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14. Abstract/Notes

The launch window studies establish a relatively significant phase in mission analysis of a given space program. The final result of such analysis is the time interval of the day during some particular days of the year in which the spacecraft launch could be achieved. The main task of the study is to identify and to formulate mathematically the several constraints concerning the launch that restrain or modify the launch window. This work presents an overview of the launch window studies made for the Data Collecting Satellite (DCS) of the Brazilian Space Program (MECB).

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The launch window studies establish a relatively significant phase in mission analysis of a given space program. The final result of such analysis is the time interval of the day during some particular days of the year in which the spacecraft launch could be achieved. The main task of the study is to identify and to formulate mathematically the several constraints concerning the launch that restrain or modify the launch window. This work presents an overview of the launch window studies made for the Data Collecting Satellite (DCS) of the Brazilian Space Program (MECB).

INTRODUCTION

Once the constraints have been identified the next step is to combine them according to the mission failure criteria, classifying the restrictions in essentials and desirables. The constraints that cause loss of mission if not obeied are essentials. Among such restrictions, it is very common to find limitation to the spacecraft attitude relative to sun just after orbit injection (when normally the satellite attitude is different from the nominal one), due to thermal or power constraints. Eclipse duration is also an important source of launch window definition despite the fact that the reduction of eclipse duration is a nonessential restriction. In geosynchronous mission it is frequent to utilize a constraint that limits the launch days to the epoch of the year when the sun is far from the equinoxes (near March and September, 21st) to avoid that the spacecraft pass through the Earth shadow just after injection in its nominal orbit. In general, desirable constraints are only considered if there are no essential constraints or if the latter were already carried out.

The Data Collecting Satellite will be launched in the middle of 1989, with the aim of receiving data coming from remote meteorological platforms (PCDs). It has a prism shape, with an octogonal base. With exception of the thermal radiator on the lower base, all the spacecraft faces are covered with solar cell arrays. The satellite orbit is near circular at 750 km altitude and 25^o inclination. It will be launched from Alcantara Launch Base (CLA), north of Brazil, whith launch azimuth equal to 65^o, approximately. Launch azimuths greater than 90^o (with orbit injection at descending node) are not possible due to CLA location. At the separation point, the spacecraft will be rotating with 180 rpm along its symmetrical axis. The spin axis direction is controlled by means of a magnetic coil actuator.

The main constraint to the launch window for the Brazilian spacecraft comes from thermal impositions, due to the lower base thermal radiator. In fact, so as to guarantee the correct temperature operation of the on-board equipment, the lower base should not be illuminated by sun. As can be seen in Figure 1, the sun must be kept on the upper spacecraft's hemisphere during the satellite lifetime of 6 months.

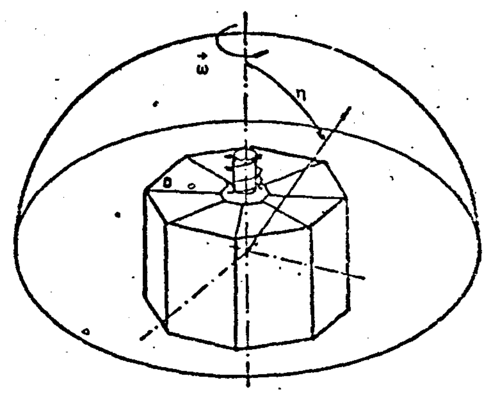

Fig. 1. DCS attitude relative to sun.

Unfortunately, the attitude control can not be utilized to adjust the satellite attitude relative to sun, if the sun initially faces the thermal radiator side. The attitude manouvers are very long (1 to 4 days) because of the weak interaction of the spacecraft coil with the Earth's magnetic field and so the temperature could be out of range before the satellite is in a safe attitude. The launch window must then guarantee that the spacecraft be injected in an attitude such that the radiator panel does not face the sun during the 6 months lifetime, as shown in Figure 2.

ATTITUDE PROPAGATION

The spacecraft attitude does not remain fixed along the time, as several perturbations accumulate their effects and deviate the inertial direction of the spin axis. Due to the symmetrical geometry of the satellite, perturbations like the atmospheric torque and solar radiation pressure have less influence on the attitude changes. Magnetic torques become then the major perturbations as they do not depend on the satellite geometry. This

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assertion is totally confirmed through the attitude analysis of the TELSTAR satellite (Yu[1]), whose geometry and mass is similar to that of DCS.

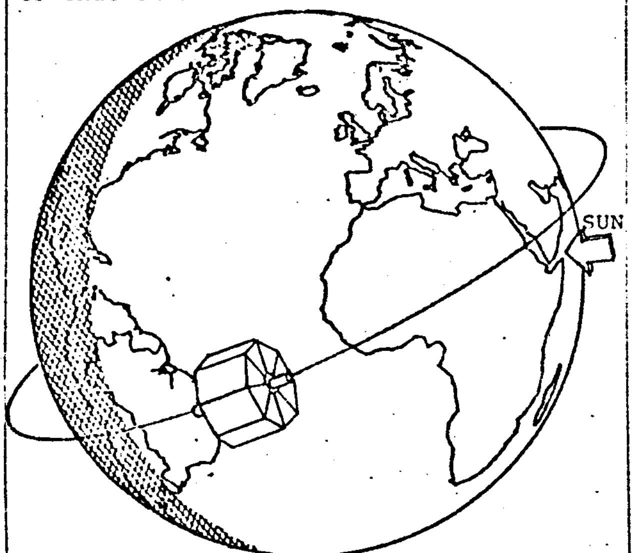

Fig. 2. Launch attitude.

The angle between the spin axis and the sun will alter due to both the perturbing torques and the motion of the sun relative to Earth. To define its orientation, two angles are adopted: the spin axis right ascension α and declination δ, as seen in Figure 3, in inertial coordinates (X and Y lie in equatorial plane, X pointing towards the vernal equinox).

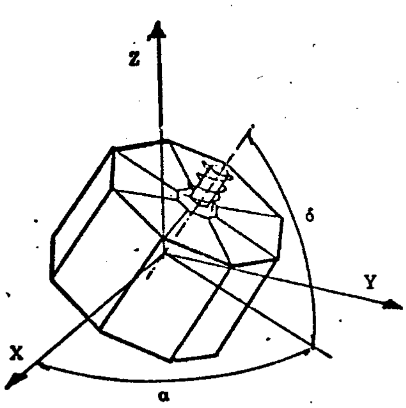

Fig. 3. Spacecraft's right ascension and declination angles.

As the rotation axis is a principal moment of inertia axis, one can neglect the transverse angular velocities and then

$$\vec{\omega} = \omega\,(\cos\delta\cos\alpha\,\hat{i} + \cos\delta\sin\alpha\,\hat{j} + \sin\delta\,\hat{k}) \qquad (1)$$

where ω is the spacecraft angular rate and $\hat{i}$, $\hat{j}$ and $\hat{k}$ denote the inertial X, Y and Z unit vectors, respectively. The following torques were considered: magnetic torque caused by the residual spacecraft magnetic moment interaction with the Earth's magnetic field and Eddy current torque due to the rotation of the satellite in the presence of the geomagnetic field. The gravity gradient torque has shown to be negligible compared to the preceeding torques.

The spacecraft attitude was numerically integrated for 6 months by Moro[2] using the residual torque formulation extracted from Wertz[3]

$$\vec{N}_r = m\,\vec{\omega} \times \vec{B} \qquad (2)$$

where m is the spacecraft's residual magnetic moment, $\vec{\omega}$ is the unit vector along the spin axis and $\vec{B}$ the Earth's magnetic field. The residual moment arises from uncompensated electric currents on the on-board equipments. It is quite difficult to calculate the moment a priori because of the obvious complexity of the equipment distribution over the spacecraft structure. In order to guarantee that the residual magnetic moment remains restricted to certain limits, the magnetic field produced by the satellite must be measured during integration and tests. To assure the measured values lie inside the design limits, it is a common procedure to fix some permanent magnets on the satellite body. For the DCS, the established values of the residual moment in the spin axis direction are:

$$-1.5\ \mathrm{Am}^2 < m < -0.5\ \mathrm{Am}^2 \qquad (3)$$

where the minus sign indicate that the moment is opposite to the angular velocity.

Eddy current torque formulation is given by Smith[4]

$$\vec{N}_E = p\,\vec{B} \times (\vec{B} \times \vec{\omega}) \qquad (4)$$

where p is a constant that depends on the spacecraft geometry. Note that the residual magnetic torque is always perpendicular to the satellite angular momentum and then it causes a precessional motion on the spin axis. On the other hand, Eddy current torque causes a rotational energy dissipation, decreasing the spacecraft angular rate. For the Data Collecting Satellite, the parameter p assumes the value (Kuga[5]):

$$p = 1916\ \mathrm{m}^4/\mathrm{Ohm}$$

The attitude integration results are shown in Figures 4 and 5: the right ascension and declination motion of the spin axis, and the angular rate decay along 6 months of attitude propagation, respectively. A residual magnetic moment equal to $-0.6\ \mathrm{Am}^2$ was used. The effects of the magnetic torque (spin axis precession in right ascension) and the Eddy current torque (exponential spin decay) are clearly identified in the figures.

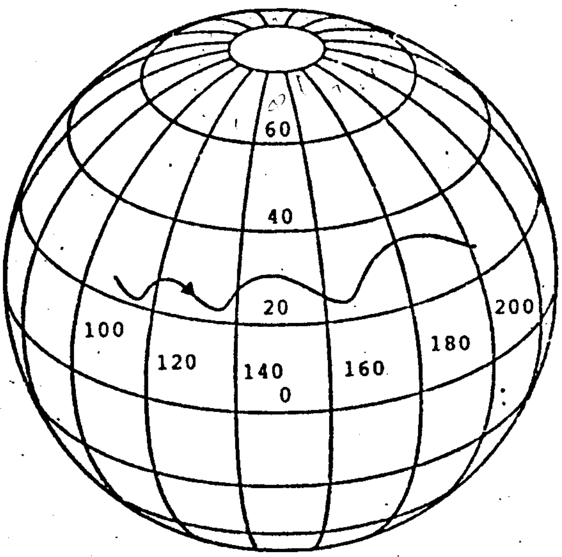

Fig. 4. Precession of the spacecraft spin axis over 6 months.

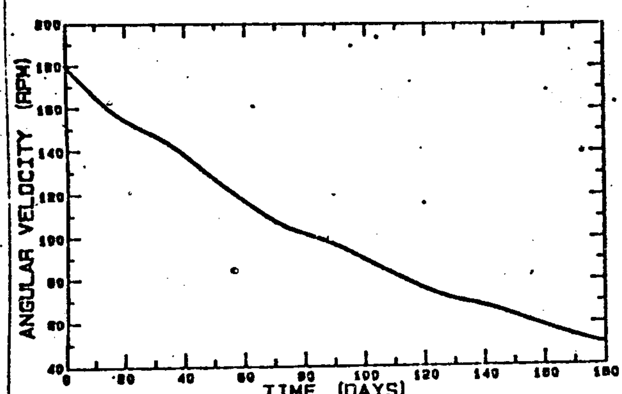

Fig. 5. Angular velocity decay.

LAUNCH WINDOW FORMULATION

Launch windows are normally calculated by verifying if the constraints are satisfied at several hours of the day and for several days of the year. Such a process presents as advantage a relatively easy implementation of new restrictions to the launch time that often appear during the satellite development design. As a disadvantage, one has a reduced precision on the window margins or a great consumption of computational effort, due to necessary verification of all constraints to evaluate a single launch window point. To avoid this disadvantage, the solution of the problem was guided to an analytical approach, by fitting simplified functions to the attitude propagated values (Figures 4 and 5), that define the spacecraft state:

$$\omega = \omega_o \, e^{-bt} \tag{5}$$

where ω_o is the initial rotation rate of 180 rpm and b is a constant adjusted by fitting the function to the rotation rate decay

$$b = 0.00716 \text{ /day.}$$

The spin axis right ascension can be given by

$$\alpha = \alpha_o + m \, K \, (e^{bt} - 1) \tag{6}$$

where α_o is the spin axis right ascension at the orbit injection and K is a proportionality constant:

$$K = -58.65^{\circ}/\mathrm{Am}^2$$

The motion of the spin axis declination was neglected and the declination value adopted was equal to 27°. This procedure was necessary to obtain an analytical solution to the problem. However, the error introduced from this simplification is small, as later studies based on numerical solutions have shown (INPE [6]).

The sun-spin angle is then given by:

$$\cos\eta = \cos\delta \cos\delta_s \cos(\alpha-\alpha_s) + \sin\delta \sin\delta_s \tag{7}$$

where α_s and δ_s are the sun right ascension and declination, respectively, at the date. To calculate the sun position, an analytical Earth orbit propagation was used. The thermal constraint prevents the angle η from exceeding 90 degrees and, therefore,

$$\cos(\alpha-\alpha_s) > -\tan\delta \tan\delta_s. \tag{8}$$

By substituting the expression for the spin axis right ascension as function of time, it has:

$$\alpha_{omin} < \alpha < \alpha_{omax} \tag{9}$$

where

$$\alpha_{omin} = \alpha_s - m \, K \, (e^{bt} - 1) - \cos^{-1}(-\tan\delta \tan\delta_s) \tag{10.a}$$

and

$$\alpha_{omax} = \alpha_s - m \, K \, (e^{bt} - 1) + \cos^{-1}(-\tan\delta \tan\delta_s). \tag{10.b}$$

As the sun right ascension and declination are also functions of the time, the condition (9) must be satisfied during the whole satellite lifetime, and then

$$\overline{\alpha}_{omin} = \max(\alpha_{omin}) \text{ for } 0<t<180 \text{ days}$$

$$\overline{\alpha}_{omax} = \min(\alpha_{omax}) \text{ for } 0<t<180 \text{ days.}$$

The equations for α_{omin} and α_{omax} are nonlinear and so a numerical procedure was developed to obtain the minimum and maximum values, by varying the time in intervals of one day. Nevertheless, $\overline{\alpha}_{omin}$

and $\overline{\alpha}_{omax}$ are still functions of the epoch year, as they depend on the launch date.

The spin axis right ascension is related to the orbital right ascension of the ascending node through the relationship (Figure 6)

$$\tan \Omega_i = \frac{-\sin \alpha_{oi} \sin\upsilon - \cos\alpha_{oi} \cos i \cos\upsilon}{-\cos \alpha_{oi} \sin\upsilon + \sin\alpha_{oi} \cos i \cos\upsilon} \quad (11)$$

as a function of the orbit inclination i, and the angle from the asceding node to the orbit injection point υ ($\upsilon = 9.2°$). It was assumed that the spin axis direction is perpendicular to the radius vector at the injection point and tangent to the satellite trajectory.

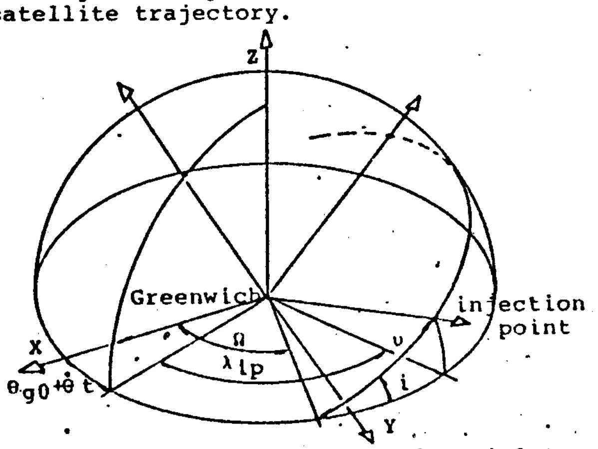

Fig. 6. Angles relating the inertial to terrestrial coordinates.

The α_{oi} values will define the two limits to the right ascension of the ascending node, in the form:

$$\Omega_{min} < \Omega < \Omega_{max} \quad (12)$$

with Ω_{min} and Ω_{max} obtained by substituing $\overline{\alpha}_{omin}$ and $\overline{\alpha}_{omax}$ in (11).

Consider now the injection point longitude λ_{ip} ($\lambda_{ip} = 326°$), the Greenwich sidereal time θ_{g0} at 0:00 hs GMT of the launch day and the Earth rotation rate $\dot{\theta}$ ($\dot{\theta} = 360.986$ °/day). From Figure 6, one gets

$$t_i = \frac{\Omega_i + \tan^{-1}(\cos i \tan\upsilon) - \lambda_{ip} - \theta_{g0}}{\dot{\theta}} \quad (13)$$

which furnishes the time interval of the launch day when the spacecraft could be injected in orbit in a way to satisfy the thermal constraint. The launch windows are shown in Figures 7 and 8, as functions of the launch date, considering the residual magnetic moment values of -0.5 Am^2 and -1.5 Am^2, respectively. The windows present a small oscillation during the year, basically due to the declination motion of the sun. It can be noted that there are launch windows in the whole year for both residual moment values and it lasts at least 2 hours around 12:00 hs GMT. The window corresponding to the -1.5 Am^2 residual magnetic moment is larger than the first one. The reason to that is the precession rate of the spin axis which is almost equal to the sun mean motion on the equatorial plane. The spin axis then follow the sun along the 6 months lifetime, increasing the right ascension limits, where the launch is possible, to almost 10 hours. To assure that the launch window is valid whatever be the spacecraft residual magnetic moment, between the limits imposed by Condition (3), Figures 7 and 8 must to be combined to produce a final window. Figure 9 presents the resulting launch window for the DCS.

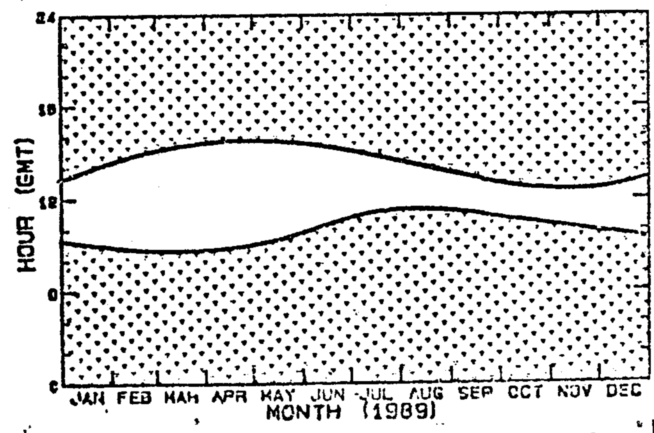

Fig. 7. Launch window for $m = -0.5$ Am^2.

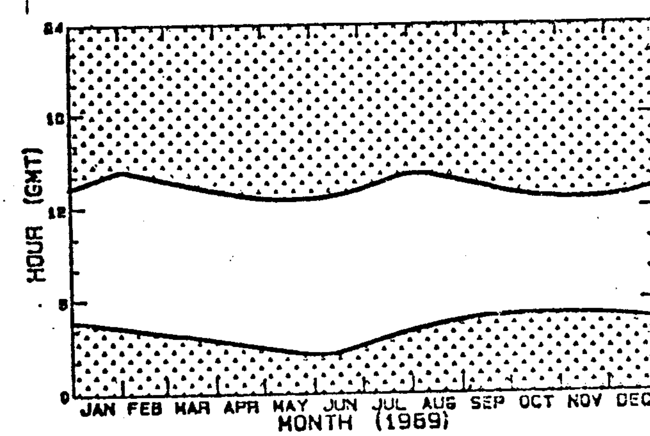

Fig. 8. Launch window for $m = -1.5$ Am^2.

CONCLUSIONS

The results showed that the best value for the residual magnetic moment should be -1.5 Am^2 approximately. For this value, the resulting launch window is of almost 10 hours around 9:00 GMT, whatever be the launch date. The equatorial plane component of the spacecraft's angular velocity precesses by an angle of 180° during the 6 months lifetime. Nevertheless, whichever be the residual magnetic moment between -1.5 Am^2 and -0.5 Am^2, there will be a large launch window, as shown in Figure 9.

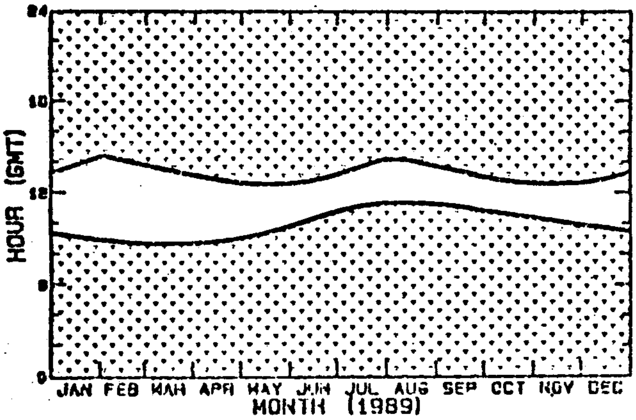

Fig. 9. Launch window for $-1.5 < m < -0.5$.

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