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The end-of-life disposal of satellites in libration-point orbits using solar radiation pressure

Stefania Soldini *, Camilla Colombo, Scott Walker

Astronautics Research Group, University of Southampton, Southampton SO171BJ, United Kingdom

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Abstract

This paper proposes an end-of-life propellant-free disposal strategy for libration-point orbits which uses solar radiation pressure to restrict the evolution of the spacecraft motion. The spacecraft is initially disposed into the unstable manifold leaving the libration-point orbit, before a reflective sun-pointing surface is deployed to enhance the effect of solar radiation pressure. Therefore, the consequent increase in energy prevents the spacecraft's return to Earth. Three European Space Agency missions are selected as test case scenarios: Herschel, SOHO and Gaia. Guidelines for the end-of-life disposal of future libration-point orbit missions are proposed and a preliminary study on the effect of the Earth's orbital eccentricity on the disposal strategy is shown for the Gaia mission.

Keywords: End-of-life disposal; Solar radiation pressure; Circular restricted-three body problem; Elliptic restricted-three body problem; Zero-velocity curves

1. Introduction

Libration-point orbit (LPO) missions are often selected for studying the Sun and the Universe. Orbits around the Libration points L_1 and L_2 of the Sun–Earth system are advantageous as they can be reached from the Earth and, since a constant sky field of view is ensured with respect to the Sun and the Earth, they are frequently used for space observation. There are also further advantages regarding the ease of Earth communication and in the thermal system design. However, they lie in highly perturbed regions; therefore, an uncontrolled spacecraft would naturally follow the unstable manifold and after several years could cross the protected regions at the Earth and the L_1/L_2 regions. In addition, since LPO spacecraft are characterised by large dry masses, it is critical to clear these regions once the mission has ended.

Possible disposal strategies for LPO missions were investigated as a result of a European Space Agency (ESA) study on end-of-life (EOL) disposal concepts for Lagrange-point and Highly Elliptical Orbit Missions. The natural multi-body dynamics in the Earth environment and in the Sun–Earth system were exploited. The main idea is to either restrict the motion of the spacecraft to specific regions or to destroy the spacecraft through an impact with another body. The options investigated were Earth's re-entry, injection onto trajectories towards a Moon impact, heliocentric parking orbit by means of Δv or

Abbreviations: AU, astronomical unit; CR3BP, circular restricted three body problem; EOL, end-of-life; ER3BP, elliptic restricted three body problem; ESA, European Space Agency; GEO, Geosynchronous Equatorial Orbit; JAXA, Japan Aerospace Exploration Agency; LPO, libration-point orbit; SRP, solar radiation pressure; ZVC, zero velocity curve.

* Corresponding author.
E-mail addresses: s.soldini@soton.ac.uk (S. Soldini), c.colombo@soton.ac.uk (C. Colombo), sjiw@soton.ac.uk (S. Walker).

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Nomenclature

L_j/SL_j jth collinear Lagrangian/Pseudo-Lagrangian point with $j = 1, \ldots, 3$
U total potential energy which includes the rotating system potential and the gravitational potential
U_s potential of SRP
$_x$, $_y$ and $_z$ derivatives with respect to x, y and z
r_{Sun-p} and $r_{Earth-p}$ spacecraft (p) and Sun and Earth distances respectively
m_{Earth} Earth's mass [kg]
M_{Sun} Sun's mass [kg]
μ mass parameter $(m_{Earth}/(M_{Sun} + m_{Earth}))$ for the Sun–(Earth+Moon) system is equal to $3.04042 \cdot 10^{-6}$
x_{Sun} and x_{Earth} Sun $(-\mu)$ and Earth $(1-\mu)$ positions respectively
μ_{Sun} and μ_{Earth} Sun $(1-\mu)$ and Earth (μ) unit masses respectively
m spacecraft's mass [kg]
A spacecraft's reflective area [m^2]
σ mass-to-area ratio (m/A) [$kg\ m^{-2}$]
σ^* Sun luminosity, 1.53 [$g\ m^{-2}$]
β lightness parameter (σ^*/σ)
c_R reflective coefficient
$P_{srp-1AU}$ Sun pressure at 1 AU
$r_{Earth-Sun}$ Earth to Sun distance
x_{SL_j} position of the collinear pseudo-Lagrangian point
E energy of the system
β_0 lightness parameter for the spacecraft's initial dry area-to-mass ratio
Δv variation in the spacecraft's velocity after a manoeuvre
ΔE variation of energy between the spacecraft and L_j or SL_j
E_{SL_j} energy of the pseudo-Lagrangian point
$\Delta\beta$ contribution of the increased area of the spacecraft after the deployment where $\beta = \beta_0 + \Delta\beta$
V magnitude of the spacecraft's velocity
V_{clsr} spacecraft's velocity required after the Δv manoeuvre
Δv_{eq} theoretical Δv
R correspond to 60,000 [km]
$d_{Earth-p}$ spacecraft-Earth distance
$\boldsymbol{r}_{Earth}$ and $\boldsymbol{r}_p$ distances of the Earth and the spacecraft from the center of mass
A_0 initial spacecraft's deployable area [m^2]
m_{dry} dry mass [kg]
A_0/m_{dry} initial area-to-mass ratio [$m^2\ kg^{-1}$]
r Earth+Moon distance from the Sun [km]
a semimajor axis [km]
f true anomaly [rad]
e Earth's orbit eccentricity
$\dot{f}$ angular velocity [$rad\ s^{-1}$]
h angular momentum [$km^2\ s^{-1}$]
$M_{Earth+Moon}$ Earth+Moon mass [kg]
G constant of gravitation [$N\ m^2\ kg^{-2}$]
$\tilde{\mu}$ $G \cdot (M_{Sun} + M_{Earth+Moon})$ [$km^3\ s^{-2}$]
ω, Ω and Ω' potential energy in the ER3BP
$\boldsymbol{r}_d$ and $\dot{\boldsymbol{r}}_d$ dimensional position [km] and velocity [$km\ s^{-1}$] coordinates
$\boldsymbol{r}$ and $\boldsymbol{r}'$ non-dimensional pulsating position and velocity coordinates
I energy integral
f_0 initial true anomaly when leaving the LPO [rad]

heliocentric parking orbit by means of solar radiation pressure (SRP) (Colombo et al., 2015, 2014).

Olikara et al. (2013) initially proposed a disposal option, which injects the spacecraft towards the inner or the outer solar system and closes the Hill's surfaces though a Δv manoeuvre at L_1 or L_2. Olikara et al. (2013) suggested that the overall return trajectories to the Earth in the circular restricted three body problem (CR3BP) and in the full-body (ephemerides) are quite similar.

In this article, an alternative disposal strategy is investigated that allows the closure of the zero-velocity curves by means of SRP. In this case, the spacecraft is disposed at the EOL onto the unstable manifold leaving the LPO from L_2. An energetic approach is used to close the Hill's curves at SL_2 (*e.g.*, the pseudo Libration-point L_2 when SRP is added (McInnes, 2000)) by increasing the energy of the system and then computing the reflective deployable area required for the EOL curves-closure. As a term due to SRP is added to the energy, the shape of the potential surfaces changes and the required reflective area is computed via numerical optimisation, imposing the condition for the curves closure at SL_2. After the closure, the spacecraft is bounded in its following motion at the right-hand side of the pseudo Libration point, thus, preventing the spacecraft's return to the Earth and protecting the L_2 region. It is also demonstrated that the spacecraft cannot be confined towards the inner solar system due to the constraint in the direction of SRP acceleration and that the disposal through SRP can only be performed at SL_2. This strategy can be achieved through a sun-pointing auto-stabilised deployable structure, such as light reflective surfaces that are already proven for attitude control applications (*e.g.*, GOES's cone solar sail[1] (DRL-101-08, 1996)), with the advantage of saving propellant.

[1] The cone shape of the sail guarantees attitude passive stabilisation.

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