Proceedings of the 5th WSEAS International Conference on Applied Computer Science, Hangzhou, China, April 16-18, 2006 (pp1024-1027)

Magnetorquer Control for Orbital Manoeuvre of Low Earth Orbit Microsatellite

A.M. SI MOHAMMED, A. BOUDJEMAI, S. CHOURAQUI
Division de Mécanique Spatiale
Centre National des Techniques Spatiales
1, Avenue de la Palestine, ARZEW
ALGERIA

M. BENYETTOU
Université des Sciences et de la Technologie «Mohamed Boudiaf »
Laboratoire de Modélisation et Simulation, ORAN
ALGERIA

Abstract:-A propulsion system is required in order for the spacecraft to be able to operate as a part of the constellation, and is used to carry out initial launcher injection corrections, spacecraft separation into their respective orbital slots, altitude maintenance and finally an end of life manoeuvre to remove the spacecraft from the operational system also these manoeuvres allow the satellite to alter its semi major axis and the eccentricity of the orbit. The propulsion system fires through the centre of gravity to perform orbit changes manoeuvres.
This paper describes the magnetorquer attitude damping propulsion disturbances of low Earth Orbit microsatellite for orbit maintenance with the following points: (1) Mathematical modelling including attitude dynamics and magnetorquer control law, (2) Simulation results will be presented to evaluate the performance and design objectives and finally the conclusion.

Keywords:- Attitude, Control, Magnetorquer, Propulsion, Microsatellite.

1 Introduction

The 3 axis magnetic torquer coils will be used primarily for attitude control and momentum damping of libration damping thruster disturbances. In this study the main focus will be on the attitude control function of the magnetic torquers. The X and Y axis torquers are air coils embedded into channels around the edges of the solar panels and the Z axis into channels within the top facet of the satellite. For redundancy and torque control reasons there are two coils per axis and they can be switched on singly or in parallel. The coils can each be fed with a constant current to generate a vector magnetic dipole moment **M**. The vector dipole moment **M** from all coils will interact with the magnetic field vector **B** to generate a magnetic torque vector $\mathbf{N_M}$ by taking the vector cross product.

Although the direction and magnitude of **M** can be controlled on average by the correct interleaving of three orthogonal coils, the **B** vector is totally dependent on the orbital location. The torque $\mathbf{N_M}$ will always be orthogonal to **B** and not favourable in certain regions of the orbit to control the attitude of a specific spacecraft axis. It is also possible that a desirable control torque for a certain attitude axis (pitch, roll or yaw), when a specific coil or combination of magnetic coils are switched, will generate undesired disturbance torques for the other axis. The main objective of the control algorithms in this study will be to optimise the control effort by maximising the desired influence and minimising the undesired disturbances.

The only known 3 axis stabilisation control algorithm for passive gravity gradient with active magnetic torquing [3], the algorithm makes use of the cross product law, this method tends to choose the most favorable magnetorquing direction at any control instant by interleaving or simultaneously switching any of the three orthogonal magnetic coils, relying on the current direction of the local geomagnetic field vector. Depending on the required torque vector and the given geomagnetic field vector, this most favorable magnetorquing vector unfortunately does not exclude the generation of unfavorable cross disturbances.

2 Mathematical Modelling

The Euler parameters are an attitude parameterization, also referred to quaternion as they are essentially the same. The quaternion is the attitude parameterization preferred in most computational aspects. It has four parameters, no singularities, and the constraint is easy to uphold. A rotation of an angle θ around an axis α gives the rotation vector $\theta\alpha$ and the quaternion is given as follows

$$\mathbf{q} = \begin{bmatrix} \eta \\ \varepsilon \end{bmatrix} = \begin{bmatrix} \cos\frac{\theta}{2} \\ \alpha\sin\frac{\theta}{2} \end{bmatrix} \tag{1}$$

Where the constraint required is described by one of the following

$$\|\mathbf{q}\| = 1 \tag{2.1}$$

$$\varepsilon^T\varepsilon + \eta^2 = 1 \tag{2.2}$$

$$\varepsilon_1^2 + \varepsilon_2^2 + \varepsilon_3^2 + \eta^2 = 1 \tag{2.3}$$

The main advantage of the quaternion is that rotations are expressed by the quaternion product. The quaternion product is defined as follows

$$\begin{aligned} \mathbf{q}_1 \otimes \mathbf{q}_2 &= \begin{bmatrix} \eta_1 \\ \varepsilon_1 \end{bmatrix} \begin{bmatrix} \eta_2 \\ \varepsilon_2 \end{bmatrix} \\ &= \begin{bmatrix} \eta_1\varepsilon_2 + \eta_2\varepsilon_1 + \mathbf{S}(\varepsilon_1)\varepsilon_2 \\ \eta_1\eta_2 \quad \varepsilon_1^T\varepsilon_2 \end{bmatrix} \end{aligned} \tag{3}$$

When $\mathbf{q}_1$ represents an attitude, and $\mathbf{q}_2$ a rotation, the new attitude is given by the quaternion product as follows

$$\mathbf{q} = \mathbf{q}_1 \otimes \mathbf{q}_2 \tag{4}$$

Differentiating the quaternion yields

$$\begin{cases} \dot{\eta} = -\frac{1}{2}\varepsilon^T\omega \\ \dot{\varepsilon} = \frac{1}{2}(\eta\mathbf{I} + \mathbf{S}(\varepsilon))\omega \end{cases} \tag{5}$$

Where ω is the angular velocity of the body frame in relation to the frame as attitude is given, given in body frame.

The dynamics of the spacecraft in inertial space is governed by Euler's equations of motion can be expressed as follows in vector form [1], [5]

$$\mathbf{I}\dot{\omega} = \mathbf{N_{GG}} + \mathbf{N_D} + \mathbf{N_M} + \mathbf{N_T} - \omega \times (\mathbf{I}\omega + \mathbf{h}) - \dot{\mathbf{h}} \tag{6}$$

Where ω, $\mathbf{I}$, $\mathbf{N}_{GG}$, $\mathbf{N}_D$, $\mathbf{N}_M$ and $\mathbf{N}_T$ are respectively the inertially referenced body angular velocity vector, moment of inertia of spacecraft, gravity-gradient torque vector, applied magnetorquer control firing, unmodelled external disturbance torque vector such as aerodynamic or solar radiation pressure.

The 3-axis stabilisation control algorithm for passive gravity gradient with active magnetic tends to choose the most favourable magnetorquing direction at any control instant by interleaving or simultaneously switching any of the three orthogonal magnetic coils, relying on the current direction of the local geomagnetic field vector.

Then the most favourable magnetorquing vector $\mathbf{M}$ is expressed [2], [4] as:

$$\mathbf{M} = \frac{(\mathbf{K_p}(\mathbf{a} - \mathbf{a}_{cmd}) + \mathbf{K_d}(\dot{\mathbf{a}} - \dot{\mathbf{a}}_{cmd})) \times \mathbf{B}}{\|\mathbf{B}\|^2} \tag{7}$$

Where

$\mathbf{B}$: Body magnetic field vector from a Magnetometer;

$\mathbf{K}_p$: Proportional gain diagonal matrix;

$\mathbf{K}_d$: Derivative gain diagonal matrix;

$\mathbf{a}$, $\dot{\mathbf{a}}$: Attitude and rate error vectors;

$\mathbf{a}_{cmd}$, $\dot{\mathbf{a}}_{cmd}$: Attitude and rate command vectors.

The attitude is obtained from a full state extended Kalman filter [6]. This filter take measurement vectors (in the frame body) from magnetometer and sun sensor and by combining them with corresponding modelled vectors (in a reference frame), estimate the attitude and attitude rate. The extended Kalman filter estimator is implemented for earth-pointing spacecraft undergoing only small rotation angles. The system model used in this estimator is based on Euler angles, and simplified in order to reduce the complexity and processing time for accommodation on an on-board processor that has limited memory space.

3 Simulation Results

The results presented in this paper were obtained with a simulator that implements the dynamics and kinematics of the satellite using C code, MATLAB and SIMULINK. The satellite in a low Earth orbit was used as an example during these simulations.

The satellite dynamics are modelled using Euler's equations for a rigid body motion under the influence of internal and external torques. A 98° inclination, circular orbit at an altitude of 860 km was used during the simulation tests.

The magnetic moment in the orthogonal X, Y and Z-axis was assumed to be equal to 10 Am^2 each.

An IGRF model was used to obtain the geomagnetic field values.
The following sources of torque are all ignored due to the fact that they are all very small compared to the main gravity and magnetic torque.

- The gravity torque model does not take into account the tidal forces created by the earth - moon system.
- The sun radiates a vast amount of particles known as solar wind and electromagnetic particles known as solar pressure. Both solar wind and solar pressure will establish a torque on the satellite
- As the satellite orbits in LEO, the atmosphere is still present and the atmospheric drag will be none-zero
- The satellite itself will generate different torques. The deployment of boom will produce great torques, but they will be short lived. All electric components on board might produce electromagnetic fields interacting with the earth's magnetic field in the same way as the control torque.

Figures 1 and 2 present the attitude magnetorquer libration damping of thruster disturbances for multiple firings.
Figure 3 present the B-Field measurement during magnetorquer libration damping thruster disturbances for multiple firings.
We assume that the thruster is in the X-axis with the following torque Nx=0.000005Nm, Ny=0.0015Nm, Nz=0.0015Nm. The firing time of the thruster is 30 seconds at 30000 seconds and 10 seconds at 36000 seconds. For 30 seconds firing the yaw angle is achieving a disturbance angle of 180 degrees and the roll angle and pitch angle are achieving a disturbance angle of 10 degrees. Damping of the attitude disturbances is achieved within 1.5 orbits.
Figure 4 and 5 present the attitude magnetorquer libration damping of thruster disturbances for a single firings.
Figure 6 present the B-Field measurement during magnetorquer libration damping thruster disturbances for single firing.
The firing time of the thruster is 1 minute at 30000 seconds. For 1 minute firing the yaw angle is achieving a disturbance angle of 180 degrees and the roll angle and pitch angle are achieving a disturbance angle of 6 degrees. Damping of the attitude disturbances is achieved within a 6 orbits.

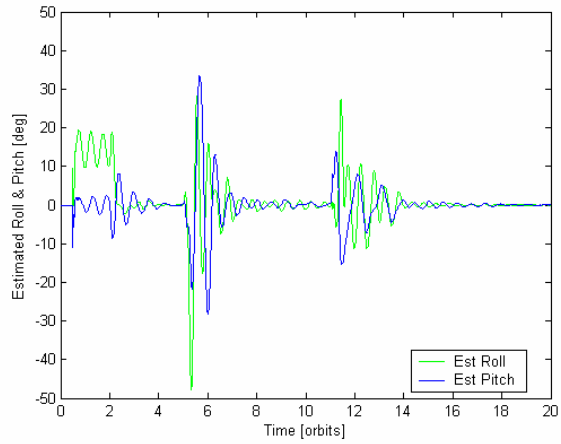

Fig. 1 Roll and Pitch Attitude during magnetorquer libration damping thruster disturbances for multiple firings

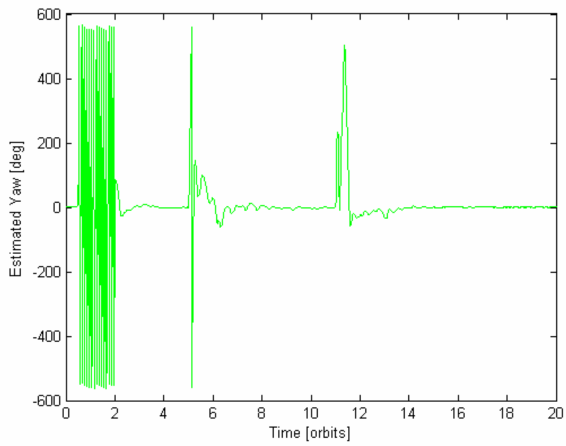

Fig. 2 Yaw Attitude during magnetorquer libration damping thruster disturbances for multiple firings

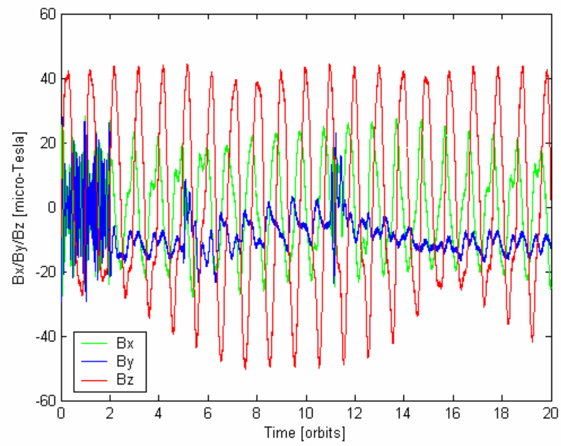

Fig. 3 B-Field measurement during magnetorquer libration damping thruster disturbances for multiple firings

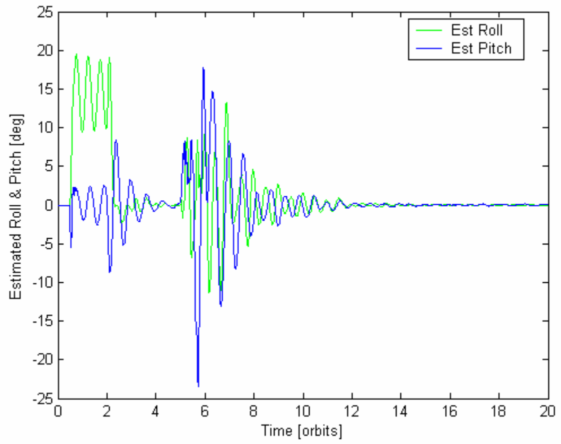

Fig. 4 Roll and Pitch Attitude during magnetorquer libration damping thruster disturbances for single firing

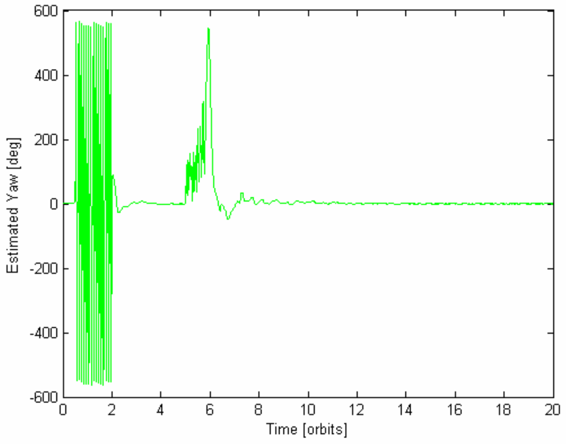

Fig. 5 Yaw Attitude during magnetorquer libration damping thruster disturbances for single firing

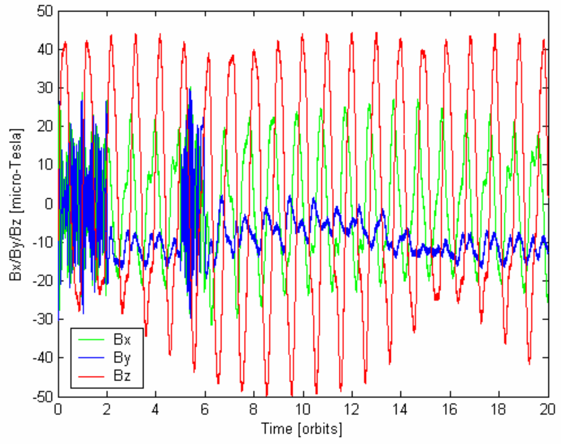

Fig. 6 B-Field measurement during magnetorquer libration damping thruster disturbances for single firing

4 Conclusion

Misalignment of the thrust vector to the center of mass of the satellite can cause significant disturbances to the attitude.

For magnetorquer libration damping of thruster disturbances, w e obtain the following results.

- For 30 seconds firing the yaw angle is achieving a disturbance angle of 180 degrees and the roll angle and pitch angle are achieving a disturbance angle of 10 degrees. Damping of the attitude disturbances is achieved within 1.5 orbit.
- For 1 minute firing the yaw angle is achieving a disturbance angle of 180 degrees and the roll angle and pitch angle are achieving a disturbance angle of 6 degrees. Damping of the attitude disturbances is achieved within six orbit.

References:

[1] W.H. Steyn, “A Multi-Mode Attitude Determination and Control System for Small Satellite”, Ph.D (Engineering), University of Stellenbosch, December 1995.

[2] Hodgart M.S. "Gravity Gradient and Magnetorquing Attitude Control for Low Cost Low Earth Orbit Satellites- the UoSAT Experience", Ph.D. Submission at University of Surrey, June 1989.

[3] Martel F., Pal P.K. and Psiaki M.,"Active Magnetic Control System for Gravity Gradient Stabilized Spacecraft", *Proceedings of the 2th AIAA/USU Conference on Small Satellites*, Utah State University, Sept.1988.

[4] Stickler A.C. and Alfriend K.T., "An Elementary Magnetic Attitude Control System", *AIAA Mechanics and Control of Flight Conference*, Anaheim California, 5-9 Aug. 1974.

[5] Wertz, J.R. 1991. *Space Mission Analysis and Design*. Space Technology Library, Kluwer Academic Publishers, Dordrecht – Boston – London.

[6] Y. Hashida, ADCS *Design for Future UoSAT Standard Platform*, Surrey Space Centre, Guilford, UK, August 2004.