Aerospace Science and Technology 33 (2014) 92–99

Contents lists available at ScienceDirect

Aerospace Science and Technology

www.elsevier.com/locate/aescte

On satellite electrodynamic attitude stabilization

K.A. Antipov [1], A.A. Tikhonov *,[2]

St. Petersburg State University, Faculty of Mathematics and Mechanics, Stary Peterhof, Universitetsky pr., 28, St. Petersburg, 198504, Russia

ARTICLE INFO

Article history:
Received 31 March 2013
Received in revised form 7 December 2013
Accepted 5 January 2014
Available online 10 January 2014

Keywords:
Satellite
Attitude stabilization
Geomagnetic field

ABSTRACT

The paper deals with a satellite in a circular near-Earth orbit. The satellite interacts with the geomagnetic field by the Lorentz and magnetic torques. The gravitational disturbing torque acting on the satellite attitude dynamics is taken into account as the largest disturbing torque. The octupole approximation of the Earth's magnetic field is used. Satellite electromagnetic parameters, namely the electrostatic charge moment of the first order and the intrinsic magnetic moment are the controlled quasiperiodic functions. Control algorithms for the satellite electromagnetic parameters, which allow the satellite attitude position to be stabilized in the orbital frame were obtained. The cases of direct and indirect equilibrium positions in the orbital frame are investigated. The total stability of the satellite stabilized orientation is proved both analytically and by PC computations.

1. Introduction

Torques from the satellite electrodynamic interaction with the geomagnetic field considerably influence satellite attitude dynamics and can be used for designing control systems of satellite attitude orientation.

It is well known that magnetic control systems can be successfully applied to long-operating spacecraft due to the fact that these control systems are quite simple, highly reliable, and do not require working body consumption [4,14,18,20]. At the same time, magnetic control systems have a functional feature restricting their capabilities on directing the vector of control torque.

The method for attitude stabilization of the satellite moving in the Keplerian circular orbit in the geomagnetic field based on applying only the electrodynamic effect of the Lorentz forces acting upon the charged part of the satellite volume was covered by patents [22,27] and published for the first time in [25]. It was proved that by the controlled change of the radius-vector $\vec{\rho}_0$ of the center of satellite charge with respect to the center of mass of the satellite, it is possible to create the controlled Lorentz torque and use it as a restoring torque in the neighborhood of the satellite direct equilibrium position in the orbital coordinate system. Application of this torque does not require working substance consumption by an actuator or moving any massive bodies, and is distinguished by the simplicity of the control law, reliability, and economy. A certain variant of the control law for the radius-vector $\vec{\rho}_0$ was proposed; it realizes this method for orbits with small inclinations i, but degenerate for mean and large values of i.

It was demonstrated in [5] that the indicated method can be generalized onto the case of arbitrary satellite position in the orbital coordinate system and the new control law suitable for orbits with any inclinations was proposed. It was noted that the application of only Lorentz torque is connected with the presence of a constraint on the direction of vector of control torque similar to the constraint on the direction of magnetic torque mentioned above. It was demonstrated that the stated shortcomings of both control systems disappear when a unified electrodynamic control system for satellite attitude orientation using both restoring torques simultaneously, i.e., of Lorentz and magnetic torques, is created. In the papers [25] and [5] the quadrupole approximation of the Earth's magnetic field was used based on the mathematical apparatus, suggested in [21]. The development of this mathematical apparatus according to [28] allowed us to construct analytically the magnetic induction $\vec{B}$ of geomagnetic field taking into account the first three multipole components (of the 2-nd, 3-rd and 4-th orders). With use of these results in the paper [6] in contrast to [25] and [5], the Earth's magnetic field was approximated by the more precise model — the octupole approximation. The total stability of the satellite's direct attitude position in the orbital frame was proved in [6] under condition that the satellite is equipped with a damper mechanism generating the dissipative torque of the model type.

In the paper [5] it was revealed that the electrodynamic attitude stabilization system allows ensuring the satellite self-oscillation damping mechanism without exceeding the bounds of functional capabilities that the electrodynamic control system

* Corresponding author at: Department of Theoretical and Applied Mechanics, Faculty of Mathematics and Mechanics, St. Petersburg State University, Stary Peterhof, Universitetsky pr., 28, St. Petersburg, 198504, Russia.

E-mail addresses: antipov_k@rambler.ru (K.A. Antipov), aatikhonov@rambler.ru (A.A. Tikhonov).

[1] Ph.D. (in Physics and Mathematics), Scientific researcher, St. Petersburg State University.

[2] Doctor of Science (in Physics and Mathematics), Professor at the Department of Theoretical and Applied Mechanics, St. Petersburg State University.

1270-9638/$ – see front matter
http://dx.doi.org/10.1016/j.ast.2014.01.004

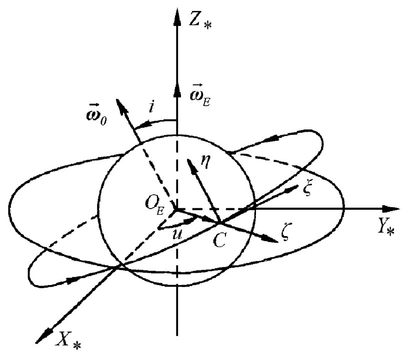

Fig. 1. Coordinate systems.

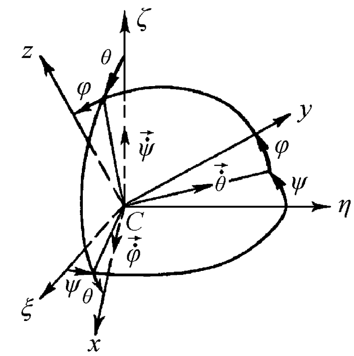

Fig. 2. "Airborne" angles.

has. So in the present paper the satellite electrodynamic attitude stabilization is investigated on the basis of mathematical model constructed with an octupole approximation of the Earth's magnetic field with the help of results, obtained in [5]. On the basis of the mentioned mathematical model we prove that there is no need to introduce the separate damping torque (as in [6]) and we modify the control law in order to ensure the damping-like effect of Lorentz and magnetic torques. The problem of satellite attitude stabilization is solved in the present paper not only for the case of direct equilibrium position (as in [5] and [6]), but also for indirect equilibrium position, corresponding to arbitrary angle $\psi_0 \neq 0$. The disturbing action of gravitational torque is taken into account. The total stability of satellite positions in the orbital frame is proved analytically and confirmed by PC computations.

2. The preset motion

A satellite whose center of mass moves in the Newtonian central Earth's gravitational field in the Keplerian circular orbit of the radius R, is considered. It is assumed that the satellite has the electrostatic charge Q distributed within some volume V with the density σ: $Q = \int_V \sigma\, dV$ and possesses the intrinsic magnetic moment $\vec{I}$. It is assumed here that the influence of the charge and the intrinsic magnetic moment on the satellite's orbit is negligibly small, i.e. the restricted satellite problem is under consideration. Thus we don't take into consideration some interesting effects that rise from the Lorentz force and which can be exploited for example in realization of the so-called Lorentz augmented orbits near the Earth [23] and Jupiter [11] and in many other potent applications for the so-called Lorentz spacecraft [12,15].

We study the satellite attitude motion with respect to the orbital coordinate system[3] $C\xi\eta\zeta$ (Fig. 1) with the origin at the satellite mass center whose axis $C\xi(\vec{\xi}_0)$ is directed along the tangent to the orbit towards the motion, the axis $C\eta(\vec{\eta}_0)$ is directed along the normal line towards orbit plane, the axis $C\zeta(\vec{\zeta}_0)$ is directed along the radius-vector $\vec{R} = \overrightarrow{O_E C} = R\vec{\zeta}_0$ of the satellite mass center with respect to the center of the Earth O_E.

The investigation is carried out taking into consideration the rotation of the orbital coordinate system with respect to the inertial coordinate system with angular velocity $\vec{\omega}_0$. As an inertial coordinate system, we consider the system $O_E X_* Y_* Z_*$, whose axis $O_E Z_*(\vec{k}_*)$ is directed along the axis of the Earth's self-rotation, the axis $O_E X_*(\vec{i}_*)$ is directed to the ascending node of orbit, and the plane $(X_* Y_*)$ coincides with the equatorial plane. Also, a system $Cxyz$ (basis vectors $\vec{i}, \vec{j}, \vec{k}$) of satellite principal central axes of inertia rigidly bound with the satellite is used. The orientation of the orbital coordinate system with respect to the system $O_E X_* Y_* Z_*$ is defined on the basis of equalities $\vec{i}_* = -\sin u\vec{\xi}_0 + \cos u\vec{\zeta}_0$, $\vec{j}_* = \cos i \cos u\vec{\xi}_0 - \sin i\vec{\eta}_0 + \cos i \sin u\vec{\zeta}_0$, $\vec{k}_* = \sin i \cos u\vec{\xi}_0 + \cos i\vec{\eta}_0 + \sin i \sin u\vec{\zeta}_0$, where $i = \widehat{(\vec{k}_*, \vec{\eta}_0)}$ is an orbit inclination; $u = \widehat{(\vec{i}_*, \vec{\zeta}_0)}$ is an argument of latitude and $u = \omega_0 t$. The orientation of the axes $Cxyz$ with respect to the axes $C\xi\eta\zeta$ is defined by a matrix $\mathbf{A}$ of direction cosines α_i, β_i, γ_i $(i = 1, 2, 3)$ so that there exist the equalities

$$\vec{\xi}_0 = \alpha_1\vec{i} + \alpha_2\vec{j} + \alpha_3\vec{k}, \qquad \vec{\eta}_0 = \beta_1\vec{i} + \beta_2\vec{j} + \beta_3\vec{k},$$
$$\vec{\zeta}_0 = \gamma_1\vec{i} + \gamma_2\vec{j} + \gamma_3\vec{k}.$$

If we determine the satellite orientation in the system $C\xi\eta\zeta$ by "airborne" angles φ, θ, ψ (Fig. 2), then the elements of the matrix $\mathbf{A}$ will have the form

$$\alpha_1 = \cos\psi\cos\theta, \qquad \alpha_2 = -\cos\varphi\sin\psi + \sin\varphi\cos\psi\sin\theta,$$
$$\alpha_3 = \sin\varphi\sin\psi + \cos\varphi\cos\psi\sin\theta, \qquad \beta_1 = \sin\psi\cos\theta,$$
$$\beta_2 = \cos\varphi\cos\psi + \sin\varphi\sin\psi\sin\theta,$$
$$\beta_3 = \cos\varphi\sin\psi\sin\theta - \sin\varphi\cos\psi,$$
$$\gamma_1 = -\sin\theta, \qquad \gamma_2 = \sin\varphi\cos\theta, \qquad \gamma_3 = \cos\varphi\cos\theta. \tag{1}$$

The orbital frame is considered to be the base coordinate system and the preset satellite orientation in this system is defined by matrix $\mathbf{A}_0$ of direction cosines. Let $\vec{\omega}' = p\vec{i} + q\vec{j} + r\vec{k}$ be the satellite angular velocity with respect to the orbital frame. The satellite position in which

$$\mathbf{A} = \mathbf{A}_0, \qquad \vec{\omega}' = 0, \tag{2}$$

will be referred to as the preset satellite motion. It is needed to construct the control torques, which ensure the preset satellite motion.

3. The control torques

The satellite motion through the Earth's magnetic field with the magnetic induction $\vec{B}$ give rise to the Lorentz and magnetic interaction torques, which have the following forms [5]:

$$\vec{M}_L = \vec{P} \times \vec{T}, \qquad \vec{M}_M = \vec{I} \times \mathbf{A}^\top \vec{B}. \tag{3}$$

In these formulae $\vec{P} = Q\vec{\rho}_0$, $\vec{\rho}_0 = x_0\vec{i} + y_0\vec{j} + z_0\vec{k} = Q^{-1}\int_V \sigma\vec{\rho}\, dV$ is the radius vector of the satellite charge center with respect to the satellite's mass center, $\vec{\rho}$ is the radius vector of the satellite's element dV with respect to its mass center, $\vec{T} = \mathbf{A}^\top(\vec{v}_C \times \vec{B})$, $\vec{v}_C = \dot{\vec{R}} - \vec{\omega}_E \times \vec{R} = R(\omega_0 - \omega_E \cos i)\vec{\xi}_0 + R\omega_E \sin i \cos u\vec{\eta}_0$ is the velocity of the satellite's mass center with respect to the geomagnetic field, and $\vec{\omega}_E = \omega_E\vec{k}_*$ is the angular velocity of the Earth's daily rotation.

The Earth's magnetic field can be considered in this case to be uniform throughout the satellite volume. Therefore, the value of $\vec{B}$ in (3) coincides with the value of $\vec{B}$ at the satellite mass center.

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