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ScienceDirect

Advances in Space Research xxx (2018) xxx–xxx

ADVANCES IN SPACE RESEARCH (a COSPAR publication)

www.elsevier.com/locate/asr

An iterative analytical technique for the design of interplanetary direct transfer trajectories including perturbations

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Received 21 June 2017; received in revised form 21 March 2018; accepted 28 March 2018

Abstract

An iterative analytical trajectory design technique that includes perturbations in the departure phase of the interplanetary orbiter missions is proposed. The perturbations such as non-spherical gravity of Earth and the third body perturbations due to Sun and Moon are included in the analytical design process. In the design process, first the design is obtained using the iterative patched conic technique without including the perturbations and then modified to include the perturbations. The modification is based on, (i) backward analytical propagation of the state vector obtained from the iterative patched conic technique at the sphere of influence by including the perturbations, and (ii) quantification of deviations in the orbital elements at periapsis of the departure hyperbolic orbit. The orbital elements at the sphere of influence are changed to nullify the deviations at the periapsis. The analytical backward propagation is carried out using the linear approximation technique. The new analytical design technique, named as biased iterative patched conic technique, does not depend upon numerical integration and all computations are carried out using closed form expressions. The improved design is very close to the numerical design. The design analysis using the proposed technique provides a realistic insight into the mission aspects. Also, the proposed design is an excellent initial guess for numerical refinement and helps arrive at the four distinct design options for a given opportunity.

Keywords: Interplanetary; Trajectory design; Analytical technique; Patched conic; Perturbations

1. Introduction

The analytical techniques for interplanetary transfer trajectory design provide quick and in-depth information about the mission scenario. In general, the analytical techniques use simple force models and provide quick approximate designs. The patched-conic approximation offers an efficient means for generating interplanetary trajectories (Bate et al., 1971; Prussing and Conway, 1993; Conway, 2010; Englander et al., 2012). By dividing the overall interplanetary trajectory into a series of two-body trajectories, it greatly simplifies the transfer trajectory design process. The patched conic design could be used as an initial guess for a numerical procedure which produces a fully integrated n-body solution. As an improvement over the conventional patched conic technique, an iterative patched conic technique has been developed and used for interplanetary trajectory design (Parvathi and Ramanan, 2017). This technique provides an improved initial guess for numerical refinement. However, all the above techniques do not include the perturbations in the design process of transfer trajectory. In general, the perturbations are included during the numerical refinement process. The aim of the current paper is to account for the perturbations

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https://doi.org/10.1016/j.asr.2018.03.034
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Nomenclature

a	semi-major axis, km
e	eccentricity
h_p	periapsis altitude, km
i	inclination, deg
J_2–J_6	non-spherical Earth gravity zonal harmonic coefficients
$\boldsymbol{r}$	geocentric position vector of the spacecraft, km
$\boldsymbol{r_m}$	geocentric position vector of Moon, km
$\boldsymbol{r_{sd}}$	heliocentric position vector of the spacecraft, km
$\boldsymbol{r_s}$	geocentric position vector of Sun, km
$\boldsymbol{r_{md}}$	selenocentric position vector of the spacecraft, km
R_e	equatorial radius of Earth, km
t_A	SOI duration for the arrival planet, days
t_D	SOI duration for the departure planet, days
t_{FD}	flight duration, days
$\boldsymbol{v}$	geocentric velocity vector of the spacecraft, km/s
Ω	right ascension of ascending node, deg
α_∞	right ascension of the asymptotic velocity vector, deg
δ_∞	declination of the asymptotic velocity vector, deg
μ	gravitational constant, km^3/s^2
ν	true anomaly, deg
ω	argument of periapsis, deg

Subscripts

e	Earth
m	Moon
s	Sun
A	arrival planet
D	departure planet
P	periapsis
∞	hyperbolic orbit

Abbreviations

AoP	argument of periapsis
APO	arrival parking orbit
B-ITRPC	biased iterative patched conic
CAA	closest approach altitude
DPO	departure parking orbit
ITR PC	iterative patched conic
POI	planetary orbit insertion
RAAN	right ascension of ascending node
SOI	sphere of influence
TCM	trajectory correction maneuver
TDB	barycentric dynamical time
TPI	trans-planetary injection

analytically in the design process and provide an improved analytical design.

The perturbations such as non-spherical Earth gravity and luni-solar gravities affect the motion of an Earth satellite to a very large extent. The equations of motion of a satellite in an orbit over an oblate earth in vacuum are solved analytically by King-Hele (1958). However, these expressions are applicable for orbits of eccentricity 0.05 or less. Many studies present closed form expressions to compute the deviations in orbital elements due to the luni-solar perturbations. Cook (1962) discussed several papers that deals with the luni-solar perturbations of the orbits of artificial satellites and their limitations also. He developed a theory to compute the perturbations due to a third body using Lagrange's planetary equations by integrating over one revolution of the satellite. Luni-solar perturbations of the orbit of an artificial Earth satellite are given by modifying the analytical theory of an artificial lunar satellite by Roy (1969). Kozai (1973) gives analytical expressions for the short-periodic perturbations due to the moon and the sun. The secular and long-periodic perturbations are derived by numerical integration. Lane (1989) developed two different procedures for analytically modelling the effects of the Moon's gravitational force on artificial Earth satellites. Recently, there are studies on the orbit of artificial Earth satellite in strongly perturbed environments (Russell, 2012). The expressions presented in all above studies are useful for closed orbits only and cannot be used for the hyperbolic orbits in which the spacecraft moves during interplanetary travel.

To include the third body perturbation effects in the interplanetary trajectory design process, an improved analytical technique known as pseudostate technique was proposed by Wilson (1970). This technique accounts for the effect of Sun within the sphere of influence of Earth. Parvathi and Ramanan (2016) developed an algorithm for interplanetary orbiter missions using pseudostate technique. Penzo (1970) derived analytical expressions for the variations in orbital elements due to Earth J_2 within five Earth radii and computed the effect of Earth oblateness on lunar and interplanetary trajectories. Byrnes and Hooper (1970) proposed a design process that uses these analytical expressions and refines the design accounting for Earth oblateness effect in the case of lunar transfer. Note that in a lunar transfer, the departure phase is a geocentric ellipse. But for an interplanetary transfer, as pointed out earlier, the departure phase is a geocentric hyperbola. Recently, Zhang et al. (2014) developed a linear approximation (LA) based fast prediction algorithm to compute the effects of the non-spherical gravity

Please cite this article in press as: Parvathi, S.P., Ramanan, R.V. An iterative analytical technique for the design of interplanetary direct transfer trajectories including perturbations. Adv. Space Res. (2018), https://doi.org/10.1016/j.asr.2018.03.034