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ATTITUDE CONTROL ASPECTS FOR SCD1 AND SCD2

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Abstract

The Data Collecting Satellites (SCD) were developed and integrated at INPE, Brazil, with an aim of retransmiting meteorological data collected by automatic platforms spread over the country. The specified orbit for SCD1 and SCD2 are circular at 750 km altitude and 25° inclination. SCD1 was launched in February 1993 by the Pegasus launcher (OSC). Both are spin stabilized, with magnetic attitude control coils. The angular velocity of the SCD2 spacecraft will be controlled by a spin plane magnetic coil, commanded by an on-board magnetometer.

Key words: Attitude control, Control system design, Minimum time control. On-off control, Satellite control, Satellite artificial, Simulation.

Introduction

Two Data Collecting Satellites (Fig. 1) were developed at INPE. The first one was launched on February 1993, and the second one is scheduled to be injected in orbit in 1995. The main goal of the mission is to receive and retransmit the environmental data collected by autonomously operated on-ground platforms. They are spread in remote regions where man-operated stations are inadequate, as the tropical rain forest, off-shore locations, etc.

The orbit was chosen to accommodate particular aspects of ground coverage (Brazil), altitude decay and transmitter power. The selected orbit is a circular 750 km altitude orbit, with 25° inclination.

Both spacecrafts are provided with a transponder as a payload. Internal subsystems are: Power Supply, Structure and Thermal Control, Telemetry and Telecommand Subsystem (TMTC), an experimental On-Board Computer and Attitude Control Subsystem. An experimental reaction wheel (only the shaft, bearings and electronics, without the wheel mass) will also be integrated in SCD2.

SCD1 satellite

The Attitude Control Subsystem (ACS) is responsible to stabilize and control the spacecraft orientation with respect to the Sun. To perform these tasks, the ACS is provided with two digital one-axis Sun Sensors, one analogue Magnetometer, one spin axis air core Magnetic Coil and a passive Nutation Damper. The stabilization is achieved by a rotation around its major principal axis imparted to the spacecraft by the launcher's last stage. Attitude determination and control are both performed on-ground, by using the telemetered sensor signals and commanding the appropriate coil polarity.

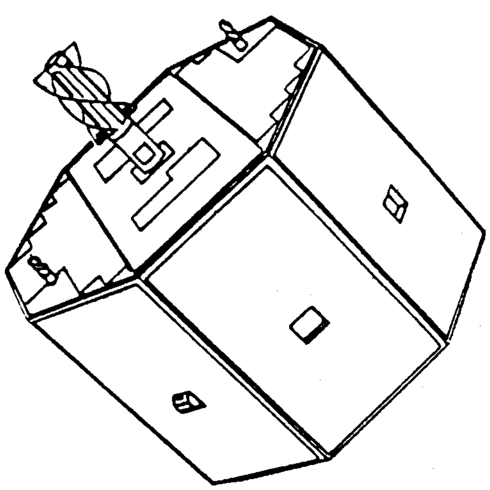

Figure 1: The Data Collecting Satellite.

SCD1 was launched successfully on February 9, 1993. All the internal subsystems are operating according to specifications. Orbit and attitude determinations are being performed properly.

Due to the launcher restrictions (Pegasus), the injection angular velocity was limited to 120 rpm, rather than the specified 140 rpm. Eddy current torques caused by the Earth's magnetic field decrease exponentialy the angular velocity of the satellite. Pre-flight calculations predicted a spin decay from 120 rpm, at the beginning of life, to 20 rpm, approximately, after 1 year (expected spacecraft life). This torque is approximately described by[1]:

$$\bar{T}_e = p\ (\bar{\omega} \times \bar{B}) \times \bar{B} \qquad (1)$$

where p is a constant coefficient, function of the spacecraft's geometry and its surface electrical properties. $\bar{\omega}$ is the satellite angular velocity and $\bar{B}$ the Earth's magnetic flux density. Theoretical approaches gave to p a value close to 1920 m^4/Ohm.

In addition, magnetic torques due to unbalanced electrical currents in the spacecraft equipments precess the satellite spin axis, around the Earth's magnetic dipole. After some period, the change in the satellite attitude relative to the Sun would illuminate the thermal bottom panel, so an attitude maneuver would be necessary to reorient the spin axis. In this case, the magnetic air core coil shall be activated, in such a way as to drive the spin axis in a safe attitude with respect to Sun. The 6 Am^2 torque coil is located in the spacecraft upper panel, with its axis parallel to the spin axis. The magnetic torque suffered by the spacecraft is given by:

$$\bar{T}_m = (m_r + u\ m_c)\ \hat{\omega} \times \bar{B} \qquad (2)$$

where m_r and m_c are the residual and the coil magnetic moment, respectively. u is the coil polarity (-1, 0, 1) and $\hat{\omega}$ is the spin axis direction. The residual magnetic moment was specified to be in the range -0.5 to -1.5 Am^2, so as to cause a spin axis precession in the same direction of the sun's motion relative to Earth. This magnetic moment was measured during satellite integration and resulted a value of -0.75 Am^2.

The SCD1 has no attitude pointing requirements. The only restriction is that the bottom radiator panel should be kept out from the Sun direct incidence. Periodic attitude maneuvers take place whenever necessary as to achieve the correct orientation with respect to the Sun. The coil polarity is selected by an attitude simulation program, in such a way as to reduce the Sun aspect angle (angle between the Sun and the spin axis). A telecommand is then transmitted to switch the coil to the specified polarity. The maneuver can be spread for several days.

SCD2 satellite

As SCD1, SCD2 will also be spin stabilized. Nevertheless, its angular velocity will be controlled close to 30 rpm, by means of a magnetic air core coil (4 Am^2 magnetic moment) located in the lateral panel. The angular velocity control will be autonomously activated whenever the rotation falls bellow 28 rpm and will be deactivated when the rotation reaches 32 rpm. Also, the SCD2 spin axis orientation will be controlled in such a way as to point the rotation axis to the ecliptic north pole. Residual magnetic torques shall slowly precess this axis, so after a few weeks the required pointing is no longer assured. To correct the attitude, periodical spin axis maneuvers are predicted, whenever the angle between the spin and the ecliptic normal is greater than 10°.

Spin Axis Maneuvers

The coil polarity as a function of time can be derived from the attitude error[2-3]:

$$\bar{E} = \hat{s}_f - \hat{H} \qquad (3)$$

where $\hat{s}_f$ is the desired spin axis direction, after the maneuver, and $\hat{H}$ is the angular momentum direction. In order to reduce the attitude error, E^2 is derived with respect to time, resulting

$$d(E^2)/dt = (-2/H)\ \bar{E} \cdot \bar{T}_m. \qquad (4)$$

It is assumed that the magnetic torque causes only precession (the magnitude of the angular momentum remains unchanged) and the perturbing torques are neglected. Substituting Equation 1 in the above equation results

$$d(E^2)/dt = -2/H\ (m_r + u m_c)\ s_w. \qquad (5)$$

The switching function s_w, given by

$$s_w = \bar{E}\ .\ \hat{\omega} \times \bar{B} \qquad (6)$$

determines the appropriate coil polarity in order to decrease the attitude error, since $m_c \gg m_r$ (the specified values are 12 Am^2 and ±0.1 Am^2, respectively). Therefore,

$$u = 1 \text{ for } s_w > 0 \qquad (7a)$$
$$u = -1 \text{ for } s_w < 0 \qquad (7b)$$
$$u = 0 \text{ if } s_w = 0. \qquad (7c)$$

Time-tagged maneuvers will be calculated on-ground, telemetered to the spacecraft and stored in the on-board computer. Each time-tagged command informs the computer the spin axis coil (located at the spacecraft's upper panel) polarity, activation and deactivation times. It is expected to point the spin axis to the ecliptic pole with accuracy better than 1°. To calculate the maneuver, the coil polarity is obtained at defined time intervals (5 minutes, in the simulations) using the above equations. The resulting maneuver provides a significant saving of the maneuvering time with respect to a QOMAC (quarter orbit magnetic attitude control) type maneuver. Both spin axis and angular velocity maneuvers shall last a few hours (6 to 18) with time interval between maneuvers around 25 days.

Simulations were carried out to check the control algorithms and to adjust an optimal coil magnetic moment. The strategy consists in driving the spin axis inside the 1° cone around the ecliptic north pole. Environmental torques will slowly precess the spin axis, and after some days it reaches the 10° cone around the ecliptic pole. A new maneuver is then calculated on-ground, transmitted and stored in the on-board computer. The maneuvers consist in a set of switching times and the corresponding coil polarities.

Spin Plane Maneuvers

To control the angular velocity of the spacecraft around 30 rpm, two (a main and a backup unit) spin plane magnetic coils will be employed. They are individually activated, by on-ground telecommand. Also, the on-board computer will monitor the angular velocity using the sun sensor outputs and autonomously will activate the coil electronics when the velocity falls below 28 rpm. To accelerate the spacecraft from 28 to 32 rpm, the maneuver time is estimated in 8 hours, approximately. The coil switching is performed automatically by a dedicated electronics that process the 3-axis fluxgate magnetometer signals. Each coil is commanded by the magnetometer sensor orthogonal to both the coil axis and the the spin direction. To avoid the interference in the magnetometer output caused by the magnetic field of the coil, a threshold activation and deactivation level was implemented in the commutation electronics. These levels differ each other to prevent feedback: the activation value is B_{on} = 120 and B_{off} = 60 mGauss.

Considering a spin plane coil rotating in a XY plane, such that the geomagnetic component in this plane lies along the X direction, the instantaneous torque experienced by the satellite is given by:

$$\vec{T} = \nu m_s (\cos\theta\ \hat{i} + \sin\theta\ \hat{j}) \times \vec{B} \tag{8}$$

where ν is the coil polarity, m_s is the spin plane magnetic moment (m_s = 4 Am²) and $\theta = \omega t$. If B_x and B_z are the components of the geomagnetic flux intensity in the X and Z directions, the commutation limits are then:

$$\theta_{on} = \arcsin B_{on}/B_x \tag{9a}$$
$$\theta_{off} = \arcsin B_{off}/B_x \tag{9b}$$

For a spin-up and a spin-down maneuvers, the coil polarity as a function of the position angle is given in Table 1

Table 1: Spin coil polarities

θ limits	Spin up	Spin down
$0 \le \theta < \theta_{on}$	0	0
$\theta_{on} \le \theta < \pi - \theta_{off}$	-1	+1
$\pi - \theta_{off} \le \theta < \pi + \theta_{on}$	0	0
$\pi + \theta_{on} \le \theta < 2\pi - \theta_{off}$	+1	-1
$2\pi - \theta_{off} \le \theta < 2\pi$	0	0

The mean torque over a revolution can now be integrated, resulting[4]:

$$T_x = -m_s B_z(\cos\theta_{on} + \cos\theta_{off})/\pi \tag{10}$$

$$T_y = -m_s B_z(\sin\theta_{on} - \sin\theta_{off})/\pi \tag{11}$$

$$T_z = m_s B_x(\cos\theta_{on} + \cos\theta_{off})/\pi \tag{12}$$

Note that the torque along the Y direction can be neglected, as θ_{on} and θ_{off} are normally small.

Results

Attitude determination of the SCD's spacecrafts is performed on-ground, using the sensor signals, telemetered to the Earth during the ground station contact. The satellite shall count with two sensors: one-axis digital sun sensor with meridian sun crossing indicator and a 3-axis fluxgate magnetometer. The sensors are sampled at a 2 Hz frequency, by the on-board TMTC subsystem. The digital sun sensor measures the sun's aspect angle, between the spin axis and the direction of the sun. The magnetometer gives the Earth's magnetic field vector, allowing the determination of the attitude by solving the ambiguous orientation. The meridian sun crossing indicator is also sampled and gives information about the spacecraft angular velocity. Both SCD1 and SCD2 are provided with nutation dampers to reduce perturbations in the angular velocity of the satellite during separation from the launcher. They are toroidal shape rings, partially filled with oil and located in an off-center position at the upper panel.

The first year results are presented in Figures 2 to 6. Figure 2 presents the attitude maneuvers times, through the coil polarities history. Since the satellite launch, the requirement for pointing the spin axis is not only to avoid the sun incidence in the bottom panel, but also to maximize the electrical energy and to adjust the equipment temperatures by placing the spacecraft in a specific orientation with respect to the sun. These requirements are achieved for sun aspect angles between 70 and 90°. Several spin axis maneuvers were performed in order to conform to this requirement, with negative coil polarity (against the angular velocity vector). The sun aspect angle history is shown in Figure 3. Except during the initial 40 days, the sun angle remained constricted to the specified range.

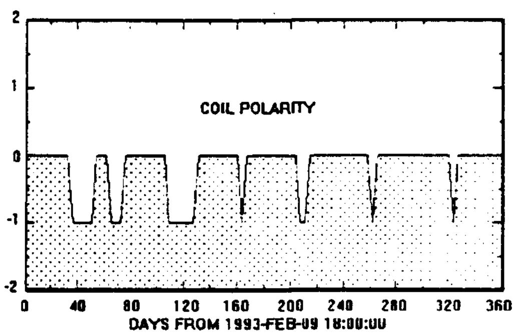

Figure 2: Coil polarity history.

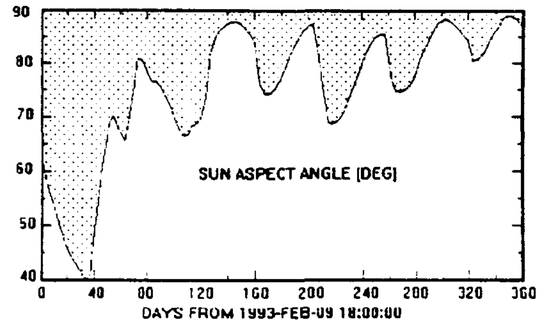

Figure 3: Sun aspect angle history.

Figure 4 presents the path of the spin direction in celestial coordinates (right ascension and declination). Note that the attitude maneuvers caused the spacecraft spin axis to align to the Earth's magnetic pole.

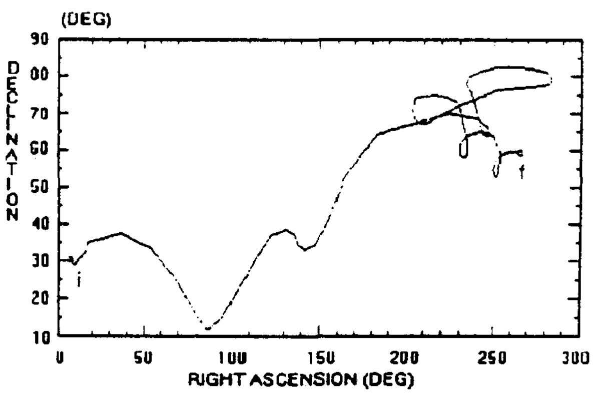

Figure 4: Spin axis motion.

Spin decay history is presented in Figure 5, where the exponential decay behavior due to the induced electrical currents in the spacecraft structure by the Earth's magnetic field is clearly seen. It should be noted the high discrepancy between the predicted and the in-flight values of the decay rate. In fact, as shown in Figure 6, the estimate of the Foucault parameter p through filtering technics resulted 500 m^4/Ω mean, smaller than the calculated 1920 m^4/Ω from theory.

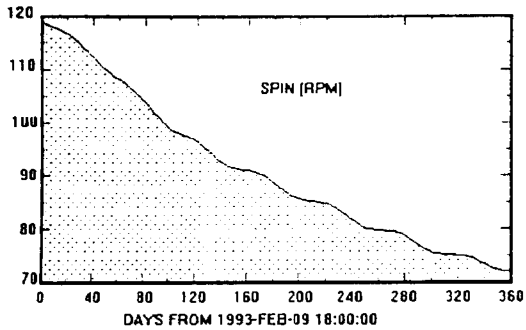

Figure 5: Spin rate history.

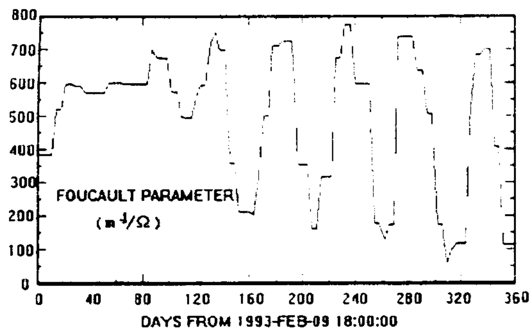

Figure 6: Foucault parameter.

The spacecraft residual magnetic moment is also estimated by the attitude determination procedures. In the Figure 7, the estimated magnetic moment of the satellite reflects the coil activation when a manouver is performed. The mean value of the residual magnetic moment (when no manouver is on way) is around -0.65 Am^2, close to the -0.75 Am^2 measured on-ground before launch.

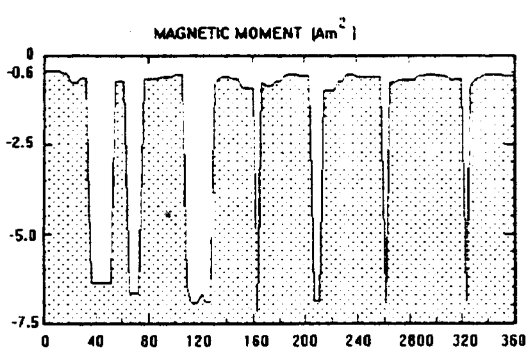

Figure 7: Magnetic moment estimation.

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