Design of High-Energy Escape Trajectories with Lunar Gravity Assist

Lorenzo Casalino[1]
[1]*Politecnico di Torino, Italy,*
lorenzo.casalino@polito.it

Keyword : Earth Escape, Lunar Gravity Assist

The proposed paper presents a procedure for design and optimization of maneuvers for Earth escape with lunar gravity assist, to be used for interplanetary missions. In particular, focus is on missions to near-Earth asteroids (NEAs) with electric propulsion. Exploration of the solar system requires relatively large values of escape energy. The escape mass that a given launcher can provide is a decreasing function of the escape energy, but the velocity upon leaving the Earth can be properly directed to reduce the propellant consumption of the heliocentric flight. Even though the heliocentric flight employes a propulsion system with larger specific impulse compared to the launcher (e.g., electric propulsion), the optimal trade-off usually requires escape C3 of a few (km/s)^2. Lunar-gravity-assist (LGA) maneuvers can be used for a free increase of the escape C3, and the design of LGA escape trajectories for interplanetary missions is the object of the proposed paper.

Preliminary design of interplanetary missions is usually carried out with the patched-conic approximation. During the heliocentric flight the spacecraft is subject to the gravity of the sun alone. In the vicinity of a planet (i.e., inside its sphere of influence) only the planet's gravity is considered. The planets' spheres of influence are small compared to heliocentric distances and are neglected in preliminary design; asteroids do not have a sphere of influence as their gravity is negligible. Heliocentric legs connect at encounters with relevant bodies (planets, asteroids, etc.). The relative velocity at encounter is the hyperbolic excess velocity.

The patched-conic approximation is adopted here. An indirect method (L. Casalino, G. Colasurdo, D. Pastrone, "Optimal Low-Thrust Escape Trajectories Using Gravity Assist", *Journal of Guidance, Control, and Dynamics*, 1999) is used to optimize the geocentric and heliocentric phases. The design procedure starts from the heliocentric trajectory and then finds suitable escape maneuvers. It is not difficult to find a suitable tentative solution for the heliocentric leg. Convergence is extremely fast and allows to easily perform trade studies (escape mass, hyperbolic excess velocity, departure date).

Analysis of the escape maneuver is treated with the same indirect optimization method, but finding a tentative solution that allows for convergence is a more difficult task, at least in the case of LGA escape. An approximate analysis is here introduced to this purpose. The analysis is based on a patched-conic approximation that neglects the dimension of Moon's sphere of influence. The trajectory is split into two geocentric legs. The inner leg goes from trajectory perigee (usually imposed by the launcher) to the Moon, the outer leg from the Moon to the boundary of Earth's sphere of influence (150 million km) where the escape velocity required by the heliocentric trajectory must be met at the escape time. Complexity arises for missions to NEA, as the hyperbolic excess velocity is usually required to rotate the orbit plane rather than change its energy, and has a large component perpendicular to the equator and Moon's plane.

The outer leg is solved first, given the estimated Moon position at encounter. A set of algebraic relations is derived and solved to obtain the specific hyperbola that matches the escape conditions while intercepting the Moon. Iterations are carried out to refine Moon's position once the time of flight is obtained. The hyperbolic excess velocity at the Moon is also found. The inner leg is defined by perigee radius and magnitude of the hyperbolic excess velocity at the Moon encounter, which must be the same as that of the outer leg. For any given inclination of the inner trajectory with respect to the Moon orbit plane, two possible trajectory exist, which intercept the Moon either at their ascending or descending node, Also in this case, a set of algebraic equations can be defined and solved to obtain the required inner leg trajectories.

For any solution, feasibility is checked by comparing the required and allowed velocity rotations at Moon's flyby. Position and velocity at perigee are obtained and can be used for a preliminary estimation of the escape mass, based on launcher performance as a function of launch C3 and azimuth. In addition, the results of this approximate analysis are also employed to build a tentative solution for the indirect optimization of the geocentric leg. It adopts an accurate dynamic model, which takes also solar gravity and radiation pressure into account, in addition to Earth and Moon's gravity. Results comparison shows that the approximate analysis makes very accurate predictions, despite its extreme simplicity, and is a rapid tool for preliminary trajectory evaluation. Refined results can the be obtained with the indirect method. Extension to more complex escape maneuvers (e.g., multiple Moon's flybys) is also feasible.