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Nadir Stabilised System of the Alsat-1 First Algerian Microsatellite in Orbit

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Abstract: The National Centre of Space Techniques (NCTS) has initiated a project to develop and propagate microsatellite technologies in Algeria. As part of this initiative, NCTS has worked with Surrey Satellite Technologies Limited (SSTL) to develop a 100 kg class enhanced microsatellite, Alsat-1. With the successful completion of this project, NCTS will be capable of designing its own satellites, covering all phases from design to production and in orbit operation. Alsat-1 was successfully launched from Plesetsk Cosmodrom in Russia in 28 November 2002. In this study the attitude determination and control system in orbit result of Alsat-1 is presented.

Key words: Microsatellite, system, alsat-1, orbit, attitude, sensor, actuator, control

INTRODUCTION

The launch of Alsat-1 in November 2002 saw the first Disaster Monitoring (DMC) satellite successfully placed in orbit. This was followed in September 2003 by the highly successful launch of 3 satellites, BILSAT (Turkey), NigeriaSat-1 (Nigeria) and UK-DMC (United Kingdom), on a Cosmos launcher from Plesetsk, Russia (Mohammad *et al*., 2006).

The Disaster Monitoring Constellation (DMC) is designed to achieve daily imaging of any part of the world from a constellation of 5 satellites. Each satellite is owned and operated by a single nation, but by cooperating together in the DMC Consortium they can achieve daily imaging worldwide, greatly enhancing the capability of their national space asset.

The DMC demonstrate the small satellites can involve a wide range of nations in worthwhile space programs. Earth Observation (EO) missions are particularly well suited to stimulate the development of sustainable resource management and environmental monitoring and can aid the development of new national industries and space capabilities (Mohammad *et al*., 2003).

Alsat-1 is 3-axis stabilised, using a pitch momentum wheel and yaw reaction wheel, with dual redundant 3-axis magnetorquers. A gravity gradient boom is employed to provide a high degree of system stability. Two vector magnetometers and four dual sun sensors are carried in order to determine the attitude to better than ±0.25 deg (Mohammed *et al*., 2004).

MATHEMATICAL MODELLING

Satellite attitude dynamics: In general, spacecraft attitude can be expressed by direction cosine matrix. Using Euler rotations to describe the spacecraft motion with respect to spacecraft body coordinates is commonly not as convenient for numerical computation as the Euler symmetric parameters known as attitude quaternion. In Euler angle expression, there are 12 possible sequence of rotation. For 2-1-3 Euler rotation used in this paper, the attitude transformation matrix can be described as follows (Wertz, 1992),

$$\mathbf{A}_{213} = \begin{bmatrix} c\psi c\theta + s\psi s\phi s\theta & s\psi c\phi & -c\psi s\theta + s\psi s\phi c\theta \\ -s\psi c\theta + c\psi s\phi s\theta & c\psi c\phi & s\psi s\theta + c\psi s\phi c\theta \\ c\phi s\theta & -s\phi & c\phi c\theta \end{bmatrix} \quad (1)$$

Where

c : cosine function; s : sine function;
ϕ: roll angle ; θ: pitch angle ; Ψ: yaw angle.

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Dynamic equations: The basic equation of attitude dynamics relates the time derivative of the angular momentum vector, dL/dt, to the external disturbance torque, N, known as Euler's equation of motion. A spacecraft equipped with reaction or momentum wheels is not a rigid body. The dynamic equation is given by (Hashida, 2004; Mohammed *et al*., 2006)

$$\frac{dL}{dt} = I\dot{\omega} = N_{dist} - \dot{h} - \omega \times (I\omega + h) \tag{2}$$

Where I is an inertia tensor of satellite, ω is an inertial angular velocity vector with respect to the body axis coordinate, h is the angular momentum of the wheel. N_d is the external disturbance torque vector given as

$$N_{dist} = N_{GG} + N_m + N_{aero} + N_{sol} \tag{3}$$

Where N_{GG}, N_m, N_{aero} and N_{sol} are respectively the gravity gradient torque, the magnetic torque, the aerodynamic torque and the solar radiation pressure torque.

If a satellite does not have any cross terms of inertia tensor, the Eq. 2 is explicitly re-written by:

$$\begin{aligned} I_x\dot{\omega}_x &= N_x - \left(I_z - I_y\right)\omega_y\omega_z - h_y\omega_z \\ &\quad + h_z\omega_y - \dot{h}_x \end{aligned} \tag{4.a}$$

$$\begin{aligned} I_y\dot{\omega}_y &= N_y - \left(I_x - I_z\right)\omega_x\omega_z - h_x\omega_z \\ &\quad + h_z\omega_x - \dot{h}_y \end{aligned} \tag{4.b}$$

$$\begin{aligned} I_z\dot{\omega}_z &= N_z - \left(I_x - I_y\right)\omega_x\omega_y - h_y\omega_x \\ &\quad + h_x\omega_y - \dot{h}_z \end{aligned} \tag{4.c}$$

The attitude of Alsat-1 was estimated using a Euler angles (small libration version) based Extended Kalman Filter (EKF) (Mohammed *et al*., 2005). This filter uses measurement vectors (in the body frame) from all the attitude sensors and by combining them with corresponding modelled vectors (in a reference frame), it estimates the attitude of the satellite.

The attitude sensors (magnetometer, sun sensor) will be used to determine the attitude of the satellite relative to the orbital frame. When using magnetic field data: A GPS receiver or an orbital propagator is used to obtain the position of the satellite. Using this position data, a model of the geomagnetic field, the International Geomagnetic Reference Field (IGRF) model, computes the geomagnetic B-field in orbit coordinates. On the other hand, the magnetic B-field is also measured by the 3-axis magnetometer in body coordinates. The attitude can then be solved from these two vectors over time.

In order to damp nutation of Alsat-1 after boom deployment, it is required to obtain roll, pitch and yaw rate and attitude knowledge. Therefore we have to design an estimator including Y/Z wheel under the restriction of the processing power of an OBC 186 (Wertz, 1992; Mohammed *et al*., 2005).

The satellite is axially symmetric and the small libration angle can be assumed, then the system equation becomes (Mohammed *et al*., 2005).

$$\begin{aligned} \ddot{\phi} &= (4\omega_0^2(\frac{I_z}{I_T} - 1) + \frac{h_y\omega_0}{I_T})\phi + (\frac{I_z}{I_T}\omega_0 + \frac{h_y}{I_T})\dot{\psi} \\ &\quad + \frac{h_z\omega_0}{I_T} - \frac{h_z}{I_T}\dot{\theta} + \frac{N_x^{MT}}{I_T} + w_x \end{aligned} \tag{5.a}$$

$$\begin{aligned} \ddot{\theta} &= 3\omega_0^2(\frac{I_z}{I_T} - 1)\theta + \frac{h_z}{I_T}\dot{\phi} - \frac{h_z\omega_0}{I_T}\psi - \frac{\dot{h}_y}{I_T} \\ &\quad + \frac{N_y^{MT}}{I_T} + w_y \end{aligned} \tag{5.b}$$

$$\ddot{\psi} = (-\frac{h_y}{I_z} - \omega_0)\dot{\phi} + \frac{h_y\omega_0}{I_z}\psi - \frac{\dot{h}_z}{I_z} + \frac{N_z^{MT}}{I_z} + w_z \tag{5.c}$$

Where

- w : $[w_x\ w_y\ w_z]^T$ zero mean system noise vector;
- N : $[N_x^{MT}\ N_y^{MT}\ N_z^{MT}]^T$ applied magnetorquer control firing;
- I : diag $[I_T\ I_T\ I_z]^T$ moment of inertia tensor of the spacecraft;
- h : $[0\ h_y\ h_z]^T$ wheel angular momentum vector;
- ω_0 : orbital rate.

Kinematic equations: For Euler angles 2-1-3 sequence, the equation is derived by using spacecraft's angular velocity ω as follows (Hashida, 2004).

$$\begin{aligned} \dot{\phi} &= \omega_x \cos\psi - \omega_y \sin\psi \\ \dot{\theta} &= \left(\omega_x \sin\psi + \omega_y \cos\psi\right)\sec\phi \\ \dot{\psi} &= \omega_z + \left(\omega_x \sin\psi + \omega_y \cos\psi\right)\tan\phi \end{aligned} \tag{6}$$

Note that Euler 213 Eq. 6 has a singularity when the roll angle ϕ equals 90°.

The kinematics (attitude) of the satellite will be modelled making use of quaternions. A quaternion does not suffer from any singularities, no trigonometric functions are needed in the transformation matrix and it can be easily be referenced to the orbit following coordinate system. The kinematics can be updated by the following vector set of differential equations (Mohammed *et al.*, 2006)

$$\dot{q} = \frac{1}{2}\begin{bmatrix} 0 & \omega_z & -\omega_y & -\omega_x \\ -\omega_z & 0 & \omega_z & \omega_y \\ \omega_y & -\omega_z & 0 & -\omega_z \\ -\omega_z & -\omega_y & -\omega_z & 0 \end{bmatrix} q \tag{7}$$

Attitude controllers: The only known (to the author) three-axis stabilization control algorithm for passive gravity gradient with active magnetic torquing was derived by (Martel *et al.*, 1988). Their algorithm tends to choose the most favourable magnetorquing direction at any control instant by interleaving or simultaneously switching any of the three magnetic coils, relying on the current direction of the local geomagnetic field vector.

The control error on all 3 axes is expressed as a Proportional and Derivative (PD) error correction vector e (Mohammed *et al.*, 2005):

$$e = K_p a + K_d \dot{a} \tag{8}$$

Where

K_p : Proportional gain matrix.
K_d : Derivative gain matrix.
a and a: Attitude and rate error vectors, respectively.

The most favourable magnetorquing vector M (Chang, 1992; Hodgart and Ong, 1994) is then

$$M = \frac{e \times B}{|B|} \tag{9}$$

Where
B: local body field vector from a magnetometer.

In orbit results: This section describes the in orbit results of Alsat-1 attitude determination and shown in Fig. 1 control system which has been demonstrated during the middle of March.

The Alsat-1 attitude requirement are tabulated below in Table 1.

The attitude, sun sensor, magnetometer measurement and Y/Z wheels profiles on 14th March 2006 are shown in Fig. 2 to 8. Standard deviation of attitude parameters are tabulated in Table 2.

Table 1: Alsat-1 ADCS requirements

Attitude Requirement	Degrees
Bore-sight pointing (Roll/Pitch)	≤ 1 deg (1σ)
Bore sight rotation (Yaw)	≤ 0.5 deg (1σ)
Attitude stability during imaging	≤ 5 mdeg/sec (1σ)

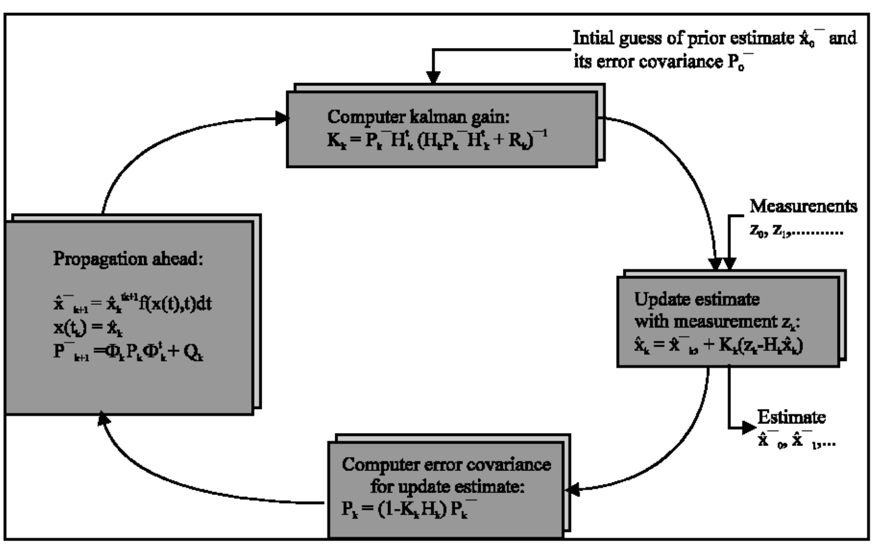

Fig. 1: Attitude determination loop

Fig. 2: Magnetometer measurement during imaging mode

Fig. 3: Sun sensor measurement during imaging mode

Fig. 4: Roll and pitch wheel performance during imaging mode

Fig. 5: Magnetorquer activity during imaging mode

Fig. 6: Attitude full state filter performance during imaging mode

Fig. 7: Attitude during imaging mode

Fig. 8: Angular rate during imaging mode

Table 2: STD of attitude parameters

Roll [deg]	Pitch [deg]	Yaw [deg]
-0.1	0.05	-0.04
Roll Rate [mdeg/sec]	Pitch Rate [mdeg/sec]	Yaw Rate [mdeg/sec]
0.02	0.05	0.5

The results shown in Table 2 meet the Alsat-1 requirement listed on Table 1. Although the roll and pitch results are quite satisfactory against the requirements.

CONCLUSION

The Imaging Mode design has been demonstrated to meet the attitude determination and control system requirements. Its key elements are:

- Zero-bias Y and Z-wheel control: For the best pointing stability, the zero-bias control mode can be entered from the Y-momentum mode by enabling the Z-wheel and selecting the zero-bias controller. The pitch angle will be controlled by the Y-wheel and the yaw angle by the Z-wheel. The Y-wheel momentum will be dumped by the magnetorquers in about half an orbit.
- A state estimation approach using Magnetometer, Sun sensor and wheels plus gravity gradient boom provide nadir pointing attitude system.
- The use of PD control design techniques to ensure simultaneous stability on all axes for the coupled system.

As a result, this paper explains the details of the attitude determination and control system of a successful LEO satellite project, Alsat-1.

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