Acta Astronautica 141 (2017) 219–227

Contents lists available at ScienceDirect

Acta Astronautica

journal homepage: www.elsevier.com/locate/actaastro

Electrodynamical compensation of disturbing torque and attitude stabilization of a satellite in J_2 perturbed orbit

A.A. Tikhonov [a,b,*], K.A. Antipov [c], D.G. Korytnikov [a], D.Yu. Nikitin [a]

[a] Saint Petersburg State University, 7–9 Universitetskaya nab., Saint Petersburg, 199034, Russia
[b] Saint Petersburg Mining University, 2, 21st Line, Saint Petersburg, 199106, Russia
[c] JSC "GYROOPTICS", 14 Chugunnaja Str., Saint Petersburg, 194044, Russia

ARTICLE INFO

2010 MSC:
70E17
70K20
70Q05
93C15
93D21

Keywords:
Satellite
Attitude stabilization
Geomagnetic field
Perturbed orbit
Electrodynamic attitude control system

ABSTRACT

The paper deals with a satellite in a circular near-Earth orbit, perturbed due to J_2 Earth's oblateness. The satellite interacts with the geomagnetic field by the moments of Lorentz and magnetic forces. The octupole approximation of the Earth's magnetic field is used. The possibility of electrodynamical attitude control for the satellite's stabilization in the orbital frame is analyzed. Once the problem of electrodynamical compensation of disturbing torque is solved, we can obtain the control algorithms for the satellite electromagnetic parameters which allows to stabilize the satellite attitude position in the orbital frame in the presence of disturbing gravity gradient torque. The total stability of the satellite programmed motion is proved analytically and verified by computer modeling.

1. Introduction

Since the advent of the space age, the variety of electrodynamic effects caused by a satellite interaction with the Earth's magnetic field has been considered a reliable basis for attitude control systems in spacecraft missions. Thus, magnetic attitude control systems (MACS) became widely accepted as they are quite simple, highly reliable, and can be successfully applied to long-operating spacecraft [1–3]. The known shortcoming of MACS is underactuation since control torque is always orthogonal to the induction vector of the Earth's magnetic field [4–6].

Another effect, caused by the interaction of a satellite electric charge with the Earth's magnetic field gave rise to Lorentz attitude control systems (LACS). Covered by patents [7,8], in which the control Lorentz torque was introduced, LACS became known after the publication [9].

Papers [10,11], where the term "Magnetocoulombic Attitude Control" is introduced for LACS, present the general approach of control theory to the problem of 3-axis active nonlinear attitude control of a charged body with the help of Lorentz torque, and address the fault-tolerance issue of this method of control.

As shown in Ref. [12], the LACS have an underactuation feature which makes them similar to MACS. At the same time, the integrated system which use simultaneously the torques of magnetic interaction and the Lorentz one, is free of the mentioned shortcomings and is more effective than each one of MACS and LACS taken alone. In Ref. [13] this integrated system, called electrodynamic attitude control system (EDACS) was used for the problem of spacecraft attitude stabilization in the orbital frame. It was shown in Ref. [13] that the EDACS allows ensuring the satellite self-oscillation damping mechanism without exceeding the bounds of functional capabilities that the EDACS has. So, it is proved in Ref. [13] that there is no need to introduce the separate damping torque because it is enough to modify the control law in order to ensure the damping-like effect of Lorentz and magnetic interaction torques.

In later publications [10,11,14–22] the approach based on the usage of EDACS was analyzed, developed, modified and used for several important problems.

Papers [9] and [13] use the quadrupole approximation of the Earth's magnetic field based on the mathematical apparatus suggested in

* Corresponding author. Saint Petersburg State University, 7-9 Universitetskaya nab., Saint Petersburg, 199034, Russia.
E-mail addresses: a.tikhonov@spbu.ru (A.A. Tikhonov), antipov_k@rambler.ru (K.A. Antipov), dgkorytnikov@gmail.com (D.G. Korytnikov), danil_nikitin94@mail.ru (D.Yu. Nikitin).

https://doi.org/10.1016/j.actaastro.2017.10.009
Received 7 July 2017; Received in revised form 11 September 2017; Accepted 4 October 2017
Available online 7 October 2017

Ref. [23]. The development of this mathematical apparatus in Ref. [24] allowed us to construct analytically the magnetic induction $\overrightarrow{B}$ of geomagnetic field taking into account the first three multipole components (of the 2-nd, 3-rd and 4-th orders). On the basis of these results, in papers [17,18] in contrast to [9] and [13], the Earth's magnetic field was approximated by the more precise model – the octupole approximation.

The problems of satellite attitude stabilization in direct and indirect equilibrium positions in the orbital coordinate system [17,18] and in the Konig coordinate system [19,20,22] are investigated with the use of EDACS. In Refs. [13,14,17,18] mathematical justification of the method is based on consideration of differential equations in linear approximation. The existence of a domain of values for parameters of a satellite and its orbit providing total stability of equilibrium position is proved with the use of the results of numerical analysis of the roots of characteristic polynomial.

The idea of electrodynamic compensation of disturbing gravity gradient torque was realized for the first time in Ref. [16] and later it was used in Ref. [20]. In Ref. [21] the general solution of the problem of electrodynamic compensation of disturbing torque acting on a satellite whose programmed motion is an equilibrium position in the Konig frame was obtained.

Earth oblateness [25–27], atmospheric drag [28–30] and solar radiation pressure [31,32] are usually taken into account as perturbations in satellite motion [33,34]. It was shown that in many cases J_2 perturbation is the most dominant one. So, the analysis of the influence of geopotential J_2 harmonic on satellite motion and methods of elimination of this influence are important research issues. They were treated in Refs. [28–35] and papers cited therein.

In the present paper we consider a satellite in a circular near-Earth orbit, osculating due to the Earth's oblateness J_2 perturbation. The satellite stabilization by means of EDACS is investigated on the basis of mathematical model constructed with an octupole approximation of the Earth's magnetic field. In previous similar studies, including those by the authors of the paper, electrodynamic compensation of disturbing torque was elaborated and used either for monoaxial stabilization of a satellite in the orbital coordinate system [16] or for 3-axial stabilization of a satellite in the Konig coordinate system [19–22]. Electrodynamic compensation of disturbing torque in the problem of 3-axial stabilization of a satellite in the orbital frame was never used before. Moreover, the idea of electrodynamic compensation of disturbing torque is used in the present paper even for more complicated problem, because the orbital frame rotates due to the evolution of the orbital plane. No one of previous publications used electrodynamic control strategy for attitude stabilization of a satellite in J2-perturbed orbit. In the present paper the disturbing action of gravity gradient torque is taken into account. The electrodynamic compensation of disturbing gravity gradient torque is realized. A precise control law has been derived that solves the problem of satellite attitude stabilization in the orbital frame. The total stability of satellite position in the orbital frame is proved analytically and verified by computer modeling.

2. Osculating orbit and coordinate systems

We consider a satellite moving along a circular orbit which is osculating due to the Earth's oblateness perturbation. Secular perturbations of the orbit caused by the second zonal harmonic J_2 of the geopotential are taken into account. These perturbations represent deviation of the longitude of the ascending node Ω with angular velocity k_Ω and deviation of the argument of perigee ω_π with angular velocity k_ω. The orbit form (determined by the radius R) is invariable, and the orbit inclination i to the equator plane is constant [36]. The attitude motion of the satellite is studied.

As an inertial coordinate system in which the satellite motion is studied we take the coordinate system $O_E X_* Y_* Z_*$ (unit vectors $\overrightarrow{i}_*$, $\overrightarrow{j}_*$, $\overrightarrow{k}_*$) with origin at the Earth's center. The $O_E Z_*$ axis of this system is

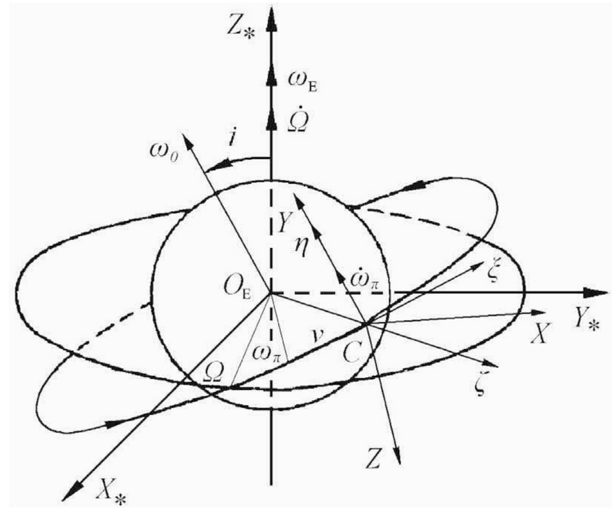

Fig. 1. Coordinate systems.

directed along the axis of the Earth's diurnal rotation (with angular velocity $\overrightarrow{\omega}_E = \omega_E \overrightarrow{k}_*$). The axes $O_E X_*$ and $O_E Y_*$ lie in the equatorial plane.[1]

Let $CXYZ$ (with unit vectors $\overrightarrow{\sigma}_1$, $\overrightarrow{\sigma}_2$, $\overrightarrow{\sigma}_3$) be the perigee coordinate system whose axis CX is parallel to the tangent to the orbit in the direction of motion of the satellite's center of mass at its perigee. The CY axis is directed along the normal to the orbit plane, while the CZ axis is parallel to the radius vector of the orbit perigee (Fig. 1). Due to the regression of the orbit the perigee coordinate system rotates in the inertial coordinate system with the angular velocity

$$\overrightarrow{k} = \overrightarrow{k}_\Omega + \overrightarrow{k}_\omega = \dot{\Omega}\overrightarrow{k}_* + \dot{\omega}_\pi \overrightarrow{\sigma}_2, \quad \text{where}$$

$$k_\Omega = \dot{\Omega} = -\omega_0 \tilde{\varepsilon}(R_E/R)^2 \cos i, \quad k_\omega = \dot{\omega}_\pi$$
$$= \omega_0 \tilde{\varepsilon}(R_E/R)^2 \left(5\cos^2 i - 1\right)/2,$$

$\omega_0 = \sqrt{\mu/R^3}$ is the orbital angular velocity of the satellite, μ is the Earth's gravitational constant, $R_E = 6.378136 \cdot 10^6$ m is the equatorial radius of the Earth, and $\tilde{\varepsilon} \approx 0.0016239$ is the dimensionless constant determined by the value of the Earth's oblateness [36].

Also we introduce the orbital coordinate system $C\xi\eta\zeta$ (unit vectors $\overrightarrow{\xi}_0$, $\overrightarrow{\eta}_0$, $\overrightarrow{\zeta}_0$) with the axis $C\xi$ directed along the positive transversal to the orbit. Its $C\eta$ axis is oriented along the normal to the orbit plane, and the $C\zeta$ axis is directed along the radius vector $\overrightarrow{R} = \overrightarrow{O_E C} = R\zeta_0$.

The system of principal central axes of inertia $Cxyz$ (unit vectors $\overrightarrow{i}$, $\overrightarrow{j}$, $\overrightarrow{k}$) is fixed to the satellite itself.

Mutual orientation of the perigee coordinate system and the orbital coordinate system is determined by the matrix of direction cosines $\mathbb{U}$.

$$\mathbb{U} = \begin{pmatrix} \cos(\omega_0 t) & 0 & \sin(\omega_0 t) \\ 0 & 1 & 0 \\ -\sin(\omega_0 t) & 0 & \cos(\omega_0 t) \end{pmatrix},$$

so that the following equalities take place:

$$\overrightarrow{\sigma}_i = U_{i1}\overrightarrow{\xi}_0 + U_{i2}\overrightarrow{\eta}_0 + U_{i3}\overrightarrow{\zeta}_0, \quad i = 1, 2, 3.$$

Mutual orientation of the perigee coordinate system and the inertial coordinate system ($O_E X_* Y_* Z_*$) is determined by the matrix of direction

[1] In this paper, direct Cartesian rectangular coordinate systems are used.