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Table of Contents

5

Development of Algorithms for the Magnetic Attitude Control of Small Spacecraft

Main problems of development of magnetic systems of satellite attitude control i.e. complexity of achievement of small spacecraft three-axis orientation; variability of magnetic field creating difficulties in development of attitude control laws with constant feedback coefficients; residual magnetic torque significantly influencing to the orientation of small satellites are considered.

5.1. INTRODUCTION

Currently small satellites (up to 100 kg) are widely used in space branch that gives the opportunity of solving various scientific and technological tasks with the lowest expenses as well as small satellites have relatively low cost and short terms of development. Experience of other countries having the program of small satellite development shows, that small satellites can fully replace large satellites when solving the tasks of detailed cartographic survey of the Earth, can solve the tasks of fire detection, carry out the survey of disaster zones, conduct the ecological monitoring, weather observations, also such satellites are designed to processing of new technologies and conduction of experiments in space.

However, in view of small mass and dimensions such satellites in a large extent are subjected to external influences provided by the Earth gravitational field, Earth atmosphere, solar pressure. Because of limitation of sizes and energetic resources saving it is practically impossible to use mechanical actuators (such as reaction wheels) on many small satellites as a part of attitude control system that are necessary for the provision of required angular position of the satellite and compensation of external influences. In connection with this fact magnetorquers generating controlling torque at the expense of interaction of own generating magnetic torque with the Earth magnetic field got a lot of popularity for the development of attitude control systems of low Earth orbit small satellites. Magnetorquers are more reliable in comparison with mechanical actuators due to the absence of mechanical elements and can be widely used for the achievement of required orientation of the satellite as well as for unloading angular momentum.

At this moment several schools in Russia and countries for a far abroad conduct their researches in the field of development of satellite attitude control system on the basis of magnetorquers that are always called as magnetic attitude systems and used their results on already launched satellites [1]-[2].

Small satellite TNS developed by Institute of applied mathematics named after M. Keldysh has successfully launched from the International space station board in 2005 year [3]. The satellite was equipped with passive magnetic attitude system consisting of strong permanent magnet placed along satellite symmetry axis and the set of hysteresis rods.

One of the missions of this satellite is the conduction of experiments on verification of satellite control technologies with the help of magnetorquers. Specialists of Institute of applied mathematics named after M. Keldysh have widely studied the task of the development of magnetic attitude systems including the attitude control laws on the basis of modern techniques.

Small satellites based on the CubeSat standard that are equipped with magnetic attitude system and that were active on the orbit during several years were developed in the laboratory of intellectual space systems of the Tokyo University [4]-[5].

Classical as well as modern methods of control system synthesis are used for the development of control laws of change of magnetic torque of these satellites magnetorquers.

The main factor affecting to the control development process for magnetoquers is their type, satellite parameters and orbit character.

By the results of applicability analysis of various control synthesis methods for the magnetic attitude systems it was determined that when developing linear control laws for magnetorquers the following peculiarities should be considered [6]:

- standard methods of linear control systems synthesis can be used for control synthesis used on short terms due to the non-linearity of mathematical model of satellite motion controlled by magnetorquers;
- developing control law should compensate all influencing to the satellite external disturbances;
- developing control law should provide the robustness in the conditions of presence of uncertainties provided by change of moments of inertia and the accuracy of mathematical model of the Earth magnetic field;
- the great difficulty is in the realization of control laws with variable coefficients of feedback onboard of the satellite;
- for the production of magnetic torque of magnetorquers electrical currents only in definite range can be used as a result of which there can arise a problem of magnetorquers saturation, especially when working with large deviations from required orientation.

Synthesis methods of nonlinear control laws in comparison with linear ones allow to consider lots of given above peculiarities and develop more flexible control systems.

In this work as a result of research of various linear and nonlinear methods for the control system synthesis there were chosen two techniques that can be adapted for the development of magnetic attitude systems of small satellites: synthesis technique of linear control law on the basis of PD-regulator and synthesis technique of nonlinear control law on the basis of usage of control theory with sliding mode. Modification of these control laws taking into account peculiarities of the Earth magnetic field and magnetorquers allowed to achieve a full controllability of the small satellite.

In many tasks performed by small satellites it is required a provision of three-axis orientation, accuracy of which depends on its mission. Magnetorquers are widely used for the attitude control of the satellites and their use is especially beneficial when there are serious constraints on mass, cost, energy consumption of satellite. In connection with this fact small satellites are often equipped with magnetic attitude control. However, accuracy of orientation and maneuverability of satellites with magnetic attitude systems is relatively smaller than control systems with mechanical actuators. Control quality in this case can be improved at the expense of enhancement of mathematical apparatus of satellites control. In connection with this the task of development of control for small satellite with magnetic attitude system is relevant and presents the scientific interest during last several decades.

The main area of research concentrated on development of control law for the provision of three-axis attitude of small satellite with required accuracy using magnetorquers and on assessment of residual magnetic torque that significantly influences to the dynamics of small satellites and work of control algorithms.

In this work there are considered main problems of development of magnetic systems of satellite attitude control: complexity of achievement of satellite three-axis orientation in case of near zero angle between directions of magnetic induction vector and magnetic torque vector of the satellite; variability of magnetic field creating difficulties in development of attitude control laws with constant feedback coefficients; residual magnetic torque significantly influencing to the orientation of small satellites.

5.2. ALGORITHMS FOR THE MAGNETIC ATTITUDE CONTROL OF SMALL SPACECRAFT

For describing the motion of a spacecraft the following coordinate systems are used:

- $OE_xE_yE_z$ is an inertial coordinate system (ICS), the origin of the system coincides with the Earth' center of mass, the z axis coincides with the axis of rotation of the Earth and is directed to the north pole. The x axis connects the Earth's center of mass and the point of the vernal equinox, the y axis complements the system;

- $CS_xS_yS_z$ is an associated coordinate system (ACS), the origin of the system coincides with the center of mass of the spacecraft, the axes coincide with the main central axes of inertia of the spacecraft;
- $CP_xP_yP_z$ is a coordinate system of guidance (CGS), the origin of the system coincides with the center of mass of the spacecraft, one of the main axes of inertia of which is directed to a point on the surface of the Earth.

5.3. THE ALGORITHM OF ATTITUDE CONTROL ON THE BASE OF PD-REGULATOR

Dynamics and kinematics of rotational motion of small spacecraft is described by equations (5.1), (5.2):

$$\dot{\vec{\omega}}_{bi}^{b} = I^{-1}\left(\vec{\omega}_{bi}^{b} \times (I\vec{\omega}_{bi}^{b}) + \vec{M}^{b}{}_{grav} + \vec{M}^{b}{}_{a} + \vec{M}^{b}{}_{res}\right) \quad (5.1)$$

$$\vec{\omega}_{\mathrm{bi}}^{\mathrm{b}} = 2\vec{Q}_{bi}{}^{*} \otimes \dot{\vec{Q}}_{bi} \quad (5.2)$$

where

$\vec{\omega}_{bi}^{b}$ is an angular velocity vector of the small spacecraft,

I is an inertia tensor of a small spacecraft,

$\vec{M}^{b}{}_{grav}$ is a gravitational torque vector of the small spacecraft,

$\vec{M}^{b}{}_{a}$ is a control torque,

$\vec{M}^{b}{}_{res}$ is a residual magnetic torque,

$\vec{Q}_{bi}$ is a quaternion that specifies the angular position of the small spacecraft in the inertial coordinate system,

$\overrightarrow{Q_{bi}}^{*}$ is a quaternion, inverse to $\vec{Q}_{bi}$,

$\otimes$ is a multiplication of quaternions.

The torques in the right side of equation (5.1) has the form:

$$\vec{M}^{b}{}_{a} = \vec{m} \times \vec{B} \tag{5.3}$$

$$\vec{M}^{b}{}_{res} = \vec{m}_{res} \times \vec{B} \tag{5.4}$$

where

$\vec{m}$ is a magnetic moment of electromagnetic actuator,

$\vec{m}_{res}$ is a residual magnetic moment of spacecraft and $\vec{B}$ is the geomagnetic induction vector.

The geomagnetic induction vector according to [7] is determined by the formula:

$$\vec{B} = \mu_0 \vec{H} \tag{5.5}$$

where

$\vec{H}$ is the vector of the geomagnetic field strength,

$\mu_0 = 4\pi \cdot 10^{-7}\, кг \cdot м \cdot A^{-2} \cdot c^{-2}$ is the magnetic constant.

The vector of the geomagnetic field strength is determined through the negative gradient of the potential function [8]:

$$\vec{H} = -\nabla V \tag{5.6}$$

The formulation of potential function in (6) depends on the model of geomagnetic field. Depending on the requirements for the control system, different models of the Earth's geomagnetic field can be adopted for the synthesis of the magnetic orientation system. There are three most commonly used models of the Earth's geomagnetic field: the International Geomagnetic Reference Field (IGRF) [9] developed by the International Association of Geomagnetism and Aeronomy (IAGA); The world model of the Earth's geomagnetic field is WMM (World Magnetic Model) [10] developed by the US National Agency for the Processing of Spatial Geodata, the National Center for Geophysical Data of the USA, and the British Geological Prospecting Society; The global geomagnetic field model - BGGM (The BGS Global Geomagnetic Model) [11], developed by the British Geological Prospecting Society. The IGRF model is updated every five years and it is recommended to use for orbits of spacecraft with a height of up to an altitude of 600 kilometers [8]. In this work, the IGRF model is used, since it most fully describes the geomagnetic field.

The expansion in the row of the field potential which is used in this model was proposed by K.F. Gauss and has the following form [8]:

$$V(r,\theta,\phi) = R\sum_{n=1}^{k}\left(\frac{R}{r}\right)^{n+1}\sum_{m=0}^{n}\left(g_n^m \cos m\phi + h_n^m \sin m\phi\right)P_n^m(\theta) \tag{5.7}$$

where

r,θ,ϕ are the spherical coordinates of spacecraft in ICS,

$R = 6371.2\cdot 10^3$ m is the average radius of the Earth,

g_n^m and h_n^m are Schmidt coefficients [9],

P_n^m are Schmidt adjoint Legendre functions.

The values of the coefficients are determined empirically by means of measurements of the magnetic field produced by the satellites.

The components of the geomagnetic field strength vector in the spherical coordinate system have the form [8]:

$$\begin{aligned}
H_r &= -\frac{\partial V}{\partial r} = \sum_{n=1}^{k}\left(\frac{R}{r}\right)^{n+2}(n+1)\sum_{m=0}^{n}\left[g_n^m \cos(m\phi) + h_n^m \sin(m\phi)\right]P_n^m(\theta),\\
H_\theta &= -\frac{1}{r}\frac{\partial V}{\partial \theta} = -\sum_{n=1}^{k}\left(\frac{R}{r}\right)^{n+2}\sum_{m=0}^{n}\left[g_n^m \cos(m\phi) + h_n^m \sin(m\phi)\right]\frac{\partial P_n^m(\theta)}{\partial \theta},\\
H_\phi &= -\frac{1}{r\sin\theta}\frac{\partial V}{\partial \phi}\\
&= -\frac{1}{\sin\theta}\sum_{n=1}^{k}\left(\frac{R}{r}\right)^{n+2}\sum_{m=0}^{n}\left[-m g_n^m \sin(m\phi) + m h_n^m \cos(m\phi)\right]P_n^m(\theta),
\end{aligned} \tag{5.8}$$

In the orbital coordinate system the geomagnetic induction vector can be written in the form:

$$\vec{B}^o = R_b^o \vec{B}^b. \tag{5.9}$$

In accordance with the principle of constructing a PD controller, we will consider the function of changing of the control torque as [12]:

$$\vec{M}^b{}_a = [u_x, u_y, u_z], \tag{5.10}$$

$$\vec{u} = K_p \vec{e} + K_d \frac{d\vec{e}}{dt} \tag{5.11}$$

where

K_p, K_d are unknown proportional coefficient and differentiation constant, respectively,

$e(t)$ is a misalignment of the angular position of small spacecraft.

Angular position of the small spacecraft is given in the form of a quaternion $\overrightarrow{Q_{bi}}$, then the misalignment with respect to the angular position of the small spacecraft is given in the form $\Delta\vec{Q}$ as the difference between the current angular position and the required one and the control signal will take the form:

$$\vec{u} = -\begin{pmatrix} K_p^1 & 0 & 0 & K_d^1 & 0 & 0 \\ 0 & K_p^{21} & 0 & 0 & K_d^2 & 0 \\ 0 & 0 & K_p^3 & 0 & 0 & K_d^3 \end{pmatrix} \begin{bmatrix} \Delta q_1 \\ \Delta q_2 \\ \Delta q_3 \\ \Delta\omega_1 \\ \Delta\omega_2 \\ \Delta\omega_3 \end{bmatrix} = \begin{bmatrix} -K_p^1 \Delta q_1 - K_d^1 \Delta\omega_1 \\ -K_p^2 \Delta q_2 - K_d^2 \Delta\omega_2 \\ -K_p^3 \Delta q_3 - K_d^3 \Delta\omega_3 \end{bmatrix} \tag{5.12}$$

where

$\Delta q_1, \Delta q_2, \Delta q_3$ are components of the vector part of the quaternion which describe deviation small spacecraft current orientation from the desired,

$\Delta\vec{Q} = [q_0, \Delta\vec{q}]$; $\Delta\omega_1, \Delta\omega_2, \Delta\omega_3$ are components of the vector of the deviation of the current angular velocity of the small spacecraft from the required.

The required orientation of the small spacecraft is given as a quaternion corresponding to the angular position of the small spacecraft when it is directed to the Earth. We call this quaternion as a required quaternion $\overrightarrow{\mathrm{Q}_{\mathrm{r}}}$.

Misalignment $\Delta\vec{\mathrm{Q}}$ between the small spacecraft current angular position $\overrightarrow{\mathrm{Q}_{\mathrm{bi}}}$ and the required angular position $\overrightarrow{\mathrm{Q}_{\mathrm{r}}}$ can be determined using the quaternion multiplication:

$$\Delta\vec{\mathrm{Q}} = \overrightarrow{Q_r}^{-1} \otimes \overrightarrow{Q_{bi}} . \tag{5.13}$$

The required quaternion characterizing the orientation of the longitudinal axis of the survey equipment on the survey object is obtained as follows. We define a unit vector specifying the axis of rotation of the small spacecraft on the survey object as:

$$\vec{\eta} = \frac{\overrightarrow{-r_{MKA}} \times \vec{R}}{\left|\overrightarrow{-r_{MKA}} \times \vec{R}\right|}, \tag{5.14}$$

where

$\vec{R} = \vec{r} - \overrightarrow{r_{MKA}}$,

$\vec{r}$ is a radius - the vector of the current point of the survey route in the ICS.

We define the rotation angle around the axis $\vec{\eta}$ by the formula:

$$\alpha = \arccos\left(\frac{\left(\overrightarrow{-r_{MKA}}, \vec{R}\right)}{\left|\overrightarrow{r_{MKA}}\right|\left|\vec{R}\right|}\right). \tag{5.15}$$

Then $\overrightarrow{Q_r}$ takes the form:

$$\overrightarrow{Q_r} = \left[\cos\frac{\alpha}{2}, \vec{\eta}\sin\frac{\alpha}{2}\right]. \tag{5.16}$$

It can be seen from formula (5.12) that to include the PD controller in the control system, it is necessary to determine the unknown coefficients K_p, K_d [13]. An approach based on the optimal arrangement of the roots of the characteristic equation of a closed-loop control system is used in this work to determine the coefficients.

Fig. 5.1 shows the results of modeling the orbital orientation of the small spacecraft using a PD controller with tuning coefficients based on the optimal arrangement of the roots of the characteristic equation of a closed-loop control system.

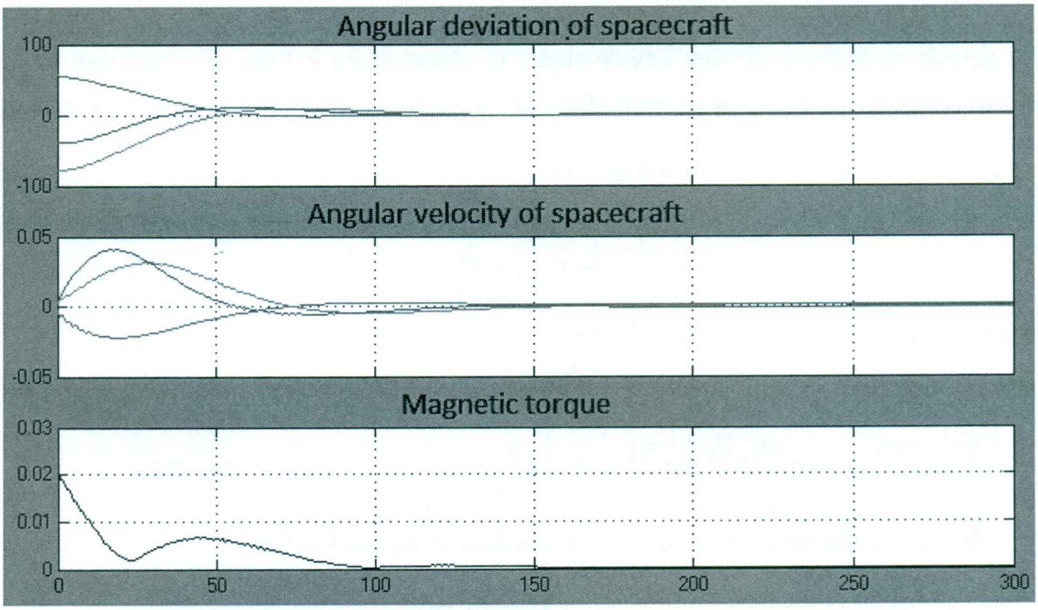

Fig. 5.1. Results of modeling the orbital orientation of the small spacecraft using a PD controller for determining the magnetic moment when adjusting its coefficients by means of the optimal arrangement of the roots of the characteristic equation of a closed control system

Source: own elaboration

5.4. THE ALGORITHM OF ATTITUDE CONTROL ON THE BASE OF THEORY OF SLIDING MODE CONTROL

Another important approach in controlling the orientation of a spacecraft with a magnetic orientation system is based on nonlinear methods. We will consider a strategy for controlling the orientation of a spacecraft on the basis of a theory of sliding mode control [14].

We consider the problem of turning the small spacecraft from an arbitrary angular position to the required [14]. Assume that the desired angular position $\overrightarrow{\mathrm{Q}_{\mathrm{r}}}$, required angular velocities $\overrightarrow{\omega_{\mathrm{r}}}$ and $\overrightarrow{\dot{\omega}_{\mathrm{r}}}$ are given. Then the dynamics and kinematics equations of the small spacecraft with respect to the deviation of the current orientation of the small spacecraft from the required orientation are given in the form:

$$I\overrightarrow{\dot{\omega}_{bi}^{b}} + \overrightarrow{\omega_{bi}^{b}} \times I\overrightarrow{\omega_{bi}^{b}} = \overrightarrow{M_{a}} + \overrightarrow{M_{\mathrm{dis}}}, \tag{5.17}$$

$$\overrightarrow{\Delta\dot{Q}} = \frac{1}{2}\Omega(\overrightarrow{\Delta\mathrm{Q}})\overrightarrow{\Delta\omega}, \tag{5.18}$$

where

I is a diagonal (3x3) matrix of the small spacecraft inertia tensor,

$\overrightarrow{\omega_{bi}^{b}}$ is an absolute angular velocity vector of the small spacecraft in the projections on the axis of the associated coordinate system $CS_xS_yS_z$,

$\overrightarrow{M_{a}}$ is a vector of the control torque in the projections on the axis of the associated coordinate system,

$\overrightarrow{M_{\mathrm{dis}}}$ is a vector of the disturbing torque in the projections on the axis of the associated coordinate system,

$\overrightarrow{\Delta Q}$ is a quaternion characterizing the deviation of the current angular position of the small spacecraft from the required,

$\overrightarrow{\Delta\omega}$ is a deviation of the angular velocity of the small spacecraft from the required,

$\overrightarrow{\Delta\omega} = \overrightarrow{\omega_{bi}^{b}} - \overrightarrow{\omega_{\mathrm{r}}}$.

$$\Omega(\overrightarrow{\Delta \mathrm{Q}}) = \begin{bmatrix} \Delta q_0 E + [\Delta q\times] \\ -\Delta q^{\mathrm{T}} \end{bmatrix} \quad (5.19)$$

$$[\Delta q\times] = \begin{bmatrix} 0 & -\Delta q_3 & \Delta q_2 \\ \Delta q_3 & 0 & -\Delta q_1 \\ -\Delta q_2 & \Delta q_1 & 0 \end{bmatrix} . \quad (5.20)$$

The synthesis of control algorithm based on the control theory with sliding mode consists of two steps [14]:

1. The construction of a surface $\vec{\mathrm{S}}(\overrightarrow{\Delta\omega}, \overrightarrow{\Delta Q}, t) = 0$ along which a point characterizing the state of the control system should move. If this point is on the surface $\vec{\mathrm{S}}(\overrightarrow{\Delta\omega}, \overrightarrow{\Delta Q}, t) = 0$ when the small spacecraft moves, the motion of the small spacecraft is considered asymptotically stable.
2. The construction of a control providing motion along a given surface.

Usually the surface is given in the form [14]:

$$\vec{\mathrm{S}} = \overrightarrow{\Delta\omega} + K_q \overrightarrow{\Delta q} \quad (5.21)$$

where

K_q is a constant, $K_q > 0$.

It must be taken into account that two types of control actions must be obtained when developing a control based on the control system with a sliding mode. The first of them $\overrightarrow{u'_{eq}}$ must provide movement on a given surface, as soon as the point characterizing the state of the control system reaches the specified surface. The second control action $\overrightarrow{u'_{\kappa}}$ must ensure the achievement of a point characterizing the state of the control system, a given surface:

$$\vec{\mathrm{u}} = \overrightarrow{u'_{eq}} + \overrightarrow{\mathrm{u}_{\mathrm{k}}} \tag{5.22}$$

The motion of the system along the given surface is characterized by the following equations:

$$\vec{\mathrm{S}} = 0,\ \vec{\dot{S}} = 0 \tag{5.23}$$

and

$$\vec{\mathrm{u}} = \overrightarrow{u'_{eq}} \tag{5.24}$$

With account of the equations of the small spacecraft motion the second equation of (5.23) takes the form:

$$\begin{aligned}\vec{\dot{\mathrm{S}}} &= \overrightarrow{\Delta\dot{\omega}} + \mathrm{K_q}\overrightarrow{\Delta\dot{q}} = \overrightarrow{\dot{\omega}^b_{bi}} - \overrightarrow{\dot{\omega}_T} + \frac{1}{2}\mathrm{K_q}\left(\Delta q_0 E + [\overrightarrow{\Delta q}\times]\right)\overrightarrow{\Delta\omega} = \\ &= -\mathrm{I}^{-1}[\overrightarrow{\omega^b_{bi}}\times]I\overrightarrow{\omega^b_{bi}} + I^{-1}\overrightarrow{u'_{eq}} + I^{-1}\overrightarrow{M_{dis}} - \overrightarrow{\dot{\omega}_T} + \\ &\frac{1}{2}\mathrm{K_q}\left(\Delta q_0 E + [\overrightarrow{\Delta q}\times]\right)\left(\overrightarrow{\omega^b_{bi}} - \overrightarrow{\omega_T}\right) = 0.\end{aligned} \tag{5.25}$$

Then $\overrightarrow{u'_{eq}}$ can be obtained directly from (5.24):

$$\overrightarrow{u'_{eq}} = [\overrightarrow{\omega^b_{bi}}\times]I\overrightarrow{\omega^b_{bi}} - \overrightarrow{M_{dis}} + \mathrm{I}\overrightarrow{\dot{\omega}_T} - \frac{1}{2}\mathrm{K_q}I\left(\Delta q_0 E + [\overrightarrow{\Delta q}\times]\right)\left(\overrightarrow{\omega^b_{bi}} - \overrightarrow{\omega_T}\right). \tag{5.26}$$

Until the point characterizing the state of the control system reaches the surface:

$$\vec{\mathrm{S}} \neq 0,\ \vec{\dot{S}} \neq 0 \tag{5.27}$$

and $\vec{\mathrm{u}} = \overrightarrow{u'_{eq}} + \overrightarrow{\mathrm{u}_{\mathrm{k}}}$.

As $\overrightarrow{\mathrm{u}_{\mathrm{k}}}$ we use the proportional control law:

$$\overrightarrow{\mathrm{u}_{\mathrm{k}}} = -\lambda\vec{\mathrm{S}} \tag{5.28}$$

where

λ is a constant, $\lambda > 0$.

Fig. 5.2 shows the results of the simulation of the orbital orientation of the small spacecraft using a sliding control for specifying the law of the change in the magnetic moment of the small spacecraft.

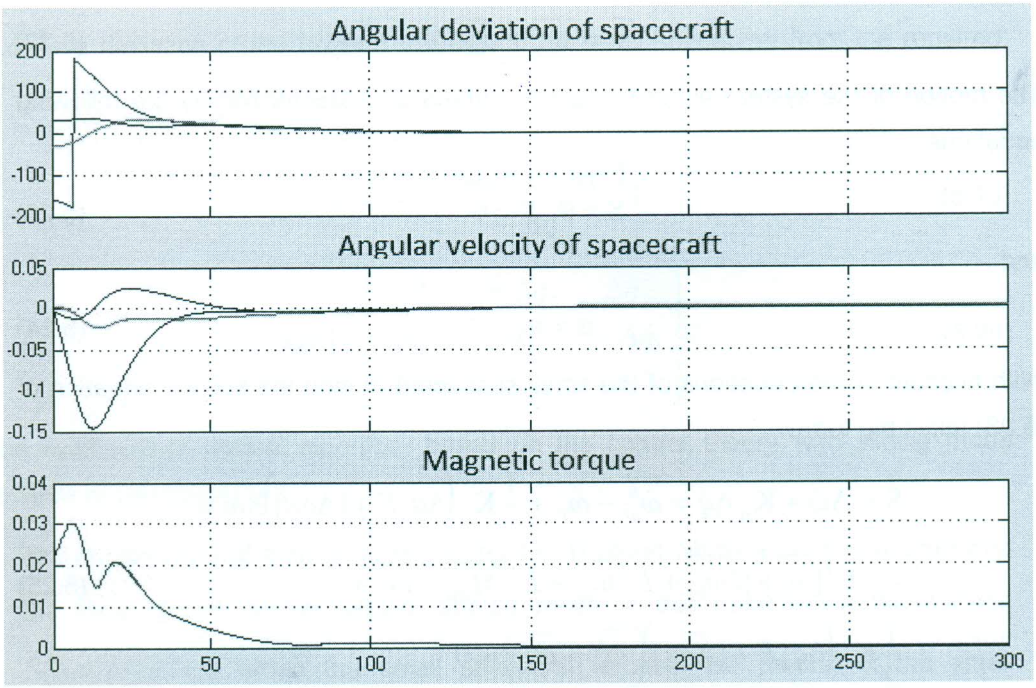

Fig. 5.2. Results of the simulation of the orbital orientation of the small spacecraft using a sliding control for determining the magnetic moment

Source: own elaboration

5.5. Conclusions

There was developed an algorithm of three-axis attitude control of small satellite on the basis of linear control law taking into account the compensation of residual magnetic torque.

There was developed an algorithm of three-axis attitude control of small satellite on the basis of control theory with sliding mode taking into account compensation of residual magnetic torque.

Obtained results on developments of algorithms of three-axis attitude control of satellite taking into account the compensation of residual magnetic torque can be used as the base for the creation and implementation of orbital orientation algorithm of small satellite at periodic effect of residual magnetic torque.

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