Post mission disposal design: Dynamics and applications

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Abstract. This research addresses post mission disposal design techniques with particular attention to long-term evolution of spacecraft orbits. Semianalytical dynamics models for orbital perturbations are developed to filter out short-periodic perturbation effects and more importantly to improve computational efficiency when dynamics models are integrated in the manoeuvre optimisation problem. The disposal of a spacecraft targeting an Earth re-entry is attained by employing natural orbital perturbations and enhancing natural effects with impulsive manoeuvres. Hamiltonian representations of the system are used to analyse dynamics behaviours and to validate the solutions obtained from optimisation processes.

Introduction

The population of space objects has been increasing more rapidly in last decades, especially with the introduction of mega-constellations. According to European Space Agency (ESA), about 16,990 satellites have been placed into Earth orbit and only about 9000 of them are still functioning. And about 35,150 debris objects are regularly tracked by Space Surveillance Networks (SSN) and maintained in their catalogue [1][2]. The increasing number of space objects in Earth orbits increases probability of collisions between objects leading to a cascade process, known as the Kessler syndrome [3]. In response to this situation, the Inter-Agency Space Debris Coordination Committee (IADC) published space debris mitigation guidelines specifying different mitigation measures [4], one of which is to design end-of-life disposal strategies for spacecrafts [5], preventing prolonged stay in geostationary orbit (GEO) and limiting passage in low Earth orbit (LEO).

A successful end-of-life disposal could make a large contribution to debris mitigation, on the one hand, and implementation of end-of-life could consume large amounts of propellant and significantly increase economic cost, on the other, which decreases feasibility of disposal strategies and discourages spacecraft operators from meeting mitigation guidelines. Therefore, the research aims to develop end-of-life disposal design techniques for spacecrafts, leveraging orbital perturbations to reduce energy consumption during end-of-life disposal.

The research focuses on applying semianalytical propagators to manoeuvre design and optimisation processes, reducing computational burden compared to optimisation with high-fidelity models. The research is composed of three main parts. First, semianalytical models for orbital perturbations are developed based on averaging techniques. This could be implemented in different reference frames with different pros and cons. The following section shows an example in the Earth equatorial frame. Then, the analytical approach to describe orbital perturbations by Kozai using Hamiltonian representations is developed considering J_2 and lunisolar perturbations. This can be exploited in analysing long-term dynamics behaviours and identifying possible disposal strategies qualitatively. It can also be used after the optimised solutions are obtained to validate the effectiveness of the solutions. The last part is to design disposal manoeuvres. This can be done with or without numerical orbit propagation. An optimisation using triple averaged model without numerical orbit propagation is firstly done as a preliminary investigation due to its higher

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efficiency but less accuracy. In this way, the optimisation require much less computational resources. The obtained results can then be used as a first guess to optimise with numerical orbit propagation using double averaged model to refine the results, since it's more accurate but computationally expensive.

In the next sections, the research performed or planned for each block is explained with the focus on simplification of dynamics models and analysis of dynamics behaviours.

Semianalytical model for orbital perturbations

One of the main difficulties of spacecraft end-of-life disposal design is high computational cost of optimisation process since numerical orbit propagation of decades is involved. Therefore, the main idea is to use semianalytical dynamics models for orbital perturbations to reduce the computational costs and hence the resources needed within the optimisation process.

The dynamics of a spacecraft orbiting the Earth is described as a perturbed two-body problem and is given by the well-known Lagrange planetary equations of motion, where the disturbing function is trigonometric functions of Keplerian elements, periodic in different time scales corresponding to different angles. The idea of semianalytical dynamics model is to separate the short-periodic, long-periodic, and secular perturbation effects to focus on long-term and secular evolution of orbital motion. One method of developing such models is to average the disturbing function over different time intervals to eliminate the fast angles.

The disturbing function of the J_2 perturbation is averaged over one orbital period of a spacecraft [6],

$$\overline{R}_{J_2} = \frac{\mu J_2 R_\oplus^2}{4a^3\eta^3}(2 - 3\sin^2 i), \tag{1}$$

and the one of third-body perturbation is averaged in the same manner,

$$\overline{R}_{3b} = \frac{\mu_3}{r_3}\sum_{l=2}^{\infty}\left(\frac{a}{r_3}\right)^l F_l(A, B, e), \tag{2}$$

and averaged again over one orbital period of the perturbing body,

$$\overline{\overline{R}}_{3b} = \frac{\mu_3}{a_3}\sum_{l=2}^{\infty}\left(\frac{a}{a_3}\right)^l F_l(\alpha_A, \beta_A, \alpha_B, \beta_B, e), \tag{3}$$

where $A, B, \alpha_A, \beta_A, \alpha_B, \beta_B$ are functions of i, Ω, ω of a spacecraft and $e_3, i_3, \Omega_3, \omega_3$ of a perturbing body.

The total single- and double-averaged disturbing functions are following,

$$\overline{R} = \overline{R}_{J_2} + \overline{R}_{Sun} + \overline{R}_{Moon}, \tag{4}$$

$$\overline{\overline{R}} = \overline{R}_{J_2} + \overline{\overline{R}}_{Sun} + \overline{\overline{R}}_{Moon}. \tag{5}$$

To further simplify the dynamics model, one can average the third-body disturbing function over period of variation of RAAN, also known as elimination of the node,

$$\overline{\overline{\overline{R}}}_{3b} = \frac{\mu_3}{a_3}\sum_{l=2}^{\infty}\left(\frac{a}{a_3}\right)^l F_l(e, i, \omega, e_3, i_3, \omega_3), \tag{6}$$

where the node of the third body's orbit is also eliminated since it is coupled with RAAN of a spacecraft.

The averaging technique allows one to eliminate fast angles in the disturbing function, hence separating long-periodic and secular effects from the short-periodic ones. This procedure is of importance since it simplifies the manoeuvre optimisation process a lot. The simplified model is validated by comparing with the high-fidelity models and actual ephemerides.

Phase space structures of the dynamics

The Hamiltonian formulation of a system allows to understand the behaviours and qualitative insights of dynamics. Thanks to simplification through averaging, a Hamiltonian formulation of dynamics of orbital perturbations only depending on e, i, ω of a spacecraft is obtained. Furthermore, the so-called Kozai parameter $\Theta = (1 - e^2)\cos^2 i$ [7] is a constant since the z-component of angular momentum is conserved. Hence, one can get a Hamiltonian of one degree-of-freedom and phase space maps of the dynamics [8].

As an example, the left figure of Fig 1 shows a phase space map of e, i, ω of a spacecraft in highly elliptical orbits (HEO) considering lunar perturbation only. It demonstrates that the phase space of the dynamics follows a layer structure where different layer corresponds to different respective values of Kozai parameter. The red line corresponds to $\Theta = 0.3$ while the black line corresponds to $\Theta = 0.1$. The figure of right-hand side is an e, ω map of orbits with a semimajor same as the INTEGRAL mission, and the red curve shows the initial phase curve of the INTEGRAL mission [9]. It is identified that spacecraft orbits evolve under natural perturbations along the phase curve. Therefore, it is possible to enhance natural evolution of orbits by impulsive manoeuvres at some point of a phase curve, driving the orbit to another phase curve and the orbit then evolves under natural perturbations and meets re-entry conditions by entering the critical region after curtain time.

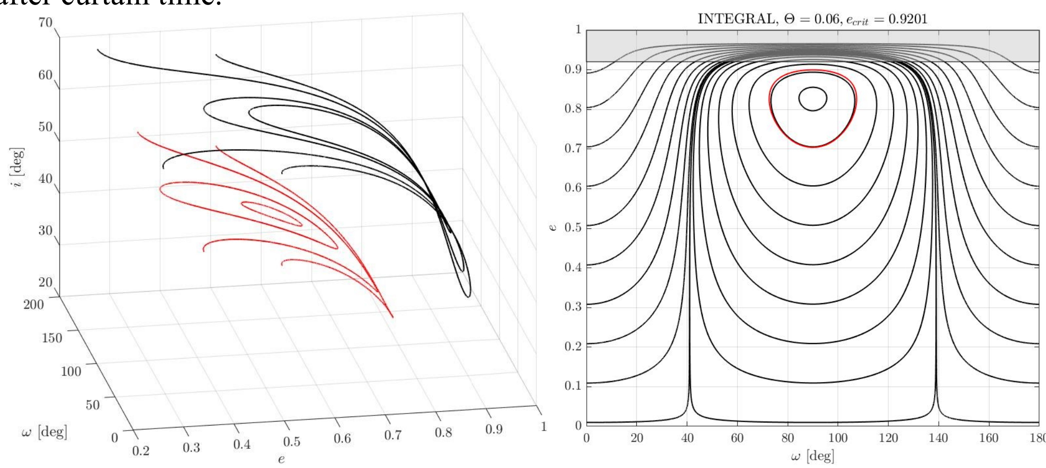

Fig 1 Left: Phase space map, lunar perturbation only, red line: $\Theta = 0.3$*, black line:* $\Theta = 0.1$*; Right: Phase space portrait of the INTEGRAL mission,* $\Theta = 0.06, e_{crit} = 0.9201$*, red curve: initial phase curve of the INTEGRAL mission.*

Disposal manoeuvres design

Following inspiration of the previous section [8], the procedures for computing disposal manoeuvres are following. For any given start point, variations of elements are obtained using Gauss' variational equations, in which Δv is formulated in tangential-normal-subnormal (TNH) frame. The post-manoeuvre elements are propagated using averaged equations to get the maximal eccentricity e_{max}. Assume that a successful re-entry is attained if perigee height is below 100 km, base on which the concept of critical eccentricity e_{crit} is defined. Therefore, the attainment of re-entry is determined by $e_{max} > e_{crit}$.

As a preliminary investigation, a grid search was carried out where magnitudes and directions of manoeuvres are meshed with limits $\Delta v = 100{\sim}150\ \mathrm{m/s}, \alpha = 0{\sim}180°, \beta = 0{\sim}180$. Fig 2 shows the result of the grid search in phase space map and demonstration of manoeuvres corresponding to successful re-entries. The result shows that manoeuvres leading to successful re-entries lie in a cone region along the tangential direction and the most effective ones are the ones in the same or the opposite direction of the tangential direction.

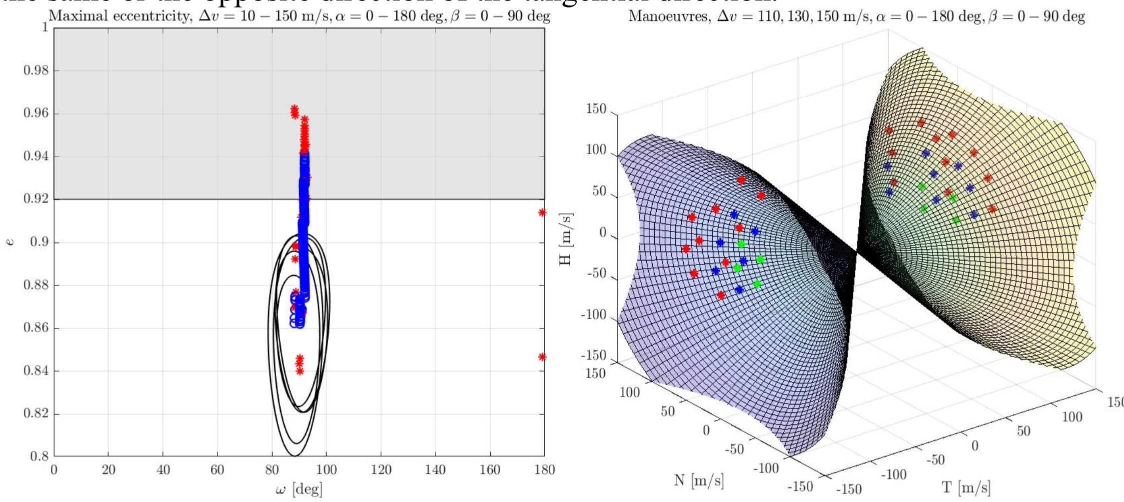

Fig 2 Left: Result of grid search shown in e, ω *phase space, black curve: initial phase curve, blue markers:* e_{max} *for successful re-entries, red markers:* e_{max} *for non-successful re-entries, grey zone:* $e_{max} > e_{crit}$*; Right: Demonstration of manoeuvres corresponding to successful re-entries.*

Conclusions and next steps

The output of semianalytical models and phase space analysis of dynamics provides possibilities for efficient disposal manoeuvre optimisation. Following the preliminary investigation, the optimisation of disposal manoeuvres with or without numerical orbit propagation can be carried out to identify the optimal solutions. The obtained solutions can then be validated through high-fidelity numerical orbit propagation and through phase space representations. This would be part of the final year research and would focus on the feasibility of analytical approaches.

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