MODELLING AND SIMULATION OF MAGNETIC CONTROL AND ITS APPLICATION ON ALSAT-1 FIRST ALGERIAN MICROSATELLITE

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KEYWORDS
Modelling, Simulation, Magnetic, Attitude, Control, Microsatellite.

ABSTRACT
The aim of this paper is the modelling and simulation techniques of a non gravitational force of the earth magnetic field and its application to the stabilisation of ALSAT-1 first Algerian microsatellite build by Surrey Satellite Technology Ltd. (SSTL), Guilford, United kingdom. This paper describes 1) the attitude dynamic, 2) the modelling of the earth magnetic field, 3) the magnetic torquer control. Simulation results will be presented.

INTRODUCTION
Attitude determination provides the information needed for attitude control. Attitude control is the process of changing the orientation of spacecraft. It roughly comprises two areas:

- Attitude stabilisation: maintaining an existing orientation;
- Attitude slew maneuver: controlling the spacecraft from one attitude to another.

However, the two requirements are not totally distinct. For example, the stabilisation of a satellite with one axis towards the Earth implies a continuous maneuver relative to its inertial orientation. The control accuracy typically depends on the actuators and control algorithms.
The limiting factor for attitude control is typically the performance of the actuator hardware and control software. Although with autonomous control systems, it may also be the accuracy of the orbit or attitude information.

An attitude control system is both the process and the hardware by which the attitude is controlled. In general, an attitude control system consists of the following four major components (as shown in Fig. 1):

- Attitude sensors ;
- Control logic ;
- Attitude actuators;
- Vehicle dynamics.

An attitude sensor locates known reference targets such as the Sun or the Earth's centre to determine when control is required, what torques are required, and how to generate them. The attitude actuator is the mechanism that supplies the control torques. Control systems can be classified as either an open loop system in which the control process includes human interactions (e.g. attitude data from the attitude sensors is analysed, and a control analyst occasionally sends command to the spacecraft to activate the control hardware), or a closed loop feedback system in which the control process is entirely electrical or computer controlled (e.g. attitude sensors sends attitude data to an on-board computer which determines the attitude and then activate the control hardware). Further there are two types of attitude control mechanisms: active attitude control in which continuous decision making and hardware operation is required (the most common sources of torques for active control systems are gas jets, electromagnetic coils, and reaction wheels) and passive attitude control which makes use of environmental torques to maintain the spacecraft orientation (gravity gradient and permanent magnets are common passive attitude control methods).

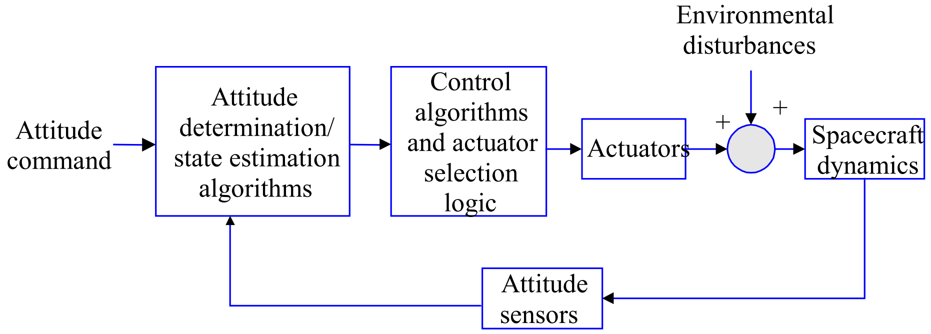

Figure 1 : Schematic Diagram of a Satellite Attitude Control System

Proceedings 18th European Simulation Multiconference
Graham Horton (c) SCS Europe, 2004
ISBN 3-936150-35-4 (book) / ISBN 3-936150-36-2 (CD)

ATTITUDE DYNAMICS

The dynamics of the spacecraft in inertial space governed by Euler's equations of motion can be expressed as follows in vector form

$$\mathbf{I}\dot{\boldsymbol{\omega}}_{\mathbf{B}}^{\mathbf{I}} = \mathbf{N}_{\mathbf{GG}} + \mathbf{N}_{\mathbf{D}} + \mathbf{N}_{\mathbf{M}} + \mathbf{N}_{\mathbf{T}} - \boldsymbol{\omega}_{\mathbf{B}}^{\mathbf{I}} \times (\mathbf{I}\boldsymbol{\omega}_{\mathbf{B}}^{\mathbf{I}} + \mathbf{h}) - \dot{\mathbf{h}} \qquad (1)$$

Where $\boldsymbol{\omega}_B^I$, $\mathbf{I}$, $\mathbf{N}_{GG}$, $\mathbf{N}_D$, $\mathbf{N}_M$ 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 rate of change of the quaternion is given by

$$\dot{\mathbf{q}} = \frac{1}{2}\boldsymbol{\Omega}\mathbf{q} = \frac{1}{2}\Lambda(\mathbf{q})\boldsymbol{\omega}_{\mathbf{B}}^{\mathbf{O}} \qquad (2)$$

Where,

$$\boldsymbol{\Omega} = \begin{bmatrix} 0 & \omega_{oz} & -\omega_{oy} & \omega_{ox} \\ -\omega_{oz} & 0 & \omega_{ox} & \omega_{oy} \\ \omega_{oy} & -\omega_{ox} & 0 & \omega_{oz} \\ -\omega_{ox} & -\omega_{oy} & -\omega_{oz} & 0 \end{bmatrix} \qquad (3)$$

$$\Lambda(\mathbf{q}) = \begin{bmatrix} q_4 & -q_3 & q_2 \\ q_3 & q_4 & -q_1 \\ -q_2 & q_1 & q_4 \\ -q_1 & -q_2 & -q_3 \end{bmatrix} \qquad (4)$$

Where,

$\boldsymbol{\omega}_B^O = \begin{bmatrix}\omega_{ox} & \omega_{oy} & \omega_{oz}\end{bmatrix}^T$ = body angular velocity vector referenced to orbital coordinates.

The angular body rates referenced to the orbit coordinates can be obtained from the inertially referenced body rates by using the transformation matrix A:

$$\boldsymbol{\omega}_B^O = \boldsymbol{\omega}_B^I - \mathbf{A}\boldsymbol{\omega}_0 \qquad (5)$$

If we assume the satellite in a near circular orbit with average orbital angular rate ω_o, then

$\boldsymbol{\omega}_0^B = \begin{bmatrix}0 & -\omega_o & 0\end{bmatrix}^T$ is a constant rate vector.

The kinematic equations can derived by using a spacecraft referenced angular velocity vector $\boldsymbol{\omega}_B^R$ as follows:

$$\begin{aligned} \dot{\phi} &= \omega_{Rx}\cos\psi - \omega_{Ry}\sin\psi \\ \dot{\theta} &= (\omega_{Rx}\sin\psi + \omega_{Ry}\cos\psi)\sec\phi \\ \dot{\psi} &= \omega_{Rz} + (\omega_{Rx}\sin\psi + \omega_{Ry}\cos\psi)\tan\phi \end{aligned} \qquad (6)$$

Where,

$\boldsymbol{\omega}_B^R = \begin{bmatrix}\omega_{Rx} & \omega_{Ry} & \omega_{Rz}\end{bmatrix}^T$ body relative angular velocity in any reference coordinate frame.

EARTH MAGNETIC FIELD MODELLING

The earth's magnetic field B can be expressed as the gradient of a scalar potential function V,

$$\mathbf{B} = -\Delta\mathbf{V} \qquad (7)$$

The nature of solenoid to Laplace's equation

$$\Delta^2\mathbf{V} = 0 \qquad (8)$$

V can be conventionally represented by a series spherical harmonics.

$$V(r,\alpha,\beta) = r_t \sum_{k=1}^{n} \left(\frac{a}{r}\right)^{n+1} \sum_{m=0}^{n} A_{mn} P_{nm}(\alpha) \qquad (9)$$

Where

$$A_{mn} = g_n^m(\alpha)\cos m\beta + h_n^m(\alpha)\sin m\beta$$

A : Equatorial radius earth (6371.2 Km adopted for the International Geomagnetic Reference Field, IGRF)

g_n^m, h_n^m : Gaussian coefficients ;

R : Geocentric distance ;

α : Coelevation ;

β : East longitude from Greenwich which define any point in space.

MAGNETIC TORQUER CONTROL

Any reaction and momentum wheel 3-axis stabilised satellite must employ a momentum management algorithm to restrict the wheel momentum within allowable limits. Momentum build-up naturally occurs due to the influence of external disturbance torques, for example, the torques due to passive gravity gradient, aerodynamic and solar forces, and active control torques from magnetorquers. These disturbances to the body of an attitude-controlled satellite cause an accumulation of momentum on the reaction and momentum wheels. The added momentum may cause saturation of the reaction and momentum wheel speed. Moreover, the existence of large angular momentum in the satellite causes control difficulties when attitude controllers are implemented, because the momentum provides the satellite with unwanted gyroscopic stability. Therefore, the management of three-axis wheel momentum is required in order to counteract the influence of persistent external disturbance torques.

The following cross-product control law is used to achieve the control objectives stated above

$$\mathbf{M} = \frac{\mathbf{e}\times\mathbf{B}}{\|\mathbf{B}\|} \qquad (10)$$

Where

e : error vector for a magnetorquer cross-product controller;

B : Magnetometer measured magnetic field vector.

The attitude is obtained from a full state Extended Kalman filter (EKF). This filter take measurement vectors (in the frame body) from magnetometer with 0.3 microTesla noise and sun sensor with 0.1 degree noise and by combining them with corresponding modelled vectors (in a reference frame), estimate the attitude and attitude rate. The EKF 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.
The assumptions of the simplified EKF estimator are listed as follows

- The spacecraft is nominally Earth pointing with either a certain spin rate in Z-.
- The spacecraft has a symmetric structure on X and Y-axes.
- The orbit of the spacecraft is near circular with an almost constant angular rate.

The system noise model has zero mean.
The state vector to be estimated is 6 dimensional such that

$$\mathbf{x} = [\phi \;\; \theta \;\; \psi \;\; \dot{\phi} \;\; \dot{\theta} \;\; \dot{\psi}] \qquad (11)$$

Using the cross-product control law with the error vector for a magnetorquer cross-product controller implemented on ALSAT-1 is given by

$$\mathbf{e} = \begin{bmatrix} Kd_x \dfrac{\omega_{ox}}{\omega_0} \\ Kd_y \dfrac{\omega_{oy}}{\omega_0} \\ \dfrac{Kd_z}{\omega_0}(\omega_{oz} - \omega_{oz\_ref}) + Kp_\psi(\psi - \psi_{ref}) \end{bmatrix} \qquad (12)$$

With Kp is the proportional gain, Kd is the derivative gain, ω_{0x},ω_{0y},ω_{0z} are the X, Y and Z orbit referenced angular rate of the satellite in radian/second, ω_{0zref} is the reference Z angular rate in radian/second, ψ is the Yaw angle in radian, and ψ_{ref} is the reference yaw angle in radian. The orbit reference angular rate and the angle in Eq. (12) are obtained from a full state Extended Kalman Filter (EKF).

SIMULATION RESULTS

A 98° inclination, circular orbit at an altitude of 860 km was used during the simulation tests. The following matrix of inertia is assumed for Alsat-1 during tests

$$\mathbf{I} = \begin{bmatrix} 153 & 0 & 0 \\ -0.25 & 153 & 0.0005 \\ 0.1 & 0 & 5 \end{bmatrix} \text{kgm}^2 \qquad (13)$$

The magnetic moment in the orthogonal X, Y and Z-axes was assumed to be equal to 10 Am^2 each. The reaction/momentum wheels has a MOI of 8.10^{-4} kgm^2 and the maximum speed is ± 5000 rpm, this gives a maximum angular momentum of 0.42 Nms. The maximum wheel torque is 5 milli-Nm.
We assume that we have gravity gradient torque and aerodynamic torque as external torque.

An IGRF model was used to obtain the geomagnetic field values. A sampling period of TS = 10 seconds was utilised for the discrete filter algorithm. To initialize the filter we use the yaw filter.
The satellite is left to nutate and librate freely for the two orbits in order to converge the filter. At the start of the third orbit the magnetorquer is activated.
The satellite is left to librate freely for the two orbits starting from an initial attitude of 3 degrees roll, 0 degree pitch, 0 degree yaw, 0 degree/second roll rate, 0 degree/second pitch rate and 0.6 degree/second yaw rate. At the start of the third orbit the magnetorquer is activated and within one orbits the pitch and roll librations are damped to nadir pointing error of less than 1 degree, the yaw angle is controlled to 0 degree. At the start of the eighth orbit the yaw angle is commanded to 170 degree for six orbits.
The total accumulated on time of magnetorquer is approximately 13000 seconds during an active control window of 12 orbits (72000 seconds). This gives an average magnetorquer power drain of 0.15 Watt from the start until the attitude is achieved.
We obtain the following results

Table 1: Lists the Euler angles RMS for the last three orbits (ψ_{ref} = 0.0 degree).

	Roll [degree]	Pitch [degree]	Yaw [degree]
Average	$0.52*10^{-2}$	10^{-2}	$0.88*10^{-1}$
STD [1-σ]	$0.24*10^{-1}$	$0.29*10^{-1}$	0.19
RMS	$0.25*10^{-1}$	$3*10^{-2}$	0.22

Table 2: Lists the Euler angles RMS for the last three orbits (ψ_{ref} = 170.0 degrees).

	Roll [degree]	Pitch [degree]	Yaw [degree]
Average	$-8.7*10^{-4}$	$-85*10^{-4}$	170.30
STD [1-σ]	$0.94*10^{-1}$	$0.38*10^{-1}$	0.33
RMS	$0.95*10^{-1}$	$0.39*10^{-1}$	170.30

Table 3: Lists the error Euler angles RMS for the last three orbits (ψ_{ref} = 0.0 degree).

	Error Roll [degree]	Error Pitch [degree]	Error Yaw [degree]
Average	$-0.3*10^{-2}$	$0.41*10^{-1}$	$-0.3*10^{-1}$
STD [1-σ]	$0.46*10^{-1}$	0.18	0.17
RMS	$0.46*10^{-1}$	0.19	0.18

Table 4: Lists the error Euler angles RMS for the last three orbits (ψ_{ref} = 170.0 degrees).

	Error Roll [degree]	Error Pitch [degree]	Error Yaw [degree]
Average	$11*10^{-4}$	$0.4*10^{-1}$	$-54*10^{-3}$
STD [1-σ]	$0.25*10^{-1}$	0.17	0.14
RMS	$0.25*10^{-1}$	0.18	0.15

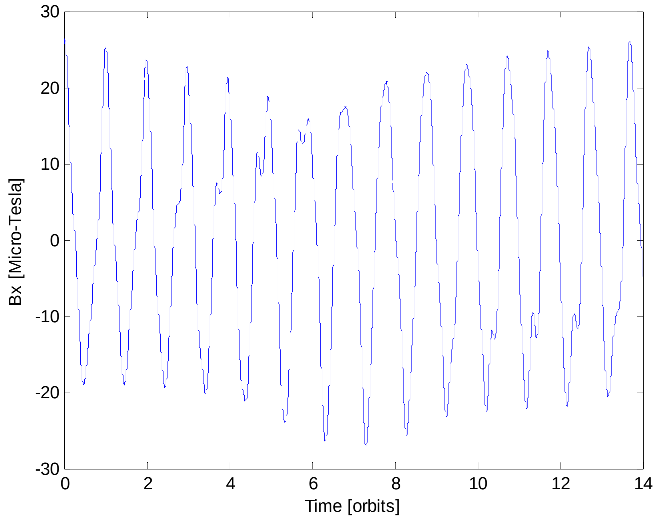

Figure 2 : B-Field (X-axis)

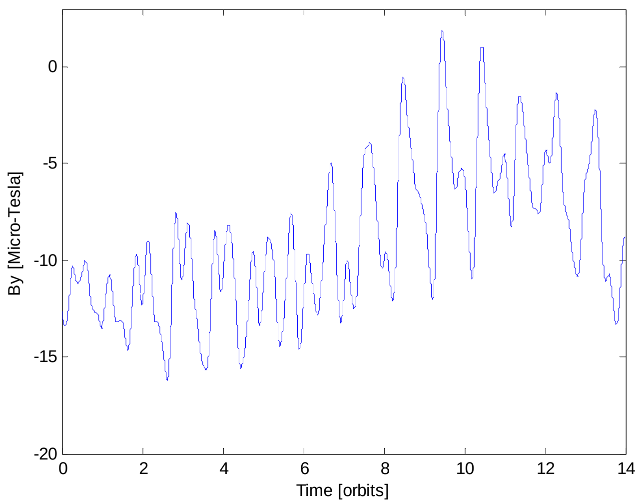

Figure 3 : B-Field (Y-axis)

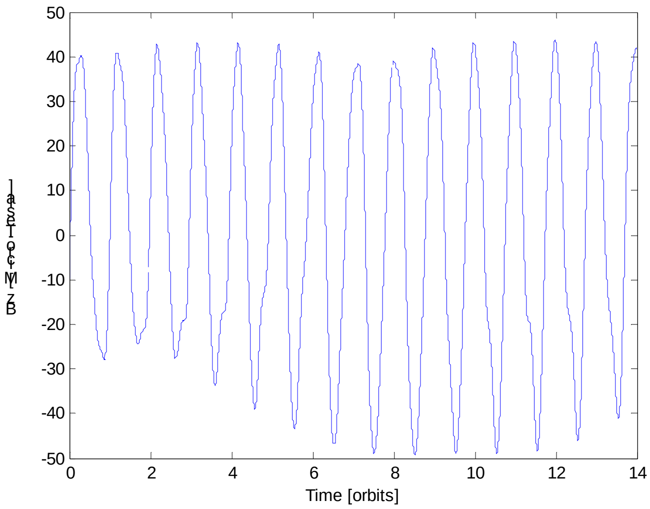

Figure 4 : B-Field (Z-axis)

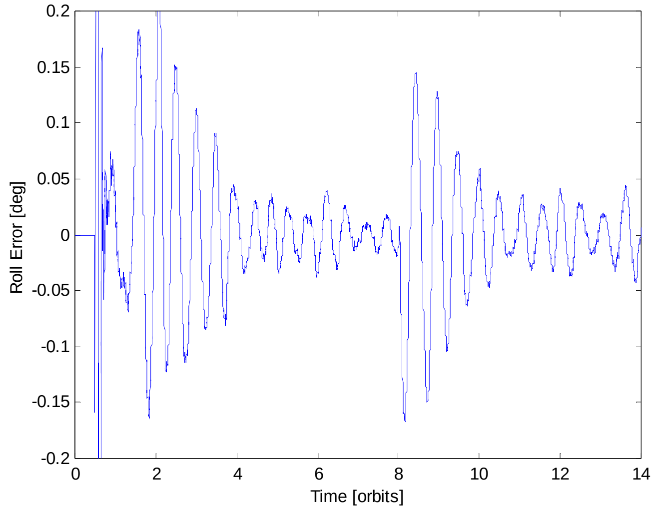

Figure 5 : Roll Attitude error during Magnetorquer Yaw Phase Control

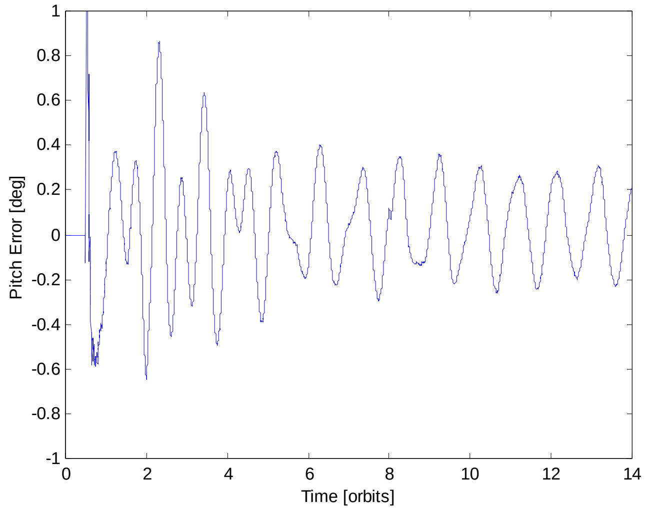

Figure 6 **:** Pitch Attitude error during Magnetorquer Yaw Phase Control

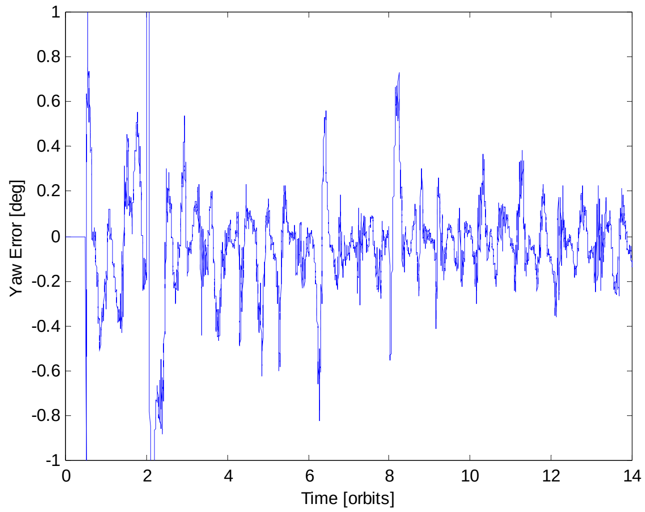

Figure 7 : Yaw Attitude error during Magnetorquer Yaw Phase Control

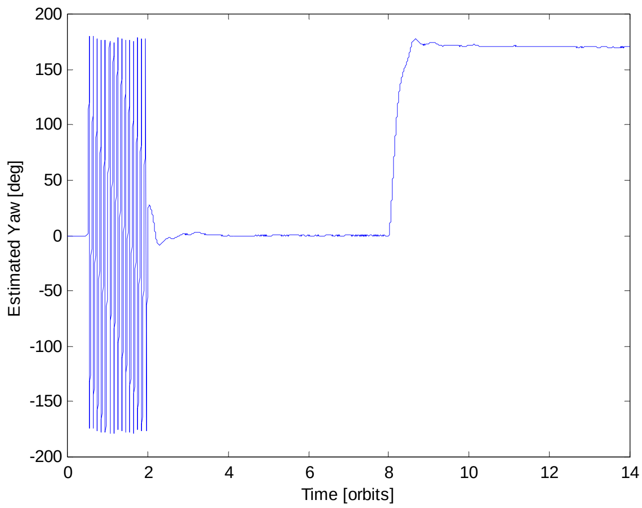

Figure 8 : Yaw Attitude during Magnetorquer Yaw Phase Control

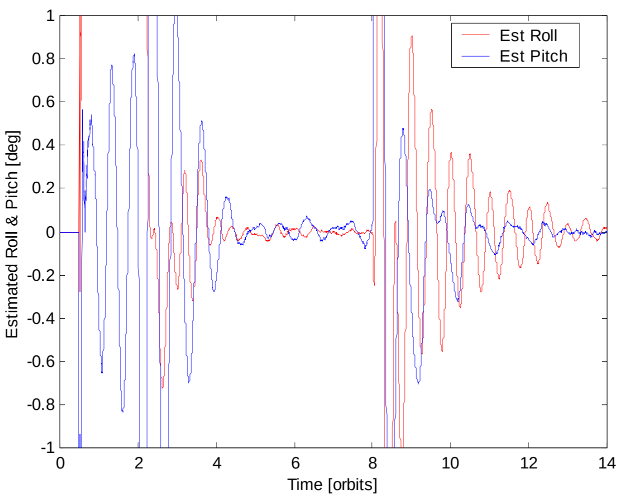

Figure 9 : Roll and Pitch Attitude during Magnetorquer Yaw Phase Control

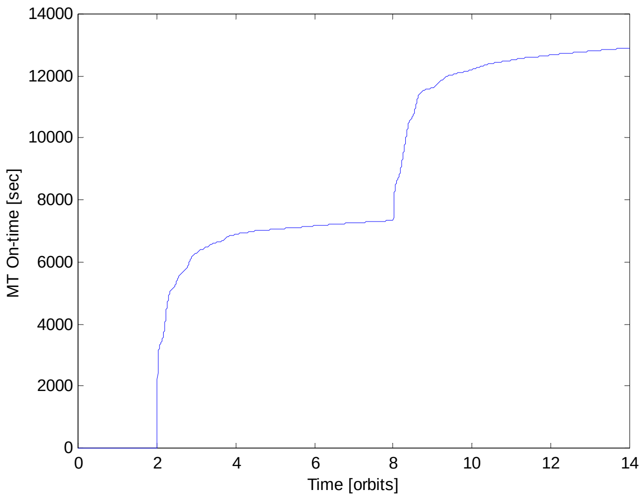

Figure 10 : Magnetorquer on time during libration damping and yaw phase control

CONCLUSION

Alsat-1 attitude requirements are tabulated below

Bore-sight pointing (Roll/Pitch)	$\leq$ 1.0 degree (1σ)
Bore-sight rotation (Yaw)	$\leq$ 0.5 degree (1σ)
Attitude stability (rate) during imaging	$\leq$ 0.005 degree/second (1σ)

The results we have obtained indicate, the roll, pitch, yaw and yaw rate achieves the requirement values by using magnetorquer cross-product.
This controller was designed to keep the microsatellite in accurate nadir pointing attitude. A cross product magnetorquer control law will damp out undesired pitch and roll libration and control either a constant yaw rate or a fixed yaw angle. This control mode will be regarded as the nominal attitude determination control system mode for the microsatellite.

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