AAS 17-456

TRAJECTORY DESIGN FOR THE LUNAR POLAR HYDROGEN MAPPER MISSION

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The presented trajectory was designed for the Lunar Polar Hydrogen Mapper (LunaH-Map) 6U CubeSat, which was awarded a ride on NASA's Space Launch System (SLS) with Exploration Mission 1 (EM-1) via NASA's 2015 SIMPLEX proposal call. After deployment from EM-1's upper stage (which is planned to enter heliocentric space via a lunar flyby), the LunaH-Map CubeSat will alter its trajectory via its low-thrust ion engine to target a lunar flyby that yields a Sun-Earth-Moon weak stability boundary transfer to set up a ballistic lunar capture. Finally, the orbit energy is lowered to reach the required quasi-frozen science orbit with periselene above the lunar south pole.

INTRODUCTION

The presented trajectory design uses an ion thruster with enough iodine propellant to give more than 2 km/s of Delta-V (ΔV) capability for a 6U CubeSat. However, this ΔV cannot be quickly imparted given the sub-millinewton thrust level of the propulsion system. Thus, to reach a lunar orbit, the Lunar Polar Hydrogen Mapper (LunaH-Map) CubeSat must approach the Moon slowly enough such that a ballistic (or near-ballistic) capture enables subsequent orbit braking to achieve a suitable science orbit.

As LunaH-Map intends to map the hydrogen at/near the lunar south pole[1], the periselene of the science orbit is required to be above the south pole. Given the precession of the argument of periselene, a slightly higher value than 270 degrees was chosen for this element, along with values of < 40 km for the periselene altitude, 90 degrees for the orbit inclination, with RAAN left as a free variable given an aposelene altitude low enough for stability purposes (less than 3,150 km is seen as an upper bound for the aposelene altitude; discussed later). Additionally, 141 passes over the lunar south polar region are needed.

The required science orbit thus represents a challenging final target, especially given the low-thrust propulsion system; regardless, a trajectory solution that closes is presented.

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ASSUMPTIONS & CONSTRAINTS

The LunaH-Map trajectory was designed with the Systems Tool Kit (STK) Astrogator module, using an N-body force model and high-order Runge-Kutta numerical integrator. For verification, the General Mission Analysis Tool (GMAT) was used, with a high-fidelity force model made to match that used by STK/Astrogator. The expected thrust is under 1 millinewton and the Isp greater than 1000 seconds; specific values are proprietary.

Critical trajectory-related constraints include:

* Latest post-TLI state vector of EM-1's Upper Stage (Oct. 7, 2018) assumed as the starting point for the LunaH-Map trajectory
* Spacecraft Initial Wet Mass <= 14 kg
* Ion Engine is Solar-Powered, thus line-of-sight with Sun needed for thrusting
* Initial Propellant Mass (iodine) = 1.5 kg
* Input Power Operating Range for Propulsion System: 56 to 65 Watts
* Maximum duration for any single maneuver = 5 days
* Minimum Range to Lunar Terrain = 5 km
* Lunar Science Orbit:

    141 orbits with periselene altitude/height < 40 km

    inclination = 90 degrees

    argument of periselene = 270 degrees

    aposelene altitude < 3,150 km

    RAAN → any value that satisfies science requirements

END-TO-END TRAJECTORY DESIGN

The LunaH-Map CubeSat shares a ride with EM-1's upper stage with a trans-lunar injection (TLI) from a roughly circular low-Earth parking orbit, as shown in Figure 1 (A). that is disposed of via a heliocentric-bound trans-lunar injection (TLI) with a close lunar swingby. The EM-1 upper stage uses a close lunar swingby (altitude about 400 km) that ejects the stage into a heliocentric disposal trajectory. Deployment of the LunaH-Map CubeSat (at ~1 m/s ΔV) is assumed to occur about 8 hours later (Fig. 1, B). After 24 hours of spacecraft checkout, the thruster will be turned on for the first time (Fig. 1, C). This thruster was assumed to operate between 55 and 65 W of input power with a minimum thrust of 0.65mN and specific impulse (Isp) of 1,200 seconds; 1.5 kg of iodine propellant is assumed to be carried by the 6U CubeSat, with maximum initial wet mass of 14 kg. After 2 days of thrusting, the CubeSat makes its final-approach to its lunar flyby (Fig. 1, D), making sure to target a B-theta and periselene altitude (that needs to be about 2700 km altitude) such that a Sun-Earth weak stability boundary (WSB) transfer can be flown back to the Moon[2,3,4] after passing through a perigee altitude of > 100,000 km (Fig. 1, E) that clears the Van Allen radiation belts. No deterministic maneuver is needed at the farthest point of the spacecraft's orbit, about 1 million km from Earth (Fig. 1, F); perigee is raised to lunar distance by solar gravity with the additional benefit of a slow-enough approach to the Moon to enable ballistic/natural capture into lunar orbit (Fig. 1, H) To further reduce the lunar orbit energy upon the ballistic capture, the spacecraft performs a braking maneuver (starting at G, Fig. 1) in the anti-velocity direction (with respect to the Moon) to further reduce the lunar orbit energy.

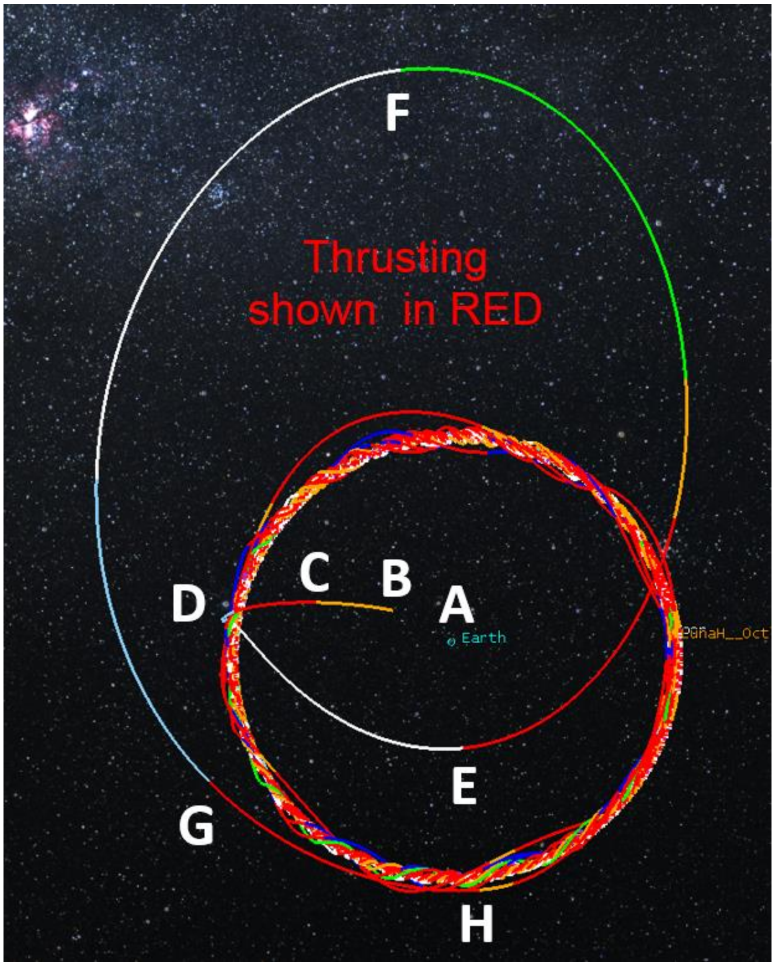

Figure 1. End-to-end trajectory solution for the LunaH-Map CubeSat with starting ride assumed to be a TLI with EM-1; Ecliptic-plane view in the Earth-centered, Earth inertial reference frame.

Perhaps the greatest challenge of this trajectory design is achieving a polar orbit with periselene located above the south pole. To accomplish this feat without a dedicated propulsion stage, the CubeSat targets a ballistic lunar capture that utilizes a WSB transfer, but now in the Moon-Earth-Sun system. An example of this transfer is shown in Figure 2; although there is significant thrusting (red arcs), only braking maneuvers are performed meaning the inclination is changed naturally in the Moon-Earth-Sun WSB region. A significant braking period is needed to reach the required stable orbit since a low-thrust propulsion system is used.

An interesting feature of this trajectory is that the orbit is circularized to a relatively stable 8,000 km lunar altitude. This circular orbit is useful for multiple reasons including: 1) the risk of a lunar impact due to a low periselene is minimized; 2) the orbit is high enough such that the low-thrust system can significantly change the orbit period to avoid mission-threatening 4+ hour eclipses of the Sun by the Earth; and 3) from the circular orbit, the argument of periselene can be chosen at will, allowing the periselene to be placed above the south pole to meet the requirements for the science orbit.

From this 8,000 km polar circular lunar orbit, most of the thrusting required for the mission is needed to reach the science orbit. Given the circular orbit strategy coupled with operations requirements to limit continuous thrusting periods, over one year of low-thrust spiral-down time to the science orbit was calculated. This spiral phase is seen in Figure 3, where once again only red-arcs representing thrust periods. It is seen that ample time is built into the trajectory for non-thrusting functions such as tracking, orbit determination, battery charging, etc.

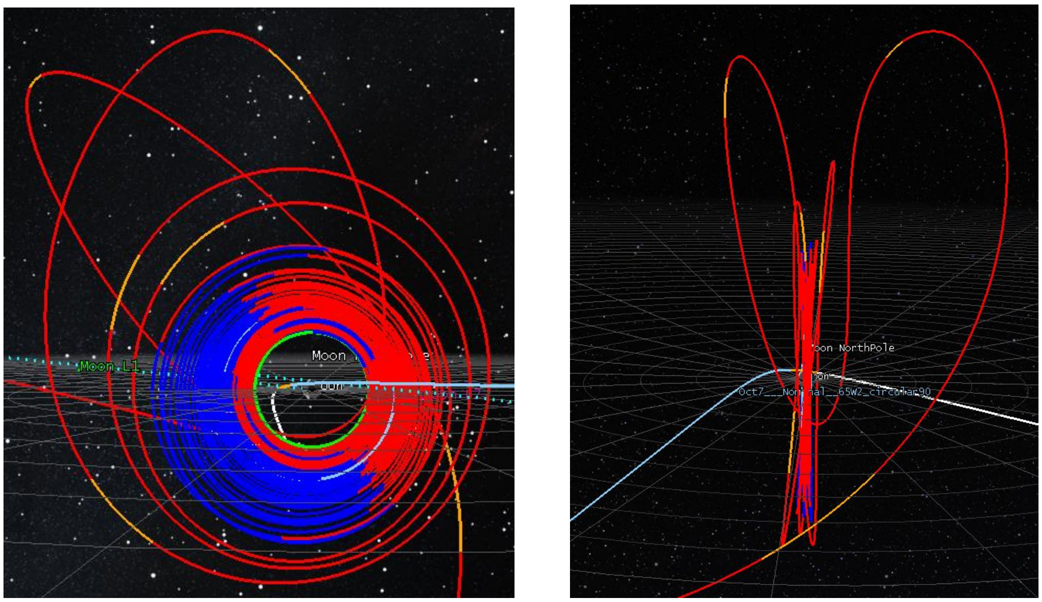

Figure 2. Transfer from a pre-ballistic lunar capture state to a relatively stable 8,000 km circular polar lunar orbit, shown in the Moon-centered, Moon-inertial reference frames (left and right).

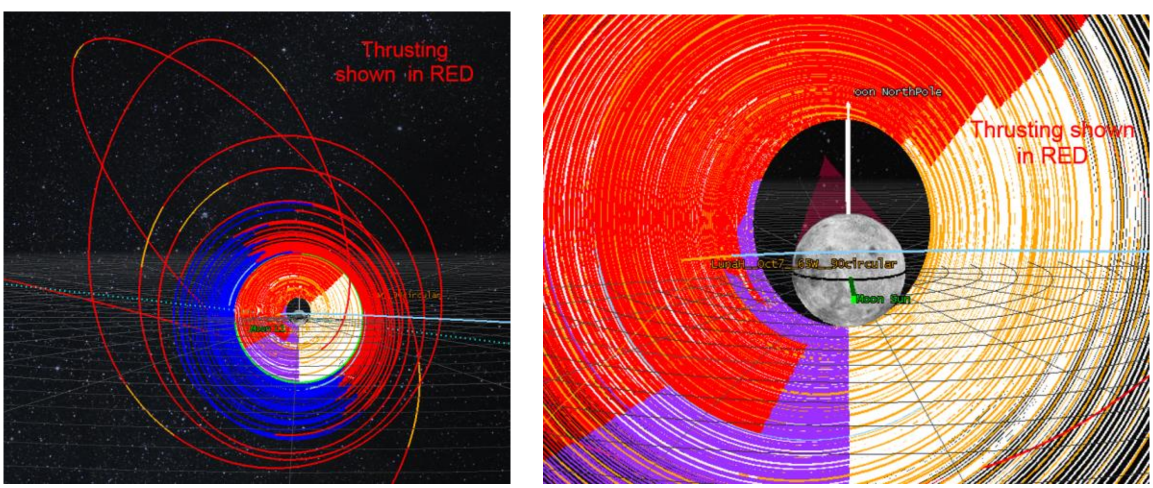

Figure 3. Transfer from a pre-ballistic capture state to the required science orbit, shown in the Moon-centered, Moon inertial frames at different zoom levels (left and right).

TRAJECTORY WITH OPERATIONAL CONSTRAINTS

After the above trajectory was computed, a detailed schedule for tracking and operations was developed for the first two weeks of the mission by the KinetX navigation team, as shown in Table 1. This allows ample time for tracking and orbit determination, to determine the initial orbit, and then assess the performance of the early maneuvers, so that adjustments can be made, as needed, to take into account the actual thruster performance relative to the pre-launch predicted performance. Also, the CubeSat deployment was changed to “Bus Stop 2” at a geocentric distance of 64,500 km, well above the Van Allen radiation belts. New thrust/Isp performance information was provided by Busek, the company that is building the spacecraft propulsion system, based on new tests that they performed. The earlier trajectory was computed for an input power of 65 watts, the maximum possible. All of these factors, including a desire to see how low the input power could be, and still have a good mission, resulted in new trajectory computations described below. The new calculations were not as thorough as those for the earlier trajectory, since revised EM-1 trajectory information was due to be provided by NASA in late January 2017, so the process would need to be repeated then.

Table 1. Tracking, Orbit Determination, and Maneuver Schedule for the first 15 days (times are relative to Launch; LunaH-Map CubeSat release is about 8h later)

Man.	DSN Track		Orbit Det.		Create Commands		Uplink	Maneuver		
#	Start	End	Start	End	Start	End		Start	End	Type
1	0d 8h	0d 23h	0d 17h	0d 19h	0d 20h	0d 22h	0d 23h	1d 0h	1d 15h	V+
2	1d 16h	1d 19h						1d 20h	2d 1h	V+
3			1d 20h	1d 22h	1d 23h	2d 1h	2d 2h	2d 3h	3d 10h	V+
4	3d 11h	3d 14h						3d 15h	3d 20h	V+
5			3d 15h	3d 17h	3d 18h	3d 20h	3d 21h	3d 22h	4d 20h	V+
6	5d 0h	5d 5h	6d 8h	6d 13h	6d 14h	6d 19h	6d 20h	6d 22h	10d 0h	V-
7	10d 1h	10d 6h	10d 7h	10d 13h	10d 14h	10d 20h	10d 21h	11d 1h	14d 15h	V-

The first column of Table 1 specifies the maneuver number. The start and end of the Deep Space Network (DSN) track preceding the maneuver, if any, is given in the next two columns. Throughout the table, the start times start at the start of the hour specified, while the end times are for the end of the hour that is given. “Orbit Det.” is orbit determination. The maneuvers for this part of the trajectory are all either along the Earth-relative velocity vector (V+), or opposite it (V-). The initial lunar swingby is about halfway between the 5^{th} and 6^{th} maneuvers. There is an additional 6h DSN track planned 24h after the listed one before the 6^{th} maneuver. The goal of the first four maneuvers is to raise the periselene altitude of the lunar swingby, to prevent escape into heliocentric orbit (as is intended for the EM1 upper stage), while the next two maneuvers (around perigee) decrease the post-lunar-swingby apogee for passage through the WSB region to raise the perigee to the lunar orbit and allow the near-ballistic capture at the 2^{nd} lunar swingby two months after deployment.

For the new trajectories, calculations were made for three different power levels: 56 watts (the lowest throttleable value for operating the thrusters), 60 watts, and 65 watts, near the expected spacecraft maximum power. The thrust and Isp values are approximately proportional to the power level, but aren’t specified here due to their proprietary and export-control nature.

The trajectories were computed by varying the durations of the 5th and 7th maneuvers. Varying the 5th maneuver changed the altitude of the first lunar swingby, while changing the 7th maneuver changed the 2nd apogee distance (about 1 million km) to achieve the approximately 2-month period of the orbit needed to reach the Moon in December 2018, For comparison, a plot of the start of the complete trajectory of the previous section (as shown in the inertial frame in Figure 1) is shown in Figure 5a in the solar rotating frame. In that trajectory, the 2nd apogee is 57° counterclockwise from the direction to the Sun (towards the left), placing it in the WSB region that allows solar perturbations to raise the trajectory's perigee to the lunar orbit. The timing is also right (the apogee is just high enough) so that the Moon is there at the perigee, allowing capture into a loose lunar orbit (with some help with LunaH-Map's thruster). Since we know it reaches the desired science orbit with ample margin (that is, the trajectory "closes"), the new trajectories should be similar to maximize their chances for also closing. Plots of the three new trajectories in the solar rotating frame are shown in Figures 4, 5b, and 6.

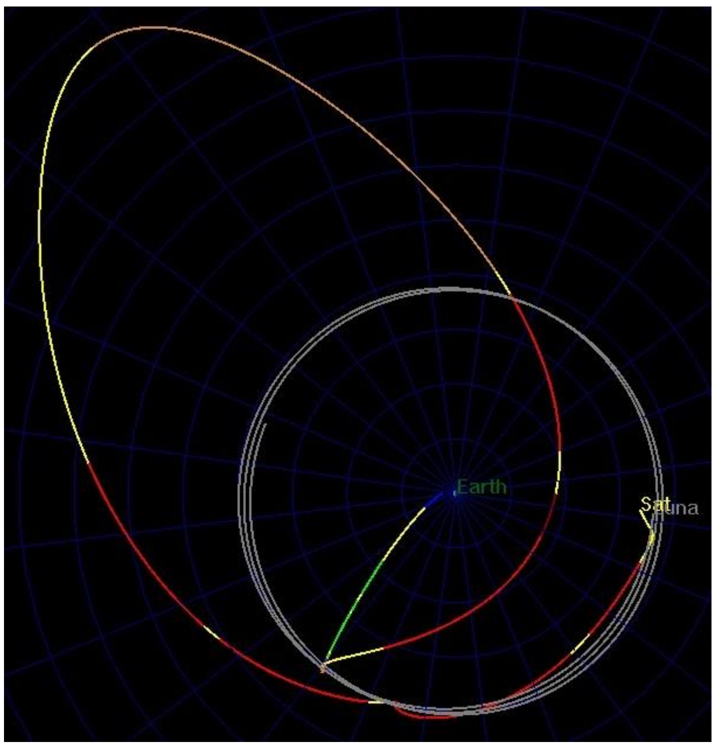

Figure 4. E12 Trajectory with power 65 Watts, ecliptic plane projection in the solar rotating frame. Thrust arcs are green, red, and purple; coast arcs are yellow and tan. The lunar orbit is gray. Deployment from the SLS upper stage occurs where the blue segment near the Earth turns yellow.

Although both the Figure 1 and E12 trajectories used the 65 Watts power level, the E12 calculations used slightly different values for the thrust and Isp, based on new tests that were completed after the Figure 1 trajectory was computed. The 2nd apogee shown in Figure 4 is 51° counterclockwise from the horizontal Earth-Sun line (the Sun is to the left in all of the solar rotating frame figures), so it passes a little deeper into the WSB area than the Figure 1 trajectory. The trajectory was computed only to the 3rd periselene and needs to match the Figure 1 trajectory better, but the effort to do that was reserved instead for the 60 Watt F19 trajectory portrayed and discussed next.

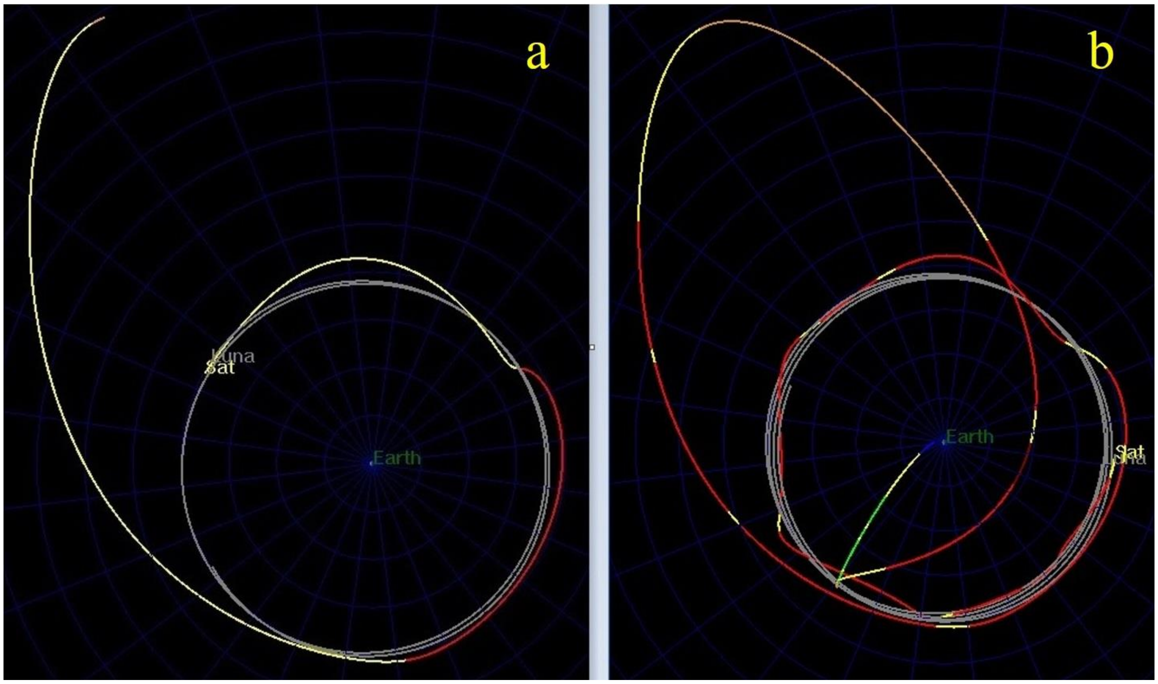

Figure 5. Trajectories in the solar rotating frame; the lunar orbit is gray. a: Part of the Figure 1 65-Watt trajectory, from the 2nd apogee to the 4th periselene. Thrust arcs are not indicated; the color changes at the periselenes. b: The new F19 60-Watt trajectory, color-coded like Figure 4.

Figure 5a (left side) shows the important lunar capture part of the complete trajectory shown in Figure 1. Part of the trajectory (2nd apogee to 4th periselene) of Figure 1, but shown in an ecliptic plane projection in the usual solar rotating frame, relative to the fixed horizontal Sun – Earth line, like in Figure 4. This shows the "loose capture" sequence that we want to try to match with the other trajectories. The 2nd apogee is in the WSB region that raises perigee to the lunar orbit that's shown in gray. The lunar orbit is gray in both plots.

Figure 5b (right side) shows the new 60-Watt trajectory, computed to show that the spacecraft can still be captured into a loose lunar orbit two months after the Bus Stop 2 release, in case the power system, for any reason, can't deliver the maximum planned 65 Watts. The durations of the 5th maneuver (the last, or lower-left-most, green segment before the first lunar swingby) and the 7th maneuver (the almost vertical red segment above and to the right of the Earth, after the first perigee) were adjusted to move the 2nd apogee into the WSB and to loosely capture at the periselene that's close to the 2nd perigee. Starting 16 days after the 2nd apogee and 19 days before arrival at the Moon, almost continuous thrusting in the anti-velocity (relative to the Moon) direction commenced. Maximum thrust durations were 5.5 days, always followed by at least a 0.5-day coast (shown as yellow interruptions in the trajectory on the plot) for DSN tracking and orbit determination.

It's apparent that the trajectories in Figures 5a and 5b are very similar. Both trajectories have no eclipses during the first four months of the mission; they miss the Earth's shadow when it passes close to the Moon in December 2018 and January 2019 (in spite of a total lunar eclipse that occurs on January 21; the spacecraft is near aposelene north of the Moon then). A comparison of some parameters of the trajectories is given in Table 2.

Table 2. Comparison of the Figure 5a and 5b trajectories.

Parameter\Trajectory	5a (& Fig. 1)	5b (F19)
2nd apogee dist., km	1,013,947	1,012,667
2nd apogee angle (°) from the Earth-Sun	57.39	59.29
2nd apogee inc., °	12.01	11.73
2nd periselene date, UT	2018Dec16, 6:32	2018Dec16, 2:54
2nd periselene dist., km	28,463	29,669
2nd periselene SMA, km	3,994,771	634,550
2nd periselene ecc.	0.9929	0.9532
2nd periselene inc., °	54.87	47.30
4th periselene date, UT	2019Jan03, 5:05	2019Jan02, 3:59
4th periselene dist., km	29,210	19,318
4th periselene SMA, km	40,088	37,041
4th periselene ecc.	0.2713	0.4785
4th periselene inc., °	89.66	96.01

The two trajectories can never be identical, since the thrust/Isp's are different. The ever-changing geometry of the four-body dynamics and long, low-thrust maneuvers greatly complicate the situation. As noted before, the angle from the Sun and distance of the 2nd apogee are important, and are rather closely matched by the F19 trajectory. The inclination of the trajectory is also important. The 12° inclination (to the ecliptic of J2000; "inc." in Table 2) is much higher than the lunar orbit inclination of 5.1°, so we investigated lowering it. A maneuver pointed south relative to the Earth-centered orbit lasting a day before the initial lunar swingby changed the B-plane location of the swingby and resulted in an inclination at the 2nd apogee of 8°, significantly closer to the lunar orbit, but after the loose lunar capture a month later, the inclination of the orbit relative to the Moon remained too low; it became clear that the higher Earth-orbit inclination was needed to end up in the desired high-inclination orbit at the Moon. Therefore, no out-of-plane thrusting was included in the new orbit calculations. We found that, as we spiraled down towards the science orbit in a roughly circular lunar orbit, we could change the inclination of that orbit by a few degrees by thrusting perpendicular to the orbit plane over arcs of 90° centered at the lunar equator (node) crossings for two weeks. Of course, the change was more effective at the higher altitudes, and very small amounts of out-of-plane thrusting during the loosely-captured phase, and even during the approach to the 2nd periselene, can change the inclination at the Moon by large amounts.

It's clear that, at the 2nd periselene, the spacecraft is very loosely captured, with a semi-major axis (SMA) much larger than the approximately 70,000 km distance from the Moon of the Earth-Moon libration points (that is, well outside the Hill sphere). But with continued anti-velocity thrusting, there are 3rd and 4th periselenes for both trajectories, with reasonably well-captured trajectories by the 4th periselene, which are not very different for the two trajectories. The aposelene of the F19 trajectory is still high enough to be near the Earth-Moon WSB, but continued thrusting results in a well-captured orbit. The F19 8th periselene occurs on 2019 Jan. 23, after the last possible Earth eclipse until June 2019; its parameters are: Distance 16,033 km, SMA 24,860 km, eccentricity (ecc. in Table 3) 0.3550, inc. 109.11°. For clarity, the F19 trajectory shown in Figure 5b ends at the 7th periselene on 2019 Jan. 18.

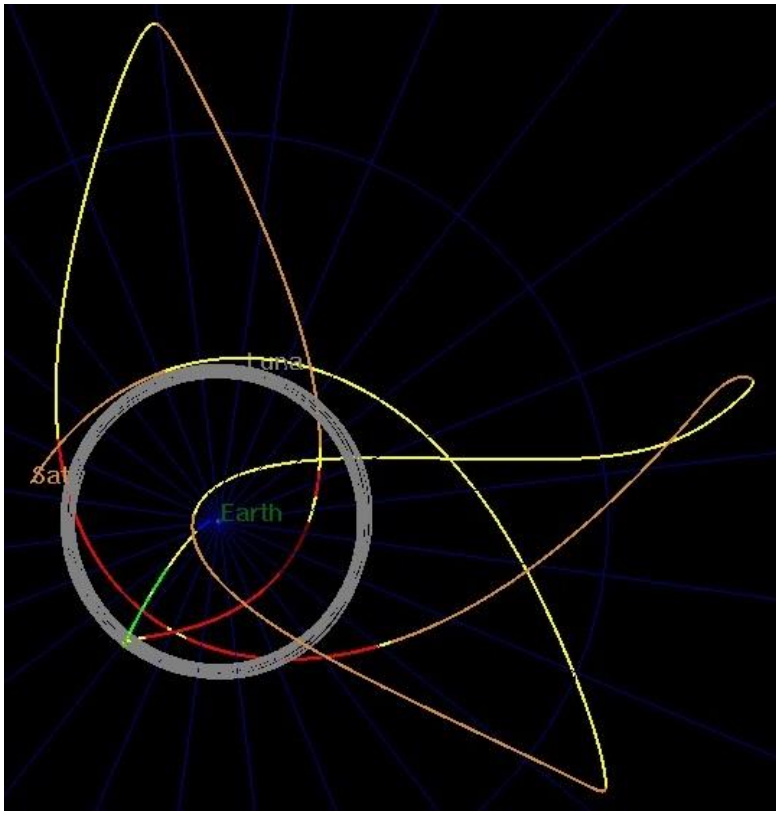

Figure 6. D10 Trajectory with the minimum power 56 Watts.

The D10 trajectory was computed with the lowest power level. The thrust was not sufficient, with any possible duration of the 5th and 7th maneuvers, to move the 2nd apogee close enough to the WSB (about 85° from the Earth-Sun line) to raise the 2nd perigee to the lunar orbit, so it was not possible to capture into the loose lunar orbit two months after the release, as was possible with the other trajectories. But after six more months and two more orbits of the Earth, the luni-solar perturbations raised the perigee to the lunar orbit, as shown near the top of the lunar orbit in Figure 6. However, the Moon was near the opposite side of its orbit, 540,000 km away, at that 4th perigee. The long anti-velocity burn around the 2nd perigee was needed to prevent escape into heliocentric orbit; adjusting its duration, and adding maneuvers around the 3rd apogee and 3rd perigee, should easily change the date of the 4th perigee enough to ensure that the Moon is there for a loose lunar capture. The six month extra cruise time, on top of the year-plus spiral down to the science orbit, would be detrimental to the mission, and it's not certain that there would be enough propellant to reach and fly the science orbit, with the lower thrust and Isp at this power level,

DISPERSED TRAJECTORIES

The trajectories in the previous section are all nominal trajectories, only dispersed (into three trajectories) in terms of the available power levels. But other trajectory dispersions need to be considered. In late January, 2017, NASA plans to provide about 200 release state vectors based on their Monte-Carlo analysis of the expected SLS dispersions for the EM-1 launch. Our analysis based on those will wait for a future paper since there is insufficient time to address them here. We look at other dispersion sources described in two subsections below.

Release Roll Angle Dispersions

LunaH-Map and 13 other cubesats will ride in an adapter ring between the Orion spacecraft and the top of the EM-1 SLS upper stage, called the Interim Cryogenic Propulsion Stage (ICPS). The adapter ring is called the MPCV Stage Adapter (MSA), where MPCV is Multi-Purpose Crew Vehicle. LunaH-Map will be deployed with 4 springs in a direction 34° from the ICPS spin axis, as shown in Figure 8. The springs should give LunaH-Map a ΔV of 1.2 m/sec, with the component in the spin axis direction then 1.2 x cosine (34°) = 0.995 m/sec. The spin axis will be pointed in roughly the Earth-relative anti-velocity direction. But there is also a component, 1.2 x sine (34°) = 0.671 m/sec, perpendicular to the spin axis. The roll angle for that component is unknown, so we calculated 4 trajectories, assuming that the roll angle was in each of 4 directions at 90° intervals of roll angle. Relative to the Earth-referenced orbit, these directions are North (perpendicular to the orbit plane), South, Left (in the orbit plane), and Right. Trajectories for each of these cases were computed that resulted in loose capture at the Moon two months later, shown in Figure 9.

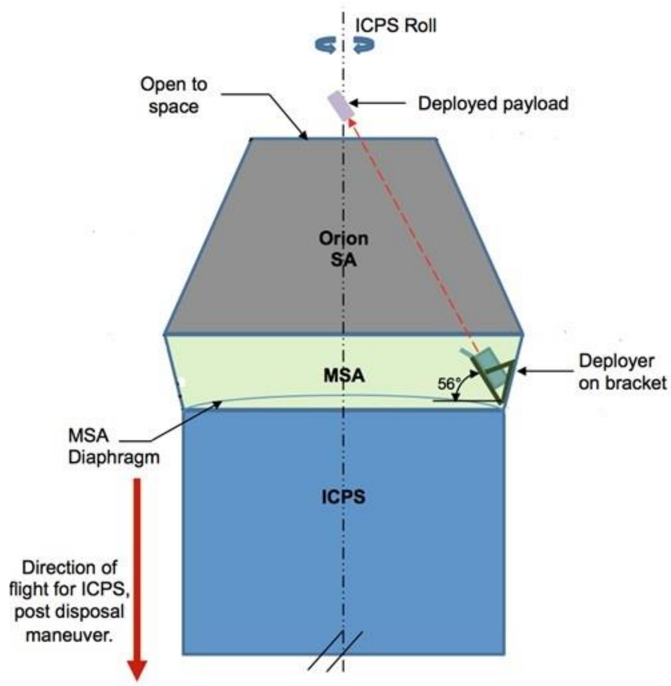

Figure 8. Diagram of the EM-1 upper stage and Orion, showing the deployment geometry.

The trajectory plots of Figure 9 use the same rotating ecliptic-plane projection and trajectory color-coding scheme that was used for Figures 4, 5b, and 6. They show that release in any roll angle direction should not be a problem for the LunaH-Map trajectory, with capture into loose lunar orbit accomplished in each case two months after the release.

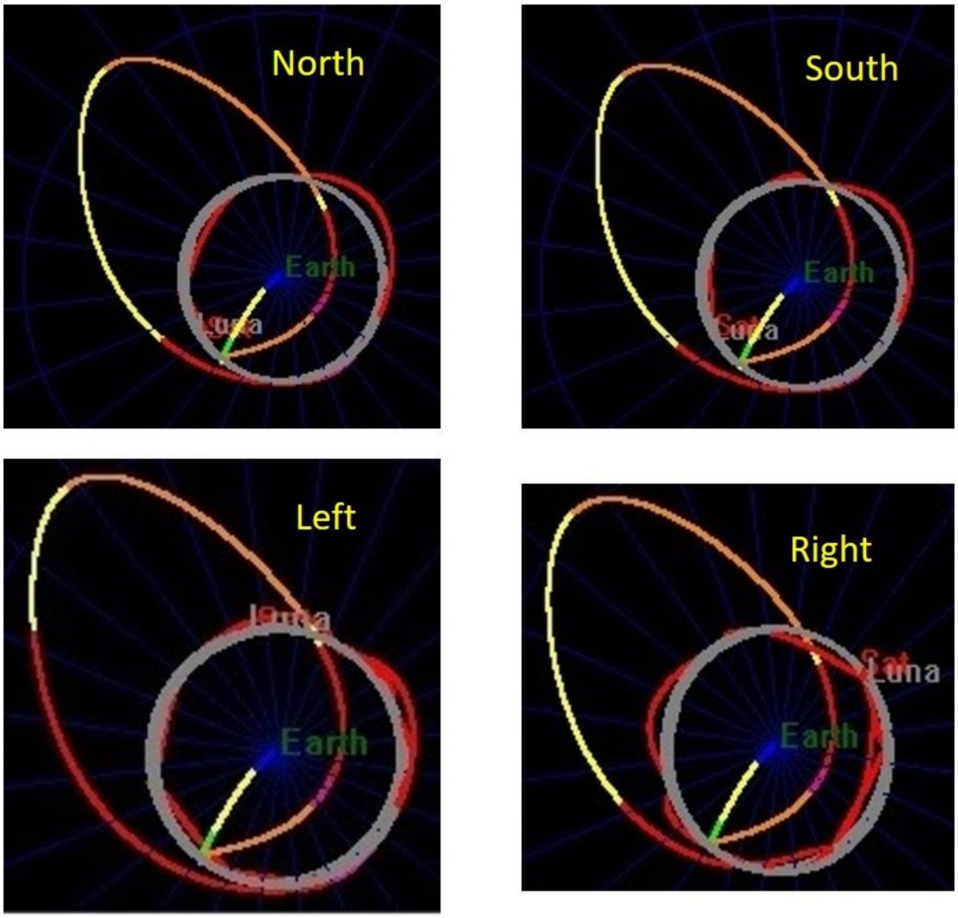

Figure 9. Solar rotating ecliptic-plane plot of trajectories computed with release in four different ICPS roll angle directions.

In addition to these four cases, a fifth case was computed that assumed no component perpendicular to the spin axis (like the nominal case), but with the along-axis component half of the nominal value, or 0.497 m/sec, in the very unlikely event that 2 of the 4 deployment springs failed. It resulted in a loose lunar capture that looked very much like those shown in Figure 9.

Cubesat Maneuver Execution and Navigation Dispersions

For the critical first phase of the mission, a Monte-Carlo analysis mapped plausible maneuver execution and orbit determination errors to 1-sigma uncertainties at the first lunar swingby on 2018 Oct. 13 at 10h UTC, as shown in Table 3. For DSN tracking, weights used were 0.25 mm/sec for Doppler and 100 meters for range. The assumed 1-sigma maneuver execution errors were ±0.2° for thrust vector control and ±5.11% for thrust magnitude.

Table 3. Uncertainties mapped to the 1st Periselene.

1 Sigma Uncertainty	Initial Uncertainty	Uncertainty at the 1st Periselene
V x H (km)	1.0436×10^4	15.62
V (km)	1.7501×10^4	50.85
H (km)	1.3391×10^3	38.40
B•T (km)	1.2997×10^4	19.84
B•R (km)	4.5029×10^3	12.08
LTOF (seconds)	3.4581×10^3	14.44

In Table 3, V is along the velocity vector, H is along the angular momentum vector (that is, perpendicular to the orbit plane), and V x H is orthogonal to the other two (that is, in the orbit plane and 90° from the velocity direction). B•T and B•R are the coordinates in the B-plane, while LTOF (left time of flight?) is the component perpendicular to the B-plane, divided by the velocity of approach to the B-plane. Errors of the size shown in Table 3 could be compensated for by changing the duration of the 7th maneuver, and following the first perigee (and after the 7th maneuver), adding an out-of-plane thrust arc near true anomalies 120° and/or 240°.

LUNAR SCIENCE ORBIT

In an attempt to eliminate the need for any RAAN constraint in the lunar orbit, a quasi-frozen orbit class was sought such that any polar lunar orbit with periselene above the south pole would be compatible with the LunaH-Map mission. Given the constraints for the science orbit (90-degree inclination, argument of periselene = 270 deg, periselene altitude/height < 40 km), the search focused on finding the maximum acceptable aposelene altitude that would yield the required 141 science passes over the south polar region for all RAAN values. This searched revealed the desired aposelene altitude with a value of 3,150 km (this exact value was investigated in detail since the corresponding period is a multiple of the lunar orbit period).

An example of this orbit for the nominal case (RAAN ~350 degrees) is shown in Figure 10. The minimum height above terrain is plotted for every periselene pass in Figure 4 as well. This minimum height above terrain is measured as the nadir-distance/altitude above terrain and is determined after calculated this parameter throughout an arc centered on periselene using the most advanced lunar terrain model available within STK/Astrogator. A depiction of a pass above Shackleton crater is seen in Figure 4; the minimum height above terrain is constrained to 7 km (to ensure that at least 5 km range to terrain), which yields an 11 km height above Shackleton's crater floor in this example.

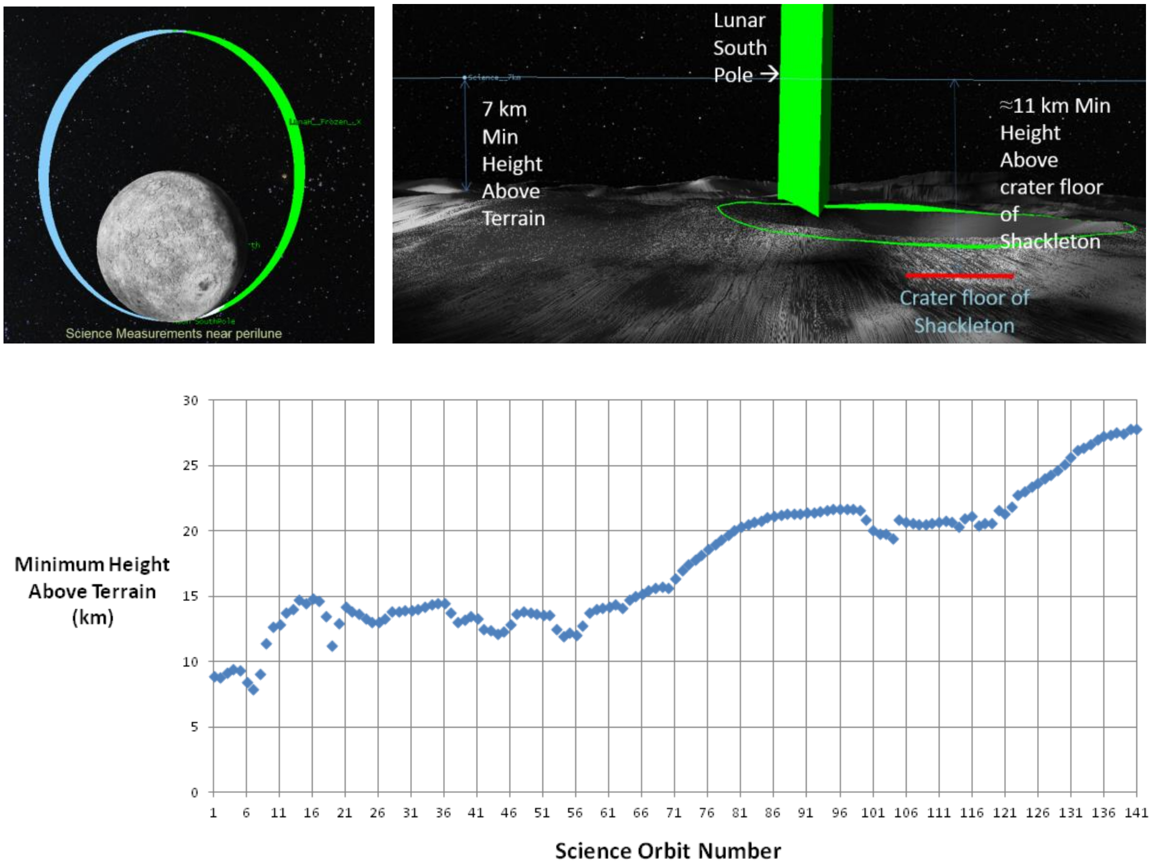

Figure 10. Minimum Height above terrain (bottom) for Nominal Science Orbit (top-left) with pass above Shackleton Crater (top-right); orbit images are shown in the Moon inertial frame.

As can be seen in Figure 11, minimum height above terrain profiles for varying RAAN values are plotted for 141 consecutive orbits (to meet the relevant science requirement). RAAN was varied in 30-degree increments from 0 to 330 degrees to cover the full range of RAAN possibilities; all analyzed minimum height profiles remain under 40 km for all 141 passes. This is how the 3,150 km aposelene altitude requirement was derived; the relative stability of this orbit allows one to eliminate a constraint on RAAN and simplify the science phase since no maneuvers would be required to obtain the necessary science via 141 passes above the lunar south pole at less than 40 km from the terrain.

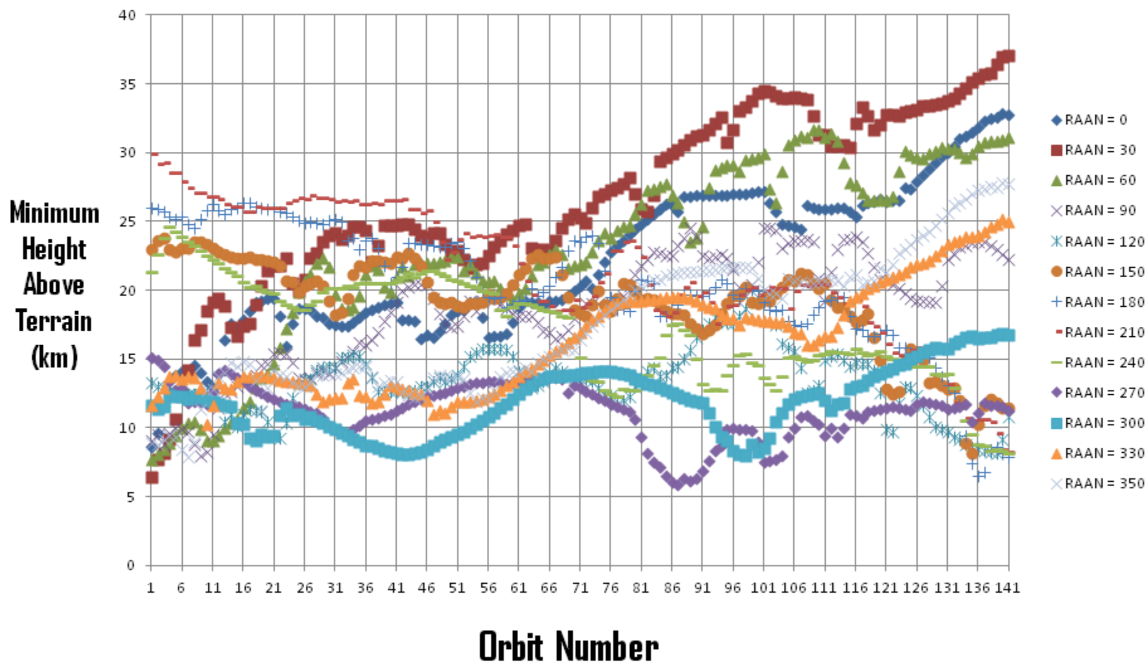

Figure 11. Minimum Height Above Terrain per pass plotted for varying RAAN profiles, in 30-degree increments from 0 to 330 degrees.

CONCLUSIONS

A trajectory design has been presented that enables a 6U CubeSat to reach a quasi-frozen lunar orbit suitable for valuable scientific-return with regard to hydrogen mapping of the lunar south pole.

A summary of the trajectory results is shown in Table 4. Only 0.058 kg of iodine propellant was calculated to reach lunar orbit given the starting ride of EM-1's disposal stage. However, such an orbit needed to be lowered to reach the required science orbit. Although the total mission duration is shown as greater than one year, a relatively safe approach was taken with regard to the spiral-down phase of the orbit. As mentioned, the post-capture circular orbit of 8,000 km is relatively stable (for > 120 days of no maneuvers), amenable to choosing the argument of periselene, and high enough to allow for eclipse avoidance via period adjustments. The total propellant expended was calculated as 1.15 kg (out of the available 1.5 kg), leaving ample room for margin, contingencies, and extended mission opportunities. However, as is the mission will plan to end after 141 science measurements by executing a lone maneuver near aposelene to target an acceptable disposal location on the lunar surface, far away from historic sites to ensure disposal requirements are met.

Table 4. LunaH-Map Nominal Trajectory Summary

Mission Event/Phase	Prop Required	Duration Required
Deployment to Lunar Capture	0.058020 kg	69 days
Transition Phase {capture --> science orbit}	0.989484 kg	472 days
Science Phase: Orbit Maintenance	0.050000 kg	26 days
Disposal: 1 maneuver to lunar impact	0.000560 kg	1 day
Navigation TCMs & Attitude Control	0.0500000 kg	n/a
Total (excluding margin)	**1.148064 kg**	**568 days**
Propellant Margin Available	**+0.351936 kg {23.4%}**	n/a

ACKNOWLEDGEMENTS

The authors appreciate the support provided for this study by NASA HQ via the SIMPLEX program as well as from their host center/institution: NASA Ames Research Center's Mission Design Division and KinetX. Additionally, the authors thank two others in KinetX's Space Navigation and Flight Dynamics Practice, Derek Nelson for providing the information in Table 1, and Jeremy Bauman for the results in Table 3, and Andres Dono Perez of Millennium Engineering and Integration Co. with regard to automating STK's height-finder function for the science orbit.

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