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# AN ELECTROMAGNETIC SYSTEM FOR ATTITUDE DETERMINATION AND CONTROL OF A NANOSATELLITE

**Розглядається система кутової орієнтації і стабілізації наносупутника. З урахуванням енергетичних обмежень оцінюються параметри системи.**

***Ключові слова***: управління кутовим положенням, електромагнітна система орієнтації і стабілізації.

**A nanosatellite Attitude Determination and Control System (ADCS) is considered. Taking into account the energy limitations, the system parameters are evaluated.**

***Key words***: nanosatellite, attitude control, electromagnetic control and stabilization.

**Рассматривается система угловой ориентации и стабилизации наноспутника. С учетом энергетических ограничений оцениваются параметры системы.**

***Ключевые слова***: наноспутник, управление угловым положением, электромагнитная система ориентации и стабилизации.

## Introduction

Dnepropetrovsk National University is the head of a project concerning the development of a university nanosatellite, in partnership with different Ukrainian Universities and the University of Brasilia. The present work deals with the development of an Attitude Determination and Control System (ADCS) for this nanosatellite using electromagnetic actuators and a set of relatively low-cost sensors. It is intended to provide required angular attitudes with errors within the range of 3 degrees for each axis.

## Satellite Attitude Model

The spacecraft (SC) dynamics and kinematics are considered for determination and control calculations. Without loss of functionality, the SC is considered a rigid body with known principal axes of inertia. Thus, the classical Euler equations describe SC dynamics:

$$\begin{aligned} I_1\omega_1' + (I_3 - I_2)\omega_2\omega_3 - M_1 &= 0,\\ I_2\omega_2' + (I_1 - I_3)\omega_3\omega_1 - M_2 &= 0,\\ I_3\omega_3' + (I_2 - I_1)\omega_1\omega_2 - M_3 &= 0. \end{aligned} \tag{1}$$

Where *diag*$([I_1, I_2, I_3])$ is the inertia tensor over the principal axes of inertia and $\vec{\omega} = [\omega_1, \omega_2, \omega_3]^T$ is the angular velocity. We calculate $I_1 = I_2 = 0{,}1521$ and $I_3 = 0{,}0375$, in $kg\ m^2$, for the SC as a uniform 10 kg cube with dimensions (400 × 150 × 150), in millimeters.

Rotations are represented by quaternions. Several factors favor this choice, like absence of singularities and the adequacy for embedded computation [8]. We relate the angular velocities of the SC with the rate of change of its attitude by the relations (2). Equation (2.b) is our quaternion definition, with $\hat{e}$ being a unit vector (Euler vector). Further details on quaternion algebra and others can be found in reference [2].

$$\dot{\vec{q}} = \tfrac{1}{2}\begin{bmatrix} \left[-\vec{\omega}^{\times}\right] & \vec{\omega} \\ -\vec{\omega}^{T} & 0 \end{bmatrix}\vec{q}\ , \quad \left[\vec{a}_{3x1}{}^{\times}\right] = \begin{bmatrix} 0 & -a_3 & a_2 \\ a_3 & 0 & -a_1 \\ -a_2 & a_1 & 0 \end{bmatrix}, \tag{2.a}$$

$$\vec{q} = \begin{bmatrix} q_1 & q_2 & q_3 & q_4 \end{bmatrix}^T \therefore \begin{bmatrix} q_1 & q_2 & q_3 \end{bmatrix}^T = \hat{e}\sin(\theta/2) \therefore q_4 = \cos(\theta/2), \tag{2.b}$$

**Attitude Determination System**

The task of attitude determination (AD), especially for a rigid small body, is a well-established and known problem [1,4,5,7]. In this sense, we evaluate different techniques to raise a good AD scheme. Some works consider this problem using only magnetometers [5]. As a vector has only 2-dof attitude information, this approach cannot resolve the attitude locally. A traditional solution is the use of line-of-sight (LOS) sensors. In this work we use sun sensors and magnetometers to retrieve SC attitude locally. The complete design will concern hardware and software implementation.

Figure 1 shows a general view of the AD hardware. In the upper part we see the Sensor Board schematics, and in the bottom part the Satellite Aided Navigation Board. The GPS antenna and sun sensors will be placed outside the SC, the last ones attached to the solar panels. All the computations are performed within the onboard computer. The sensor boards send data through USB or RS protocol only upon request by the CPU algorithms.

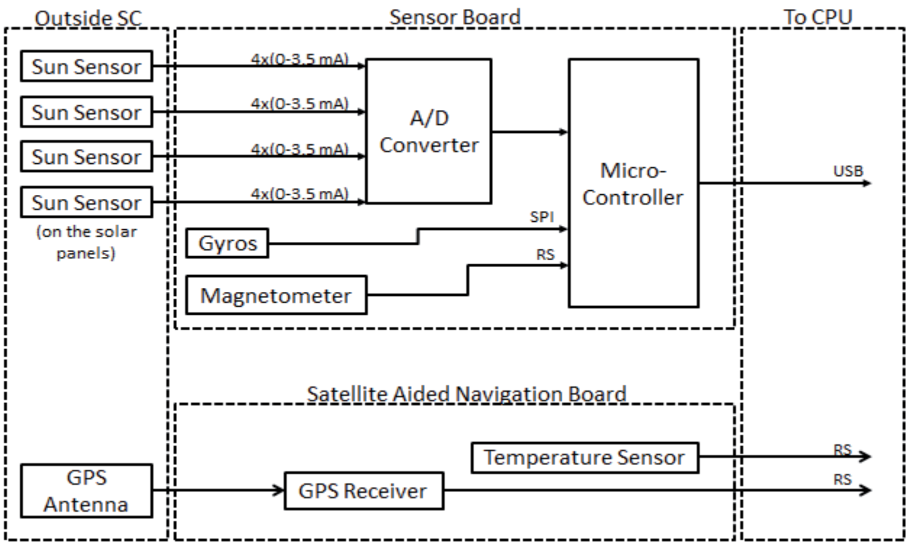


**Fig. 1. Attitude Determination Hardware Functional Schematics**

A general view of the AD algorithm is presented in Fig. 2. The core of the system is the Nonlinear Kalman Filter, which will have its prediction step fed by gyro data and correction step triggered by a complete 3 degrees of freedom (3-dof) attitude measurement.

**Sensor Models**

*Gyro and Magnetometer* – Errors arise from misalignments, biases and noises. Misalignments and some biases can be modeled and compensated by calibration, but random noises and bias random drifts should be dealt with using a stochastic filter. In our system the estimation state vector is augment with the biases. In a mathematical model [3]:

$$s^{sensor} = \left(T_{sensor}^{SC}\right)^{-1} s^{SC} + b^{sensor} + v^{sensor}. \tag{3}$$

Where, $\boldsymbol{s}$ are sensor measurements; $\boldsymbol{T}$ is a frame transformation; $\boldsymbol{b}$ is the bias; and $\boldsymbol{v}$ the noise vector. Superscripts are the reference frame of the magnitudes.

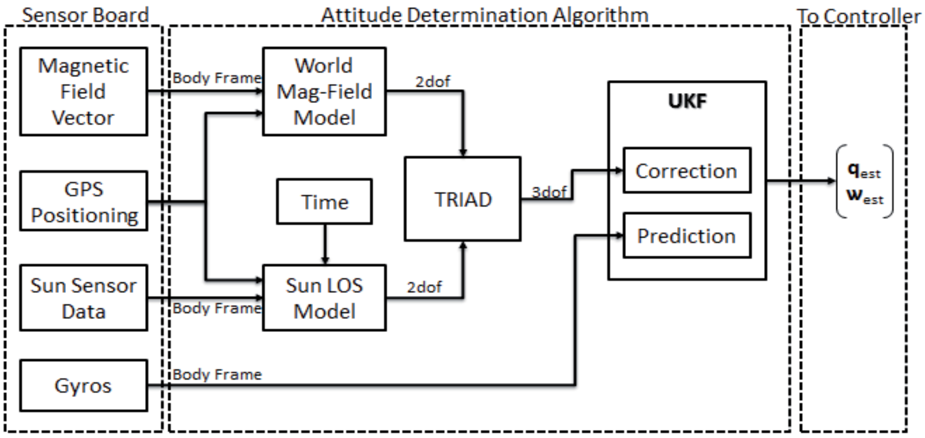


**Fig. 2. Attitude Determination Algorithm Schematics**

*Earth Magnetic Field* – The AD system models the expected geomagnetic measurements. For this we use geomagnetic models maintained by scientific organizations. In the simulations, we use the World Magnetic Model (WMM) from the U. S. National Geospatial-Intelligence Agency and the United Kingdom's Defense Geographic Center.

*Sun Sensors* – The sun sensors are general light transducers. If the angle between the sun LOS and the normal defining the surface of the sensor is $\beta$, the intensity measured is:

$$I_i = I_{\max} \cos(\beta). \tag{4}$$

Index $\boldsymbol{i}$ varies 1-4 corresponding to each sensor, and $\boldsymbol{I_{max}}$ corresponds to $\beta = 0$. If the solar panel $\boldsymbol{i}$ define an angle $\alpha_i$ to the SC body, a geometric model can recover LOS from sensor readings. If we have sensors $\boldsymbol{i}$ and $\boldsymbol{k}$ opposite, $\boldsymbol{I_k} < \boldsymbol{I_i}$ and sensor $\boldsymbol{j}$ pointed by the y-axis:

$$\vec{s}_{LOS} = \begin{bmatrix} v_x \\ v_y \\ v_z \end{bmatrix}, v_z = \frac{I_k}{\sin(\alpha_k)}, v_y = \frac{I_i - (v_z)\sin(\alpha_i)}{\cos(\alpha_i)}, v_x = \frac{I_j - (v_z)\sin(\alpha_j)}{\cos(\alpha_j)}. \tag{5}$$

For 4 sensors excited, the problem is over-determined and we take a simple mean to recover the LOS. If less than 3 sensors are enlighten, we would have to consider $\boldsymbol{I_{max}}$ known.

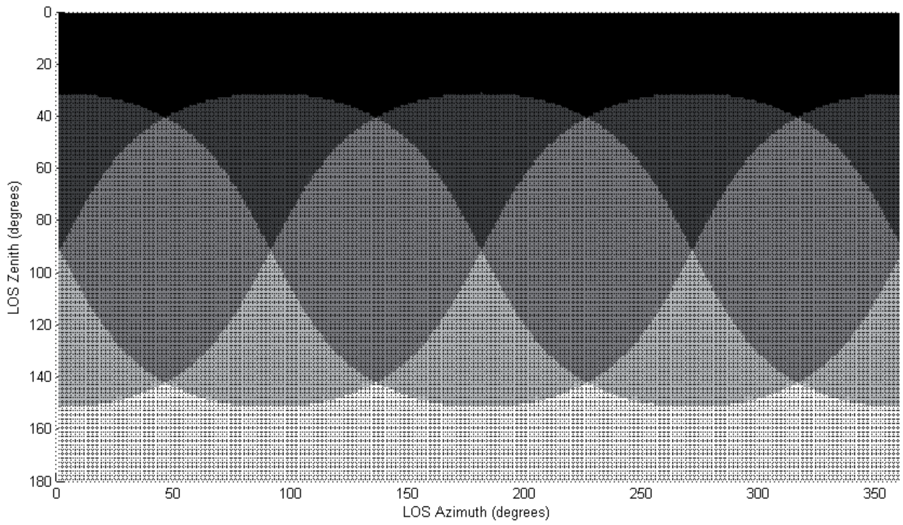


**Fig. 3. Sun Sensors Activity by Sun LOS**

In Figure 3, simulated results show the number of active sensors by sun LOS's.

In this simulation the satellite is with its z-axis pointing up; x-axis pointing east (ENU); and each solar panel makes a 30 degrees angle with the SC body. In the figure, the horizontal axis is the LOS azimuth angle in degrees from north to east. The vertical axis is the LOS zenith angle from up to down. Darker colors mean more active sensors (white in the bottom - 0 sensors; black - 4 sensors). The simulation results show that we have 3 or 4 sun sensors excited by the sun in 33.1% of the LOS sphere.

*Sun LOS Model* – The AD system models the expected sun LOS measurements. We use approximated models as developed in [9], with precisions of 0.01 degrees, due to truncations on the model development.

The Table 1 shows a proposal for the hardware selection for the ADCS system.

*Table 1*

**Attitude Determination System Hardware**

| Sensor | Brand | Model | Description | Interface | Error |
|---|---|---|---|---|---|
| Magnetometer | Honeywell | HMC2300 | 3-axis digital magnetometer | RS | <0,5% F.S. |
| Sun Sensor | - | - | General Light transducer | (0-3.5mA) | < 5º |
| Gyrometer | Analog Devices | ADIS16375 | Low power and mass IMU | SPI | < 0,45º/s |
| GPS | Surrey – SST | SGR-05U | COTS based, suitable for satellites | UART | < 15 m |

**Attitude from Vector Observations**

There are several approaches to problem of attitude determination from vector observations [9]. The TRIAD is a simple deterministic method to be used with two noncolinear vectors, largely used for SC attitude estimation. There are stochastic counterparts to the TRIAD algorithm, but for the case where only two vectors are available, the TRIAD has almost no shortcomings compared to them. The TRIAD algorithm is described below:

$$R_T^b = \left[ s^b \quad \frac{\left(s^b \times m^b\right)}{\left\|\left(s^b \times m^b\right)\right\|} \quad s^b \times \left( \frac{\left(s^b \times m^b\right)}{\left\|\left(s^b \times m^b\right)\right\|} \right) \right], R_r^T = \left[ s^r \quad \frac{\left(s^r \times m^r\right)}{\left\|\left(s^r \times m^r\right)\right\|} \quad s^r \times \left( \frac{\left(s^r \times m^r\right)}{\left\|\left(s^r \times m^r\right)\right\|} \right) \right],$$

$$R_r^b = R_T^b R_r^T . \tag{6}$$

Where $\boldsymbol{s}$ are sun LOS vectors; $\boldsymbol{m}$ are magnetic field vectors; $\boldsymbol{R}$ are rotation matrices; and superscripts are frames where the magnitude is presented ($b$ – body, $r$ – reference, $T$ –auxiliary frame).

**Data Fusion and Kalman Filter**

Extended Kalman Filters (EKFs) are highly used for SC attitude estimation. The EKFs principle is to perform a system linearization during the computation. On the other hand, the Unscented Kalman Filters (UKFs) approach the problem not by attacking the nonlinear function but the state vector probability distribution. For highly nonlinear systems as SC attitude, a well projected UKF should present lower error, a better behavior in situations where the SC have to recover orientation (which is our case as the spacecraft will fly disoriented for larges periods of time), and reduce the computational burden [4]. The filter system model is given by:

$$\begin{aligned} x_{k+1} &= f(x_k, u_k, k) + w_k, \\ y_k &= h(x_k, u_k, k) + v_k. \end{aligned} \tag{7}$$

With $\boldsymbol{w}$ and $\boldsymbol{v}$ being zero-mean Gaussian noises with covariance matrices $\boldsymbol{Q}$ and $\boldsymbol{R}$, respectively. In our implementation, $\boldsymbol{f}$ is given by a discretization of Eq. (2.a) and $\boldsymbol{h}$ is a direct quaternion measurement taken from the state vector. The filter is based in the propagation of deterministically chosen sigma points $\chi$ through the nonlinear functions as described by equations (8) and (9). A complete description of the UKF can be found in [4]:

$$\chi_k(0) = \hat{x}_k^+, \quad \chi_k(i) = \hat{x}_k^+ + \sigma_i, \quad \sigma_i = column_i(\pm\sqrt{(n+\lambda)\hat{P}_k^+}), \quad i = 1,\ldots,2n, \tag{8}$$

$$\begin{matrix} \chi_{k+1}(i) = f(\chi_k(i), u_k, k) + w_k \\ \gamma_k(i) = h(\chi_k(i), u_k, k) + v_k \end{matrix}, \quad i = 0,\ldots,2n. \tag{9}$$

In Figure 4 we can see some results from the estimation simulation. The simulation show that the errors were within the range of 0.5 degrees.

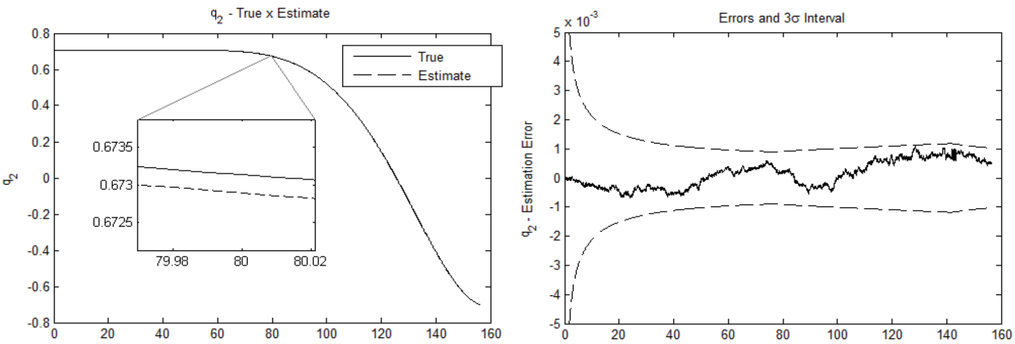


**Fig. 4. Simulated results for Attitude Estimation**

### Attitude Control

The control will be performed by the magnetorquers actuated by a board connected to de CPU. There are 8 bits available for communication between CPU and actuation board.

*Table 2*

**Magnetorquers Bit Addressing and Actuation by Bit Configuration**

| Bits | Actuator | | Bits Configuration | Actuation |
|---|---|---|---|---|
| 0-1 | magnetorquer-1 | | 0-0 | Null Induction |
| 2-3 | magnetorquer-2 | | 0-1 | Max. Positive Induction |
| 4-5 | magnetorquer-3 | | 1-0 | Max. Negative Induction |
| 6-7 | Synchronization/ Reserve Actuator | | 1-1 | Forbidden (Short Circuit) |

As seen on table 2, restrictions with the hardware interface impose a relay-type controller design. Yet, our main limitation is the earth's magnetic field itself. The controller inductions have to interact with geomagnetic field to produce torque. Thus, we can't achieve torques around the direction of the magnetic field of the Earth. This implies local controllability problems. However, as the satellite moves in its trajectory the direction of the magnetic field changes and we can stabilize it, as long as there is enough time available.

Figure 05 shows a schematic of the controller. For the control law, we calculate the required torque (Eq. 10). Then a block calculates the relay inductions that will be applied (Eq. 11). In this sense, we see that the actual applied torque (Eq. 12) can be slightly different from the required torques.

$$\vec{M}_{req} = \operatorname{sgn}\left([K]\vec{q}q_4 + [C][\vec{\omega};0]\right) \cdot \max\left(\left\|\left([K]\vec{q}q_4 + [C][\vec{\omega};0]\right)\right\|, M_{\max}\right). \tag{10}$$

$$\vec{B} = \left(\vec{H} \times \vec{M}_{req}\right) \Big/ \left(\left\|\vec{H}\right\| \cdot \left\|\vec{H}\right\|\right). \tag{11}$$

$$\vec{M}_{app} = \left(\vec{B} \times \vec{H}\right), \text{ with} \tag{12}$$

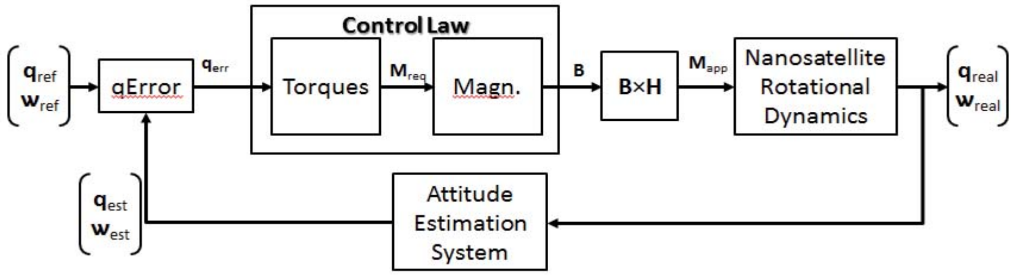


**Fig. 5. Attitude Control Algorithm Schematics**

In Equation (10), the functions **sgn** and **max** implement saturation. **K** and **C** are gain matrices to be adjusted. Figures 6–8 show some simulation results for the controller.

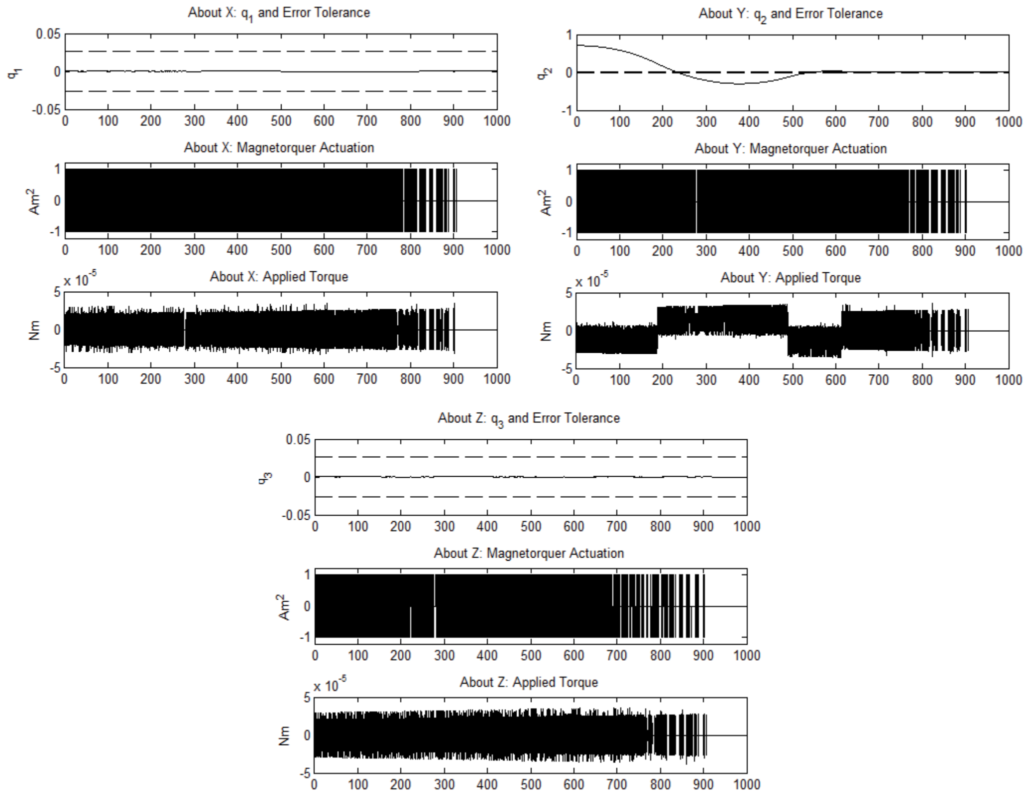


**Fig. 6. 90 deg rotation around y-axis, with no initial velocities**

In the simulations the controller could stabilize to a required position in less than 800 seconds in the absence of initial angular velocities, and in less than 3500 seconds in the presence of initial velocities. Off course these results are dependent on the geomagnetic field and initial conditions, but they represent very good preliminary outcomes.

**Conclusions**

The developments for the ADCS for the Ukrainian University Nanosatellite are presented with some preliminary results. The first strategies for attitude determination and control are defined. Suitable hardware, models and algorithms were selected to grant the system with the data and quality necessary to achieve its objectives. Algorithms are now being continuously simulated to attest their functionality or show necessary adjustments.

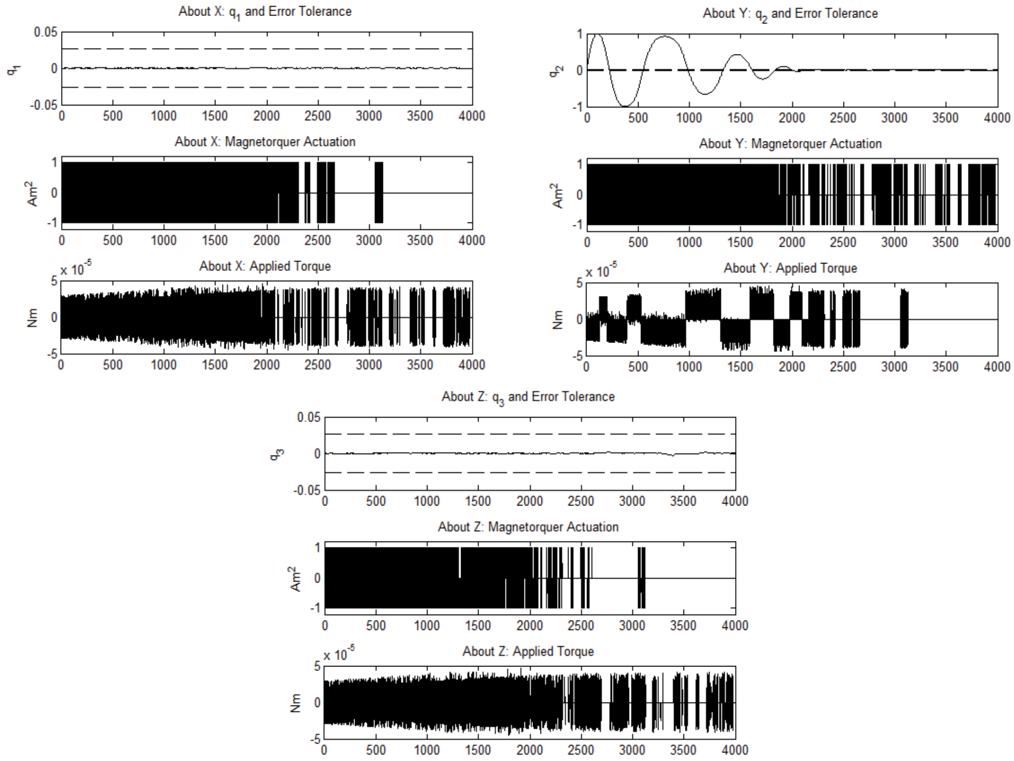


**Fig. 7. Cancel angular velocities (2deg/s around y-axis) maintaining the pointing**

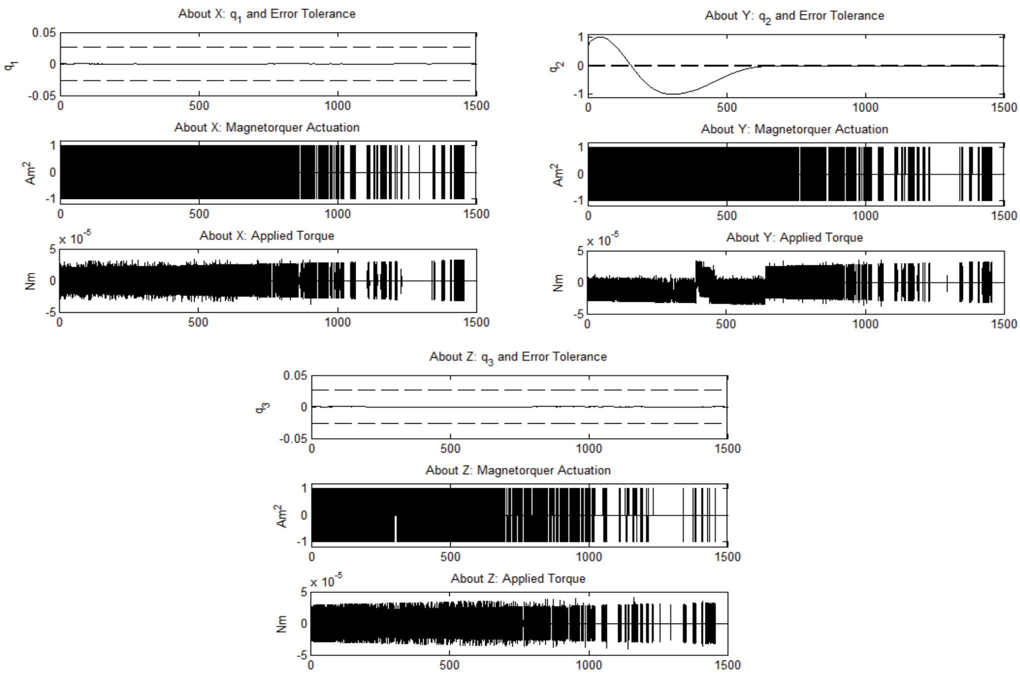


**Fig. 8. 90 deg rotation around y-axis, with 2deg/s y-axis initial velocities**

It is clear that, as the project is meant to use only low cost hardware, the system requires a lot of design effort. Despite the relatively low pointing accuracy, the low cost and power consumption justify the implementation of the system.

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# DESIGN OF GROUND COMMUNICATION ANTENNA CONTROL SYSTEM FOR A NANOSATELLITE MISSION

**Розглядається система програмного супроводження супутника наземною станцією. Наводиться алгоритм розрахунку параметрів супроводження, розрахунок лінії зв'язку, вибір антен і приводу стеження.**

***Ключові слова*: супутниковий зв'язок, наземна станція, програмний спосіб супроводження супутника, алгоритми керування супроводженням.**

**A system of program tracking of a satellite by a ground station is considered. Algorithms of calculating parameters of tracking, calculation of the communication line, choice of the antenna and the tracking drive motors are described.**

***Key words*: satellite communication, ground station, programmed way of tracking, algorithms of control of tracking.