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Recovery of halo orbit missions in case of contingent station-keeping maneuver delay

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Abstract

This paper investigates a recovery strategy for halo orbit missions in case of contingent station-keeping maneuver delay. It is assumed that (1) the delay is temporal, and the spacecraft control becomes available after the delay, (2) the thruster is not producing thrust during the delay. The idea behind the recovery strategy is to deliver the spacecraft into the "cheapest-to-get" halo orbit rather than into the reference one. This approach reduces the transfer costs and saves fuel that can be used for future station-keeping maneuvers, thereby increasing the mission lifetime. Monte Carlo trials are used to estimate the savings and their scattering for each delay time. Families of halo orbits around the Sun–Earth L_1/L_2 points and the Earth–Moon L_1/L_2 points are considered.

Keywords: Recovery strategy; Station-keeping; Circular restricted three-body problem; Libration point; Halo orbit

1. Introduction

Libration point missions are of high interest today. The dynamics around these equilibrium points provide families of periodic orbits which are convenient for conducting astrophysical and solar observations, establishing communication links etc. Among past and present missions, there are the International SunEarth Explorer 3 (ISEE-3), the Solar and Heliospheric Observatory (SOHO), the Wilkinson Microwave Anisotropy Probe (WMAP), the Genesis mission, the Gaia mission, and the Acceleration, Reconnection, Turbulence and Electrodynamics of the Moons Interaction with the Sun mission (ARTEMIS), which consist of two spacecraft: P1 and P2. Leading space agencies have proposed a number of future promising projects: the James Webb Space Telescope (NASA/ESA/CSA), Euclid (ESA/NASA), Millimetron and Spektr-RG (Roscosmos/ESA).

Since the dynamics near collinear libration points (CLPs) are unstable, station-keeping control is required. For this reason, accurate trajectory determination and regular control-law updates are essential. The optimal placement of two statistical control maneuvers for keeping a spacecraft near a CLP was studied by Renault and Scheeres (2003). Under the linear approximation, explicit formulas for spacecraft control and the mean Δv are found. The investigation reveals the relation between the optimal maneuver spacing and the characteristic time of instability: 23 days for the Sun–Earth L_1/L_2 points and 1.5–2.0 days for the Earth–Moon L_1/L_2 points. Taking into account the periods of typical halo orbits (180 days for the Sun–Earth system and 12 days for the Earth–Moon system), one can see that optimal station-keeping takes 5–8 corrections per orbit. Similar results for a continuous thrust are obtained by Gustafson and Scheeres (2009). The authors discover the same relations between the optimal control-law updates and the characteristic time of instability.

Typical values of Δv for station-keeping control are presented in Table 1 and can be found in Farquhar (2001) and

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Table 1
Missions and station-keeping requirements.

Mission	CLPs	Type of orbit	$A_y, A_z, 10^3$ km	Δv, m/s/year
ISEE-3	SE L_1	Halo	666.67, 120.0	8.5
Wind	SE L_1	Quasi-halo	640.0, 170.0	1.0
SOHO	SE L_1	Halo	666.67, 120.0	2.4
WMAP	SE L_2	Lissajous	264.0, 264.0	1.2
Genesis	SE L_1	Quasi-halo	800.0, 450.0	9.0
ARTEMIS (P2)	EM L_1	Quasi-halo	67.71, 4.68	5.09
ARTEMIS (P1)	EM L_2	Quasi-halo	63.52, 35.20	7.39
Gaia	SE L_2	Lissajous	350.0, 100.0	2.0

Dunham and Roberts (2001) for ISEE-3, in Roberts (2011) and Brown and Petersen (2014) for Wind, in Roberts (2003) for SOHO, in Limon et al. (2003) for WMAP, in Williams et al. (2000, 2005) for Genesis, in Folta et al. (2014) for ARTEMIS, and in Renk and Landgraf (2014) for Gaia. It should be noted that the difference in Δv values is formed by several factors: the three-body system considered, the station-keeping method used, and the specific attitude and orbit control system design.

Missing a correction maneuver leads to the growing deviation of the spacecraft trajectory from the nominal orbit. So, thruster failure or loss of communication with the spacecraft can result in contingent maneuver delays and threaten the mission scenario.

According to the study of Tafazoli (2009), the largest percentage of all the control system-related failures occured are due to thruster failures. In those cases, the control is allocated to a redundant set of thrusters (attitude control thrusters or a backup orbital thruster). It is worth noting that most of the publications devoted to the thruster failure issue are related only to collision avoidance during rendezvous and docking (e.g. Pong, 2010; Breger, 2007). To the authors' knowledge, the problem of libration point mission recovery has not been deeply studied yet.

Loss of communication is another problem that can cause the correction delay. On 25 June 1998, communication with the SOHO spacecraft was lost during the planned extension of the mission (Vandenbussche, 1999). It was found that divergence from the nominal halo orbit would be small only till mid-November 1998. Fortunately, the recovery of the mission took precisely the allowable time.

For the case of temporal correction maneuver delays, the authors have previously introduced two recovery strategies: periodic orbit targeting (POT) and stable manifold targeting (SMT) (Shirobokov and Trofimov, 2014). The POT strategy proposes a transfer to the best (in terms of Δv) backup orbit after the delay rather than to the reference one; the more general SMT strategy places the spacecraft onto the stable manifold of the best backup orbit. It was assumed that the thruster is not producing thrust during the delay. To illustrate the two strategies, we considered a family of planar Lyapunov orbits around the Sun–Earth L_2 point. We showed that both POT and SMT strategies can significantly save Δv for future station-keeping when a transfer is performed to the best backup orbit (or its stable manifold) rather than to the reference orbit. In addition, the difference between the POT and SMT strategies in terms of Δv appeared to be negligible. In any case, since the return transfer Δv exponentially grows as the delay increases, the transfer to the best backup or reference orbit should be performed as soon as possible (Shirobokov and Trofimov, 2014).

In the present paper, we investigate the periodic orbit targeting strategy with halo orbits around the Sun–Earth L_1/L_2 points and the Earth–Moon L_1/L_2 points. We first begin with the theory background on the circular restricted three-body problem, the dynamics around the CLPs, and the construction of halo orbits around them. Then we move to the problem statement and describe a two-impulse transfer optimization problem. In the Results section, we present the computed savings in Δv when transferring to the best backup orbit instead of to the reference one. The savings and their scattering are estimated in a series of the Monte Carlo trials.

2. Theory background

Throughout the paper, the circular restricted three-body problem model (CR3BP) is used. According to the CR3BP model, two masses m_1 and $m_2 \leqslant m_1$ move in circular orbits about their barycenter C, and a spacecraft of negligible mass moves under the gravitational attraction of m_1 and m_2. In the CR3BP, the system of two masses m_1 and m_2 defines the force field uniquely. If m_1 is the Sun and m_2 is the Earth,[1] then we obtain the Sun–Earth system; otherwise, if m_1 and m_2 represent the Earth and the Moon, respectively, then "the Earth–Moon" system is under consideration.

The equations of motion are usually written in the standard rotating coordinate frame (Fig. 1) with the origin at C; the x-axis connects the masses m_1 and m_2 towards m_2, the z-axis is directed along the angular velocity of the orbital motion of m_2 around m_1, and the y-axis completes the right-handed system.

It is also convenient to use a dimensionless system of units in which (1) masses are normalized so that

[1] Sometimes, m_2 also contains the mass of the Moon; in this case, the system is called the Sun–Earth/Moon system or the Sun–Barycenter system.