High-Precision Repeat-Groundtrack Orbit Design and Maintenance for Earth Observation Missions

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Abstract The focus of this paper is the design and station keeping of repeat-groundtrack orbits for Sun-synchronous satellite. A method to compute the semimajor axis of the orbit is presented together with a station-keeping strategy to compensate for the perturbation due to the atmospheric drag. The results show that the nodal period converges gradually with the increase of the order used in the zonal perturbations up to J_{15} . A differential correction algorithm is performed to obtain the nominal semimajor axis of the reference orbit from the inputs of the desired nodal period, eccentricity, inclination and argument of perigee. To keep the satellite in the proximity of the repeat-groundtrack condition, a practical orbit maintenance strategy is proposed in the presence of errors in the orbital measurements and control, as well as in the estimation of the semimajor axis decay rate. The performance of the maintenance strategy is assessed via the Monte Carlo simulation and the validation in a high fidelity model. Numerical simulations substantiate the validity of proposed mean-elements-based orbit maintenance strategy for repeat-groundtrack orbits.

Keywords Repeat-groundtrack orbits; Semi-analytical design; Differential correction algorithm; Maintenance strategy; Monte Carlo; High fidelity dynamical model

1 Introduction

A repeat-groundtrack orbit enables a spacecraft to retrace its groundtracks on the surface of the Earth over a certain period of time, thus allowing the spacecraft to reobserve any selected area within the same scheduled time interval (Lara 1999; Aorpimai and Palmer 2007; Nadoushan and Assadian 2015). Designing this type of orbit for target access, from the perspective of responsive space, is a key area of ongoing research (Sengupta et al. 2010). In practice, the low Earth orbit (LEO) with repeat-groundtrack pattern is of interest for various Earth observation missions, such as Landsat (USA), ICESat&ICESat-2 (USA), ENVISAT (Europe), SPOT (France), and IRS (India)

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(Aorpimai and Palmer 2007; Fu et al. 2012; Nadoushan and Assadian 2015).

During the repeat-groundtrack orbit design procedure, the effect of Earth's non-spherical gravitational perturbations cannot be neglected, that is to say, the Earth cannot be treated as a perfect sphere. In many previous studies (e.g., Abramson et al. 2001; Mortari et al. 2004) only the J_2 perturbation was considered for the orbit design and control, which can be used only for the mission without strict requirement of precision, such as China-Brazil Earth Resource Satellite (CBERS-1) mission (Orlando and Kuga 2003). However, the J_2 zonal harmonic perturbation model is not suitable for missions with higher precision and reliability requirements. For example, missions like TOPEX/Poseidon (Aorpimai and Palmer 2007), ICESat (Schutz et al. 2005), and TerraSAR-X/TanDEM-X (D'Errico 2013) require the groundtracks to be within a threshold of ± 1 km from the reference trace. The Chinese-French Oceanic Satellite (CFOSAT) (Xu and Huang 2014; Zhu et al. 2015) currently under development has a stringent demand for the gridding division of the ground to enable precise mapping. Therefore, it follows the need of designing the nominal orbit in an accurate dynamical model, including higher order zonal harmonics.

Among the approaches developed for the solution of the perturbation problem, semi-analytical methods have the desirable feature of combining the efficiency of analytical methods and accuracy of numerical integration (Wittig and Armellin 2015). Depending upon the order of desired accuracy, the semi-analytical satellite theory, such as Draper Semi-analytical Satellite Theory (DSST) and Universal Semi-analytical Method (USM), can be used to generate variations of parameters equations of motion due to perturbations (Danielson et al. 1994; Yurasov 1996; Cefola et al. 2003). In the investigation concerning the design and maintenance of Earth repeat-groundtrack orbits, the semi-analytical techniques, by eliminating the short-periodic perturbations, can compute solutions that match well with those obtained from numerical simulations. Lara and Russell have published a series of celebrated works on the repeat-groundtrack orbit, especially considering the effect from high order perturbations. By means of automated numerical algorithm, the initial conditions of repeat-groundtrack orbits have been found for the high-order zonal perturbations of gravity field (Lara 1999). From an abstract viewpoint, Lara (2003) even studied that the repeat-groundtrack orbits arise from bifurcations of equatorial periodic-orbit families by taking into account the high-order tesseral terms of perturbations. Further, with the differential corrections computation of periodic orbits, Lara and Russell (2008) demonstrated the existence of repeat-groundtrack orbits and provided a fast approach to obtain the entire families of orbits in full geopotentials. The same method was also applied to the search of repeat-groundtrack orbits around the Moon and Europa under the effect from a combination of high-order gravity potential and the restricted three-body

perturbations (Lara and Russell 2007; Russell and Lara 2007; Lara 2011). However, the above-mentioned studies did not include the orbit maintenance problem to take advantage of high-precision orbit design.

Without necessary orbit control or sufficient precision during the orbit design procedure, the repeat-groundtrack orbits would lose the repetition property over the long term due to the perturbations in the real space environments. Several works studied the problem of orbit maintenance adopting different methods. Using the epicycle elements with the zonal harmonics of geopotential up to J_4 as well as J_2^2 , Aorpimai and Palmer (2007) showed how to acquire the near-circular Earth repeat-groundtrack orbit and to maintain the satellite around the repetition condition. With the semi-analytical techniques, the design and maintenance of the low eccentricity orbits for terrestrial coverage was investigated by Sengupta et al. (2010), but only the J_2 perturbation was taken into consideration in the design process. Fu et al. (2012) studied a strategy for maintenance of a low Earth repeat-groundtrack successive-coverage based on an analysis of drift over the entire groundtrack, but they selected the two-body orbit as the reference orbit.

The present work deals with the design and maintenance of Sun-synchronous Earth repeat-groundtrack orbits. In particular, a conventional semi-analytical method is employed to compute a nominal orbit which is then used as reference for a new maintenance strategy. Different from previous works, the proposed maintenance strategy includes the effect of errors in the orbital measurements and control, as well as in the semimajor axis decay rate estimation due to inaccuracies in the drag model.

The remainder of this paper is organized as follows. In Section 2, the semi-analytical method to compute the nodal period and semimajor axis for repeat-groundtrack orbits is presented in a zonal gravitational model up to J_{15} . Then, the model for orbit groundtrack drift is derived in Section 3.1 and the maintenance strategy for repeat-groundtrack orbits is described in Section 3.2. Afterwards, in Section 4 the simulations to assess the performances of the proposed design and maintenance strategy are conducted, including the discussion of the results obtained with a Monte Carlo simulation and a validation in a high fidelity dynamical model (full geopotentials with degree and order 200 and atmospheric drag). Finally, in Section 5 the conclusions are presented.

2 Semi-analytical design of reference repeat-groundtrack orbits

To acquire the high-resolution image, Earth observing satellites are mainly located in the LEO environments, where the Earth's geopotentials and atmospheric drag are the major source of perturbations (Sengupta et al. 2010; Fu et al. 2012). The effects of the other perturbations, like solar radiation pressure and third-body perturbation are negligible. In general, the reference

repeat-groundtrack orbit is defined by the set of mean nominal orbital elements of the satellite. The calculation of these mean nominal orbital elements is done trading off accuracy of the solution and computational complexity (and thus execution time). On the one hand, the reference trajectory should be computed in a high fidelity dynamical model such that the real trajectory will experience a minimum deviation from repeat-groundtrack condition (De Florio and D'Amico 2009). On the other hand, limiting the computational burden associated with orbit design and, particularly, with orbit maintenance is highly desirable to limit operational costs. Consequently, it is reasonable to consider only the zonal harmonics associated with Earth's oblateness to design the reference orbit; whereas the deviation of the actual orbit from the reference due to the drag is corrected by the orbit maintenance strategy.

As only the zonal perturbation terms of geopotentials are considered in the design of reference repeat-groundtrack orbit, the Earth gravity field is axially symmetric (Kozai 1959; Ammar et al. 2012). Thus, the disturbing potential acting on the satellite can be formulized as the series of zonal harmonics from Kaula (1963, 1966)

$$R = -\frac{\mu}{r} \sum_{l=2}^{\infty} J_l \left(\frac{R_e}{r} \right)^l P_l(\sin \varphi), \quad (1)$$

where μ is the Earth's gravitational constant, R_e is the mean equatorial radius of the Earth, and r and φ are the geocentric distance and latitude of the satellite, respectively. J_l are the zonal harmonics coefficients, indicating the degree of irregularity of the Earth. $P_l(\sin \varphi)$ is the Legendre associated polynomials of degree l , which refers to the potential varying with latitude.

To investigate the repeat-groundtrack orbits in high-precision gravitational field, the current paper expands the non-spherical gravitational disturbing function from J_2 up to J_{15} as

$$\begin{aligned}
R = & \frac{3\mu J_2 R_e^2}{2a^3} \left(\frac{a}{r} \right)^3 \left(\frac{1}{3} - \frac{1}{2} s^2 + \frac{1}{2} s^2 \cos 2(\omega + \theta) \right) + \frac{-\mu J_3 R_e^3}{a^4} \left(\frac{a}{r} \right)^4 \left[\left(\frac{15}{8} s^3 - \frac{3}{2} s \right) \sin(\omega + \theta) - \right. \\
& \left. \frac{5}{8} s^3 \sin 3(\omega + \theta) \right] + \dots + \frac{-9694845 \mu J_{15} R_e^{15}}{2048 a^{16}} \left(\frac{a}{r} \right)^{16} \left[- \left(\frac{143}{215441} s - \frac{1001}{50692} s^3 + \frac{1001}{5336} s^5 - \right. \right. \\
& \left. \frac{35035}{42688} s^7 + \frac{7007}{3712} s^9 - \frac{105105}{44544} s^{11} + \frac{45045}{29696} s^{13} - \frac{6435}{16384} s^{15} \right) \sin(\omega + \theta) - \left(\frac{1001}{152076} s^3 - \right. \\
& \left. \frac{1001}{10672} s^5 + \frac{21021}{42688} s^7 - \frac{7007}{5568} s^9 + \frac{25025}{14848} s^{11} - \frac{135135}{118784} s^{13} + \frac{5005}{16384} s^{15} \right) \sin 3(\omega + \theta) - \\
& \left(\frac{1001}{53360} s^5 - \frac{7007}{42688} s^7 + \frac{1001}{1856} s^9 - \frac{25025}{29696} s^{11} + \frac{75075}{118784} s^{13} - \frac{3003}{16384} s^{15} \right) \sin 5(\omega + \theta) - \\
& \left(\frac{1001}{42688} s^7 - \frac{1001}{7424} s^9 + \frac{25025}{89088} s^{11} - \frac{15015}{59392} s^{13} + \frac{1365}{16384} s^{15} \right) \sin 7(\omega + \theta) - \left(\frac{1001}{66816} s^9 - \right. \\
& \left. \frac{5005}{89088} s^{11} + \frac{4095}{59392} s^{13} - \frac{455}{16384} s^{15} \right) \sin 9(\omega + \theta) - \left(\frac{455}{89088} s^{11} - \frac{1365}{118784} s^{13} + \right. \\
& \left. \frac{105}{16384} s^{15} \right) \sin 11(\omega + \theta) - \left(\frac{105}{118784} s^{13} - \frac{15}{16384} s^{15} \right) \sin 13(\omega + \theta) - \frac{1}{16384} s^{15} \sin 15(\omega + \theta) \Big], \quad (2)
\end{aligned}$$

in which we abbreviate $s \triangleq \sin i$ and use the same abbreviation in Eq. (4); and i is the inclination, a is the semimajor axis, e is the eccentricity, ω is the argument of perigee and θ is the true anomaly. Note that only the head and the tail terms are listed in Eq. (2) (as well as in Eq. (4)) while the others are omitted for convenience and simplification of presentation.

The relationship between the orbit radius and the orbital elements are presented below:

$$r = \frac{a(1-e^2)}{1+e \cos \theta}, \quad (3)$$

where $\mathbf{r}$ is the position vector, and $r = \|\mathbf{r}\|$. According to the Eqs. (2) and (3), the disturbing perturbation is derived as a function of the orbital elements. As shown from Eq. (2), neither the time t nor the right ascension of ascending node (RAAN) Ω exists explicitly in the function R as a result of only considering zonal harmonics.

In the design of Earth repeat-groundtrack orbits, we only consider secular and long-periodic perturbations. The short-periodic variations, therefore, are averaged out. According to the idea of averaging from nonlinear mechanics (Kozai 1959; Liu 1974; Liu et al. 2012), the gravitational perturbations on the orbital elements can be averaged in terms of mean anomaly M from 0 to 2π , and thus secular and long-periodic terms, denoted by R_C and R_L , respectively, are separated from the gravitational perturbations, i.e., $\bar{R} = \frac{1}{2\pi} \int_0^{2\pi} R dM = R_C + R_L$, resulting in:

$$\begin{aligned}
\bar{R} = & \frac{3\mu J_2 R_e^2}{2a^3} (1-e^2)^{-\frac{3}{2}} \left(\frac{1}{3} - \frac{1}{2} s^2 \right) + \frac{-\mu J_3 R_e^3}{a^4} (1-e^2)^{-\frac{5}{2}} e \left(\frac{15}{8} s^3 - \frac{3}{2} s \right) \sin \omega + \dots + \\
& \frac{-9694845\mu J_{15} R_e^{15}}{2048a^{16}} (1-e^2)^{-\frac{29}{2}} \left[-7e \left(1 + \frac{39}{2} e^2 + \frac{715}{8} e^4 + \frac{2145}{16} e^6 + \frac{9009}{128} e^8 + \frac{3003}{256} e^{10} + \right. \right. \\
& \left. \left. \frac{429}{1024} e^{12} \right) \left(\frac{143}{215441} s - \frac{1001}{50692} s^3 + \frac{1001}{5336} s^5 - \frac{35035}{42688} s^7 + \frac{7007}{3712} s^9 - \frac{105105}{44544} s^{11} + \frac{45045}{29696} s^{13} - \right. \right. \\
& \left. \left. \frac{6435}{16384} s^{15} \right) \sin \omega - 91e^3 \left(\frac{1}{2} + \frac{55}{16} e^2 + \frac{99}{16} e^4 + \frac{231}{64} e^6 + \frac{165}{256} e^8 + \frac{99}{4096} e^{10} \right) \left(\frac{1001}{152076} s^3 - \right. \right. \\
& \left. \left. \frac{1001}{10672} s^5 + \frac{21021}{42688} s^7 - \frac{7007}{5568} s^9 + \frac{25025}{14848} s^{11} - \frac{135135}{118784} s^{13} + \frac{5005}{16384} s^{15} \right) \sin 3\omega - 1001e^5 \left(\frac{1}{16} + \right. \right. \\
& \left. \left. \frac{3}{16} e^2 + \frac{9}{64} e^4 + \frac{15}{512} e^6 + \frac{5}{4096} e^8 \right) \left(\frac{1001}{53360} s^5 - \frac{7007}{42688} s^7 + \frac{1001}{1856} s^9 - \frac{25025}{29696} s^{11} + \frac{75075}{118784} s^{13} - \right. \right. \\
& \left. \left. \frac{3003}{16384} s^{15} \right) \sin 5\omega - 143e^7 \left(\frac{3}{16} + \frac{63}{256} e^2 + \frac{35}{512} e^4 + \frac{7}{2048} e^6 \right) \left(\frac{1001}{42688} s^7 - \frac{1001}{7424} s^9 + \frac{25025}{89088} s^{11} - \right. \right. \\
& \left. \left. \frac{15015}{59392} s^{13} + \frac{1365}{16384} s^{15} \right) \sin 7\omega - 91e^9 \left(\frac{11}{256} + \frac{11}{512} e^2 + \frac{3}{2048} e^4 \right) \left(\frac{1001}{66816} s^9 - \frac{5005}{89088} s^{11} + \right. \right. \\
& \left. \left. \frac{4095}{59392} s^{13} - \frac{455}{16384} s^{15} \right) \sin 9\omega - 91e^{11} \left(\frac{1}{512} + \frac{1}{4096} e^2 \right) \left(\frac{455}{89088} s^{11} - \frac{1365}{118784} s^{13} + \right. \right. \\
& \left. \left. \frac{105}{16384} s^{15} \right) \sin 11\omega - \frac{7}{4096} e^{13} \left(\frac{105}{118784} s^{13} - \frac{15}{16384} s^{15} \right) \sin 13\omega \right] . \quad (4)
\end{aligned}$$

As can be seen from Eq. (4), the secular and long-periodic perturbations are presented in terms of the slowly varying elements: a , e , i and ω .

Figure 1 shows a schematic drawing of an orbit, where A_1 and A_2 are the two subsequent ascending nodes of two orbital cycles, and the arrow S indicates the direction of motion of the satellite. From A_1 to A_2 , the mean argument of latitude $\lambda = \omega + M$ varies from 0 to 2π . The nodal period T_Ω , the time between two successive equator crossings in ascending node is

$$T_\Omega = \frac{2\pi}{\dot{\omega} + \dot{M}} . \quad (5)$$

Equation (5) shows the relation between the nodal period and the variational rates of the argument of perigee and the mean anomaly. Under the influence of secular and long-periodic perturbations, the mean nodal period is defined from the nodal period as

$$\overline{T}_\Omega = \frac{2\pi}{\overline{\dot{\omega}} + \overline{\dot{M}}} . \quad (6)$$

Fig. 1 Schematic diagram of the nodal period

In Eq. (6), $\overline{\dot{\omega}} + \overline{\dot{M}}$ stands for the averaged variational rate of the mean argument of latitude under the effects from secular and long-periodic perturbations. These values are derived from the Lagrange planetary equations by using the averaged disturbing function $\overline{R}$ (Kozai 1959; Kaula 1966):

$$\begin{cases} \overline{\dot{\omega}} = -\frac{\cos i}{na^2 \sqrt{1-e^2} \sin i} \frac{\partial \overline{R}}{\partial i} + \frac{\sqrt{1-e^2}}{na^2 e} \frac{\partial \overline{R}}{\partial e}; \\ \overline{\dot{M}} = n - \frac{1-e^2}{na^2 e} \frac{\partial \overline{R}}{\partial e} - \frac{2}{na} \frac{\partial \overline{R}}{\partial a}, \end{cases} \quad (7)$$

where the mean motion n is related to a and decided by $n^2 a^3 = \mu$.

After obtaining the averaged variational rate equations for argument of perigee and mean anomaly from three partial derivatives of secular and long-periodic perturbations with respect to a , e and i , namely $\frac{\partial \overline{R}}{\partial a}$, $\frac{\partial \overline{R}}{\partial e}$ and $\frac{\partial \overline{R}}{\partial i}$, we can derive the analytical expression of the mean nodal period under the influence of secular and long-periodic perturbations considering the zonal harmonics up to J_{15} .

In order to compute the semimajor axis of reference repeat-groundtrack orbit, a differential correction algorithm is proposed. The inputs of the algorithm are the desired nodal period ($T_{\Omega 0}$), eccentricity (e), inclination (i) and argument of perigee (ω). At the end of numerical procedure, the value of semimajor axis that fulfills the groundtrack repetition requirements is obtained.

Generally, for a defined $n_M:n_N$ repeat-groundtrack orbit pattern, where n_M is the number of rotations that the Earth completes in one period of repetition and n_N is the number of revolutions of the satellite during this period of repetition (Lara 2003; Vtipil and Newman 2012; Mortari and Lee

2014), the desired nodal period can be decided by n_M and n_N as $T_{\Omega 0} = \frac{2\pi n_M}{n_N(\omega_e - \dot{\Omega})}$ (ω_e is the

Earth's rotation rate and $\dot{\Omega}$ is the drift rate of RAAN). In the two-body problem, the orbital period

is $T = 2\pi \sqrt{\frac{a^3}{\mu}}$, which is used to initialize our algorithm. From the desired nodal period $T_{\Omega 0}$, the

initial semimajor axis can be derived as $a_0 = \sqrt[3]{\mu \left(\frac{T_{\Omega 0}}{2\pi}\right)^2}$ or $\sqrt[3]{\mu \left(\frac{n_M}{n_N(\omega_e - \dot{\Omega})}\right)^2}$. The derivative

of nodal period with respect to the semimajor axis, used for correction of the semimajor axis, is

$\frac{dT}{da} = 3\pi \sqrt{\frac{a_0}{\mu}}$. The procedure to compute the nominal semimajor axis can be summarized in the

following steps:

(1) From the initial semimajor axis a_0 and the given orbital elements (e , i , and ω), the new nodal period T_{Ω} is obtained from Eq. (6).

(2) The difference between the computed nodal period and the desired one is $\Delta T = T_{\Omega 0} - T_{\Omega}$.

(3) The already obtained derivative of the nodal period with respect to the semimajor axis $\frac{dT}{da}$

is used to compute the semimajor axis correction as $\Delta a = \Delta T / \frac{dT}{da}$.

(4) Correct the semimajor axis as $a_0 = a_0 + \Delta a$ and repeat the Steps (1), (2), and (3) until the convergence condition $\Delta a < \varepsilon$ is met. ε is a small value, for example, $\varepsilon = 0.001$ m is used in this algorithm.

In summary, our algorithm for the computation of the nominal semimajor axis of a repeat-groundtrack orbit is illustrated in Fig. 2.

Fig. 2 Flowchart of nominal semimajor axis determination via the differential correction algorithm

3 Maintenance strategy of repeat-groundtrack orbit

In the previous section, the semi-analytical approach for the determination of the semimajor axis has been presented. Based on the reference semimajor axis, the focus of this section is the derivation of an analytical strategy for repeat-groundtrack orbit maintenance. To begin with, in Section 3.1 the orbit groundtrack drift due to the drag is defined. After that, in Section 3.2 the mean-elements-based maintenance strategy of the repeat-groundtrack is proposed. This strategy includes the effects due to orbital measurements and control error as well as the estimation of semimajor axis decay rate estimation error.

3.1 Orbit groundtrack drift due to atmospheric drag

The dominant perturbations acting on the LEO satellites originate from the Earth's non-spherical mass distribution and atmospheric drag. As the effects of Earth's oblateness are included in the design procedure of reference repeat-groundtrack, the orbit maintenance will consider only the effect of atmospheric drag.

The repeat-groundtrack orbit allows a spacecraft to observe a specific area multiple times in a given time frame. Figure. 3 illustrates an image acquisition pattern in a typical Earth observation

mission. The images are acquired as strips of rectangular shape, and they will then go through programming and processing before they are provided to customers as imaging products (Lemaître et al. 2002). The width of a strip depends on the properties of an on board camera (see the rectangular region in Fig. 3) and the repeat-groundtrack orbit enables the groundtracks to divide the Earth surface uniformly with as little image overlap as possible. However, due to effects of orbital perturbations, mainly from the atmospheric drag causing a slow decay of the orbit, without necessary orbit maintenance the actual groundtrack of the satellite will not reproduce the reference groundtrack pattern. In particular, an eastward drift relative to the reference will manifest, due to the reduced semimajor axis and associated nodal period.

To guarantee the repeatability and effective stitching of two adjacent images of the Earth observing satellite, the maintenance strategy is required to keep the satellite in the proximity of reference repeat-groundtrack orbit, thus avoiding the violation of the corridor boundaries (see the exaggerated parabola-like curve in Fig. 3). The orbit maintenance is performed by adjusting the semimajor axis to compensate for its decay due to the atmospheric drag.

Fig. 3 A typical Earth observation mission in the repeat-groundtrack orbit. In this figure, the blue dashed line is the exaggerated drawing of the actual groundtrack to illustrate its drift relative to the reference groundtrack.

More specifically, as shown in Fig. 1, the separation in distance between two subsequent equator crossings $L = \widehat{A_1 A_2}$ during one orbital nodal period T_Ω is expressed as a function of the rotation rate of the Earth ω_e and the drift rate of RAAN $\dot{\Omega}$ due to the zonal terms:

$$L = T_\Omega (\omega_e - \dot{\Omega}) R_e. \quad (8)$$

In the LEO environments, the influence of three-body perturbation is so small that the perturbation makes a negligible contribution to the groundtrack drift associated with variations in inclination. Therefore, the drag is the main responsibility for the drift of groundtrack. The drift of groundtrack in one nodal period ΔL with respect to the reference trace can be approximately determined by

$$\Delta L = \frac{\partial L}{\partial a} \Delta a = 3\pi \sqrt{\frac{a_0}{\mu}} (\omega_e - \dot{\Omega}) R_e \Delta a, \quad (9)$$

in which a_0 is the nominal value of semimajor axis on the repeat-groundtrack condition and Δa is the deviation of semimajor axis from its nominal value. Note that the groundtrack drift at the equator ΔL describes the maximum value among the drifts at all latitudes, which is the foundation of our maintenance strategy. The mean groundtrack drift rate can be derived by dividing the nodal period from the groundtrack drift, which is shown as

$$\Delta \dot{L} = 3\pi \sqrt{\frac{a_0}{\mu}} \frac{1}{T_\Omega} (\omega_e - \dot{\Omega}) R_e \Delta a. \quad (10)$$

As stated above, the dominate perturbation causing the drift of satellite groundtrack is the atmospheric drag. Suppose the decay of semimajor axis due to the drag varies at a constant rate $\dot{a}$, the drift acceleration is thus linearized as follows:

$$\Delta \ddot{L} = \frac{3}{2} \frac{R_e}{a_0} (\omega_e - \dot{\Omega}) \frac{\Delta a}{\Delta t} = \frac{3}{2} (\omega_e - \dot{\Omega}) R_e \frac{\dot{a}}{a_0}. \quad (11)$$

By the integration of Eq. (11) over the time in two times, the mean groundtrack drift rate and the drift in one nodal period are derived as (Aorpimai and Palmer 2007)

$$\Delta \dot{L} = \frac{3}{2} (\omega_e - \dot{\Omega}) R_e \frac{\dot{a}}{a_0} t + \Delta \dot{L}_0, \quad (12)$$

and

$$\Delta L = \frac{3}{4} (\omega_e - \dot{\Omega}) R_e \frac{\dot{a}}{a_0} t^2 + \Delta \dot{L}_0 t + \Delta L_0, \quad (13)$$

where $\Delta \dot{L}_0$ and ΔL_0 , respectively, are the initial drift rate and drift in groundtrack. Without loss of generality, let it be assumed that the initial groundtrack displacement ΔL_0 is zero. In other words, the satellite is initially at the repetition condition with the groundtrack according with the reference trace. Afterwards, the trace will deviate from the reference repeat-groundtrack due to the effect of the atmospheric drag, as indicated in Eq. (13).

The initial groundtrack drift rate is actually an averaged value, which can be derived from the

groundtrack displacement in one period as

$$\Delta \dot{L}_0 = \frac{3}{2}(\omega_e - \dot{\Omega})R_e \frac{\Delta a}{a_0}, \quad (14)$$

where Δa is the deviation in semimajor axis from the reference repeat-groundtrack orbit and it also indicates the value of semimajor axis pre-offset to maintain the trace within the maximum drift threshold.

Substituting Eq. (14) into Eq. (13) to yield the variation of displacement in groundtrack, expressed in terms of time t , deviation in semimajor axis from the reference repeat-groundtrack orbit Δa and semimajor axis decay $\dot{a}$ as

$$\Delta L = \frac{3}{2a_0}(\omega_e - \dot{\Omega})R_e \left(\frac{1}{2}\dot{a}t^2 + \Delta at \right). \quad (15)$$

Therefore, with the real-time estimation of semimajor axis and its decay rate, the orbit maintenance strategy can be performed by adjusting the semimajor axis to stationkeep the actual orbit in the vicinity of repeat-groundtrack condition, whose objective is to keep the groundtrack drift not exceeding a desired threshold.

3.2 Orbit maintenance strategy

The acceptable drift threshold of groundtrack L_d can be determined from mission requirements. As illustrated in Fig. 4, the admissible region around the nominal groundtrack is defined by the eastern $(-\frac{L_d}{2})$ and western boundary $(\frac{L_d}{2})$, respectively. As already stated, the decay of the semi-major axis changes the orbital period and this causes the groundtrack an eastern drift relative to the reference track starting from the initial repeat-groundtrack orbit condition. When the eastward drift arrives at the eastern boundary, a maneuver is performed to increase the semimajor axis above the nominal value by an amount Δa . After the maneuver, as the value of the semimajor axis is greater than that of the nominal one, the trace starts to drift westward due to its longer orbital period than that of the reference orbit. As shown by Eqs. (12) and (14), the drift rate of groundtrack appears to decrease progressively as time goes on. Note that if the positive offset of semimajor axis Δa is appropriate, the drift will cease at the western boundary where the semimajor axis just equals the nominal value. At this point the groundtrack will start to drift eastward until it reaches the eastern boundary again, when a second impulsive maneuver is implemented to restore the control loop. The groundtrack drift maintained under this strategy is depicted in Fig. 4 and the corresponding semimajor axis adjustment procedure is displayed in Fig. 5.

Fig. 4 Groundtrack drift varying with time controlled by the maintenance strategy

Fig. 5 Semimajor axis adjustment of the maintenance strategy

As can be seen from Eq. (15), the drift of groundtrack relative to the reference is a quadratic form in time. Thus, there exists an instant t_s at which the drift stops ($\dot{L} = 0$). The value of t_s can be obtained from Eqs. (12) and (14) as

$$t_s = -\frac{\Delta a}{\dot{a}}. \quad (16)$$

At t_s the westward drift of groundtrack reaches the maximum value, namely the western boundary. Subsequently, the trace starts to drift in the opposite direction. The westward drift with the pre-offset of semimajor axis Δa can be given by inserting t_s into Eq. (15):

$$\Delta L_{\max} = -\frac{3}{4}(\omega_e - \dot{\Omega})R_e \frac{\Delta a^2}{a_0 \dot{a}}. \quad (17)$$

Since the maximum drift threshold, noted as L_d , has already been decided by the mission requirements in advance, the pre-offset value of semimajor axis relative to that of reference orbit for maintenance can be solved

$$\Delta a = \sqrt{-\frac{4a_0 \dot{a} L_d}{3(\omega_e - \dot{\Omega})R_e}}. \quad (18)$$

The time separation between two impulsive maneuvers by adjusting the semimajor axis can be expressed as

$$T_s = 2t_s = \sqrt{-\frac{16a_0 L_d}{3(\omega_e - \dot{\Omega})R_e \dot{a}}}. \quad (19)$$

It should be noted that the above-mentioned strategy considers an ideal scenario in which there is no error in the estimation of semimajor axis decay $\dot{a}$ and the implementation of semimajor axis adjustment Δa . However, for a high-precision Earth repeat-groundtrack orbit maintenance, the strategy needs to be modified to account for the above-mentioned errors.

In the ideal strategy, the atmospheric drag and semimajor axis decay are assumed to be constant. However, the uncertainties associated with drag perturbation affect the estimation of the semimajor axis decay rate in a real scenario. To describe the error in the estimation of $\dot{a}$ caused by these uncertainties, a relative error in the rate of semimajor axis decay is defined by $\xi = \frac{\delta \dot{a}}{\dot{a}}$, where $\delta \dot{a}$

indicates the absolute error in the semimajor axis decay rate, and $\dot{a}$ the value of decay rate obtained in the constant atmospheric drag. In addition, the orbital measurements and control accuracies are major factors that influence the accuracy of Δa . Thus, the relative error in the orbital

measurements and control is defined as $\eta = \frac{\delta \Delta a}{\Delta a}$, where $\delta \Delta a$ denotes the absolute error in the

orbital measurements and control, and Δa the value of semimajor axis adjustment in the ideal maintenance. Considering these two sources of errors, the actual value of semimajor axis decay rate would be $\dot{a}_{\text{actual}} = (1 - \xi)\dot{a}$, and the actual implementation of semimajor axis adjustment be

$\Delta a_{\text{actual}} = (1 - \eta)\Delta a$. The actual semimajor axis adjustment can be then rewritten from Eq. (18) as

$$\Delta a_{\text{actual}} = (1 - \eta) \sqrt{-\frac{4a_0(1 - \xi)\dot{a}L_d}{3(\omega_e - \dot{\Omega})R_e}} = (1 - \eta)\sqrt{1 - \xi} \sqrt{-\frac{4a_0\dot{a}L_d}{3(\omega_e - \dot{\Omega})R_e}}. \quad (20)$$

Likewise, the actual maneuvering period is derived from Eqs. (16), (19) and (20) as

$$T_{\text{actual}} = -\frac{2\Delta a_{\text{actual}}}{\dot{a}_{\text{actual}}} = \frac{1-\eta}{\sqrt{1-\xi}} \sqrt{-\frac{16a_0L_d}{3(\omega_e - \dot{\Omega})R_e\dot{a}}}. \quad (21)$$

By assuming small errors, the semimajor axis adjustment value and maneuvering period can be simplified at first order to

$$\Delta a' = (\sqrt{1-\xi} - \eta) \sqrt{-\frac{4a_0\dot{a}L_d}{3(\omega_e - \dot{\Omega})R_e}}, \quad (22)$$

and

$$T'_s = (1 + \frac{\xi}{2} - \eta) \sqrt{-\frac{16a_0L_d}{3(\omega_e - \dot{\Omega})R_e\dot{a}}}. \quad (23)$$

Therefore, the actual maneuver to adjust the semimajor axis is $2\Delta a'$, with maneuvering period of T'_s .

4 Numerical simulations

This section presents several numerical simulations to illustrate the performance of the proposed approaches for Earth repeat-groundtrack orbit design and maintenance. In Section 4.1, the effectiveness of the high-precision gravitational perturbation model is verified and the nominal semimajor axis is determined based on the nodal period via the differential correction algorithm. In Section 4.2, Monte Carlo simulation and the simulation in a high fidelity model (full potentials and atmospheric drag) are run to validate the effectiveness of the proposed orbit maintenance strategy.

4.1 Reference orbit design

The simulations presented in this section take the orbital data from four different aerospace missions (listed in Table 1) to assess the accuracy of the zonal model selected for the design of repeat-groundtrack orbits. These reference missions are IRS-P6, Landsat-8, TerraSAR-X and SPOT-7, which all fly in sun-synchronous repeat-groundtrack orbits.

Using Eqs. (6) and (7), the relation between mean nodal period and order of the zonal harmonic can be determined, as shown in Fig. 6.

Table 1 Orbital parameters of repeat-groundtrack orbits*

Orbital parameters	IRS-P6	Landsat-8	TerraSAR-X	SPOT-7
Semimajor axis a (m)	7195141.565	7077753.980	6883513.0	7073059.757
Eccentricity e (-)	0.00116877	0.00116888	0.00125	0.001106
Inclination i (°)	98.690503	98.226400	97.446	98.147
Argument of perigee ω (°)	71.987501	90.009232	90.0	85.586
Repetition pattern $n_M:n_N$ (-)	24:341	16:233	11:167	26:379

(*: Data are collected from the website of Space-Track)

Fig. 6 Determination of mean nodal period versus the order of non-spherical gravitational perturbations (J_2 - J_{15}) for Earth repeat-groundtrack orbits

Figure 6 reveals that the mean nodal period becomes stationary for a high values of the zonal harmonics, showing that considering only zonal terms up to J_4 (a typical assumption in Nadoushan and Assadian (2015)), is not enough to achieve the convergence. Convergence is reached approximately when terms up to J_8 are included, above which the nodal period exhibits small amplitudes of fluctuations.

In Fig. 7 the nominal semimajor axis values obtained by the differential correction algorithm are displayed when including zonal harmonics up to J_{15} .

Fig. 7 Determination of nominal semimajor axis versus the order of non-spherical gravitational perturbations (J_2 - J_{15}) for Earth repeat-groundtrack orbits

As for the nodal period, the nominal semimajor axis becomes stable when higher order of zonal perturbations are included. Compared with the conventional technique for repeat-groundtrack orbit based on J_2 perturbation, by using the higher order of zonal perturbation the present method allows us to improve the accuracy in the semimajor axis determination by about 5 m. This gain in accuracy is relevant if one considers that the magnitude of the corrections in the semimajor axis required for orbit maintenance is of the order of tens of meters, as it will be shown in Section 4.2.

Moreover, it is worth noting that, from the engineering implementability point of view, considering the capability of existing orbit thrusters (Gou et al. 2013) providing thrust of 5 N with pulse width of 50 ms, the adjustment of semimajor axis can be maintained at 0.5 m level (with an approximate semimajor axis of 7200 km and spacecraft mass of 1000 kg), which is compatible with the magnitude of the correction in semimajor axis required by the present semi-analytical design and maintenance strategy.

4.2 Orbit maintenance

The objective of this section is to assess the performances of the maintenance strategy described in Section 3.2. The IRS-P6 mission is used as a test case throughout this section. The IRS-P6

spacecraft (1350 kg of mass) can be modeled as a central body with cross-sectional area of about 1m^2 and two solar arrays each made by 3 solar panels $1.4\times 1.8\text{ m}^2$ of area. The nominal semimajor axis is computed via differential correction algorithm illustrated in Section 2 and it is equal to 7195109.604 m. It is considered that the maximum admissible error on groundtracks repetition is $\pm 0.25\text{ km}$, translating into a maximum drift distance $L_d = 500\text{ m}$. The MSIS model described by Hedin (Hedin 1991) is used to compute the atmospheric density. The solar flux $F_{10.7}$ is assumed to be 100. A constant value of drag coefficient $C_D = 2.5$ is chosen, which is the maximum value that can be expected from empirical tests. With these data the decay rate of semimajor axis is $\dot{a} = -0.293\text{ m/day}$. Based on Eq. (18), the maximum pre-offset value of semimajor axis relative to that of the reference orbit is $\Delta a = 5.934\text{ m}$, which is the ideal value without considering errors. A more robust method is presented in this paper, in which the errors in the orbital measurements and control as well as in the estimation of the semimajor axis decay rate are taken into account in the maintenance strategy. The absolute error in the orbital measurements and control is assumed to be 0.5 m , which is used to obtain the relative error $\eta = 0.08427$. Moreover, the relative error in semimajor axis decay rate is chosen as $\xi = 0.1$. Through Eq. (22), the modified offset of the semimajor axis is $\Delta a' = 5.129\text{ m}$, which is less than the actual value. Thus, the actual semimajor axis adjustment at each maneuver is $2\Delta a' = 10.258\text{ m}$, with a maneuvering period of $T'_s = 33.772$ days.

From these results it is apparent the difference in the nominal semimajor axis computed with the J_2 model and the J_{15} model (i.e., 4.2 m) is of the same order of the offset value ($\Delta a' = 5.129\text{ m}$) for maintenance, demonstrating that an accurate semimajor determination is of critical importance. In fact, if the nominal semi-major axis is poorly determined, the groundtrack maintenance may fail at keeping the satellite in the repeat-groundtrack condition when high accuracy is demanded.

Figures 8 and 9 show the profiles of groundtrack drift and semimajor axis in a window of half a year. As demonstrated from these two figures, the actual groundtrack drift is kept within the allowable threshold when the proposed maintenance strategy is applied. During this period, six maneuvers to adjust the semimajor axis are implemented at the eastern boundary of the admissible region. Moreover, Fig. 8 shows that the semimajor axis adjustment computed by Eq. (22) is less than the ideal value given by Eq. (18), thus reserving a certain margin to account for the effect of possible errors in the orbital measurements and control together with the error in the estimation of semimajor axis decay rate.

Fig. 8 Comparison between the actual and ideal groundtrack drift varying with time

Fig. 9 Comparison between the actual and ideal variation in semimajor axis with time

The values of parameter η and ξ in the maintenance strategy are obtained from an empirical estimation. To further validate the effectiveness of the proposed maintenance strategy, a Monte Carlo simulation is run in which 1000 scenarios with different errors $\delta\eta$ and $\delta\xi$ are analyzed. $\delta\eta$ and $\delta\xi$ are assumed to be Gaussian distributed, with statistical properties reported in Table 2. For each run the actual absolute errors are computed as $\eta = \eta_0 + \delta\eta$ and $\xi = \xi_0 + \delta\xi$.

Table 2 Statistical dispersions of initial conditions in the Monte Carlo simulation

Errors	Dispersion	
	Mean	Standard deviation($3\text{-}\sigma$)
$\delta\eta$	0	0.09
$\delta\xi$	0	0.08

The statistical distribution of the semimajor axis adjustments is illustrated in Fig. 10. It is concluded that the proposed maintenance strategy will produce an adjustment lower than the ideal value (11.867 m), which guarantees the actual drift of the groundtrack within the allowable boundary.

Fig. 10 Statistical distribution of semimajor axis adjustment for maneuver

The proposed mean-elements-based maintenance strategy in zonal perturbations and atmospheric drag is now validated in a high fidelity dynamical model that includes the zonal and cross (sectorial and tesseral) terms of non-spherical perturbations, and the atmospheric drag. The high-resolution EGM2008 global geopotential model with degree and order 200 is selected. The initial condition for simulation is the one of IRS-P6 satellite. The errors in the orbital measurements and control, and estimation of the semimajor axis decay rate are also taken into account. Finally a simulation window of 180 days is analysed. The value of semimajor axis adjustment for maneuver is obtained from the mean orbital elements in Eq. (22) and the objectives of this validation are to check whether:

- the drift of groundtrack is maintained within the boundary of 500 m;
- the maneuvering period T'_s derived based on the mean orbital elements, coincides with

the one achieved in the high fidelity model.

Fig. 11 Groundtrack drift under the maintenance strategy in the high fidelity model

Fig. 12 Relative error in groundtrack drift under the maintenance strategy in the high fidelity model

Figures 11 and 12 show the results of the maintenance strategy in the high fidelity model. In Fig. 11, we indicate the direct results of the mean-elements-based strategy with the dotted blue line (the same with the solid red lines in Fig. 8), and the evaluation of the control strategy in the high fidelity model with the solid red line. Figure. 12 displays the relative error in groundtrack drift of the strategy in the high fidelity model. As can be seen from Figs. 11 and 12, by implementing the

semimajor axis adjustment obtained from the mean-elements-based strategy in the high fidelity model, the groundtrack drift can be maintained within the specific boundary (and the relative error is controlled not exceeding 30 m), and the maneuvering period is almost equal to the one obtained with zonal perturbations and drag. However, it has to be mentioned that, as expected, due to the accumulation of errors, the actual maneuvering period would decrease in the high fidelity model compared with that in the zonal perturbations and drag. This slight difference would increase with time, especially after the fourth maneuver. Therefore, our strategy works well until the cross terms and short-periodic perturbations are negligible. On the other hand, in the long-term maintenance the maneuvers would be performed ahead of schedule due to the reduced maneuvering period in the real perturbation scenario as shown by this simulation.

5 Conclusions

The present paper investigates the design and maintenance of high-precision Earth repeat-groundtrack orbits. A conventional semi-analytical method is adopted to derive the analytical nodal period based on the mean orbital elements and considering zonal terms up to J_{15} . To obtain the nominal value of the reference repeat-groundtrack orbit, a differential correction algorithm is used in which the inputs are the desired nodal period, eccentricity, inclination and argument of perigee. As for the orbit maintenance a mean-elements-based strategy is proposed in which the errors in the orbital measurements and control as well as in the estimation of semimajor axis decay rate are taken into account. The robustness of the maintenance strategy with respect to uncertainties in the error parameters is assessed with a Monte Carlo simulation; whereas the effects of unmolded perturbations is studied with a simulation in a high fidelity model (full geopotentials and atmospheric drag). The results confirm that the mean-elements-based maintenance strategy can keep the drift of the groundtrack within allowed limits with the same semi-major axis adjustments and maneuvering periods similar to those required by in the high fidelity perturbation model.

Acknowledgments

The authors appreciate the editor and reviewers for their valuable suggestions to improve this manuscript. They are grateful to Dr. Martin Lara for providing his works and instructive overview on repeat-groundtrack orbit design, and to Laura Pirovano of the Universidad de La Rioja for her helpful discussion. The work is supported by National Natural Science Foundation of China (11172020), Beijing Natural Science Foundation (4153060), the Fundamental Research Funds for the Central Universities and the Academic Excellence Foundation of BUAA for PhD Students.

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