Acta Astronautica 151 (2018) 228–242

Contents lists available at ScienceDirect

Acta Astronautica

journal homepage: www.elsevier.com/locate/actaastro

A lunar flyby for a tridimensional Earth-to-Earth mission

Luiz Arthur Gagg Filho*, Sandro da Silva Fernandes

Mathematics Department, Instituto Tecnológico de Aeronáutica, Praça Marechal Eduardo Gomes 50, Vila das Acácias, 12228-900, São José dos Campos, SP, 12228-900, Brazil

A R T I C L E   I N F O

Keywords:
3D lunar patched-conic approximation
Flyby maneuver
Earth-to-Earth mission
Changing plane maneuver
Orbital transfer
Spatial three-body problem

A B S T R A C T

The present work formulates an orbital transfer for an Earth-to-Earth mission between non coplanar orbits with different altitudes with a special feature: the occurrence of a lunar flyby during the transfer orbit. This lunar flyby is intended to help change the plane of motion of the spacecraft without fuel consumption. Only two-impulsive trajectories are considered with the velocity increments applied at the initial and final orbits. In order to solve this problem, a 3D patched-conic approximation associated with a two-point boundary value problem is proposed. The same transfer problem is formulated considering the spatial circular restricted three-body problem (SCR3BP). The results of the patched-conic approximation is compared with the results of the SCR3BP showing a good agreement between the models. This work also determines several trajectories in order to perform a study of the fuel consumption considering several inclinations and altitudes of both initial and final orbits around the Earth. The longitude of the ascending node of the initial orbit, and, the altitude of close approach with the Moon during the flyby are also analyzed. According to the total velocity increment analysis, the changing plane assisted by a lunar flyby can be very favorable. Despite the increase of the time of flight, the saving of fuel is considerable. Indeed, the total velocity increment of this kind of maneuver is in some cases better than the velocity increment provided by the bi-parabolic transfer.

1. Introduction

The space maneuvers assisted by gravity (flyby or swing-by maneuvers) are very common in space missions, since it allows the spacecraft to gain or lose mechanical energy without fuel consumption. This kind of maneuver is very useful for interplanetary and cometary missions [1]. Earth and Venus are bodies commonly used as primaries to accomplish this maneuver [2] [3]. Flandro [4] has determined a favorable epoch to send a spacecraft to further planets with a Jupiter flyby maneuver. However, in this same work, the Jupiter flyby maneuver was considered to be instantaneous in the context of the complete interplanetary trajectory during the preliminary analysis. In this way, there are discontinuities in the velocity vector of the spacecraft. D'Amario et al. [5] have elaborated a procedure to determine optimal interplanetary trajectories with multiple flyby maneuvers, which are also considered to be instantaneous. These studies, that focus on the flyby maneuver in an independent way of the complete trajectory, have provided valuable informations about the effects of this maneuver. Broucke [1] has classified the flyby maneuvers according to the incoming and outcoming conic. Prado [6] has utilized the patched-conic approximation to study the possibility of an intermediary impulse during the flyby maneuver. In Ref. [7], the patched-conic approximation is compared to the restricted three-body problem. Yi Qi et al. [8] have derived an energy expression to study the lunar flyby in the context of the planar three-body problem, and they have obtained double lunar flyby maneuvers for lunar and interplanetary missions. Yi Qi et al. [9] have studied the influence of the Sun in the lunar flyby maneuver utilizing the planar restricted four-body problem. Yi Qi et al. [10] have combined the possibility of double lunar flyby maneuvers with the restricted four-body problem to determine optimal Earth-Moon trajectories.

In the context of tridimensional trajectory, the flyby maneuver can also assist the spacecraft to change its plane of motion. Without the flyby maneuver, a single impulse changing plane maneuver can have a large fuel consumption [11]. For instance, the velocity increment applied to a satellite to produce a changing of 60° in the inclination of its orbit around the Earth is equivalent to its circular velocity in the context of the two-body dynamics [12]. Therefore, in order to decrease the fuel consumption during the changing of inclination between two orbits around the Earth, tridimensional models with the gravitational influence of the Moon can be utilized. Ivashkin et al. [13] was the first to propose a lunar flyby for an Earth-to-Earth mission. In 1998, the AsiaSat-3 spacecraft (Hughes Global Services 1 Spacecraft – HGS-1) has performed an unusual orbital maneuver to reposition it into a

* Corresponding author.
E-mail addresses: luizarthur.gagg@gmail.com (L.A. Gagg Filho), sandro@ita.br (S. da Silva Fernandes).

https://doi.org/10.1016/j.actaastro.2018.06.006
Received 14 November 2017; Received in revised form 27 April 2018; Accepted 3 June 2018
Available online 07 June 2018
0094-5765/

geosynchronous orbit [14]. This unusual maneuver was motivated due to the failure in the launching phase which left the spacecraft with not enough propellant. Basically, the maneuver has performed a flyby with the Moon in order to change the orbital plane of motion of the spacecraft. As the changing plane of the maneuver was assisted by the gravitational field of the Moon, an amount of fuel consumption was saved. Several authors have studied the effects of flyby maneuvers in tridimensional models. Ocampo [15] has utilized the lunar gravity to reduce the high inclination of a geoestationary orbit of a spacecraft. In this last work, an analytical approximation was developed to calculate, in a simple way, the effect of the flyby maneuver. Felipe et al. [16] have utilized the spatial restricted three-body problem to classify trajectories developed by different sets of flyby maneuvers. Prado [17] has studied analytically the possibility of changing the inclination of a spacecraft orbit around the Earth by using the lunar gravity. A similar work was made by Prado and Felipe [18], where the effect on the inclination of motion of the spacecraft is analyzed when a tridimensional flyby maneuver is performed.

Mathur et al. [19] have developed an algorithm to determine optimal Earth centered orbit transfers using lunar gravity assist. Their algorithm utilized two-body initial guess to determine the optimal trajectory, in which the lunar flyby maneuver is considered to be instantaneous, *i.e.*, there is an abrupt change in the velocity vector direction of the spacecraft when the lunar flyby occurs. The present work extends the idea of [19] by utilizing the two-body problem to formulate the entire geometry of the Earth centered transfers using lunar flyby. In this way, the geometry of the selenocentric trajectory is determined considering the complete mission without velocity discontinuities, which can be advantageous to agree with mission requirements as, for instance, the prescribing of the flyby altitude. Also, the proposed model provides a more accurate initial guess for more complex models. Similarly to [19], the present work considers that, initially, the spacecraft is at a low Earth orbit and it is desired to perform an orbital transfer assisted by a lunar flyby to another orbit around the Earth with a different inclination. Two impulsive velocity increments are applied to perform the transfer maneuver. The first velocity increment accelerates the spacecraft inserting it into a transfer trajectory which performs a flyby maneuver with the Moon that changes the plane of motion of the spacecraft. When the spacecraft returns close to Earth, the second impulsive velocity increment is applied to decelerate and establish the motion of the spacecraft in a prescribed low or high Earth orbit with a given eccentricity, altitude, inclination, and longitude of the ascending node. In order to solve the transfer problem, two models are considered: the first one is an extension of the tridimensional lunar patched-conic approximation [20], and, the second one is based on the spatial circular restricted three-body problem (SCR3BP) [21]. Trajectories are obtained by solving two-point boundary value problems considering both models. The results of the patched-conic approximation is compared with the results of the SCR3BP. As the proposed models have general purpose, several types of Earth-to-Earth missions can be analyzed, which can include, for instance, missions to insert the spacecraft from an orbit with low inclination into a sun-synchronous orbit with the same or not the same altitude, missions to insert the spacecraft from a high inclination orbit with low altitude to a geosynchronous orbit with low inclination, or missions to insert the spacecraft into orbits with inclination values of the Molniya orbit. Therefore, this work intends to determine several trajectories in order to perform a study of the fuel consumption considering several inclinations and altitudes of both initial and final orbits around the Earth. The influence of the altitude of close approach with the Moon during the flyby in the fuel consumption is also analyzed.

2. Formulation

In this section, the spatial patched-conic approximation for an Earth-to-Earth mission is formulated by means of a two-point boundary value problem (TPBVP). The same transfer problem is also formulated considering the spatial circular restricted three-body problem (SCR3BP). The formulation of this transfer problem under these two models (patched-conic approximation and SCR3BP) is an extension of the formulation of the Earth-Moon mission as presented by Ref. [20].

2.1. The transfer problem

For an Earth-to-Earth mission the spacecraft is, initially, inserted at a low Earth orbit (LEO) O_i with a given pericenter altitude h_i, inclination I_i, longitude of the ascending node Ω_i, and eccentricity e_i. The pericenter argument ω_i of the LEO is a variable to be solved with the transfer problem. The subscript i indicates that the quantities are related to the initial parking orbit O_i. With an application of an accelerating velocity increment Δv_1, the spacecraft is inserted into a transfer trajectory which performs a flyby maneuver with the Moon. When the spacecraft returns close to Earth, a second velocity increment but decelerating, Δv_2, is applied in order to establish its motion around the Earth at the final orbit O_f with a given pericenter altitude h_f, inclination I_f, longitude of the ascending node Ω_f, and eccentricity e_f. The pericenter argument ω_f of the final orbit is given by the solution of the transfer problem. The subscript f indicates that the quantities are related to the final orbit O_f around the Earth. It is assumed that the first and second velocity increments are applied, respectively, at the pericenter of the O_i and O_f orbits. The total velocity increment, given by

$$\Delta v_{Total} = \Delta v_1 + \Delta v_2 \tag{1}$$

represents the fuel consumption of the spacecraft.

The radial distance r_0 from the center of the Earth to the spacecraft is known at the moment of the first velocity increment. In the same way, the radial distance r_f from the center of the Earth to the spacecraft is known at the moment of the second velocity increment. If the initial orbit O_i is known, then all Keplerian elements related to this orbit is known but the true latitude u_i of the spacecraft, which characterizes the point of application of the first velocity increment. In the same way, the true latitude u_f of the spacecraft at O_f, which characterizes the point of application of the second velocity increment in the orbit O_f, is not known a priori. Therefore, the determination of these true latitudes is part of the solution of the Earth-to-Earth transfer as it will be seen in the next sections. The geometry of departure and the geometry of arrival are presented in Fig. 1. As the velocity increments are applied at the pericenters of the O_i and O_f orbits; thus, the determination of the true latitudes u_i and u_f by the transfer problem corresponds to the determination of the pericenter arguments ω_i and ω_f.

2.2. The patched-conic approximation

For the patched-conic approximation considering the Earth-to-Earth mission, the following hypotheses are assumed:

1. The Earth is taken as a fixed body;
2. The Moon's orbit around the Earth is circular;
3. The gravitational force fields of the Earth and the Moon are central and obey the inverse square law;
4. The transfer trajectory has three distinct phases: a first elliptic geocentric trajectory that describes the departure of the spacecraft from the initial orbit around the Earth; an hyperbolic selenocentric trajectory that describes the lunar flyby; and a second elliptic geocentric trajectory that describes the arrival of the spacecraft at the final orbit around the Earth. The characterization of these phases is done utilizing the concept of the sphere of influence (SOI); in this way, the selenocentric phase is described inside the SOI of the Moon.
5. The two-impulsive model is considered with the velocities increment applied at the initial and final orbits O_i and O_f, respectively. The impulses are not, necessarily, tangential to these terminal orbits