Journal of Zhejiang University SCIENCE A
ISSN 1673-565X (Print); ISSN 1862-1775 (Online)
www.zju.edu.cn/jzus; www.springerlink.com
E-mail: jzus@zju.edu.cn

Attitude stabilization of a pico-satellite by momentum wheel and magnetic coils*

Tao MENG†, Hao WANG, Zhong-he JIN, Ke HAN

(*Department of Information Science & Electronic Engineering, Zhejiang University, Hangzhou 310027, China*)

†E-mail: mengtao@zju.edu.cn

Received June 6, 2008; Revision accepted Feb. 25, 2009; Crosschecked Sept. 10, 2009

Abstract: The three-axis active attitude control method with a momentum wheel and magnetic coils for a pico-satellite is considered. The designed satellite is a 2.5 kg class satellite stabilized to nadir pointing. The momentum wheel performs a pitch-axis momentum bias, nominally spinning at a particular rate. Three magnetic coils are mounted perpendicularly along the body axis for precise attitude control through the switch control mechanism. Momentum wheel start up control, damping control and attitude acquisition control are considered. Simulation results show that the proposed combined control laws for the pico-satellite is reliable and has an appropriate accuracy under different separation conditions. The proposed strategy to start up the wheel after separation from the launch vehicle shows that its pitch momentum wheel can start up successfully to its nominal speed from rest, and the attitude convergence can be completed within several orbits, depending on separation conditions.

Key words: Pico-satellite, Attitude control, Magnetic coils, Momentum wheel
doi:10.1631/jzus.A0820425 **Document code:** A **CLC number:** V448.22

INTRODUCTION

The achievements of modern advanced technologies, such as micro-electronics, micro-electro-mechanical systems (MEMS), and precision machining, allow the creation of pico-satellites which are capable of solving problems usually for a big and expensive spacecraft. During the past 10 years pico-satellites were widely studied because of their relative simplicity and short period of design as well as low cost (Schilling, 2006).

The development of pico-satellite requires an attitude control system that is inexpensive, lightweight, meanwhile with a small volume and low power consumption. This motivated the use of magnetic coils as the sole actuator and the momentum wheel as a stabilizer providing a pitch-axis momentum bias. In (Smirnov, 2001; Silani and Lovera, 2005), three-axis magnetic stabilization of a spacecraft can be achieved using feedback from magnetic field measurement and angular velocity.

The use of magnetic coils for control has been the subject of extensive study since the early years of satellite missions (Ergin and Wheeler, 1965; Wang and Shtessel, 1998; Wisniewski and Blanke, 1999; Bushenkov *et al*., 2002; Silani and Lovera, 2005; Guelman *et al*., 2005; Jan and Tsai, 2005). Different from the existing control mechanism, only polarities of currents in the coils are control parameters in our design. Meanwhile, they are working separately in time to prevent the distortion of magnetic field measurement by magnetic coils. Within each second, 0.25 s are allotted for measurement, 0.5 s for control, and the rest for coils demagnetization.

The momentum wheel performs a pitch-axis momentum bias, which is nominally spinning at a particular rate to provide gyroscopic stiffness. It is hard for pico-satellites to find a small volume, low power consumption and rate controllable wheel. We employed a micro-momentum wheel spinning at a

* Project supported by the Program for New Century Excellent Talents in University (No. NCET-06-0514), China, and the Postdoctoral Science Foundation of China (Nos. 20081458 and 20080431306)

particular rate without rate control; i.e., the attitude is controlled by magnetic coils only. The disturbance induced by the rate instability of the momentum wheel is also considered in this paper.

As the limitation of launch with other satellites for cost-saving, the design of pico-satellites must be flexible and robust for the change of launch program and separation condition. Here we considered the variation of wheel start-up time and the initial separation states.

Satellites equipped with a momentum wheel usually spin up the wheel before being separated for the launch vehicle, or go through a start up after separation by slowly bringing up the wheel speed while keeping the satellite stable from generated disturbances using magnetic torques (Chang *et al*., 2006). In this paper, the wheel cannot be started up slowly since the rate is uncontrollable. After power-up, the wheel spins up to its nominal rate within 1 s. The new wheel start-up method proposed does not require attitude stabilization before a momentum wheel starts up. The satellite is despun by magnetic coils after the wheel starts up.

The control laws presented in this paper with a joint control action for momentum wheel start-up, damping, and nominal operation provide an attitude control system which can achieve global three-axis stabilization using solely magnetic coils for actuation.

CONFIGURATION

The satellite, restricted to 2.5 kg, confines its own size within a cube measuring 0.15 m on all sides. The main mission of the satellite is to carry out a new way for space exploration as well as to test the onboard micro scientific experiments. Its orbit is the sun-synchronous circular orbit with an altitude ranging from 400 to 700 km. Fig.1 shows the prototype of the satellite and the corresponding body axis definition. Because of the limited space inside the cube, the magnetic coils are structured as a ring, mounted perpendicularly along the body axis. The magnetic coils are operated through a switch control mechanism, which can provide a maximum dipole moment of 18 mA·m^2. The momentum wheel is mounted along the pitch axis to provide a momentum bias.

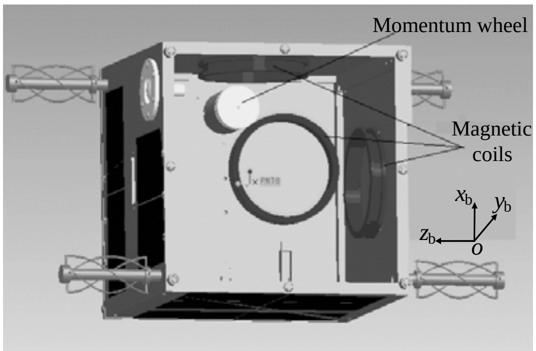

Fig.1 The prototype of the pico-satellite

The attitude and angular velocity of the spacecraft are determined by measuring the magnetic field and the sun vector in an extended Kalman filter. The magnetic field is measured using a digital magnetometer while the sun vector is calculated by monitoring the current output from the solar panels which cover all the sides of the cube (Yu *et al*., 2007).

SATELLITE MODEL

Coordinate frames

1. Earth-centered inertial (ECI) reference frame

The origin of this reference frame is located in the center of the mass. The frame is denoted as i, with its *x*-axis pointed towards the vernal equinox, and the *z*-axis as Earth's rotation axis.

2. Orbit reference frame

The orbit frame origin coincides with the spacecraft center. The origin, rotating at an angular velocity relative to the ECI frame, has its *z*-axis pointed towards the center of the earth. Its *x*-axis points in the spacecraft's direction of motion tangentially to the orbit. The satellite attitude is described by roll, pitch and yaw which are the rotations around *x*-, *y*-, *z*-axis, respectively. The orbit reference frame is denoted as o.

3. Body reference frame

The body frame which shares its origin with the orbit frame is denoted as b. The rotation between the orbit frame and the body frame is used to represent the spacecraft's attitude. Its axes are locally defined in the spacecraft, with the origin in the center of the volume (Fig.1). The nadir side of the spacecraft, intended to point towards the earth, is in z_b-axis.

Satellite dynamics and kinematics

The dynamic equations of motions of the satellite with momentum bias configuration are

$$\dot{\boldsymbol{\omega}}_{bi}^{b} = \boldsymbol{I}^{-1}\left[-\left[\boldsymbol{\omega}_{bi}^{b}\times\right]\left(\boldsymbol{I}\boldsymbol{\omega}_{bi}^{b}+\boldsymbol{h}\right)-\dot{\boldsymbol{h}}+\boldsymbol{T}_{c}+\boldsymbol{u}_{d}\right], \quad (1)$$

where $\boldsymbol{\omega}_{bi}^{b}$ is the satellite's angular velocity relative to inertial frame described in body frame; $\boldsymbol{h}$ is the angular momentum of the wheel; $\boldsymbol{I}$ is the inertia matrix of the satellite; $\boldsymbol{T}_c$ is magnetic control torque vector and $\boldsymbol{u}_d$ is disturbance torque vector.

Using quaternion, the kinematic equations of satellite attitude motion are

$$\dot{\boldsymbol{q}} = \boldsymbol{q}\otimes\boldsymbol{\omega}_{bo}^{b}/2, \quad (2)$$

$$\boldsymbol{\omega}_{bo}^{b} = \boldsymbol{\omega}_{bi}^{b} - \boldsymbol{A}_{bo}\boldsymbol{\omega}_{oi}^{o} = \boldsymbol{\omega}_{bi}^{b} - \boldsymbol{A}_{bo}[0 \quad -\omega_0 \quad 0]^{T}, \quad (3)$$

where $\boldsymbol{q}=[q_0 \quad q_1 \quad q_2 \quad q_3]^{T}$, $q_0^2+q_1^2+q_2^2+q_3^2=1$, $\boldsymbol{\omega}_{bo}^{b}$ is the satellite's angular velocity relative to orbit frame described in body frame; $\boldsymbol{\omega}_{oi}^{o}$ is the orbital angular velocity relative to inertial frame described in orbit frame; $\boldsymbol{A}_{bo}$ is the attitude transformation matrix between orbit frame and body frame, and ω_0 is the orbital angular rate.

MAGNETIC CONTROL UNIT DESIGN

Magnetic coils

The magnetic coils that align with the body axis are used for attitude control. The vector dipole moment $\boldsymbol{m}$ of magnetic coils interacts with the geomagnetic field vector $\boldsymbol{B}$ to generate a magnetic torque $\boldsymbol{T}_c$,

$$\boldsymbol{T}_c = \boldsymbol{m}\times\boldsymbol{B}, \quad (4)$$

where $\boldsymbol{m}$ is determined by on-board controller and created by the magnetic coils.

The constraint dictated by the magnetic coils is such that only the component of $\boldsymbol{T}_d$ orthogonal to the magnetic field vector $\boldsymbol{B}$ can be actually applied to the spacecraft, so the best possible control policy is to apply the magnetic dipole moment (Silani and Lovera, 2005; Chang *et al*., 2006):

$$\boldsymbol{m} = (\boldsymbol{B}\times\boldsymbol{T}_d)\big/\|\boldsymbol{B}\|^2, \quad (5)$$

where the ideal torque desired $\boldsymbol{T}_d$ is obtained from the following control laws.

Momentum wheel start-up control

The momentum wheel stabilizes the attitude through the production of angular momentum by spinning at a nominal speed. However, if the wheel is suddenly spun up to its nominal speed, the large momentum produced together with the existing momentum due to the nominal rotational motions will result in a high disturbance and unstable motion (Chang *et al*., 2006). The momentum wheel is thus set to spin up before being separated from the launch vehicle. By contrast, when the wheel is not allowed to work during the launch time, the wheel may spin up after separation from the launch vehicle, which requires the magnetic torques to provide the moment so as to keep the satellite stable.

The control law is

$$\boldsymbol{T}_d = -k_u\dot{\boldsymbol{h}}_w, \quad (6)$$

where k_u is the wheel control gain, and $\dot{\boldsymbol{h}}_w$ is the disturbance torque introduced by start-up of the wheel.

Eq.(6) simply needs the rate feedback of the wheel. However, only pitch attitude control is offered in this mode. Eq.(7) gives the control law which is combined by two parts: one is for pitch attitude control when no angular velocity information is available, and the other for whole attitude control during the wheel start-up, which can be adopted when the attitude filter works well.

$$\boldsymbol{T}_d = -k_u\dot{\boldsymbol{h}}_w - k_d\dot{\boldsymbol{\theta}}_b, \quad (7)$$

where $\dot{\boldsymbol{\theta}}_b = [\dot{\phi},\dot{\theta},\dot{\psi}]^{T}$, $\dot{\phi}$, $\dot{\theta}$, $\dot{\psi}$ are the change rates of Euler angle with time.

Damping control

After being released from the launch vehicle, the satellite is tumbling randomly with known bounds at the initial angular rate. The objective of this phase is to reduce angular velocity and the only sensor needed in this mode is the magnetometer. The control adopted here is the B-dot damping law:

$$\boldsymbol{m}(t_k)=\begin{bmatrix} m_x(t_k) \\ m_y(t_k) \\ m_z(t_k) \end{bmatrix}=\begin{bmatrix} -m_{\max}\,\mathrm{sign}(\dot{B}_x^{\mathrm{b}}(t_k)) \\ -m_{\max}\,\mathrm{sign}(\dot{B}_y^{\mathrm{b}}(t_k)) \\ -m_{\max}\,\mathrm{sign}(\dot{B}_z^{\mathrm{b}}(t_k)) \end{bmatrix}, \quad (8)$$

$$\dot{\boldsymbol{B}}_{\mathrm{b}}(t_k)\equiv\begin{bmatrix} \dot{B}_{\mathrm{b}x}(t_k) \\ \dot{B}_{\mathrm{b}y}(t_k) \\ \dot{B}_{\mathrm{b}z}(t_k) \end{bmatrix}=\frac{\boldsymbol{B}_{\mathrm{b}}(t_k)-\boldsymbol{B}_{\mathrm{b}}(t_{k-1})}{t_k-t_{k-1}}, \quad (9)$$

where $\dot{\boldsymbol{B}}_{\mathrm{b}}(t_k)$, the change rate of the measured geomagnetic vector in the body axis at t_k, can be estimated with magnetometer measurement; $\boldsymbol{m}_{\max}$ denotes the maximum dipole moment generated by the magnetic coil.

Attitude acquisition

After the phase of damping, we will adjust both the attitude angle and angular velocity to zero. The control law is

$$\boldsymbol{T}_{\mathrm{d}}=-k_{\mathrm{p}}\boldsymbol{\theta}_{\mathrm{b}}-k_{\mathrm{d}}\dot{\boldsymbol{\theta}}_{\mathrm{b}}, \quad (10)$$

where $\boldsymbol{\theta}_{\mathrm{b}}=[\phi,\theta,\psi]^{\mathrm{T}}$, and the control gains k_{p} and k_{d} are optimized by simulation.

Control flow design

To obtain optimal performance, the actual control algorithm is a combination of the control laws mentioned above. There are three criterions in the control flow. After the satellite's departure from the launch vehicle, the first criterion is executed to verify whether the wheel is started up or not. If the answer is no, turn on the momentum wheel immediately and perform momentum wheel start up control mode. If the wheel is already start up or the wheel start up control mode is finished, the second criterion is executed to decide the following control mode (Fig.2). The transition between the damping mode and attitude acquisition mode depends on attitude angles and attitude angular rates, which is achieved by simulation.

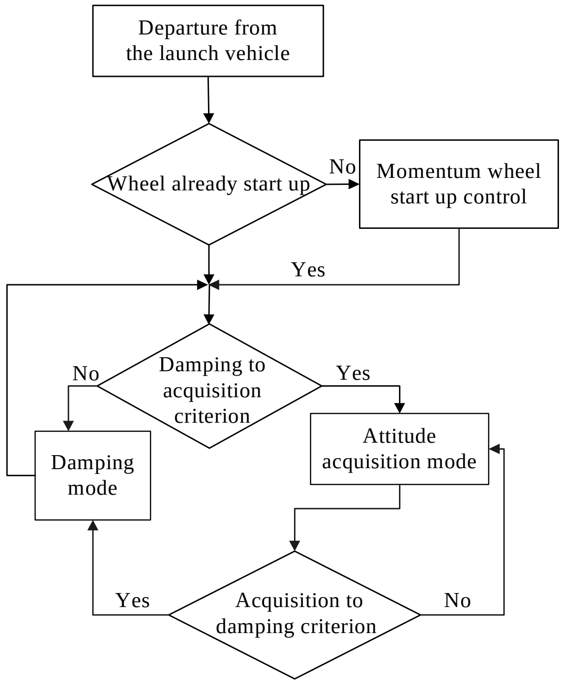

Fig.2 Control flow of the pico-satellite

SIMULATION RESULTS AND ANALYSIS

In this section, we present some results obtained by numerical simulation to study the performance of the above-described control algorithm.

The following modules are included in our system simulation: Satellite module consisting of both dynamics and kinematics equations; Attitude determination module for angle and angular velocity measurement with the magnetometer and sun sensor; On-board CPU module for house-keeping and attitude control module.

This simulation assumes that the momentum wheel will be started up 10 s after its separation from the launch vehicle. The wheel start-up time is set to 1 s; hence, the start-up rate of the wheel is equal to its nominal rate. The parameters of the model used in the simulations are listed in Table 1. Feedback from the magnetic field is given using the International Geomagnetic Reference Field (IGRF). The IGRF specifies the numerical coefficients of a truncated spherical harmonic series, and together with an orbit estimator, an estimate of the magnetic field is made.

The simulation results are shown in Figs.3~8. Fig.3 reports the time when the wheel starts up after the separation and the angular momentum instability of the working wheel. The pitch angular velocity change is also presented in Fig.3. The angular velocity goes up to 6°/s during the wheel start-up. Figs.4~6

Table 1 Satellite system simulation parameters

Parameter	Value/Description
Satellite mass (kg)	2.5
Inertia matrix (kg·m^2)	$\begin{bmatrix} 12.19\times10^{-3}, 0, 0 \\ 0, 14.06\times10^{-3}, 0 \\ 0, 0, 9.375\times10^{-3} \end{bmatrix}$
Orbit period (s)	5850
Nominal angular momentum of the wheel (N·ms)	1.5×10^{-3}
Nominal rate of the wheel (r/min)	7800
Start-up time of the wheel (s)	1
Rate instability (r/min)	10
Maximum magnetic moment (A·m^2)	18×10^{-3}
Magnetic coil control	Switch control, 30 mW per coil
Control gains	$k_p = \text{diag}\{0.001\ 0.02\ 0.001\}$
	$k_d = \text{diag}\{1\ 3\ 2\}$
	$k_u = 10$
Initial attitude after separation	$\varphi_0 = \theta_0 = \psi_0 = 3°$
	$\dot{\varphi}_0 = \dot{\theta}_0 = \dot{\psi}_0 = 0.1°/s$

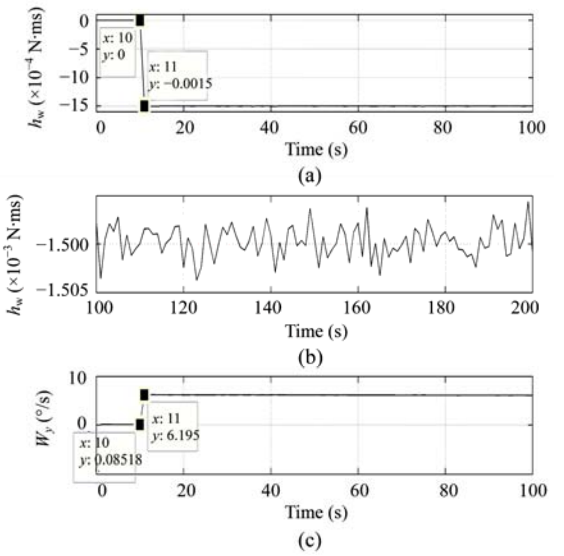

Fig.3 The momentum wheel start-up process.
(a) Angular momentum of the wheel; (b) Angular momentum instability of the wheel; (c) Pitch angular velocity change during the start-up period

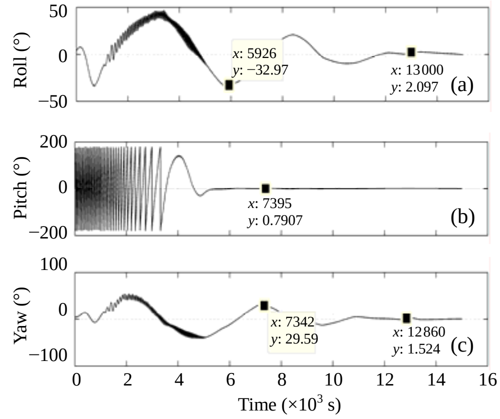

Fig.4 The variation of attitude angle without wheel start-up control. (a) Roll; (b) Pitch; (c) Yaw

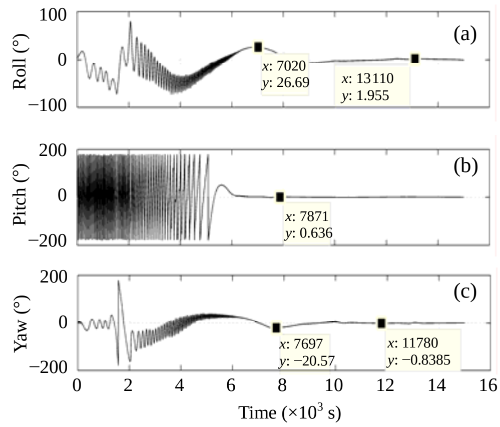

Fig.5 The variation of attitude angle using wheel start-up control. (a) Roll; (b) Pitch; (c) Yaw

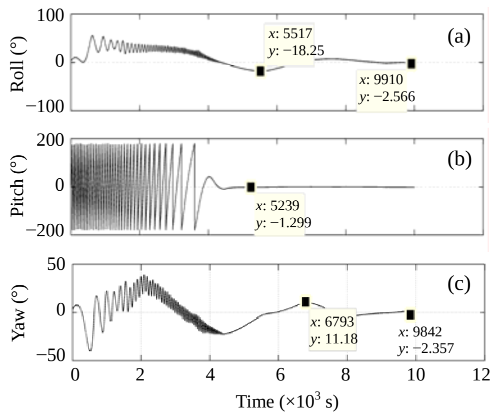

Fig.6 The variation of attitude angle using combined start-up control. (a) Roll; (b) Pitch; (c) Yaw

describe the variation of attitude angle using a different control law for comparison. Fig.4 illustrates the one without any wheel start-up control. The nutation is about 33° after one orbit control, and the attitude converges in 12 000 s. Fig.5 shows that the start-up control law introduced in Eq.(6) can fasten the attitude convergence speed effectively as compared with the one without start-up control, though only the pitch attitude is under control during the first 1000 s. If the angular velocity information is available, using the combined control law (Eq.(7), Fig.6), the nutation of the attitude can be decrease to 18° after one orbit control, the attitude becomes convergent in 10 000 s. Fig.7 outlines the variation of attitude angular velocities using the combined control law. Fig.8 shows the magnetic coils' operation correspondingly, where 0 indicates turn-off state and 1 for turn-on state. We can draw conclusions that the proposed wheel start-up control by magnetic coils for the pico-satellite is effective. The attitude angular velocities can be converged below 0.05°/s and the attitude angle can be converged down to 5° less than 2 orbits. The average power consumption during stabilizing is about 80 mW.

Meanwhile, different initial conditions are considered to demonstrate the flexibility of the proposed control algorithm. Our numerical simulation indicates that the initial attitude angle has little influence on the time of stabilizing, so only the attitude angular velocity is discussed.

Fig.9a illustrates the time of stabilization according to different initial attitude angular velocity

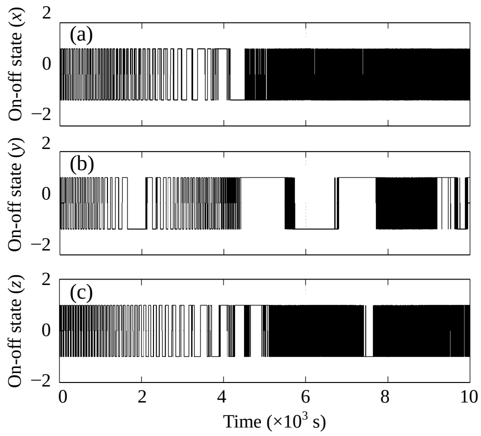

Fig.8 The operation states of magnetic coils using combined start-up control. (a) Operation state of magnetic coil in x_b-axis; (b) Operation state of magnetic coil in y_b-axis; (c) Operation state of magnetic coil in z_b-axis

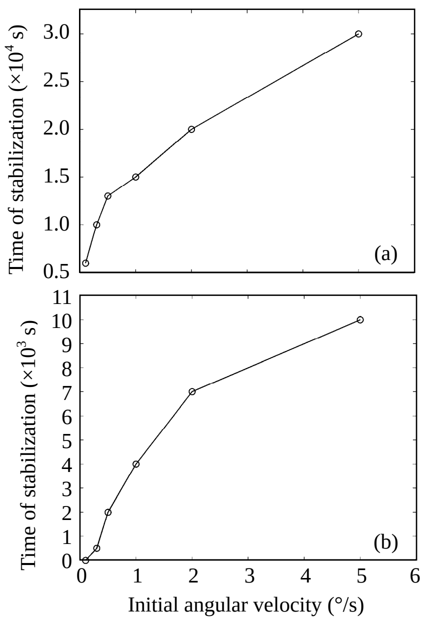

Fig.9 Time of stabilizing under different initial angular velocity. Wheel starts up (a) after and (b) before separation

when the initial angle is $\varphi_0=\theta_0=\psi_0=3°$. The maximum time needed of stabilizing is 30 000 s, nearly 6 orbits, which is obtained at the initial angular velocity of 5°/s. On the other hand, Fig.9b shows the time of stabilization when the momentum wheel starts up before separation in the same separating conditions. The maximum time needed of stabilizing is 10 000 s, nearly 2 orbits, which is also obtained at the initial angular velocity of 5°/s.

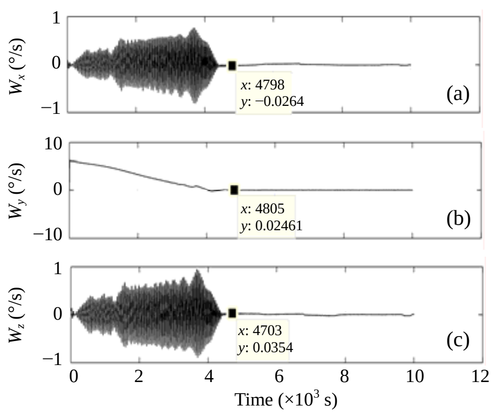

Fig.7 The variation of attitude angular velocities using combined start-up control. (a) Roll angular velocity; (b) Pitch angular velocity; (c) Yaw angular velocity

CONCLUSION

The attitude control problem for pico-satellites using magnetic coils was analyzed. A set of control laws used in different modes were discussed and combined to perform a robust control system with respect to uncertain launch conditions and initial parameters.

Our numerical simulation shows that this combined structure can stabilize attitude efficiently no matter whether the wheel was started up before or after the separation from launch vehicle. The simulation also shows the time of achieving stability under different initial conditions. This proposed method requires only magnetic coils and a momentum wheel with a particular rate for controlling the satellite, thus making it possible to be implemented on other pico-satellites.

References

Bushenkov, V.A., Ovchinnikov, M.Y., Smirnov, G.V., 2002. Attitude stabilization of a satellite by magnetic coils. *Acta Astronautica*, **50**(12):721-728. [doi:10.1016/S0094-5765(02)00011-5]

Chang, Y.K., Lee, B.H., Kim, S.J., 2006. Momentum wheel start-up method for HAUSAT-2 ultra-small satellite. *Aerospace Science and Technology*, **10**(2):168-174. [doi:10.1016/j.ast.2005.07.011]

Ergin, E.I., Wheeler, P.C., 1965. Magnetic attitude control of a spinning satellite. *Journal Spacecraft and Rockets*, **2**(6):846-850. [doi:10.2514/3.28302]

Guelman, M., Waller, R., Shiryaev, A., Psiaki, M., 2005. Design and testing of magnetic controllers for satellite stabilization. *Acta Astronautica*, **56**(1-2):231-239. [doi:10.1016/j.actaastro.2004.09.028]

Jan, Y.W., Tsai, J.R., 2005. Active control for initial attitude acquisition using magnetic torquers. *Acta Astronautica*, **57**(9):754-759. [doi:10.1016/j.actaastro.2005.03.067]

Schilling, K., 2006. Design of pico-satellites for education in systems engineering. *IEEE Aerospace and Electronic System Magazine*, **21**(7):S9-S14. [doi:10.1109/MAES.2006.1684269]

Silani, E., Lovera, M., 2005. Magnetic spacecraft attitude control: a survey and some new results. *Control Engineering Practice*, **13**(3):357-371. [doi:10.1016/j.conengprac.2003.12.017]

Smirnov, G.V., 2001. Attitude determination and stabilisation of a spherically symmetric rigid body in a magnetic field. *International Journal of Control*, **74**(4):341-347. [doi:10.1080/00207170010010560]

Wang, P., Shtessel, Y.B., 1998. Satellite Attitude Control Using only Magnetorquers. American Control Conference, Philadelphia, PA, USA, p.222-226. [doi:10.1109/ACC.1998.694663]

Wisniewski, R., Blanke, M., 1999. Fully magnetic attitude control for spacecraft subject to gravity gradient. *Automatica*, **35**(7):1201-1214. [doi:10.1016/S0005-1098(99)00021-7]

Yu, F.X., Meng, T., Han, K., Zhuang, R., Jin, Z.H., 2007. Design and error analysis of pico-satellite sun sensor. *Acta Energiae Solaris Sinica*, **28**(12):1295-1299 (in Chinese).