MAGNETOMETER CALIBRATION MANEUVERING MODE OF SPORT CUBESAT

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The SPORT Cubesat is a joint mission between several institutions from United States and Brazil, with main purpose to study the space weather around Earth's orbit. The satellite will carry four experiments: an IVM (Ion Velocity Meter), from UTD (University of Texas at Dallas), a CTECS (Compact Total Electron Content Sensor, based on GPS signal travelling through atmosphere), from Aerospace Corporation, a MVMS (Miniature Vector Magnetometer System) from GSFC (Goddard Space Flight Center), and a set of three detectors: SLP (Sweeping Langmuir Probe), EFP (Electric Field Probe) along with SIP (Sweeping Impedance Probe), from Utah State University (USU). NASA provides the financial support for payload besides launching service on ISS (International Space Station). In Brazil, ITA (Aerospace Technological Institute), is responsible for satellite platform and subsystems specification and design, together with INPE (National Institute for Space Research) which will provide mission control and in orbit operation. Besides the FME, the sensors from USU also incorporate a magnetometer of different technology that requires in orbit calibration. For this purpose the satellite shall perform attitude maneuvers periodically, by rotating the entire platform around its coordinate axes. The calibration process also requires keeping high pointing accuracy during the maneuvers, with a minimum angular rate of 2 rpm. Unfortunately, those requirements can't be both fulfilled with the on board star sensor (SS), since it can point with the required accuracy but the maximum angular rate the star sensor can afford without losing the tracking stars is less than 0.2 rpm. The proposed solution is to integrate the attitude based on the measurements of the gyroscopes (GYR), whose accuracy can be improved by appropriated on ground calibration. The attitude controller is based on reaction wheels (RW) and controls the angular velocity vector with respect to the inertial frame. This paper describes the control algorithm developed to perform the maneuvers with the gyroscopes and reaction wheels, as well as the attitude simulation during each maneuver. Results have shown that the maximum duration of the maneuvers shall be less than 3 minutes for roll and yaw axes and less than 7 minutes for pitch axis.

1. Introduction

The SPORT Cubesat is a mission designed to study the space climate, through various scientific experiments. The mission involves several institutions in the United States and Brazil, including Goddard Space Flight Center (GSFC-NASA), University of Texas at Dallas (UTD), Aerospace Corporation (AC), Utah State University (USU), in USA, in addition to the Technological Institute of Aeronautics (ITA) and the National Institute for Space Research (INPE), both from Brazil. The satellite's scientific payloads are:

- Ion Velocity Meter (IVM) from UTD,
- Miniature Vector Magnetometer System (MVMS) by GSFC,

- CTECS (Compact Total Electron Content Sensor) from Aerospace Corporation,
- Sweeping Langmuir Probe (SLP), Electric Field Probe (EFP), and Sweeping Impedance Probe (SIP) from Utah State University (USU).

ITA is responsible for the platform, while INPE will be responsible for in orbit operation, payload telemetry, data storage and distribution. The platform consists of a Power Supply Subsystem (PSS), a communication system for downlink in UHF and uplink in VHF (TT&C), an X band transponder for payload data telemetry (X-BAND Tx), an Attitude Determination and Control System (ADCS) and an on-board computer & data handling (OBDH). The platform equipments are COTS (Commercial Off-The-Shelf), with flight heritage. The IVM experiment aims to determine the velocity of ions in the upper atmosphere, and, due to the directional aspect of the ion trajectory, this experiment requires precise knowledge of attitude, specified in 0.02º. The attitude determination will be carried out by a star sensor (SS) and MEMS (Micro Electro-Mechanic Systems) gyros (GYR), while the attitude control employ reaction wheels and magnetic torque coils. The attitude determination and control system, ADCS, also has a magneto-resistive magnetometer and a digital sun sensor. In addition to this magnetometer, there are two others on the satellite: the flux gate magnetometer (GSFC payload), and other magneto-resistive for USU payloads. This one needs to be calibrated in orbit to reduce the influence of static bias, which can compromise the quality of measurements. Calibration in orbit must be performed by attitude maneuvers, in which the satellite must rotate at an angular rate of 12 º/s around the magnetometer axes, and shall keep this rate at least for two complete revolutions. To accomplish with this requirement, an attitude operation mode called MICM (Magnetometer Instrument Calibration Manouver) was established. The MICM operating mode needs to be carried out autonomously, with small supervision and ground intervention.

This paper will present the control law for performing these maneuvers, in Section 3, while Section 2 describes the models of the sensors and actuators. Section 4 will present a simulation of the maneuvers.

2. Attitude Sensor and Actuator Models

The MICM operating mode will rely only in the Star Sensor (SS) and gyroscopes (GYR) to accomplish to the attitude maneuvers. The star sensor has a short baffle that provides a 35° half-cone angle of exclusion from the Sun. This means that if the Sun makes an angle of less than the half-cone angle with the optical axis of the Star Sensor, the sensor could not be able to perform attitude determination. In addition, this sensor can also loose the attitude if the angular rate of the satellite is above 1^{o}/s, due to image blurring. Due to restrictions on the satellite faces available for the SS, both the sensor's optical axis and the IVM axis are aligned with the satellite roll axis, as can be seen in Figure 1. In this Figure, and all other figures of this report, the red, green and blue colors refer to the satellite yaw, roll and pitch axes, respectively.

The sensor model considers also that the attitude determination can't be achieved if the Earth's limb enters in the sensor's field of view. The model also takes into account that the Sun can not cause blindness in the SS if the satellite is in the Earth's shadow. Therefore the SS model assumes that the sensor can provide a valid attitude unless the Sun is inside the sensor's exclusion cone, or the Earth's is in the sensor's field of view, or the satellite angular rate is above 1^{o}/s.

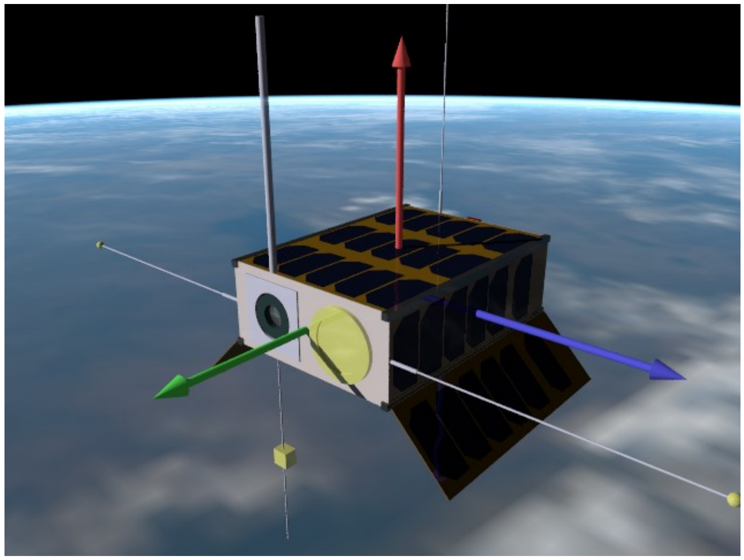

Figure 1: Nominal attitude of SPORT Cubesat.

The Star Sensor measurements are modeled by a Gaussian white noise added to the satellite attitude. If $\boldsymbol{\upsilon}$ is a random vector of the error angles around the SS axes, with σ_υ non correlated standard deviation vector, then the attitude measured by the Star Sensor, considering that υ is small, is given by:

$$\mathbf{q}_{ss} = \frac{1}{2}\mathbf{q}_{b\_i} \otimes \begin{pmatrix} \boldsymbol{\upsilon} \\ 2 \end{pmatrix}, \tag{1}$$

where $\otimes$ indicates the quaternion product [1, 2], and $\mathbf{q}_{b\_i}$ is the attitude quaternion of the satellite with respect to geocentric inertial system. It should be noted that the standard deviation of star sensors differs between the optical axis and the axes located in the focal plane of the image. The standard deviation around the optical axis is, in general, 5 to 10 times greater than the deviations in the focal plane.

The high accuracy of the star sensors allows them to estimate the angular velocity of the satellite, through the numerical derivative of the attitude. The angular velocity is calculated using the kinematics equation in quaternions [3]:

$$\dot{\mathbf{q}} = \frac{1}{2}\boldsymbol{\Xi}(\mathbf{q})\,\boldsymbol{\omega} = \frac{1}{2}\begin{pmatrix} \eta\,\mathbf{1} + \boldsymbol{\varepsilon}^{\times} \\ -\boldsymbol{\varepsilon}^{T} \end{pmatrix}\boldsymbol{\omega}, \tag{2}$$

in which $\mathbf{q} = (\varepsilon \quad \eta)^T$ is the attitude quaternion [2], ω is the angular velocity, $\times$ superscript stands for the cross product matrix, and T superscript is the transpose of the vector. The differential equation of the kinematics can be inverted, since $\boldsymbol{\Xi}^T(\mathbf{q})\,\boldsymbol{\Xi}(\mathbf{q}) = \mathbf{1}$, resulting in the finite difference equation

$$\boldsymbol{\omega}_{ss} = 2\,\boldsymbol{\Xi}^T(\mathbf{q}_{ss})\,\frac{\mathbf{q}_{ss}(t_k) - \mathbf{q}_{ss}(t_{k-1})}{t_k - t_{k-1}}, \tag{3}$$

which obtains the angular velocity of the satellite from two sequential attitude quaternions. The BST ST-200 star sensor [4], selected for the SPORT satellite, has the following characteristics, according to the manufacturer:

- Accuracy in axes orthogonal to the optical axis: < 0.0083º.
- Accuracy on the optical axis: < 0.055º.
- Maximum sampling rate: 5 Hz.
- Field of view (diagonal): 22º.
- Maximum angular speed: 1º / s.

The SPORT satellite has a MEMS Invensense MPU3030 gyros [5]. The mathematical model of the gyro considers an addition of a zero mean, non correlated Gaussian noise η_g with standard deviation vector σ_η and a vector bias $\mathbf{b}_g$, given by [3]:

$$\mathbf{v}_g = \boldsymbol{\omega} + \mathbf{b}_g + \boldsymbol{\eta}_g , \tag{4}$$

where ω is the satellite angular velocity. The gyro bias is not constant, and can be modeled using a stochastic process given by

$$\dot{\mathbf{b}}_g = \boldsymbol{\mu}_g , \tag{5}$$

where μ_g is a zero mean, non correlated Gaussian noise with standard deviation vector σ_μ.

The gyro parameters were obtained by processing the measurements in order to extract the main characteristics using an Allan Variance curve, and resulted in σ_η = (0.0139 0.01450 0.0130) º/s, and σ_μ = (0.00350 0.00087 0.00066) º/s^2.

The attitude determination algorithm assumes that both attitude and angular rate are provided by the Star Sensor. If the attitude is valid, them the gyro bias is estimated by a first-order low pass filter with the SS angular rate:

$$\tilde{\mathbf{b}}_g(t_k) = \alpha\, \tilde{\mathbf{b}}_g(t_{k-1}) + (1-\alpha)\,(\mathbf{v}_g - \boldsymbol{\omega}_{ss}) , \tag{6}$$

in which α = 0.995 is the low pass filter gain, adjusted so that the time constant of the filter is close to 100 s, also close to the measured time constant of the gyro biases. When the Star Sensor does not provide a valid attitude, the gyro is used instead. The attitude is then computed by

$$\mathbf{q}_g(t_k) = \mathbf{q}_g(t_{k-1}) + \frac{1}{2}\boldsymbol{\Xi}(\mathbf{q}_g(t_{k-1}))\, \boldsymbol{\omega}_g\, (t_k - t_{k-1}) , \tag{7}$$

such that $\mathbf{q}(0)$ is provided by the last valid attitude of the Star Sensor, and

$$\boldsymbol{\omega}_g = \mathbf{v}_g - \tilde{\mathbf{b}}_g , \tag{8}$$

where $\tilde{\mathbf{b}}_g$ is the last estimated bias of the gyro, given by Equation 6.

The reaction wheels will be commanded by speed control. The RW electronics controls the motor current, therefore the torque, with a closed loop controller based on the error between the measured wheel speed and the commanded reference speed. This closed loop control reduces the non linear friction effects of the bearings on the wheel, but does not eliminate them. Since there is no information concerning the dynamics of the RW, they were modeled by the angular momentum law:

$$g_c = I_w \frac{w_w - v}{\Delta t} , \tag{9}$$

in which g_c is the motor torque, w_w is the RW reference speed, I_w is the flywheel inertia moment, and v is the measured wheel rate, given by

$$v = \begin{cases} \omega + \upsilon, \text{ se } |\omega| > \omega_{min} \\ 0, \text{ se } |\omega| \leq \omega_{min} \end{cases}, \tag{10}$$

where ω is the rotor's speed, ω_{min} is a dead band speed due to Coulomb friction, and υ is a Gaussian noise due to the measurement process. Besides this both the torque g_c and the angular rate ω are limited in a fixed range due to design constraints. SPORT satellite will have 4 RW arranged in a pyramidal configuration.

3. Magnetometer Instrument Calibration Maneuvers

The operation mode for the magnetometer calibration arises from a requirement from USU probes, which needs that the satellite rotates around its Cartesian axes. This requirement states that:

a) The platform must provide the means to carry out satellite rotation maneuvers around the Cartesian axes, that is, around the satellite's roll, pitch and yaw axes.
b) The rotation speed must be greater than or equal to 12 º/s.
c) Rotations on each axis must not exceed 5 minutes.

Additionally, the following requirements were added:

d) The maneuvers must be carried out in both positive and negative axes.
e) At least two complete rotations must be carried out on each axis, starting from the establishment of the reference angular speed, unless there is a conflict with requirement c.
f) The maneuvers must be carried out using the reaction wheels, without the use of magnetic torque coils to avoid magnetic contamination of the platform.
g) Due to the strong magnetic dependence, a magnetic cleanness requirement was imposed to the platform, which led to the avoidance of magnetic materials on board.

To perform in orbit magnetometer calibration, one should note that the Earth's magnetic field describes a conning motion around the spin axis, during the maneuver, so that the projection of the vector on the plane orthogonal to this axis rotates around the origin describing a circle. If the magnetometer is biased, the vector is displaced from origin, and the center of the circle indicates the bias value in two axes. By repeating the same procedure on the other spacecraft axes the total bias vector can be determined.

The necessary condition for entering the MICM operating mode is that the satellite has a pointing error less than 0.1º in all axes and an angular velocity below 0.001 rad/s with respect to the orbital coordinate system. This condition must be verified by telemetry on the ground or, possibly, also by the ADCS software. In addition, it must be ensured by telemetry that the star sensor is providing valid measurements before starting the maneuver. After ending the maneuver, the attitude controller shall maintain the satellite nominal pointing up to receiving a ground command to enter the safe or nominal mode.

To avoid possible automatisms in the embedded attitude control program, which could increase the complexity of the software and reduce reliability, the maneuvers must be carried out under ground monitoring, with separate commands for executing the maneuver on each of the three axes, and an interruption command. Earth presence in the SS field of view can be avoided if the maneuvers are carried out exclusively on vertical axes or on the roll axis (Figure 1). Likewise, the presence of the Sun in the exclusion zone of this sensor can be avoided by performing night time maneuvers. The required angular rate for MICM is higher than the maximum speed expected the star sensor can perform attitude determination, and, therefore, a significant part of the maneuver should be expected to occur under exclusive control of the gyros (Equations 7 and 8). This will undoubtedly be a critical factor, as MEMS gyros usually do not have the stability necessary to ensure good performance of the attitude control.

The MICM maneuvers require inertial attitude stabilization due to gyroscope stiffness during spacecraft spin. Therefore, there shall be a transition from geocentric to inertial and back to geocentric stabilization in this operating mode. The maneuvers shall be performed on the orbital reference system (local horizontal and vertical axes). Rotation around roll and yaw axes shall begin in nominal attitude, as shown in Figures 1. In this situation the star sensor optical axis will always be close to the horizontal plane during maneuver. As for MICM around pitch axis, a previous maneuver shall be performed to point this axis to local zenith, as can be seen in Figure 2. Then, a regular rotation shall be commanded around the satellite pitch axis. At the end of the maneuver the pitch axis shall return to its initial orientation. Therefore, the optical axis of the star sensor remains inertially fixed during roll maneuver, while it describes a rotational motion in the local horizontal plane in yaw and pitch maneuvers. Attitude disturbances like aerodynamic, solar pressure, gravity gradient and residual magnetic moment were considered in simulation, although their effects on attitude can be barely seem due to small magnitude of the torques.

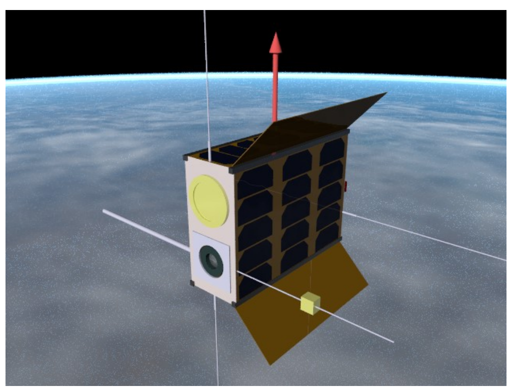

Figure 2: Previous pointing maneuver for MICM around pitch axis.

The maneuvers shall therefore be carried out according to the following sequence:

1) Step 1 – Initial pointing: The attitude shall be controlled as in nominal mode, with geocentric stabilization, and attitude determination being provided by the star sensor. Final condition is reached when the attitude pointing error obtained from an angle of an Euler angle and axis attitude is less than 0.5°. A quaternion feedback PD control law is adopted for this step, given by

$$\mathbf{u}_{pd} = -\mathbf{K}_p \, \boldsymbol{\varepsilon}_{o\_b} - \mathbf{K}_d \, \boldsymbol{\omega}_o , \qquad (11)$$

in which $\mathbf{u}_{pd}$ is the torque to be applied to reaction wheels, $\mathbf{K}_p$ and $\mathbf{K}_d$ are diagonal gain matrixes, $\boldsymbol{\varepsilon}_{o\_b}$ is the vector part of the attitude quaternion $\mathbf{q}_{o\_b} =$

$\mathbf{q}_{o\_i} \otimes \mathbf{q}_{i\_b}$ and $\omega_o = \omega_i - \mathbf{C}_{i\_o}\, \Lambda_o$ is the satellite angular rate with respect to the orbital frame. On the other hand, $\mathbf{q}_{o\_i}$ is the quaternion of the orbital frame relative to the inertial system, while $\mathbf{C}_{i\_o}$ is the direct cosine matrix corresponding to $\mathbf{q}_{o\_i}$. Λ_o is the orbit angular rate, such that $\mathbf{\Lambda}_o = (0 \quad 0 \quad \dot{f})^T$ where $\dot{f}$ is the time derivative of the true anomaly angle. Now $\mathbf{q}_{i\_b}$ (attitude quaternion) and ω_i (inertial angular rate) are equal to $\mathbf{q}_{ss}$ and ω_{ss}, respectively, when the Star Sensor provides a valid attitude, or equal to $\mathbf{q}_g$ and ω_g otherwise. The control gains $\mathbf{K}_p$ and $\mathbf{K}_d$ were adjusted, as well as other control gains present in this report, by trial based on analysis of the achieved performance in simulations.

2) Step 2 – Axis reorientation: This step aims to perform an attitude maneuver in order to orient conveniently the satellite maneuvering axis to the appropriated direction. Axis reorientation is necessary for pitch axis only. Final condition is reached when the maneuvering axis makes an angle less than 0.5° with the inertial direction for the rotating axis (local horizontal or vertical). The attitude control law is the same one of Step 1, except that the reference attitude is rotated 90° around the roll axis (Figure 2) to point the spacecraft pitch axis to zenith, for pitch maneuvering axis. Roll and yaw maneuvers remain unchanged.

3) Step 3 – Acceleration: This step will increase the spacecraft angular rate around the maneuvering axis, until it reaches w = 12°/s, using a velocity vector attitude control. During this step the attitude determination is switched from star sensor to gyros. Final condition is achieved when the error of the satellite angular velocity magnitude is less than 0.1°/s. The velocity vector control is a proportional controller on the satellite angular velocity vector, such that

$$\mathbf{u}_{am} = k_{amd}\, I_{nn}\, (w\, \mathbf{e}_n - \boldsymbol{\omega}_i)\,, \tag{12}$$

where $\mathbf{u}_{am}$ is the control torque for reaction wheels, k_{amd} is the control gain, I_{nn} is the inertia moment of the satellite in the maneuvering axis, and $\mathbf{e}_n$ is the unit vector of the maneuvering axis in spacecraft coordinates (x, y or z axis). The geocentric stabilization is changed to an inertial one in this step.

4) Step 4 – Coasting: In this step the spacecraft maneuvering axis remains inertially fixed and controlled at 12°/s. In this step the magnetometer will be sampled for in orbit calibration. Final condition is reached after two complete revolutions of the satellite around this axis, computed by numeric integration of the angular velocity:

$$\varphi = \sum_t \boldsymbol{\omega}_i^T\, \mathbf{e}_n\, \Delta t\,, \tag{13}$$

The control law for this step is the same one used in Step 3.

5) Step 5 – Rate reduction: This step decelerates the satellite up to a complete stop with respect to the inertial frame. Final condition is accomplished when the magnitude of the angular velocity is less than 0.3°/s, and the control law is just an angular rate control, given by

$$\mathbf{u}_{am} = -k_{avp}\, I_{nn}\, \boldsymbol{\omega}_i\,, \tag{14}$$

where k_{amd} is the control gain for speed reduction.

6) Step 6 – Final pointing: To orient the spacecraft to the nominal attitude. Final condition is reached when the attitude error measured by the attitude pointing error is less than 1°. However, the attitude control remains in this step up to receiving a ground command to commute to safe or nominal mode. Attitude control law is identical to Step 1.

3. Results from simulation

The MICM maneuvers were simulated in Matlab with the PROPAT toolbox [6]. Although the maneuvers were simulated on x, y and z axis satellite axes, only the MICM around z will be presented. The x and y axes have similar behavior and performance when compared to the z axis.

The simulation of the maneuver around the pitch axis (z) is shown in Figures 3 to 6. This maneuver takes longer (380 s) than on the remaining axes (180 s), due the reorientation of the pitch axis to zenith. The attitude Euler angles of a x-y-z sequence is show in Figure 3. In this figure the numbers and the vertical dashed lines indicate the Steps 1 to 6 for the maneuver, as explained above. Initial pointing takes only few seconds, since the initial condition for the attitude is already the nominal one. The angle on y axis (green) grows up to 90° during the maneuver, and remains almost constant during Step 4, as well as the angular velocity, which is shown in Figure 2. The reaction wheels produce some oscillations in the angular rate, in this figure, due to dead band crossings [7]. Figure 5 shows the simulated Star Sensor output flag for the MICM maneuver on pitch axis. In this figure the Earth presence in sensor's FOV is shown in blue, and the velocity flag (maximum angular rate of 1°/s) for this sensor appears in green. This last flag is critical for MICM maneuvers, since the angular velocity is higher than 1°/s during most of the time, and therefore attitude propagation and control is done mainly by the gyros. Figure 6 presents a simulation of the magnetometer output during the whole maneuver. It can be noted that the component on the pitch axis (blue curve) is approximately constant, and, on Step 4, it is clear the expected 90° out of phase behavior of the x and y magnetometer channels. Due to the high gyro bias and noise, the spin axis direction, which should remains fixed, presents a drift motion and describes a circle of 6° radius in inertial frame.

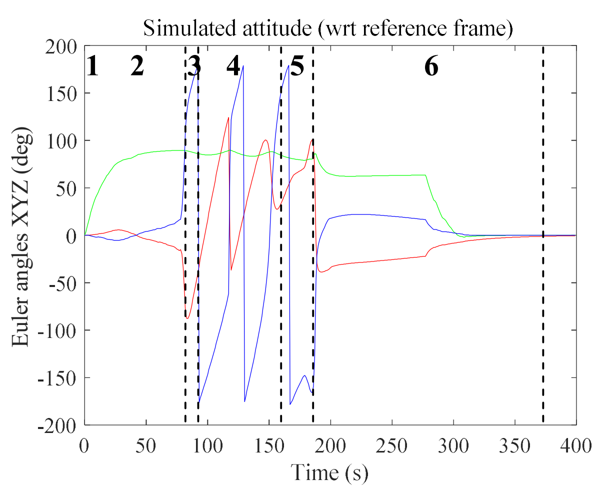

Figure 3: Euler angles for MICM around pitch axis.

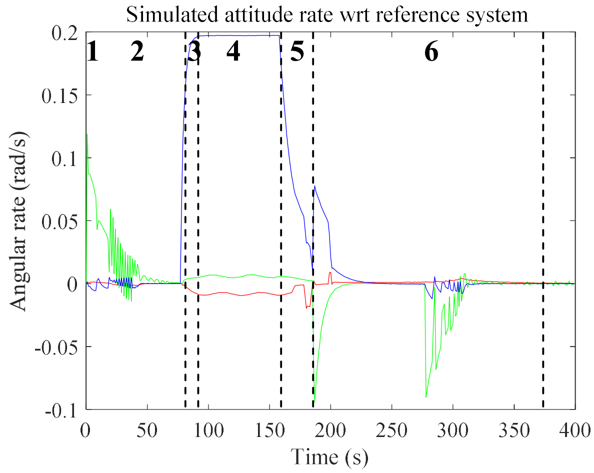

Figure 4: Satellite angular velocity during MICM around pitch axis.

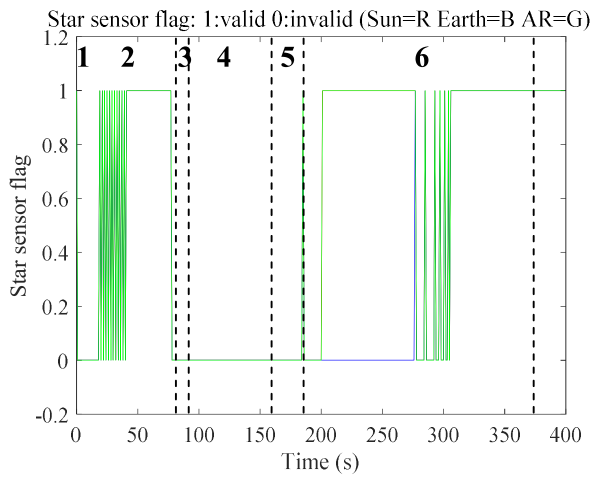

Figure 5: Star Sensor output flag during MICM maneuvering on pitch axis.

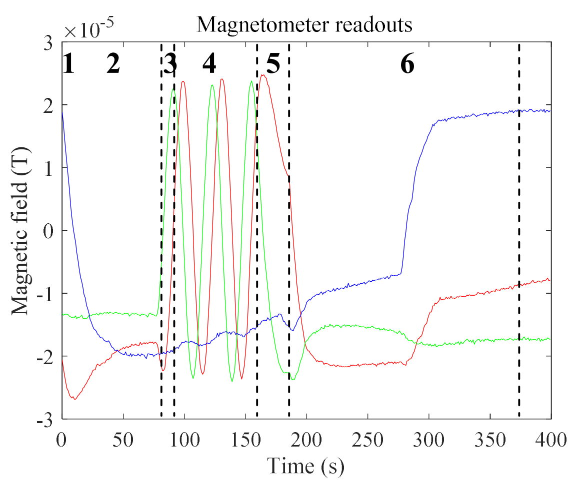

Figure 6: Simulation of the magnetometer during MICM around pitch axis.

Figure 7 shows the RW speed during maneuver. It is clear that any wheel reaches the maximum speed of 4500 rpm during the maneuver. The torque also remains bellow the maximum available torque (not shown in figures).

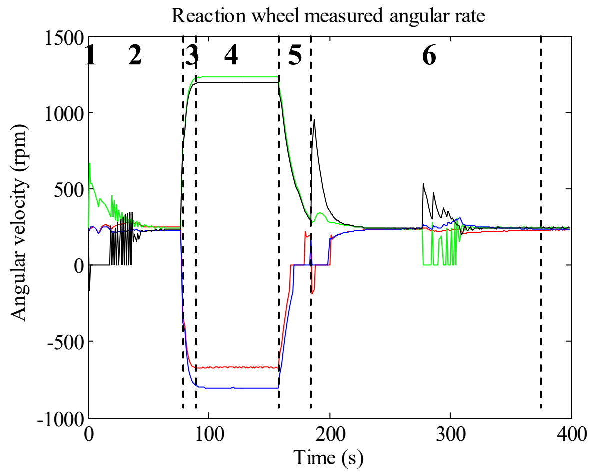

Figure 7: Simulation of the reaction wheel speeds.

5. Conclusions

Cubesats satellites are in fast expansion, due to its low cost, fast prototyping, standard interfaces, modularized components and easy access to space. The SPORT mission is a Cubesat that aims to confirm that they can be used for big science data acquisition. However, the demanding high reliability of this mission makes the ADCS (Attitude Determination and Control System) more complex and more expensive. In addition, design shall be more robust, and software validation shall be more elaborate in order to accomplish with mission needs. This includes the MICM (Magnetometer Instrument Calibration Mode) operating mode of ADCS, which shall demonstrate by simulation that the required attitude maneuver can be accomplished even under any critical situation, like Star Sensor temporary unavailability, for instance.

This work showed how the MICM operating mode was conceived and designed, as well as the control law at each step of the maneuver, and simulation results. Although this mode requires some in orbit autonomy, adopted procedures for ground operation and command can mitigate the risks. Simulation has confirmed that MICM attitude maneuvers for in orbit magnetometer calibration can be performed with the required accuracy. Care was taken to precisely model the sensors (Star Sensor and Gyros) as well as the actuator (Reaction Wheels). The MICM maneuvers lasts from 3 up to 6 minutes, which confirms that they can be monitored from ground to minimize any hazard to the spacecraft.

References

[1] Wertz, J. R. “Spacecraft attitude determination and control”. D. Reidel Publishing, 1978..

[2] Hughes, P. C. “Spacecraft Attitude Dynamics”. Mineola: Dover, 1986.

[3] Lefferts, E. J.; Markley, F. L.; Shuster, M. D. “Kalman Filtering for Spacecraft Attitude Estimation”, Journal of Guidance, Vol. 5, No. 5, 1982, pages 417-429.

[4] Berlin Space Technologies GmbH. “ST200/ST400 Family - Interface Control Document”, BST, Berlin, 2017. ST200-TN04-004.

[5] Invensense, MPU-3300 Product Specification (Revision 1.1). Invensense Inc. Datasheet. 2011.

[6] Carrara, V. An Open Source Satellite Attitude and Orbit Simulator Toolbox for Matlab. DINAME 2015 – XVII International Symposium on Dynamic Problems of Mechanics. Proceedings… Natal, RN, Brazil, Feb 22-27, 2015. (ISSN 2316-9567). Disponível em: http://www.dem.inpe.br/~val/publicacoes/DIN-2015-0055.pdf.

[7] GomSpace. “NanoTorque GSW-600 Manual”. GomSpace A/S, Denmark, 2017. gs-man-nanotorque-gsw600-1.1