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UW CubeSat, L5 Mission Proposal – Write-up

Mission Overview:

This mission aims to launch a CubeSat from a parent craft on route to or on return from the Moon, such as the SLS, and then to attain capture by one of the two stable Sun-Earth Lagrange Points, L4 or L5. The target will be determined by conditions set by the launch date of the parent mission and calculated Delta-V requirements. Per the feasibility assessment to follow, a CubeSat mission to L5 is more easily achievable. Released just outside of the Moon’s sphere of influence, the CubeSat will use its own propulsion to then path to L5. Upon reaching L5, the CubeSat will perform a minimal maneuver to be captured and begin to survey for asteroids in the immediate area.

Mission Purpose:

The intent of this mission is to explore the L4 and L5 points of the Sun-Earth system. L4 and L5 are stable balance points between the gravities of the Earth and of the Sun and the rotational forces of orbit, located in the orbit of the Earth, ahead and behind by 60 degrees respectively. An object or spacecraft that reaches one of these points with low enough relative velocity will be captured and will experience a restorative force when moved from equilibrium. This results in an orbit apparently ‘around nothing’, in a frame rotating with the orbit of the Earth. This property means that spacecraft can be placed at one of these points without the requirement of periodic station-keeping maneuvers.

Captured asteroids known as Trojans, extensively observed in the Sun-Jupiter system, have scarcely been found in the Sun-Earth system. Only one Trojan has been observed and catalogued at each the Sun-Earth stable points, but this does not mean that there are only two Trojans of the Earth and Sun. The Sun-Earth L4 and L5 points are relatively unobserved, due to their generally sunward direction and the difficulty of ground-based daylight observations. Additionally, previous space missions have never targeted the Sun-Earth stable points and have encountered them only incidentally, at velocities too great for capture and prolonged study.

This mission seeks to fill this gap in our collective knowledge of the Solar System by achieving capture by the target point and surveying for Trojans over an extended period. If no new asteroids are discovered at the target stable point, the implication would be that that point is a low hazard location for spacecraft to be “parked” for a wide variety of applications, chief among them being space-based observations, particularly of the Earth. If the point is found to be more heavily populated with Trojans than has so far been shown, then the point will be a convenient point of interest for future study and/or exploitation of asteroids. Additionally, in the process of designing the CubeSat and throughout the course of study, this mission will aim to provide insight into improving techniques for the remote detection of asteroids and other small bodies.

Initial Conditions, Assumptions and Method of Assessment of Feasibility:

Initial conditions for this mission were taken from publicly available NASA documentation for the Space Launch System, namely the ICPS Disposal State. Those conditions were then propagated forward using GMAT to find velocity and position relative to the Earth at the time of CubeSat release, just outside the Moon’s sphere of influence. It was assumed that these parameters would be generalizable to any day of the lunar synodic month (i.e. any position of the Moon relative to the Earth in its orbit), so that Delta-V requirements to reach L4 or L5 can be found and optimized for any potential launch window of the host mission. It was assumed that spacecraft will be equipped with a Pulsed Plasma Thruster capable of sustaining an acceleration of (conservative estimate) of 5\*10-5 m/s2, with sufficient fuel for a total Delta-V of 1 km/s. Keeping 40% of fuel in reserve for maneuvers and corrections will leave a Delta-V-to-target requirement of no more than 600 m/s, limiting desirable launch windows.

GMAT, as a single body propagator, does not model Lagrange points and was therefore found to be insufficient for feasibility assessments of this mission. A simplified two-body propagator was written in MATLAB for this assessment, modeling the Earth, the Sun and a general spacecraft object. In the model, the Earth and the Sun orbit a shared barycenter located at the origin. These two orbits are maintained through setting appropriate tangential velocities, distances from barycenter, and applying gravitational acceleration due to the other body to each body at every time step. This resulted in a model in which the Earth and Sun are in circular orbits, approximating the low eccentricity of Earth’s orbit. The spacecraft object can be placed anywhere with any initial velocity, and at each timestep will experience the combined gravitational accelerations of the other two bodies. The spacecraft’s mass is assumed to be very small compared the Sun or the Earth, and the effects of its gravity on those two bodies are ignored. Motions in this model are restricted to the orbital plane of the Earth and the Sun. Calculations are done in Cartesian coordinates in the inertial frame of the system’s barycenter. Results are rotated into a non-inertial frame in which the Earth and Sun appear stationary for display purposes and analysis.

The propagator functions by numeric approximation. At every time step, positions and velocities of all objects are recorded. Then, gravitational accelerations, based on relative position and mass, are calculated. In the case of the spacecraft, acceleration due to thrust is also calculated, if applicable during the timestep. Thrust profiles are set by the user before the propagator is run. The propagator then integrates accelerations into change in velocity, by multiplying the calculated accelerations by the length of the timestep. This velocity change is added to current velocity. Similarly, the new velocity is integrated into change in position and the position is updated. The accuracy of this technique improves with shorter timesteps. A timestep length of 480 seconds was used for most of the analysis to provide a good balance between accuracy and computation time for timescales of a year or multiple years.

For the assessment, the spacecraft was placed and initialized with the position and velocity relative to Earth as found from the ICPS Disposal State and GMAT propagation. This position data was generalized to any time of the lunar month by rotation using the following system: using the synodic month of 29.5 days so that day 0 will mark the moon’s return to the same initial position relative the Earth and the Sun, day 0 was set to the position at which the moon is collinear with Earth and Sun and outside the Earth’s orbit (full moon position) and counting days positively counter-clockwise and negatively clockwise to a maximum/minimum at ±14.75 days (new moon position).

Data and Assessment:

In the initial assessment, the viability of 6 separate launch days, roughly evenly spaced throughout the lunar month, were tested. First, the propagator was run without a defined thrust profile, to see how close the craft would come to either target point under no propulsion. From there, trial-and-error techniques were used to find thrust profiles that would result in a successful encounter with the most convenient target for a minimized Delta-V. Encounters were judged to be successful if the spacecraft passed within 0.15 AU of the target center at a velocity relative to the target of no more than 50 m/s. It was assumed that a corrective maneuver could be performed under those conditions without the requirement for additional reserve fuel. As this document’s aim is to assess first-order viability of the proposed mission, once a successful encounter was achieved in a timely (< 3 yr.) manner with satisfactory fuel consumption, little more refinement was done, though the results listed below did prompt additional assessment in a narrower range between -8 and +8 days.

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| --- | --- | --- | --- | --- | --- |
| Starting Position (Day) | Delta-V to Best Encounter | Time to Encounter | L4/L5? | Relative Speed During Encounter | Successful? |
| ±14.75 | 230 m/s | 1 yr | L4 | 260.7 m/s | N |
| -10 | 526 m/s | 2 yr | L4 | 134 m/s | ~Y |
| -5 | 350 m/s | 2 yr | L5 | 6.1 m/s | Y |
| 0 | 865 m/s | 1 yr | L5 | 16.4 m/s | ~Y |
| +5 | 409 m/s | 1 yr | L5 | 11.8 m/s | Y |
| +10 | 877 m/s | 20 yr | L4 | 0-400 m/s | N\* |

Assessment 1, general:

~ = (Within parameters, but not ideal)

\* = (Captured in horseshoe orbit very early, but approaches target too slowly)

Per this data, a mission to L4 may not be possible without a larger craft capable of a greater Delta-V. On the other hand, a mission to L5 using a CubeSat probe is very plausible.

|  |  |  |  |  |
| --- | --- | --- | --- | --- |
| Starting Position (Day) | Delta-V to Best Encounter | Time to Encounter | Relative Speed During Encounter | Successful? |
| -8 | 702 | 9 yr | 334 | N |
| -6 | 740 | 2 yr | 50 | ~Y |
| -4 | 374 | 2 yr | 50 | Y |
| -2 | 468 | 1 yr | 16 | Y |
| 0 | 877 | 1 yr | 16 | ~Y |
| +2 | 573 | 1 yr | 17 | Y |
| +4 | 222 | 1 yr | 7 | Y |
| +6 | 608 | 1 yr | 49 | Y |
| +8 | 667 | 2 yr | 4 | ~Y |

Assessment 2, L5-focus:

~ = (Within parameters, but not ideal)