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CONTROL SYSTEM DESIGN FOR THE FRENCH-BRAZILIAN SATELLITE

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Paper presented at the 2nd International Aerospace Congress IAC'97, Moscow, Russia, Aug.31 – Sept.5 1997.

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CONTROL SYSTEM DESIGN FOR THE FRENCH-BRAZILIAN SATELLITE

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Abstract

In this paper some aspects of the French-Brazilian Micro-Satellite attitude and orbit control system (AOCS) are presented. Basically, the mission is divided in two phases. In the first one, called aerobraking phase, the AOCS is responsible for the reduction of the orbit. In the second one, called routine phase, the satellite must be pointing to the sun. The AOCS performance is evaluated when the satellite is subjected to environment perturbation torques during the routine phase mode. The requirements of pointing necessary to perform the experiments of the French-Brazilian scientific mission are satisfied.

Introduction

The initiative of building a micro-satellite aims basically the development of low cost spacecraft programs, which means significant reductions in time and schedule while advancing the state of the art in spacecraft performance. Use of the latest in sensor, actuator, processor, and structure technology enables the French-Brazilian Micro-Satellite guidance, navigation and control subsystem to be built with significant improvement in its pointing capabilities over most lightsat programs. In accordance to the CNES-INPE cooperation protocol, it has been decided to start from a preliminary multimission platform concept, 3-axis stabilised with propulsion to be launched by Ariane 5 in a GTO orbit. A sequence of orbital manoeuvre mission called Aerobreaking phase need to be performed to reduce the apogee of the GTO orbit up to LEO in less than 3 months. After that phase the satellite will be put in an inertial orbit position pointing to the sun, characterising the routine phase mode.

The Aerobraking Phase

Among the aeroassist technologies, the aerobraking manoeuvre consists of utilising aerodynamic drag in the outer layers of a planet's sensible atmosphere as a "tool" to change the magnitude and, possibly, the direction of the velocity vector of an aerospace vehicle made to "bounce" off those layers one or more times. Quite clearly, by changing the velocity, the vehicle's orbit gets automatically changed, as well. This is a direct application of Hohmann's "braking ellipses" concepts.[1]

A rather serious problem, which limits the application of aerobraking, resides in the fact that the first passage through the atmosphere is the one where the vehicle experiences the most drastic heating. This is because, during the first pass, the vehicle must dissipate enough kinetic energy to ensure that it will exit the atmosphere at a speed lower than the planet's local escape speed, $V_{esc} = (2\mu/r)^{1/2}$, where μ is the planetary constant and r, the magnitude of the local position vector. Otherwise, the target-orbit (braking ellipse) is missed.[1]

To preserve the very idea underlying the aeroassist technologies (performing orbital manoeuvres with the least fuel consumption), the heating problem must be attacked via solutions which will not imply an excessive increase of the vehicle mass.[2] Notwithstanding, the greatest difficulty in applying aerobraking lies in determining accurately enough the atmospheric density in the outer layers at the instant of entry injection.[1,3] As those layers may exhibit very active dynamics, the entry corridor may result critically narrow. In fact, at any given altitude the value of the density is always subjected to a degree of uncertainty, which ultimately affects the design and behaviour of the braking system,[3] requiring specific studies aiming at quantifying how much the atmospheric density varies from one orbit to the next and what the effects of such variation on the vehicle and the manoeuvre may be.[4] These studies must include, among other things, simulations which take into account the altitude interval to be spanned, the kinds of breaking system under consideration, materials used, temperature range, atomic oxygen concentration along the entry corridor, accumulated radiation, etc.[4,5]

The greatest potential risk arises from the fact that, should the perigee be too low (high atmospheric density), the thermokinetic flux (thermal-power flux per unit area) on the vehicle will ultimately lead to a catastrophic increase in surface temperature.[6] The problem is compounded if the braking cycles occur under solar flux. Even more critical is the lunar perturbation, which also contributes to raise the vehicle temperature. As the deceleration is a function of the maximum temperature tolerated by the vehicle's materials, lunar perturbation effects must be countered by a control system.[6]

Now, as mentioned, France and Brazil are developing a joint scientific micro-satellite project (FBMS). As a part of this mission, the French will be flying an aerobraking experiment. The satellite will be initially injected in a geosynchronous orbit. Using this as the apogee altitude for a transfer ellipse, and roughly 125 km around the Earth (outer atmospheric layer) as the perigee altitude for the same ellipse, the satellite will be made to graze the atmosphere several times, thus having its apogee lowered to somewhere between 800 and 1500 km. Propellant consumption shall be minimal, as the required retrograde ΔV's will all be applied at the apogee, constituting a classical example of aerobraking. To optimise the breaking, the manoeuvre will be extended over three months, with sundry atmospheric fly-through and various intermediate orbits (whose semi-major axes will be progressively shorter). To complete the manoeuvre, a suitable, minimal posigrade ΔV will be applied at the desired apogee altitude to raise the perigee to about 400-800 km. From then on, the satellite will be in the routine phase, scheduled to last some 12 months.

Routine phase normal mode

In the normal mode the AOCS is responsible for keeping the satellite stabilised in three axes using the reaction wheel. Besides, the satellite must be maintained in an inertial pointing with accuracy of a maximum error of 0.5° on each axis and 0.05 deg/seg in terms of stability. The AOCS subsystem comprises three reaction wheels, three gyros, one star sensor, one solar sensor, a three-axis magnetometers sensor and three magnetic torque coils, their associated electronic and interface subsystems of the satellite.

In the inertial pointing mode the satellite nominal attitude is with the x– axis pointing toward the sun, the y-axis is in the north ecliptic direction and z=x^y, which in terms of satellite body is as follows: **x-axis** is longitudinal axis directed from the launcher/spacecraft interface toward the payload, **y-axis** is normal to one face and towards the aerobraking panel, where y+ correspond to the star sensor line of sight and **z-axis** completes the right hand system.

The star sensor field of view is 21° by 31°, and its accuracy is 0.02°, around the two-axis perpendicular of the star sensor line of sight. Measurement accuracy is determined by the expression around the star sensor line of sight, that is $\Delta\psi = \delta\sqrt{2}\ /\Theta$, where δ =0.02° and Θ is the angular range between two stars, typically 10 deg. The gyro accuracy is (the worst case): Drift = 50°/h (3σ), Noise = 0.05°/s (1σ). The solar sensor accuracy used is 0.1°.

Although the routine orbit will be circular (800km), in the simulations it has been used an eccentric orbit in order to have a worst case in terms of perturbation torque, that orbit has apogee of 1500 km and perigee of 400 km; inclination of 7°, argument of perigee 90°, and ascending node of – 90°.

Mathematical model and control laws.

The control law adopted is a proportional-derivative (PD), with gains tuned by simulation, trying to optimise the control performance in terms of pointing accuracy. A Kalman filter was designed to update attitude and gyro drift rate estimates, from star sensor attitude outputs. The estimated angular pointing error and its derivative are feedback by the control-closed loop. If we consider the satellite like a rigid body with null inertia products and a PD controller, the rotational dynamics can be represented in each axe by the following differential equations:

$$\begin{aligned} \dot{\xi} &= \omega \\ \dot{\omega} &= -K_p * \hat{\xi} - K_d * \hat{\omega} + T_P(t) \end{aligned} \qquad (1)$$

where ξ, ω and $\hat{\xi}, \hat{\omega}$ are the position and angular velocity and their respective estimates, and T_p (t) the disturbing torque. In order to simplify the moment of inertia has been absorbed by the terms of the equation. The model of gyros can be represented by:[7]

$$u = \omega + d + n \qquad (2)$$

where d and n represent drift and noise, respectively. The drift and noise are propagated by:

$$\begin{aligned} \dot{d} &= -\lambda_l d + q_l(2\lambda_l \sigma_d^2) \\ \dot{n} &= -\lambda_c n + q_c(2\lambda_c \sigma_n^2) \end{aligned} \qquad (3)$$

where $q_{l,c}$, $\lambda_{l,c}$ and $\sigma_{d,n}$ represent the noise, the correlation, and the covariance of the drift and noise, respectively. The model of sensor can be written as:

$$y = \xi + \nu(\sigma^2) \qquad (4)$$

where ν is the sensor noise and σ^2 is the covariance due to star sensor and solar sensor. The estimator has been modelled as:

$$\begin{aligned} \dot{\hat{\xi}} &= \hat{\omega} = u - \hat{d} \\ \dot{\hat{d}} &= -\lambda_l \hat{d} \end{aligned} \qquad (5)$$

Defining the $\delta\xi$, δd and δy as:

$$\begin{aligned} \delta\xi &= \xi - \hat{\xi} \\ \delta d &= d - \hat{d} \Rightarrow \\ \delta y &= y - \hat{\xi} \end{aligned} \left\{ \begin{aligned} \hat{\xi} &= \xi - \delta\xi \\ \hat{\omega} &= \omega + \delta d + n \end{aligned} \right\} \qquad (6)$$

The errors of estimator can be modelled as:

$$\begin{aligned} \delta\dot{\xi} &= -\delta d - n \\ \delta\dot{d} &= -\lambda \delta d + q_l(2\lambda_l \sigma_d^2) \\ \delta y &= \delta\xi + \nu(\sigma^2) \end{aligned} \qquad (7)$$

Considering T_p (t) =0 and using Eq. (6), the Eq. (1) can be written as:

$$\begin{aligned} \dot{\xi}_h &= \omega_h \\ \dot{\omega}_h &= -K_p(\xi_h - \delta\xi) - K_d(\omega_h + \delta d + n) \end{aligned} \qquad (8)$$

Defining the state vector Z = $[\xi_h \quad \omega_h \quad \delta\xi \quad \delta d \quad n]$, we can write Eqs. (7) and (8) in the following linear form:

$$\begin{aligned} \dot{Z} &= AZ + BN(Q) \\ \delta y &= HZ + \nu(\sigma^2) \end{aligned} \qquad (9)$$

where $A = \begin{bmatrix} 0 & 1 & 0 & 0 & 0 \\ -K_p & -K_d & K_p & -K_d & -K_d \\ 0 & 0 & 0 & -1 & -1 \\ 0 & 0 & 0 & -\lambda_l & 0 \\ 0 & 0 & 0 & 0 & -\lambda_c \end{bmatrix}$,

$B = \begin{bmatrix} 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 \end{bmatrix}^t$, $Q = \begin{bmatrix} 2\lambda_l \sigma_d^2 & 0 \\ 0 & 2\lambda_c \sigma_n^2 \end{bmatrix}$,

and $H = [0 \quad 0 \quad 1 \quad 0 \quad 0]$.

Once Eq. (9) has been obtained, a Kalman filter can be implemented in the following form:[8]

$$\begin{aligned} P &= \vartheta P \vartheta^t + \Gamma \\ K &= PH^t(HPH^t + \sigma^2)^{-1} \\ P &= (I - K^0 H)P(I - K^0 H)^t + K^0 K^{0^t} \sigma^2 \end{aligned} \qquad (10)$$

where:

$\vartheta = \exp(A\Delta t)$,

$\Gamma = \frac{\Delta t}{4}(BQB^t + \vartheta BQB^t \vartheta^t + 2\vartheta^{1/2} BQB^t \vartheta^{1/2^t})$

and K^0 is a reduced Kalman gain, since only $\delta\xi$ and δd are being estimated

$$K^0 = \begin{bmatrix} 0_{2x2} & 0_{2x2} & 0_{2x1} \\ 0_{2x2} & 1_{2x2} & 0_{2x1} \\ 0_{1x2} & 0_{1x2} & 0 \end{bmatrix} K$$

The K_d and K_p gains are selected by Multiple Criterion Method.[3] The performance criteria are normalised by:

$$\begin{aligned} \delta\xi(i, j) &= \delta\xi(K_{di}, K_{pi}) \\ \delta\omega(i, j) &= \delta\omega(K_{di}, K_{pi}) \end{aligned}$$

and

$$\begin{aligned} F\xi &= (\delta\xi - \delta\xi_{bad})/(\delta\xi_{good} - \delta\xi_{bad}); \\ F\omega &= (\delta\omega - \delta\omega_{bad})/(\delta\omega_{good} - \delta\omega_{bad}; \end{aligned}$$

It is important to note that $\delta\xi$(i,j) and $\delta\omega$(i,j) represents a linear sum of the effects due to disturbance and estimation errors on pointing and angular rate errors, respectively, for the control gain pair K_{di} and K_{pi}. The critical criterion is defined like:

$$F = \min(F\omega, F\xi) \qquad (11)$$

Therefore, the optimal gains K_d and K_p, can be obtained by:

$$K_d\text{opt} = K_d(i) \qquad (12)$$
$$K_p\text{opt} = K_p(j) \qquad (13)$$

where i and j are such that maximize F. Considering a simulation where the gain vary in the range from 0.01 to 0.12 in K_d and from 0.06 to 0.12 in K_d. with step of 0.01, the optimal gains selected were: K_d = 0.12 and K_p = 0.06.

Pointing accuracy verification

A simulation has been run, considering a gyro and star sensor, plus sun sensor ($\sigma = 0.1°$) with sampling period of 1s. Figure 1 shows the angular pointing total error and its derivative for each axis, which satisfies the mission requirements in the normal mode. That is, in all three axis the point error is better than 0.5° and its derivative is better than 0.05°/s. Besides, considering that the simulation has used a very conservative data, the results indicate that the mission specifications can be achieved, even without the use of sun sensor.

Final comments and conclusion

One presented an overview of both aerobraking and routine phase of French-Brazilian scientific micro-satellite. Problem description and its solution approach were given, as well as some numerical results predicting the achievement of control goals.

Other important aspects under investigation include the reaction wheel dessaturation control mode and the number of stars at star sensor FOV along the satellite lifetime.

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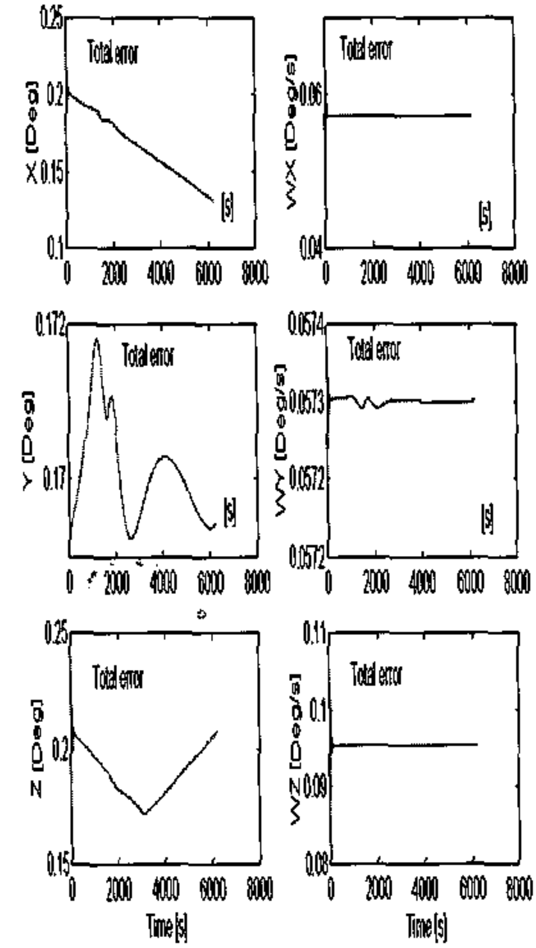

Figure 1 - Angular pointing total error

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