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LUNAR CUBE TRANSFER TRAJECTORY OPTIONS

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Numerous Earth-Moon trajectory and lunar orbit options are available for Cubesat missions. Given the limited Cubesat injection infrastructure, transfer trajectories are contingent upon the modification of an initial condition of the injected or deployed orbit. Additionally, these transfers can be restricted by the selection or designs of Cubesat subsystems such as propulsion or communication. Nonetheless, many trajectory options can be considered which have a wide range of transfer durations, fuel requirements, and final destinations. Our investigation of potential trajectories highlights several options including deployment from low Earth orbit (LEO), geostationary transfer orbits (GTO), and higher energy direct lunar transfers and the use of longer duration Earth-Moon dynamical systems. For missions with an intended lunar orbit, much of the design process is spent optimizing a ballistic capture while other science locations such as Sun-Earth libration or heliocentric orbits may simply require a reduced Delta-V imparted at a convenient location along the trajectory.

INTRODUCTION

Cubesats are being proposed for many missions as a low cost efficient alternative to traditional mission and spacecraft architectures. Cubesats can be launched as secondary payloads into many classes of deployment orbits from which they can easily be transferred to their science orbit, and in our application, a lunar orbit. Several NASA studies, including a recent investigation at GSFC, have demonstrated that the cubesat paradigm can be used to support high priority science goals. Subsystem design from these studies includes state of the art attitude control, propulsion (for transportation from LEO, GTO or Earth escape to lunar capture), communication, power, thermal and radiation protection systems which provide lunar orbital operations of a cubesat bus. Based on this work, it was concluded that a 6U bus with state of the art cubesat systems available now or being built and tested can support a high priority science orbiter delivered to lunar orbit.[1] The standard 10×10×10 cm basic Cubesat is often called a "one unit" or "1U" Cubesat and are scalable along only one axis, by 1U increments. A 6U bus dimension could then be 10x20x30 cm. Based on basic lunar orbital missions; a series of progressively more challenging missions appears feasible including an impactor and a pathfinder observatory with consideration of designs using technology available in 2014, in five years, and in ten years. Particular challenges for orbiters or impactors are communication, navigation and tracking in a volume, power, and bandwidth constrained environment. Thermal and radiation protection will likely be the principal challenges for landed cubesat. The end result is generic design(s) for a cross-section of future high priority payloads for planetary, heliophysics, and astrophysics disciplines.

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Opportunities to propose deep space cubesat missions (HTIDeS for Exploration Mission-1) demonstrate that the onboard control systems required for operating outside of low Earth orbit (e.g., propulsion, active attitude control) as well as adequate communication capability, thermal and mechanical design, are becoming available. This expanding capability offers an opportunity for pathfinder or multi-platform distributed missions for relatively low cost at a time when the cost constrained environment has rendered serious curtailment of more conventional mission options. In fact, the availability of distributed measurements will allow observations of the truly dynamic nature of interactions on bodies throughout the solar system. Thus, as funding declines and costs increase for conventional space missions, low-cost cubesat technology has matured over the last decade to the point where the increasing capability of this approach to support science-driven applications has translated into its emergence as a significant method for access to space for the NASA Science mission Directorate (SMD) Heliophysics (e.g., CINEMA, DICE, RAX-2), Astrophysics (e.g., ExoPlanetSat, CXBN), and Earth Application (e.g., QuakeSat, Firefly) communities.

To exploit cubesat considerations and designs, options using limited deployment and propulsion subsystem drivers have been analyzed. The high-priority designs for investigation included those in which both impulsive and low-thrust Solar Electric Propulsion (SEP) engines are employed to place the cubesat into a highly eccentric Earth orbit or an orbit dynamically similar to a libration point orbit transfer trajectory, that subsequently delivers the vehicle to the Moon's sphere of influence, and finally achieves a highly eccentric lunar orbit. Such low-thrust transfers are feasible with a realistic microthruster model, assuming that the cubesat can generate sufficient power for the SEP. Three deployment examples are highlighted: LEO orbit, Geosynchronous Transfer Orbit (GTO), and a high energy Exploration Mission 1 (EM-1) trajectory used for a direct, free-return lunar trajectory. From these deployed orbits, employing low-thrust or impulsive maneuvers we can jump onto a local Earth-Moon transfer manifold or in the EM-1 injected initial design, we change the EM-1 targeted lunar flyby distance to alter the energy of the lunar flyby to match that of a typical Sun-Earth/Moon system heteroclinic manifold or to achieve a highly elliptical Earth orbit. Low-thrust maneuvers are located along the trajectory to raise perigee to that of a lunar orbit, adjust the timing with respect to the Moon, rotate the line of apsides, and target a ballistic lunar encounter. In the EM-1 design a second flyby can further decrease the orbital energy with respect to the Moon, so that $C3 < -0.1\ km^2/s^2$. Other designs emanating from a LEO or a GTO use impulsive maneuvers to phase onto a local Earth-Moon manifold, which then transfers the cubesat to a lunar encounter in several weeks. Our investigation concludes with several design options which provide estimated Delta-V (ΔV) requirements, achieved lunar orbit parameters, and associated transfer trajectory information. The use of GSFC dynamical systems and high fidelity mission design tools are also demonstrated in generated these results.

CONSTRAINTS, APPLICATIONS

The low thrust levels investigated vary from μN to mN to remain within the aforementioned 6U bus with state of the art cubesat systems. These thrust levels permit control of a primary mission trajectory and correction of major injection energy errors. For an impulsive ΔV design, we simply modeled the ΔV magnitude without consideration of the propulsion system volume or propellant to provide a ΔV comparison to low-thrust feasibility. The selection of the propulsion system can also limit the control authority and trajectory modifications. With possible power limitation, (power levels of less than 75W have been recommended), the low thrust level can be reduced to an inefficient level. There is also a concern on attitude control and pointing constraints of a low thrust system, which may impede use or drive thruster designs as well as placement and number. Based on a simple rocket equation, an impulsive ΔV design drives fuel mass and propulsion system volume, and the deterministic ΔV drives location and the related efficiency, thus a full trade between low-thrust and impulsive ΔV is recommended for any real mission design.

In addition to the propulsion and ΔV requirements, the deployment method further constrains the transfer design. Launch vehicle capabilities and the ensuing profiles and related primary trajectories cannot be easily modified since secondary payloads cannot drive primary mission goals. The primary mission design constrains the launch/ injection parameters while the injection energy can vary over launch period or window. Additionally the number of launch opportunities can limit a successful design. For example, for our three injection options, limitations can include launch dates, parking orbit inclination and nodal precession and related atmospheric drag, line of apsides alignment of an elliptical orbit, and for EM-1 varying injection energy over the launch window and an unknown trajectory (apoapsis) direction as the launch date changes.

Design Methodology

The transfer designs in this analysis use a combination of more traditional orbital analysis, that is, elliptical Earth orbits modified by low-thrust or impulsive accelerations and state-of-the-art dynamical systems applications to meet the endpoint lunar orbit goals. The traditional approach of applying maneuvers at an optimal location to modify the transfer orbit to target end conditions was completed using a differential corrector and an SQP optimizer.[2] These two approaches are well established and widely accessible. The dynamical systems approach is summarized since the tool used to design the initial transfer trajectories has only recently been developed under a GSFC Internal Research and Development (IRAD) effort for application to astrodynamics transfer problems.

The present article builds upon a presentation by some of the authors at a Lunar Cubes 2013 Workshop[3]. The name of that presentation was inspired by a 2000 survey paper by Biesbroek and Janin[4]. The literature on low-energy Earth-Moon transfers is extensive, and we can only give a partial survey here. In 1968, a seminal paper by Conley[5] described how to use dynamical systems theory to achieve low-energy transfer via a corridor about the Earth-Moon L1 point. In 1987 Belbruno[6] described a way to combine low thrust with the L1 corridor to reach the Moon. In Sept 2003 the European Space Agency launched the SMART-1 mission into Earth orbit. The SMART-1 spacecraft then employed a combination of low thrust, resonant orbits and the L1 corridor to reach lunar orbit in February 2005 after a 13-month transfer[7]. Belbruno and Miller[8] developed an alternative transfer method in the multibody regime, where a spacecraft is sent from Earth toward the Sun-Earth L1 or L2 point in order to raise perigee and so approach lunar orbit. This technique was applied to the Japanese spacecraft Hiten, and has since been applied to the GRAIL and ARTEMIS missions. The ARTEMIS mission is discussed further below.

In 1991 Sweetser[9] estimated the minimum Delta-V required to reach the Moon from Earth orbit. In 1995 Pernicka et al.[10] performed a search for low-energy trajectories. Kluever and Pierson[11] and Herman and Conway[12] focused on the optimal control problem of low-thrust transfer. Koon et al.[13] extended the dynamical systems approach to low-energy transfer. One of the challenges in the design of a low-thrust trajectory is that there can be several measures of performance, such as time-of-flight and mass delivered, that can compete with each other. To allow multiobjective optimization in trajectory design, Schütze et al.[14] used search space pruning, while Vavrina and Howell[15] used a genetic algorithm. Howell and Ozimek[16] investigated low-thrust trajectories in the Earth-Moon system, focusing on transfer to a libration point orbit. Mingotti, Topputo and Bernelli-Zazzera[17,18] have examined low-energy transfers using either high- or low-thrust propulsion. The book **Space Manifold Dynamics**[19] contains a collection of articles on low-thrust, low-energy transfer. More recently Parker, Anderson and Lo have undertaken an extensive study of low-energy Earth-Moon transfers, documented in the book **Low Energy Lunar Trajectory Design** [20] and references therein.

Dynamical Systems Approach

Overall, a dynamical systems approach is focused on the long-term qualitative behavior of a complex system. It employs continuous and discrete differential equations to model the behavior of the system and, as such, it remains a deterministic system in which the nonlinearity leads to complexity but not necessarily a loss of predictability. In a preliminary design of deployment trajectories for Lunar Cubes (Cubesats), we are not focused on precise solutions, but on general exploration of the space (periodic orbits, quasi-periodic motion, chaos).The description of a lunar transfer trajectory as a dynamical systems application is useful. A brief examination of a Lorentz attractor (Figure 1), illustrates a nonlinear system. It is a first representation in phase space, in 1963, as an example that highlights the sensitivity of a small change in the initial conditions on the final state. In such a system, the response is loosely predictable, but can yield a wide range of end conditions.[21] Furthermore, knowing that we are working with the Sun-Earth/Moon dynamical region, where small initial energy changes from a maneuver or lunar flyby allows a large variety of transfer trajectory design options, permits the analyst to use invariant manifolds and Poincaré maps of these manifolds defined by crossings of the hyperplane, Σ, to locate long-term capture trajectories about the smaller primary in a Circular Restricted Three Body Problem (CR3BP); see Figure 2. [21,22,23,24]

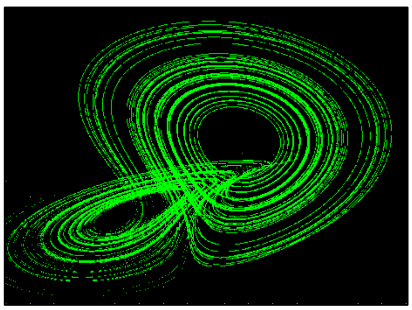


Figure 1. Lorentz Attractor of a nonlinear system sensitivity

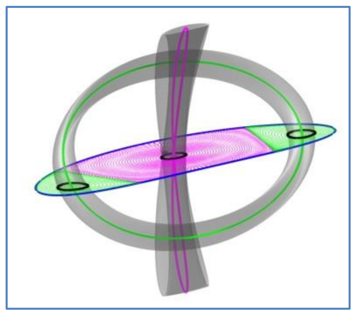


Figure 2. Earth-Moon Poincaré map of manifolds on a hyperplane

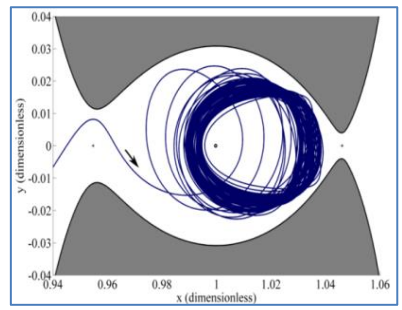


Figure 3. CRTB trajectory design for a lunar orbit

With recently developed tools and an understanding of the overall Jacobi energy level, a preliminary, but accurate design can be constructed to demonstrate a Circular Restricted Three-Body (CRTB) lunar capture concept, as shown in Figure 3.[25] The figure illustrates the design of a weakly stable lunar orbit from analyzing a trajectory on a manifold that is delivered to a given energy level. Using preliminary designs as an input to higher-fidelity tools which incorporate accelerations with an ephemeris model, operational level trajectory designs can be realized.

In designing Earth to lunar transfers that utilize a dynamical systems approach, one can envision the use of manifolds in the Sun-Earth/Moon or in a local Earth-Moon environment. With the EM-1 higher energy injection and with its specific lunar targets, a trajectory configuration can be considered as using the Moon to get to the Moon by augmenting the first post-lunar flyby energy to attain a particular Sun-Earth/Moon manifold that returns the Lunar Cube to the lunar gravity well with a low ballistic energy. For the Earth-Moon local manifolds, one can focus on manifolds that approach and depart from the lunar environment and can be modeled with respect to the energies of Earth-Moon libration point orbits or unstable lunar orbits. These manifolds use a scenario that can be easily modified by a low thrust or impulsive propulsion system. In achieving the desired lunar orbit, lunar capture energy and geometry conditions must also be considered along with the lunar orbit science drivers, e.g. terminator altitude, lifetimes.

Adaptive Trajectory Design

At the level of a simplified CRTB model, an autonomous system is used to first categorize and fit possible trajectories.[26] A CRTB model provides useful information about fundamental solutions (libration point orbits, stable/unstable invariant manifolds, retrograde orbits …) from which solutions are transitioned to an ephemeris model, which can generally maintain orbit characteristics. Shown in Figure 4 is a sample Adaptive Trajectory Design (ATD) result which was implemented in the GSFC developed General Mission Analysis Toll (GMAT). ATD is a GSFC / Purdue tool used for initial designs of trajectories in the Sun-Earth-Moon region. ATD provides the user with value added data on generation and stability of multi-body orbits, connecting trajectories and any associated manifolds.

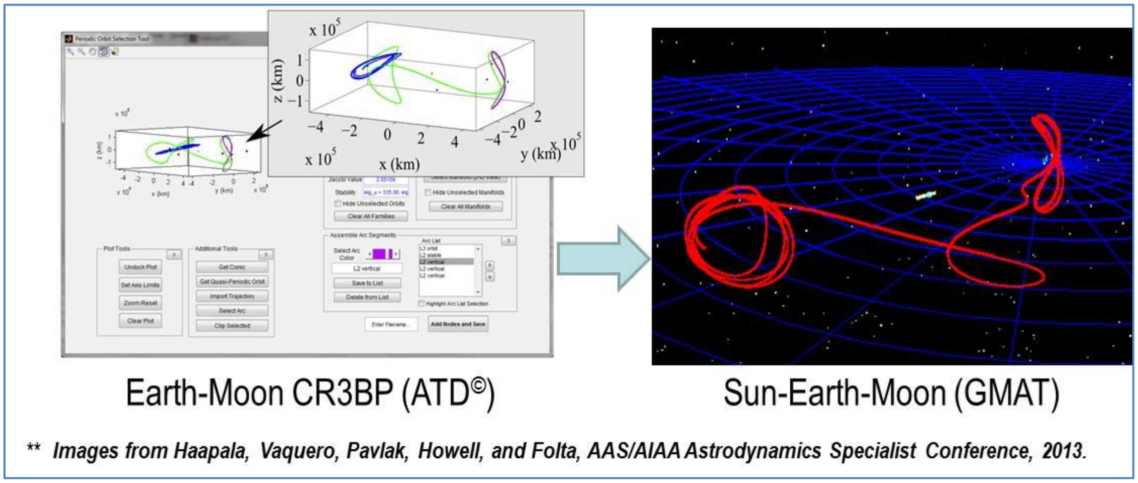


Figure 4 Adaptive Trajectory Design (ATD) Tool

As an example of how ATD is used for Lunar Cube trajectory analysis and to provide some background on manifold generation, consider a lunar libration point orbit to get a representative energy level and environmental conditions. A local stable/unstable manifold is computed by introducing a perturbation in the orbit state (fixed point), $x^* = T(\tau)$, $0 < \tau < T$, along a periodic orbit in the direction of the stable/unstable eigenvector associated with the monodromy matrix, $\Phi(\tau+T,\tau)$, corresponding to x^*. Assume that $\lambda_S < 1$ and $\lambda_U = 1/\lambda_S$ are stable and unstable eigenvalues of the monodromy matrix associated with a periodic orbit. Let w^U and w^S be their associated eigenvectors, and define W^{u+}, W^{u-}, W^{S+}, W^{S-} as the two directions associated with each eigenvector. The local half-manifold, $W^{U-}_{x^*,loc}$ ($W^{S-}_{x^*,loc}$), is approximated by introducing a perturbation relative to x^* along the periodic orbit in the direction W^{U-} (W^{S-}). Likewise, a perturbation relative to x^* in the direction W^{U+} (W^{S+}) produces the local half-manifold $W^{U+}_{x^*,loc}$ ($W^{S+}_{x^*,loc}$). The magnitude of the step along the direction of the eigenvector is denoted d, and the initial states along the local stable manifolds are computed as $x^{S+} = x^* + d \,.\, w^S$, $w^{S-} = x^* + d \,.\, w^{S-}$, where w^{S+} and w^{S-} are normalized so that the vector containing the position components of the eigenvector

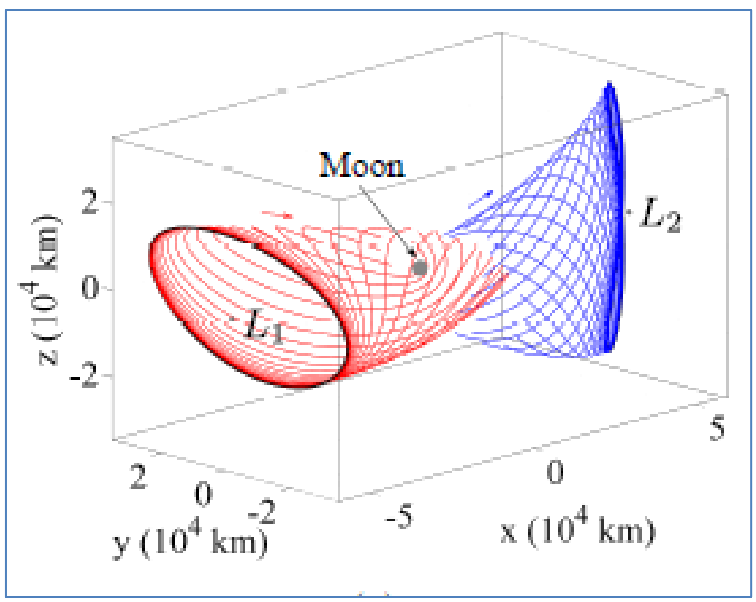


Figure 5: Sample stable (blue) and unstable (red) manifolds associated with Earth-Moon libration point orbit

is of unit length; this normalization provides a physical meaning for the value of *d* as a distance. The local stable/unstable manifolds are globalized by propagating the states x^{S+} (x^{S-})/ x^{U+} (x^{U-}) in reverse-time/forward-time in the nonlinear model. This process yields the numerical approximation for the global stable manifolds, W^{S+} (W^{S-}), and unstable manifolds, W^{U+} (W^{U-}). The value of *d* is critical because it determines the accuracy with which the global manifolds are approximated. Selecting *d* too small yields manifold trajectories that require long integration times before departure from the vicinity of the periodic orbit, leading to accumulation of numerical error. If *d* is too large, then the approximation to the local manifold is poor. Here, a value of *d* = 20 km is selected so that propagating the initial state along the manifold back toward the periodic orbit, i.e., propagating W^{S-}, W^{S+} in forward-time and W^{U-}, W^{U+} in reverse-time, yields a manifold trajectory that remains in the vicinity of the periodic orbit for at least two revolutions. The collection of all unstable manifolds forms the surfaces W^{U+} and W^{U-} that reflect asymptotic flow away from the periodic orbit. Likewise, the collection of all stable manifolds forms the surfaces W^{S+} and W^{S-} that reflect asymptotic flow toward the orbit. In Figure 5, a subset of trajectories on the unstable/stable manifold associated with an L1 northern halo/L2 vertical orbit in the Earth-Moon system are propagated for a fixed time interval, and are plotted in red/blue. In this analysis, families of quasi-periodic tori and their associated manifolds are computed numerically using techniques and is applied in the ATD tool. [27,28,29]

OPERATIONAL MISSION MANIFOLDS

Evidence of the benefit of dynamical systems & manifolds applied to mission design was accomplished in 2009-2010 with the "Acceleration, Reconnection, Turbulence and Electrodynamics of the Moon's Interaction with the Sun" (ARTEMIS) mission. Two spacecraft were transferred from elliptical Earth orbits into elliptical lunar orbits.[30,31] To accomplish this transfer, the operations team used a dynamical systems (manifold) approach with numerical targeting. The spacecraft employed a "low" thrust, 4N, propulsion system with a thrust direction constrained to the southern ecliptic hemisphere on a spinning spacecraft. Orbit-raising maneuvers were performed near periapsis of the initial elliptical orbit to raise apoapsis to lunar distance from which Lunar Gravity Assists (LGAs) were used to raise periapsis and to align the trajectory for Earth-Moon libration point orbit insertion. Figure 6 presents the ARTEMIS transfer trajectories that encompass the similar dynamics as that of the proposed EM-1 deployed Lunar Cube design. The ARTEMIS P1 spacecraft operational trajectory shown on the left used the dynamics of the Sun-Earth L1 region while the P2 spacecraft operational trajectory, shown on the right, used the dynamics of the Sun-Earth L2 region. Both used lunar gravity assist to increase the energy of the system to permit the solar gravity interaction to raise perigee to the lunar orbit radius. These trajectories represent one of the design options from the computed Sun- Earth/Moon manifolds.

To underscore the optional manifolds available to ARTEMIS, consider only the P1 outbound arc away from Earth. Figure 7 shows the related manifolds (green trajectories) and the executed P1 trajectory (red). Following the outbound path to the location of a correction maneuver which shifted the spacecraft onto a more optimal (orange) manifold, presents how the trajectory 'jumped' from one manifold to another which resulted in an optimal libration point orbit target arrival. Subsequent to and along the outbound trajectory, two outbound manifold arcs emerge which represent potential outcomes from flow along the optimal path and the alternative that incorporates a possible correction maneuver. These two manifolds are exaggerated in the Z-axis direction to highlight the difference and where they intersected (which also represents the correction maneuver location).

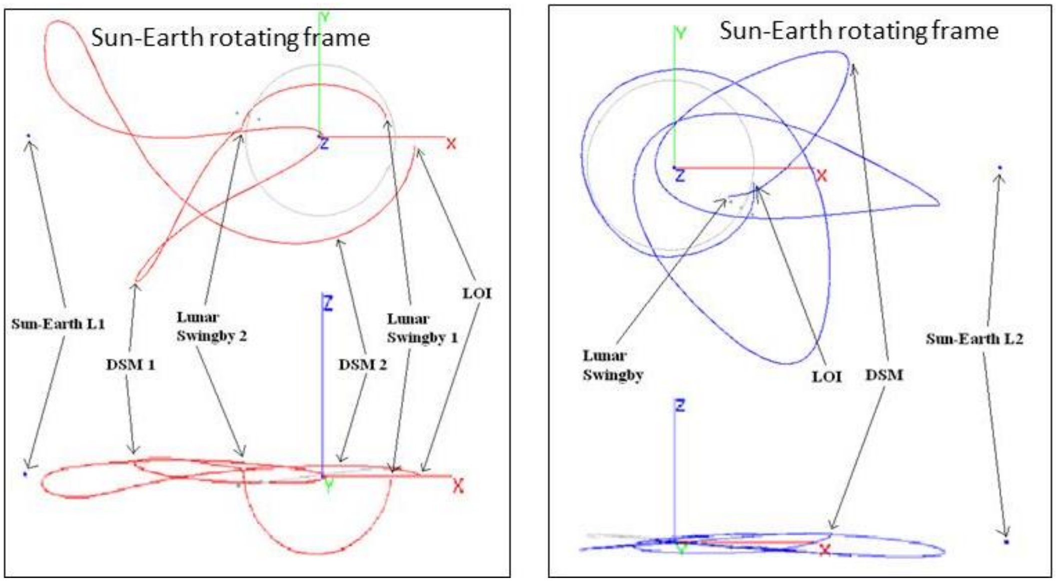


Figure 6. ARTEMIS P1 (left) and P2 (right) Operational Trajectories Shown in a Sun-Earth/Moon Rotating System

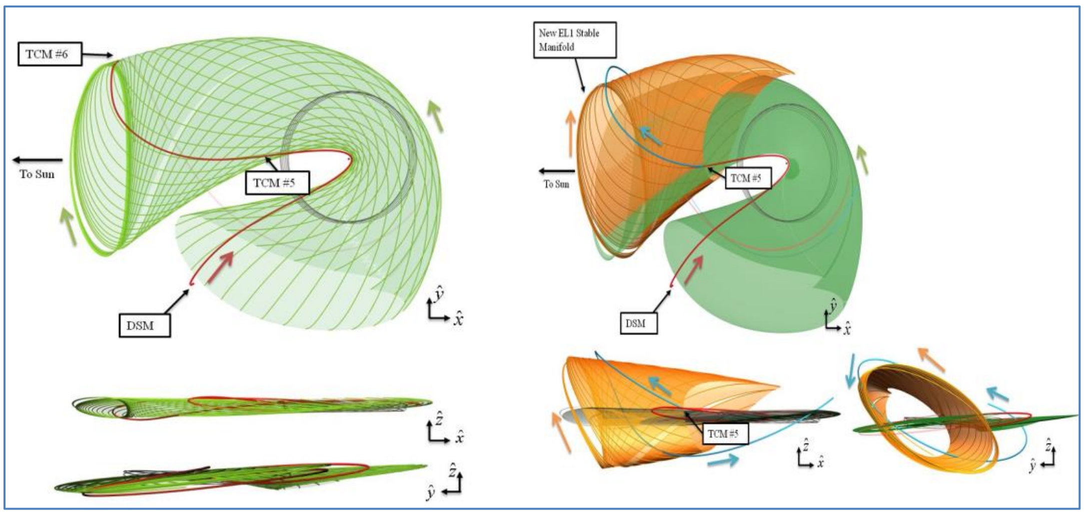


Figure 7. P1 Manifolds; Intermediate Manifold (right), Correction Manifold showing Maneuver Locations (left)

DEPLOYMENT OPTIONS

Several deployment strategies are analyzed; LEO orbit, Geosynchronous Transfer Orbit (GTO), and a high energy Exploration Mission 1 (EM-1) trajectory. The EM-1 option is the prime design used for a direct, ~ 5 day, free-return lunar transfer. From these orbits employing low-thrust or impulsive maneuvers we can jump onto a local Earth-Moon transfer manifold or in the EM-1 injected initial design, we change the EM-1 targeted lunar flyby distance to alter the energy of the lunar flyby to match that of a typical Sun-Earth/Moon system heteroclinic-type manifold or achieve an elliptical Earth orbit. While several examples are provided below, it should be pointed out that these examples reflect only a subset of the numerous and feasible options that exist using these deployment choices. Other deployment options can be used with the motivation of the paper as well.

The local Earth-Moon manifold has a particular geometry and design that is based on the Earth and Moon dynamics. This manifold, as illustrated in Figure 8, provides a background on the types of trajectories desired for a natural flow towards either the Moon or the Earth-Moon libration point orbits, EML_1 or EML_2. The premise is that a spacecraft is inserted onto an intermediate orbit which asymptotically converges onto the manifold (a best case scenario) or intersects with the manifold which then requires a manifold matching ΔV to place the spacecraft onto one of the trajectories of the manifold which then flows to the region of lunar interest. The coordinate frame used in Figure 8 is an Earth-Moon rotating system applying the dynamics of a CRTB system.

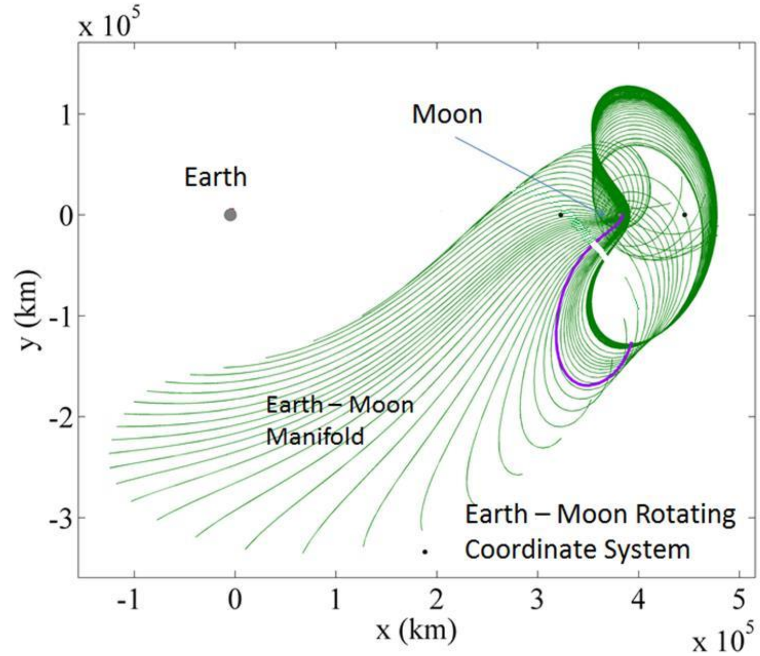


Figure 8. Example of Earth-Moon Local Manifold

LEO Deployment

The first option investigated for a Lunar Cube transfer to the Moon was for a design from a deployment in a 400-km circular orbit with an inclination of 24 deg. These designs required impulsive ΔVs to attain an intermediate trajectory that will intersect a local Earth-Moon manifold which then permits a natural flow to the Moon or into an Earth-Moon libration point orbit from which one could transfer to a lunar orbit. A low-thrust propulsion option is also available, but tends to result in a long duration (~ 1 year) outward spiraling trajectory that ballistically encounters the Moon sphere of influence region. While by definition, the spiral orbit will intersect (or converge) with the local manifold and then using the low thrust, maneuver onto a lunar bound manifold, this option was not examined and analysis was limited to impulsive maneuvers for shorter transfer durations to minimize expendables and radiation effects.

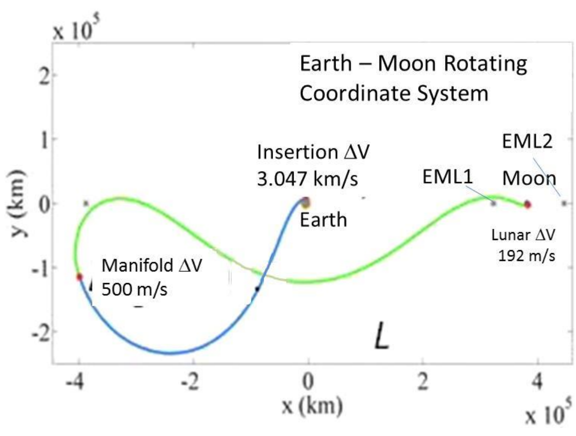


Figure 9. LEO with Intermediate Transfer which Asymptotically Matches the Manifold

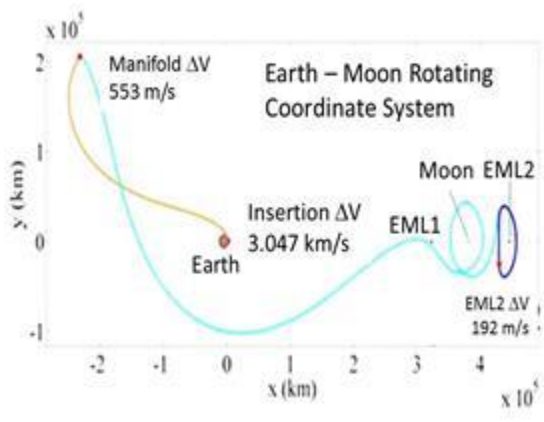
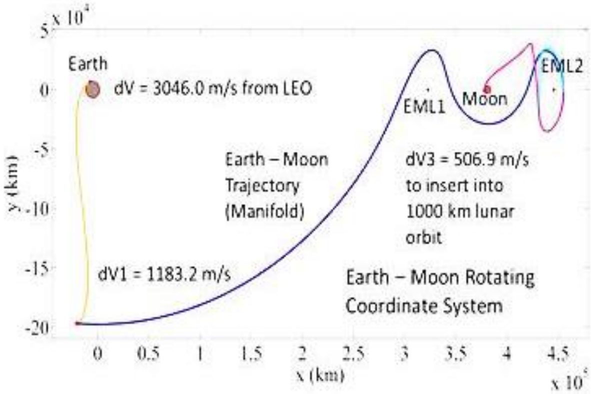


Figure 10. LEO with a Maximum Apogee Transfer and EML_2 arrival



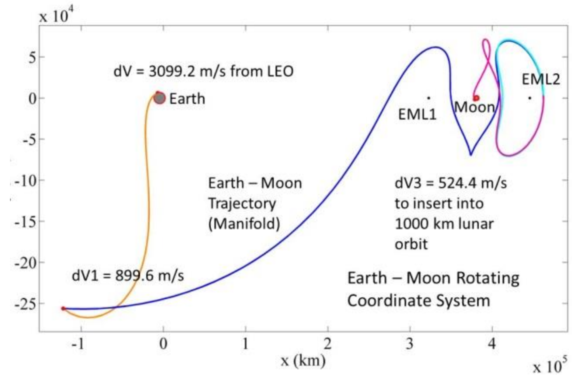


Figure 11. LEO with Manifold Matching ΔV and Lunar Orbit via EML2

Figure 12. LEO with Manifold Matching ΔV and Lunar Orbit via EML2, second option

In these LEO designs, a departure (insertion) ΔV to encounter the local Earth-Moon manifold followed by a manifold matching ΔV are required. These manifolds flow towards the Earth-Moon libration point orbits or the Moon from the maneuver location, and as one can see from the figures, take on a variety of transfer geometries. A final maneuver will be required for lunar orbit insertion, although a pure ballistic capture is notionally available. This final maneuver is also used for corrections to the final lunar orbit. Four designs are presented in Figures 9 through 12. Each figure shows the trajectory in an Earth-Moon rotating frame with the locations of maneuvers. Note that in these designs, the insertion ΔV can be provided by the launch vehicle and may not be required by the cubesat. Also note that the intermediate trajectory to arrive at a manifold is usually in a direction that is roughly opposite of the lunar direction as viewed in the Earth-Moon rotating system, providing for a minimal manifold matching ΔV magnitude. Figure 9 presents a design which minimizes the manifold matching ΔV by orienting the departure intermediate transfer in a direction which results in an asymptotical approach to the manifold. Figure 10 shows another orientation that permits a lower ΔV as well, indicating that numerous transfer options exist. Figures 11 and 12 show other non-optimal designs that require a large apogee distance and velocity vectors which do not align with the natural manifold direction resulting in larger manifold matching ΔVs.

Table 1 provides data for the four feasible designs that use an insertion ΔV to place the cubesat onto an intersection transfer with a manifold. The maneuvers to jump onto the manifold and any maneuvers required to attain the lunar or Earth-Moon libration point orbit are also shown along with the transfer duration. The total ΔV requirements for these examples ranged from 692 m/s to 1690 m/s.

Table -1. LEO Deployment Strategies and Associated ΔVs

	Design-1 (Figure 9)	Design-2 (Figure 10)	Design-3 (Figure 11)	Design-4 (Figure 12)
Insertion ΔV (m/s)	3137	3047	3046	3099
ΔV 1 (m/s)	500	553	1183	899
ΔV 2 (m/s)	192	192	507	524
S/C ΔV total (m/s)	692	745	1690	1423
Manifold duration (days)	10	27	25	32
Transfer Duration (days)	13	30	27	36

GTO Deployment

The second option investigated for a Lunar Cube transfer to the Moon was for a design from a deployment in a Geosynhronous Transfer Orbit (GTO). Similar to LEO, these designs required impulsive ΔVs to attain an intermediate trajectory that will intersect a local Earth-Moon manifold which again, permits a natural flow to the Moon or into an Earth-Moon libration point orbit from which one could transfer to a lunar orbit. Also similar to the LEO design, a low-thrust propulsion option is available, but would require a perigee increase, thus for efficiency, some thrusting would be performed only at apogee and the transfer durations would significantly increase. Once a higher perigee is attained, an outward spiraling trajectory that ballistically encounters the Moon sphere of influence region would be required. In defining the insertion ΔV for the GTO deployment strategy, optimization is required for the best location in the GTO from which to attain the intersection with the manifold. Several options are provided to demonstrate feasibility, with two separate non-optimal GTO geometries with the line of apsides not ideally aligned for an optimal ΔV. Figure 13 shows a sample GTO transfer overlaid on top of the local Earth-Moon manifold depicted in Figure 8. The principle design mechanism is to align the GTO onto a nearest manifold trajectory. Since a preferred alignment may not be feasible due to the low priority of a secondary payload, Figure 14 shows another design by shifting GTO apogee to the upper ½ plane. The manifolds must be propagated longer here to intersect with the GTO apogee. The result is a higher energy level for the manifold to reach GTO apogee altitude. Note that this whole trajectory is a manifold; the manifolds from Figure 13 were simply propagated longer (with multiple Earth passages or 'perigees') until they reach the apogee of the departure GTO.

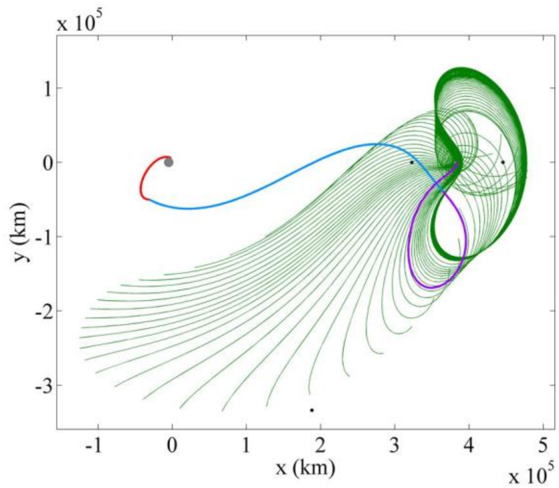


Figure 13. Example GTO Transfer Overlaid on Manifold

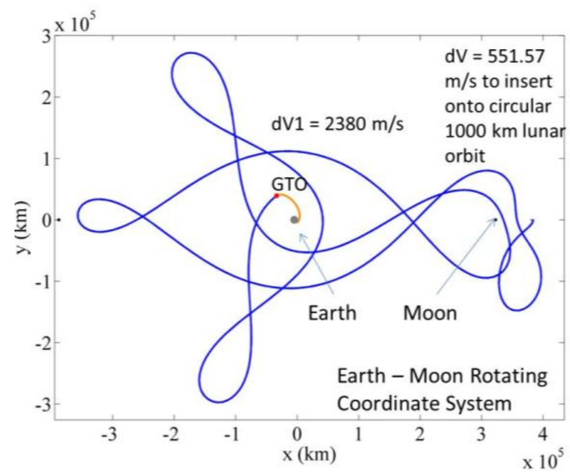


Figure 14. Manifold Extended to GTO intersection

To demonstrate the flexibility of using a GTO – manifold combination, a design is shown in Figure 15 which assumes a GTO with a departure ΔV located at apogee (which also indicates the location of the manifold intersection). The GTO has a line-of-apsides orientation such that the departure ΔV places the cubesat on a direct trajectory to the Moon. This trajectory which uses an inclination of ~ 24 degrees, is similar to a direct LEO to Moon design, and can be initialized using a ΔV either at GTO apogee or GTO perigee, as shown in Figure 16. Both of these designs are feasible but the perigee ΔV is lowest due to the benefit of a departure (or insertion) ΔV placed closer to the perturbing body.

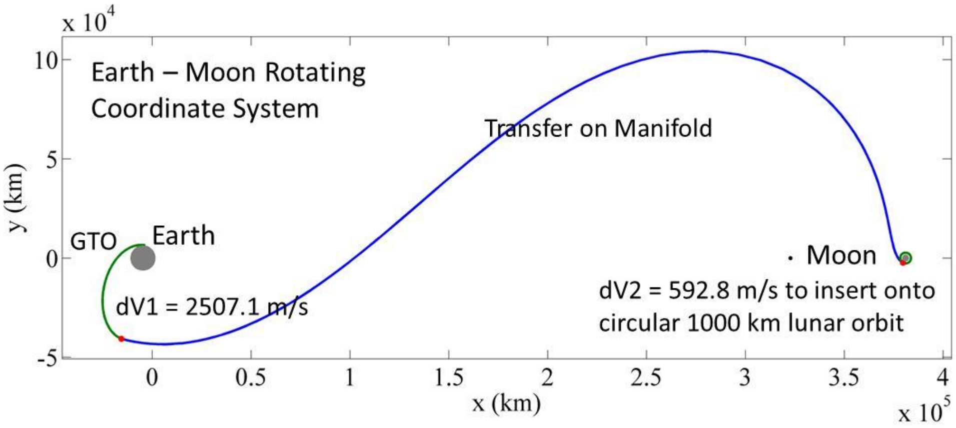


Figure 15. GTO Departure at Apogee

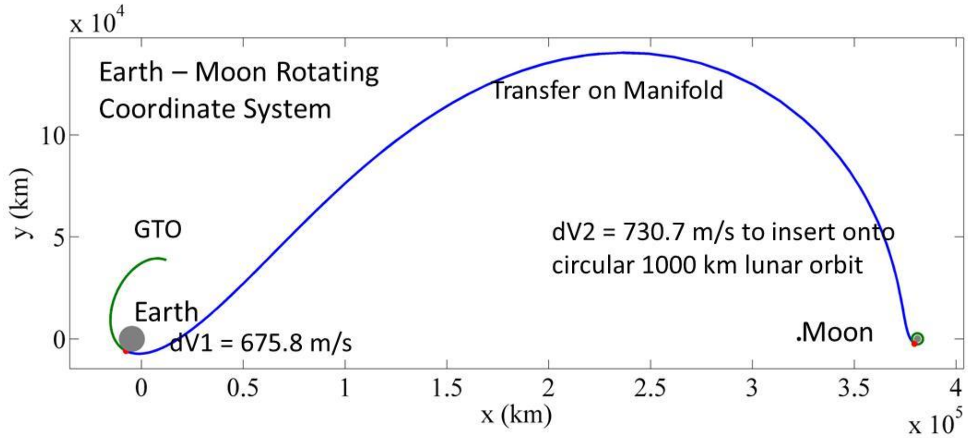


Figure 16. GTO Departure at Perigee

To demonstrate a design which is not optimally aligned and to show other feasible options, a GTO line-of-apsides was selected that would demonstrate a non-optimal launch with respect to a cubesat desire for a lunar orbit mission. A simple change to the above GTO orbit was effected by rotating the line-of-apsides by 180 degrees, providing for such a difference in initial conditions. In the following cases in Figure 17 and 18, a departure ΔV was placed at perigee for effectiveness with two options to attain the lunar transfer manifold.

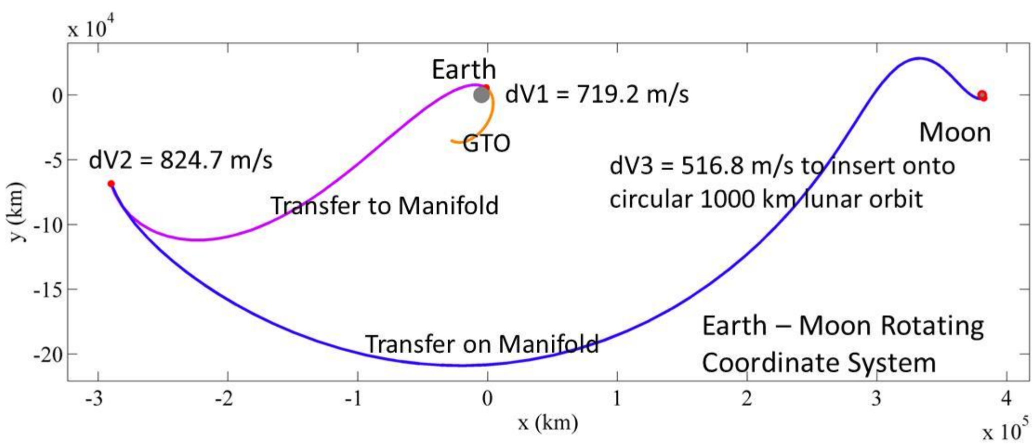


Figure 17. GTO option with Intersecting Manifold at EML$_3$ Distance

While both options place the spacecraft on a lunar bound manifold, the trajectories have different energies and distances from Earth. Figure 17 shows a design which requires an intersecting intermediate transfer with the manifold and therefore a larger manifold matching ΔV, though this ΔV is lower than the manifold matching ΔV performed at the GTO apogee in Figure 15 (again an effectiveness calculation, in this case, overall lower velocities). Figure 18 shows a design with an asymptotically approaching intermediate transfer trajectory which reduces the manifold matching ΔV by half. This option was found by selecting the appropriate manifold to transfer to the Moon in the ATD catalog option. The lunar orbits achieved are 1000 km circular orbits in the plane of the incoming velocity vector.

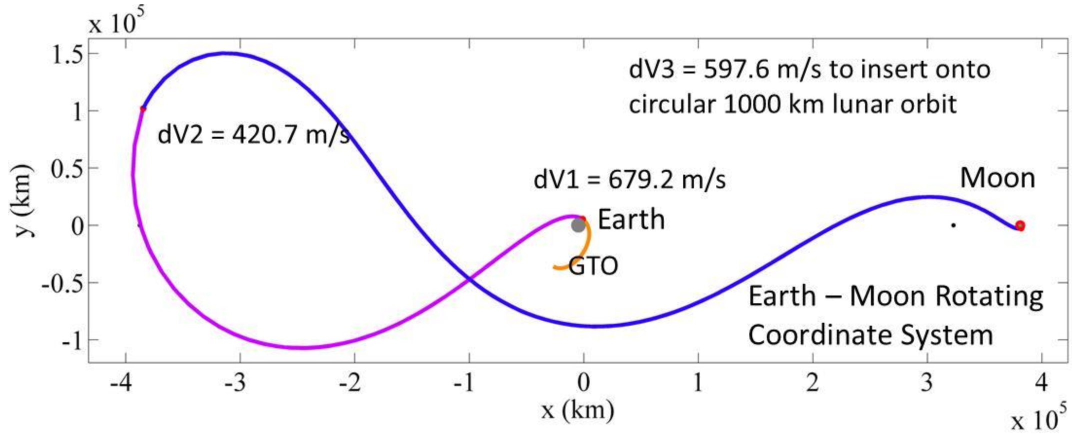


Figure 18. GTO option with Asymptotical Approach to Manifold

Table 2 shows the required impulsive maneuver magnitudes and the transfer duration for these feasible designs. Note that the optimal location of the maneuver to raise apogee to a local manifold is dependent upon the local manifold orientation and distance from the GTO orbital plane. In the cases shown, the GTO orbital plane is aligned to be near the lunar orbit plane to reduce any planar changes, though manifolds intersecting with any GTO orientation exist.

Table -2. GTO Deployment Strategies and Associated ΔVs

	Design-1 (Figure 15)	Design-2 (Figure 16)	Design-3 (Figure 17)	Design-4 (Figure 18)
ΔV 1 (m/s)	2507	676	719	679
ΔV 2 (m/s)	0	0	824	421
ΔV (m/s) (Lunar)	593	731	517	598
S/C ΔV total (m/s)	3100	1407	2060	1698
Transfer Duration (days)	5.4	4.7	20.3	16.2

EM-1 Deployment

The third option investigated for a Lunar Cube transfer to the Moon was a design starting with an EM-1 insertion state.[32] In this configuration; the Lunar Cube is deployed shortly after the insertion onto the high energy free-return lunar trajectory. Without altering the EM-1 injection energy or trajectory, a Lunar Cube would perform a lunar flyby at an altitude near 4000 km and depart into heliocentric space on a drift-away orbit, having increased its energy from the lunar gravity assist (LGA). The options to alter this LGA energy and velocity vector direction include changing the flyby distance and incoming (planar) orientation, permitting transfer trajectories to

Sun-Earth L_1/L_2, Earth-Moon L_1/L_2, and lunar orbits. There are two efficient methods to alter the post EM-1 deployed trajectory; increase or decrease the post EM-1 injection velocity approaching the lunar flyby. These maneuvers should begin soon after deployment from the upper stage structure either thrusting against or along the velocity vector relative to Earth. This maneuver duration will cover several days if using a low thrust system with the start and end time optimized for performance while it is more efficient to perform an impulsive maneuver shortly after deployment.

Examples of an altered EM-1 trajectory are shown in Figures 19 and 20 in a Sun-Earth rotating coordinate frame. The trajectory begins with launch and insertion with the nominal design for a high energy transfer to the Moon. In these cases, a low thrust propulsion system is then employed to decrease the velocity with respect to to the Earth, which in turn increases the LGA distance by several thousand kilometers. The LGA distance is adjusted such that the post flyby energy matches the energy required to attain a Sun-Earth L_1 distance with an orientation that places the Lunar Cube onto an ARTEMIS-like trajectory. This trajectory is actually one of many trajectories available on a manifold (refer to Figure 7 for a sample manifold). In this transfer design there are several low thrust propulsion and coast segments. The thrust levels, thrust direction, and durations are optimized to achieve the timing and to allow the solar gravity perturbation to raise the periapsis to the lunar orbit radius. The return trajectory timing and perigee is adjusted using the LGA and low thrust segments to arrive at the lunar orbit radius distance when the Moon is at the proper location, permitting a ballistically captured lunar orbit.

In addition to the above cases which alter the post LGA energy for a trajectory design similar to ARTEMIS, another process is to immediately thrust along the velocity vector relative to Earth to achieve an LGA that places the Lunar Cube into a highly eccentric Earth orbit with an inclination close to the Moon's orbit plane. This case relies on the low thrust system to raise perigee and lower apogee, and adjust the orbit period and the timing to approach the Moon. For the above designs, once within the lunar sphere of influence, one can thrust against the velocity vector (relative to the Moon) to capture / spiral into a distant lunar orbit or change elliptical eccentricity.

Depending on the LGA distance and orientation, numerous varieties of transfers can be achieved. Figure 19 shows a transfer following an increase in the LGA distance as a result of decreasing the post EM-1 insertion velocity by approximately 50 m/s over 4 days. The actual ΔV and fuel volume depends on the propulsion system thrust and spacecraft mass which varied from 15 m/s to 50 m/s for the cases studied. This change in velocity increases the LGA distance (moving the location of the trajectory apogee while the Moon moves in its orbit) and achieves the transfer as shown. Note that the apogee approaches the Sun-Earth L_1 distance and evolves about the Earth on a manifold similar to that shown in Figure 7. Other designs that change the post EM-1 insertion velocity are shown in Figures 20 to 23 and indicate the variability in this option. Figure 19 shows a design that places the post LGA apogee off the Sun-Earth axis in a rotating frame while Figure 20 shows a design that converges toward a Sun-Earth libration point orbit before departing on a transfer (an unstable Sun-Earth manifold, but a stable Earth-Moon manifold) to the Moon. These designs indicate that the post insertion EM-1 velocity change and resultant LGA conditions are key in attaining the proper transfer trajectory energy and orientation to target a lunar encounter. The difference in the LGA energy is small, but similar to the aforementioned Lorentz attractor illustration; this can result in a very large difference in the transfer trajectory. Using this method one can construct transfers into Sun-Earth libration point orbits or even homoclinic or heteroclinic transfers if the EM-1 initial direction is not in a desirable direction due to launch period, launch window, or performance issues.

Another more extreme option, depicted in Figure 21, is to significantly decrease the velocity over 4 days during the transfer to the Moon to enter into a highly eccentric Earth orbit by increasing the lunar distance to a larger distance that minimizes any lunar perturbations on the transfer trajectory, then reshape that orbit to match the Moon's. To reshape the orbit we use a sequence of low thrust burns. Near apogee we burn in the velocity direction to raise perigee toward the Moon's. Near perigee we burn in the anti-velocity direction to lower apogee. When the Cubesat orbit is close to the Moon, we burn along the anti-velocity direction relative to the Moon to enter lunar orbit.

Other designs, shown in Figures 22 and 23, increase the post EM-1 insertion velocity, achieving an LGA that is on the opposite side of the Moon with respect to the nominal EM-1 design. This changes the post LGA energy such that the post LGA orbit is now a highly eccentric Earth orbit from which one must use the benefits of the Sun's gravity effects while on a manifold to raise periapsis or to raise periapsis via low thrust or impulsive propulsion. The change in the flyby to the opposite side of the Moon with respect to to a nominal EM-1 trajectory changes the Earth related C3 energy, thus the apogee remains near the lunar orbit radius. In Figure 22, the LGA is designed to permit a larger elliptical orbit with apogee almost twice the lunar orbit radius and is on a manifold that increases perigee naturally, while Figure 23 (plotted in both a rotating and an Earth-centered inertial coordinate system) shows the effect of an LGA distance that places the cubesat on an elliptical orbit with perigee close to the deployment perigee. In this option, we target a set of B-plane parameters that not only removes energy but also changes the cubesat orbit plane to more closely align with the Moon's. This approach makes it much easier to achieve a lunar orbit with non-polar inclination. Again when the cubesat orbit is close to the Moon's, we burn along the anti-velocity direction relative to the Moon to enter lunar orbit. In this second option we design the lunar orbit insertion not only to achieve the desired science orbit shape and inclination, but also to achieve the desired Sun lighting condition. One hundred low-thrust arcs were needed to achieve the science orbit. To achieve the desired Sun lighting requires timing such that science orbit arrival is correct. To time the arrival, we adjust the thrust level appropriately. In these cases, once the perigee has been sufficiently raised, a local manifold can once again be used to approach the Moon on a ballistic transfer, minimizing the required insertion ΔVs. An example of the change in the LGA B-planes for the all the above cases is shown in Table 3.

Table 3. Baseline and Modified Lunar B-Plane Components

Case	B.T (km)	B.R (km)	Periapsis LGA C3 (km^2/sec^2)
Baseline EM-1 Simulation	7287	1450	0.62
Decreased Velocity Option 1	12144	1500	0.59
Decreased Velocity Option 2	10016	1318	0.60
Decreased Velocity Option 3	40381	1151	0.60
Increased Velocity Option 1	-6432	151	0.73
Increased Velocity Option 2	-8000	950	0.70

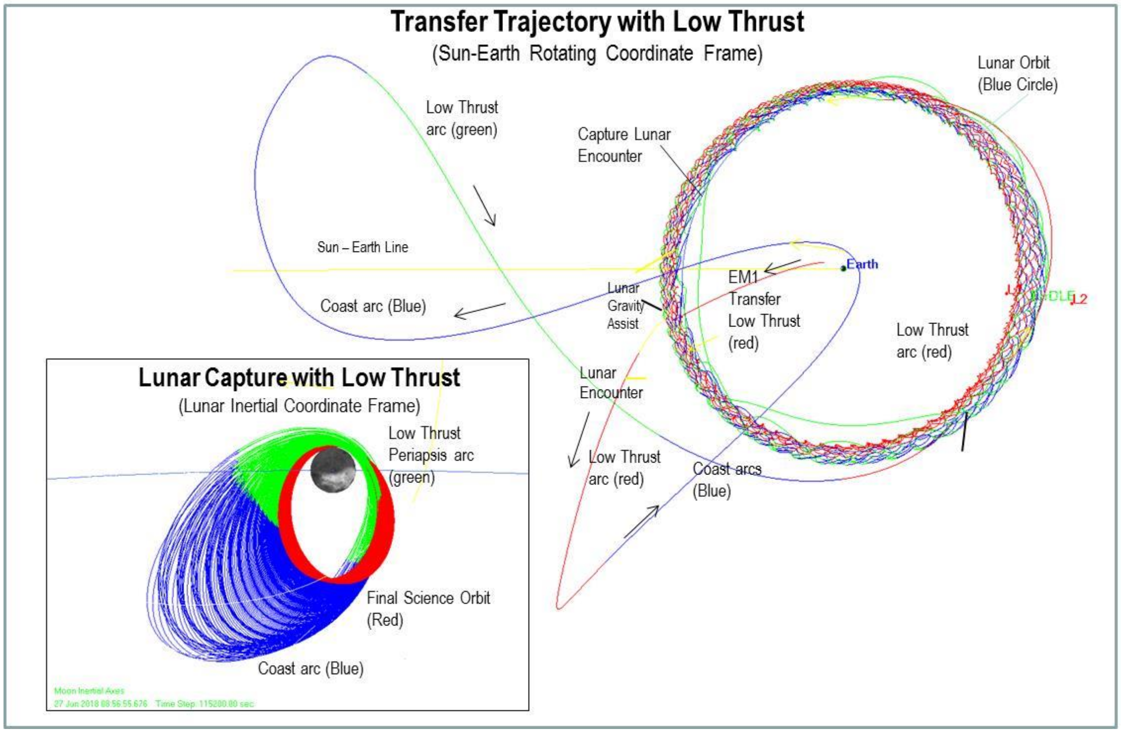


Figure 19. ARTEMIS-Like Transfer Design with Decreased EM-1 Velocity

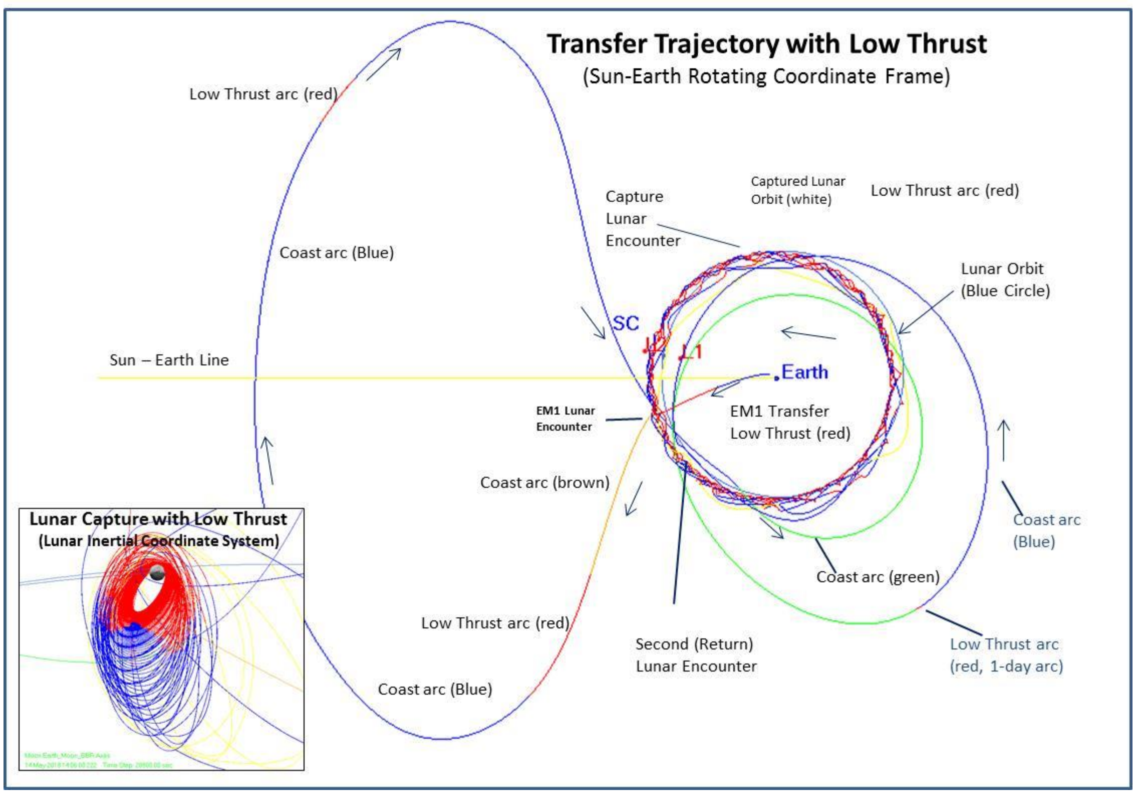


Figure 20. Libration-Like Transfer Design with Decreased EM-1 Velocity

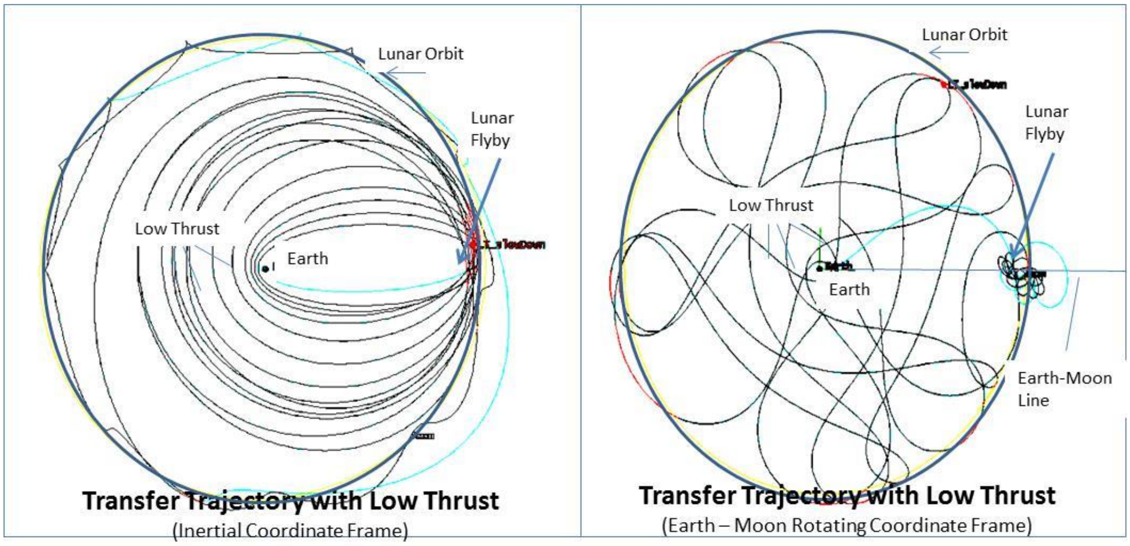


Figure 21. Decreased Velocity: Left is Inertial, Right is Earth-Moon Rotating. Red Indicates Thrusting Along Velocity Vector, Blue Thrusting Opposite Velocity Vector, Black is Ballistic

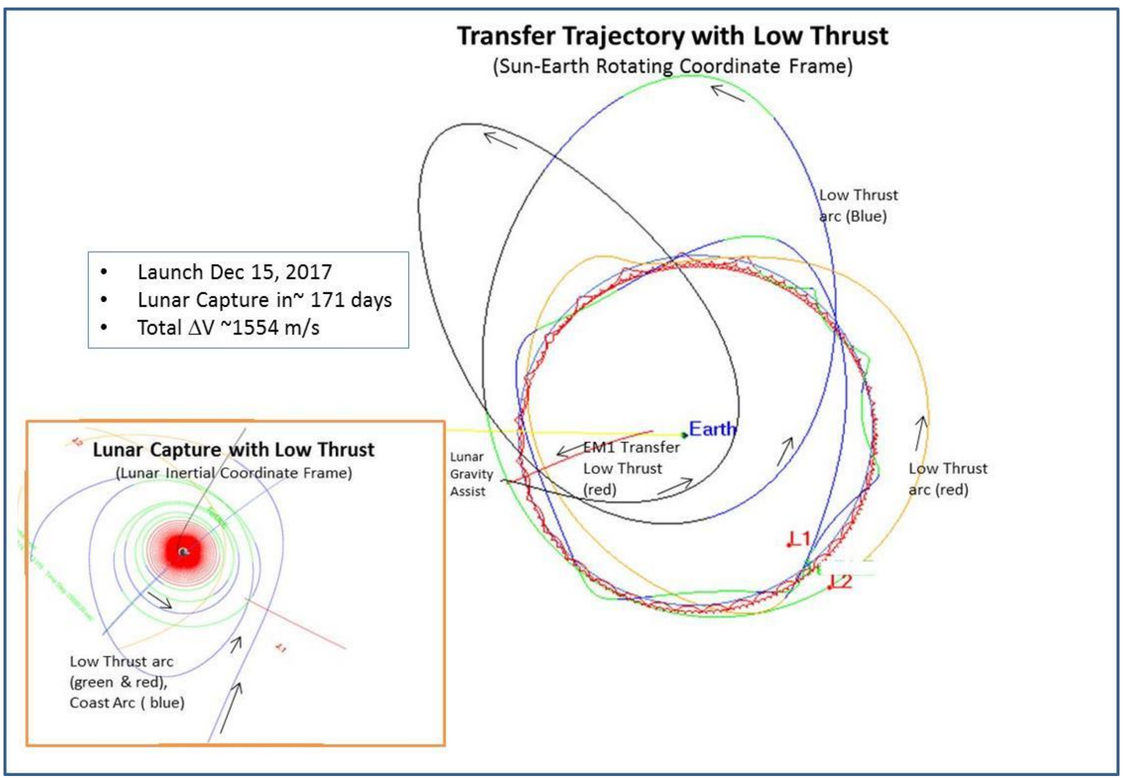


Figure 22. Leading Edge Transfer Design with Increased EM-1 Velocity

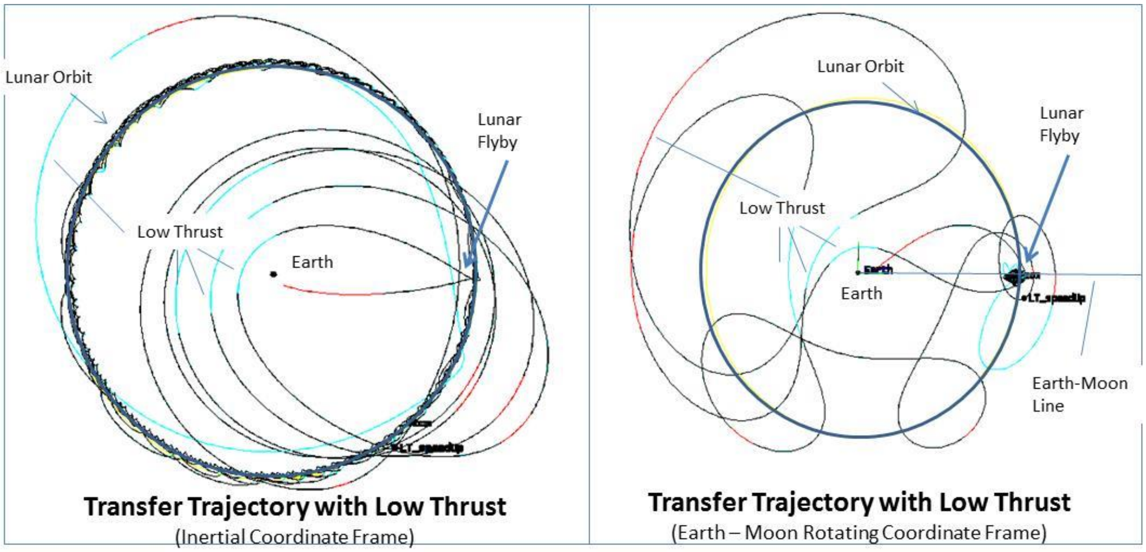


Figure 23. Increased Velocity: Left is Inertial; Right is Earth-Moon Rotating. Red Indicates Thrusting Along Velocity Vector, Blue Thrusting Opposite Velocity Vector, Black is Ballistic

Table 4 provides the trajectory, ΔV, and propulsion system characteristics for the above cases. The low thrust propulsion system assumptions for spacecraft mass and thrust level are giving at the top of the table and the ΔVs follow. Depending on the design, the ΔV ranges from a manageable level near 700 m/s to over twice that amount, and would suggest a trade of subsystems and propulsion designs. Another consideration in using these transfers is that the apogee may be at a distance that impacts the selection or capability of the communication system. In all these cases the lunar orbit at arrival was limited only to the desire to capture within a year after launch.

Table 4. Example Low-Thrust Transfer Parameters

	Decreased Velocity	Decreased Velocity	Decreased Velocity	Increased Velocity	Increased Velocity
Related Fig	19	20	21	22	23
Initial Mass (kg)	9	12	10	12	10
Thrust Level (mN)	0.5	2	3	2	3
Total DV (m/s)	869	629	1326	1595	1141
Transfer DV (m/s)	673	190	1161	557	860
Lunar Capture DV (m/s)	196	439	165	1038	581
Lunar Flyby Radius (km)	6763	5025	33,143	2510	3661
Max Transfer Range (Km)	1,524,000	1,719,925	447,959	1,154,950	467,698
Total Transfer Duration to Capture (days)	231	250	223	171	214
Lunar Capture Duration (days)	60	27	119	65	15
Maximum Lunar Eclipse Duration (hrs)	1.0	4.6	5.0	4.0	3.3
Lunar Orbit apoapsis x periapsis (km)	6800 x 100	9993 x 1545	6513x139	350 x 50	5571x101
Lunar Orbit Inclination (deg)	20	144	156	165	32

LUNAR ORBIT

A variety of lunar science orbits can be achieved from any of these analyzed transfers. Using a low thrust capture and insertion from a ballistically captured lunar orbit, one can perform an alignment of periapsis (apsides) to meet science goals, target a given periapsis altitude or periapsis decay over time, target various values of eccentricity, semi-major axis, inclinations, or achieve various science parameters, e.g., solar angles. Shown in the figures are examples of capture and final lunar orbits. Depending on the lunar capture energy and the orientation of the capture apsides with respect to the Earth-Moon line, which has an effect on the orbit perturbations, perilune will increase or decrease giving a unique lowering phase without a maneuver requirement. The lunar orbit in Figure 24 shows a capture that result in a spiral-like transfer to the desired lunar orbit, while Figure 25 shows a capture with a higher energy that required maneuvers near the perilune to reduce the apolune before achieving the desired lunar orbit. In these lunar transfers, no optimization was employed, but rather a maneuver(s) were executed to arrive into a stable lunar circular or elliptical orbit. The lunar capture and durations using a low thrust system are provided in Table 4.

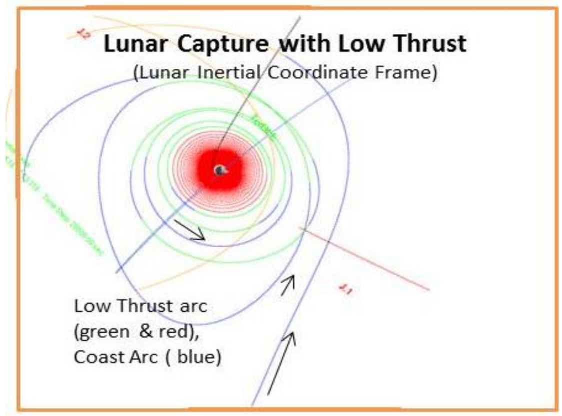


Figure 14. Lunar Orbits from EM-1 Deployed Transfers, Increased LGA Velocity

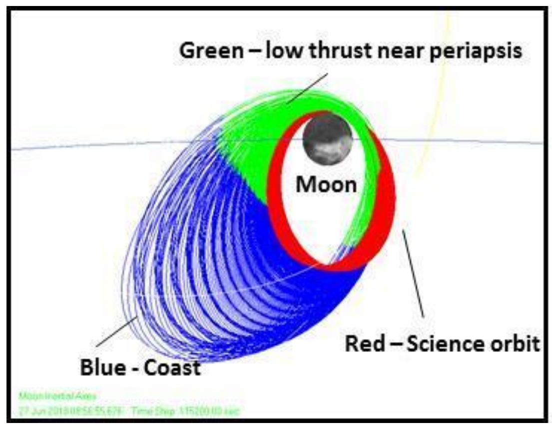


Figure 25. Lunar Orbits from EM-1 Deployed Transfers, Decreased LGA Velocity

CONCLUSIONS

There are numerous Lunar Cube Transfer Trajectory Options available for a variety of deployment strategies as a secondary payload. The use of a dynamical systems (manifolds) approach can greatly aid in the transfer design and provide an intuitive approach in addition to customary optimization techniques. These manifolds, and associated design capabilities such as Poincaré maps allow an analyst to quickly and accurately survey feasible transfer options. For LEO and GTO deployment strategies, it was found that impulsive maneuvers to move from the deployed orbit onto the local Earth-Moon manifold provided for the shortest transfer durations with reasonable ΔV requirements ranging from ~ 500 m/s to over 1500 m/s. The LEO and GTO orbital orientation was a consideration, but there are local manifolds that pass close to almost any reasonable deployment orbit as a result of the primary launch azimuth. The more interesting deployment options were found using the EM-1 strategy whereby a simple change to the outbound LGA distance generated numerous trajectories (manifolds) that could be modified by a low thrust system for an easy capture into lunar orbit. Both low thrust and high performance propulsion systems can be used in any of these deployment options, but a trade is highly recommended for detailed mission design which must consider the total cubesat subsystem options. High thrust can result in mass / volume considerations, but short durations, whereas low thrust ranging from μ-N to m-N can augment the trajectory given the proper initial conditions. These trajectory designs may drive the available subsystem options since the transfer trajectory may impose additional constraints beyond the science orbit requirements. Both the transfer and the lunar capture into a science orbit can be time-consuming and was found to approach yearlong durations. In a final consensus, combining dynamical systems techniques with high or low thrust propulsion systems provides versatile, efficient techniques for transfers to the Moon, especially for low-thrust options on high energy deployment trajectories. Given a lower cost and many secondary payload opportunities, Lunar Cubes can be the next step for flexible trajectory designs, to the Moon and beyond.

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