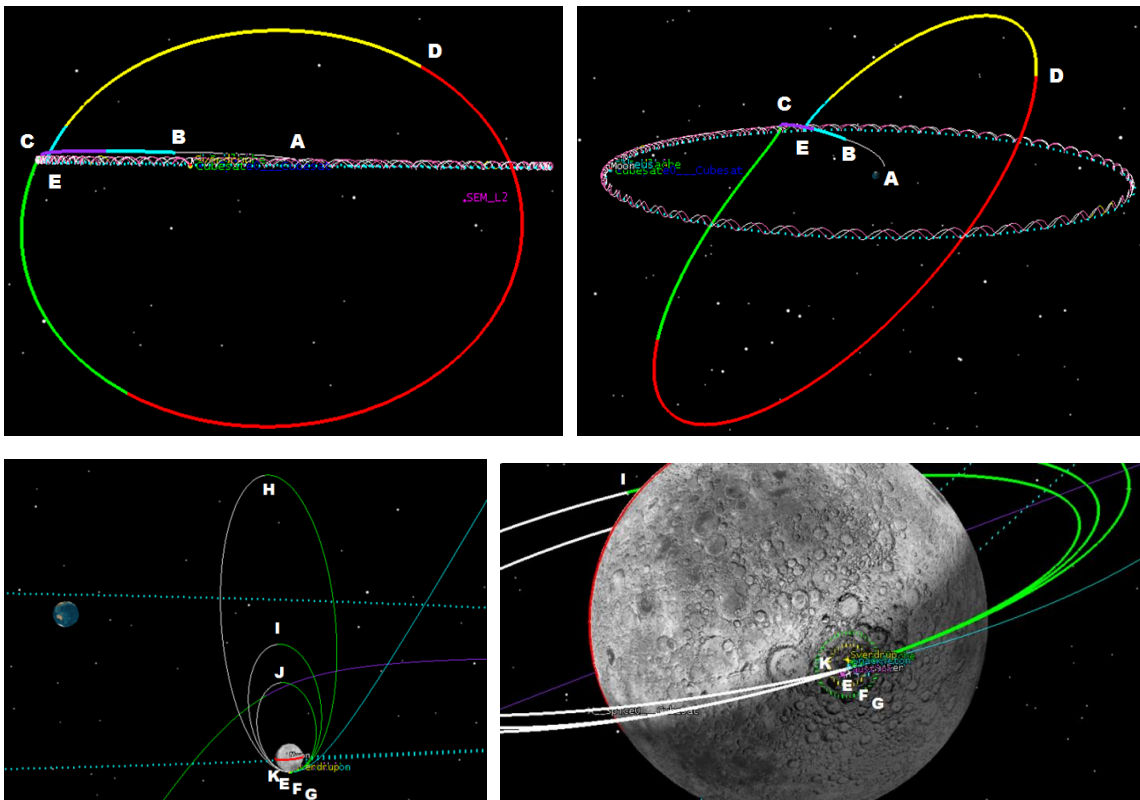


Figure 19. One-Month Transfer trajectory from TLI separation to 10-hour elliptical polar lunar orbit with perilune above south pole. Trajectory shown in Earth-centered, Earth inertial frames (top-left and top-right) and in Moon-centered, Moon inertial frames (bottom-left and bottom-right).

Two subsequent braking maneuvers (ΔV of 80 and 62 m/s) are performed at perilune (Fig. 19, F & G) to incrementally lower the apolune altitude (Fig. 19, H, I, & J) to an assumed 7,500 km since this is the planned altitude for the LunaH-Map mission. From this orbit, a CubeSat can fly its instrument(s) at 2.17 km/s above Shackleton and every other nearby crater that may host permanently shadowed ice. The ground track will be fixed over Shackleton crater if the natural rotation of the line of apsides is frozen via station-keeping maneuvers. Furthermore, due to the rate of the Moon's rotation, an incremental 360-degree longitude "sweep" centered around the south pole (assuming a controlled line of apsides) is completed every 14 Earth days. In this lunar orbiter applications example, the total deterministic ΔV requirement was calculated to be 424 m/s.

Another potential application is that of an Earth-Moon cycler orbit. Such an orbit would be similar in appearance to that designed by Uphoff ², although with lunar encounters every month. The lunar encounters would be of higher energy as well given the steep approaches, yet the sample-return trajectory shown in Fig. 1 would be connected to Earth-Moon cycler orbits in the lunar orbit plane (e.g., from Ref. 5 and much of its referenced literature) to allow crew transfers to and from Earth.

CONCLUSIONS

A proof of concept trajectory design has been presented that enables a return of a lunar sample from a permanently shadowed polar crater with two maneuverable spacecraft that do not enter lunar orbit (or land softly on the lunar surface). A similar concept utilizing a single maneuverable spacecraft (equipped with a deployable tungsten impactor mass) was presented as well for comparison purposes.

For the two-spacecraft concept, trajectories were designed so that the sample-return spacecraft (i.e., the CubeSat) would require most of the ΔV , given the centaur's relatively large mass. The ΔV loads can be reversed if needed. For the main solution presented in Fig. 1, the centaur would need to change its velocity by 27 m/s while the CubeSat would require a 309 m/s velocity change (deterministic ΔV requirement assuming instantaneous maneuvers). This assumes the perilune altitude and plume fly-through altitude must be equal, which is a valid constraint when the plume fly-through altitude needs to be very low (e.g., < 10 km). However, it was seen that such a low altitude was not favorable for cases with too much ejecta debris in the flight path of the sample-collection spacecraft, thus other solutions were presented that require significantly less ΔV . Such ΔV requirements will vary based on the assumed initial conditions (e.g., type of TLI, deployment timing, etc.) and geometric constraint (e.g., minimum fly-through altitude, impact angle, etc.).

Finally, it is noted that given the assumed instantaneous, a challenge for a modern-day CubeSat would be to impart up to 78 m/s of ΔV quickly enough (i.e., within a few days) to target the first lunar flyby, assuming the bulk of the ΔV load has been placed on the CubeSat. The observed 78 m/s ΔV maneuver in the presented solutions was assumed to occur one day after the TLI; such an amount of ΔV would be challenging (if not impossible) to quickly impart for low-thrust propulsion systems, although a cold gas or other higher thrust system (e.g., hydrazine) may be used to enable this trajectory. All other maneuvers are performed outside of gravity wells thus a low-thrust propulsion system is more applicable for these mission legs.

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