ORBIT MAINTENANCE STRATEGY FOR THE BRAZILIAN REMOTE SENSING SATELLITE

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ABSTRACT: The problem of orbit control in a remote sensing satellite is analyzed, with the aim to determine the major perturbations on the orbit and their effects. The mission requirements of the Brazilian Remote Sensing Satellite are applied to the problem, and the corrections and maintenance of the orbital elements are identified. The results showed that only semimajor axis corrections will be necessary to keep the ground trace repetitivity and the orbit circularity.

1. INTRODUCTION

A low Earth orbit like the orbit of the Brazilian Remote Sensing Satellite (BRSS) is subject to the action of several forces such as drag, solar radiation pressure, gravitational (including Earth, Sun and Moon), etc. Some of these forces modify one orbital parameter in a lesser extent than others in such a way that it can be assumed each parameter as being perturbed by the action of a single force. Table 1 shows the principal factor that changes each orbital parameter of a sun–synchronous orbit.

The influence of these forces on the orbit must be determined in such a way to compute the hydrazin consumption during orbital maneuvers and maintenance. An important result is also to provide values for the Orbit Control System design. The effects of these perturbations are analyzed separately in the following sections.

R. Vieira Martins, D. Lazzaro and W. Sessin (eds.), Orbital Dynamics of Natural and Artificial Objects, 85–97. Observatório Nacional, Rio de Janeiro, 1989.

2.1. SEMIMAJOR AXIS

The semimajor axis decay due to the atmospheric drag can be linearized near the nominal altitude:

$$a = a_c + a_1 t \quad (1)$$

where a_c is the semimajor axis just after a rising maneuver and a_1 is given by (Medeiros, 1983, equations 2.43, 2.45 and 2.48):

$$a_1 = -(\mu a_0)^{1/2} \rho C_d A_r / m \quad (2)$$

for circular orbits. In this expression, ρ is the atmospheric density, C_d is the drag coefficient, A_r is the spacecraft frontal area and m is the spacecraft mass.

μ = 398600 km^3/s^2
a_0 = 7017.89 km (nominal semimajor axis)
C_d = 3.8
A_r = 0.665 m^2
m = 150 kg (at end of life)

Table 1. The main perturbations on the BRSS orbital parameters

Semimajor axis	a	Aerodynamic drag
Eccentricity	e	Drag, solar radiation
Inclination	i	Sun and Moon gravity
Right ascension of the ascending node	Ω	Earth gravitational field (flatteness)
Perigee argument	ω	Earth gravitational field
Mean anomaly	M	Earth gravitational field

The atmospheric density is related to the exospheric temperature by (Jacchia, 1977):

$$T_i = 5.48 F_m^{0.8} + 101.8 F^{0.4} \quad (3)$$

For the launch date, the solar flux F (at 10.7 cm) and the averaged solar flux F_m are supposed to be equal to 260 (10^{-22} W/m^{2}Hz) maximum, with 97% of confidence. To the exospheric temperature should be added the contribution due to the geomagnetic activity. The geomagnetic index Kp is strongly affected by the solar storms, rising from the quiet daily values (less than 2) to the geomagnetic storms (during solar flare events), reaching 5 to 8. Jacchia gives the expression for the exospheric temperature variation as a function of the geomagnetic activity.

$$\Delta_G T_i = 57°.5 \, Kp \, (1 + 0.027 \, e^{0.4Kp}) \sin^4 \phi' \quad (4)$$

where ϕ' is the magnetic latitude:

$$\sin \phi' = 0.9792 \sin \phi + 0.2028 \cos \phi \cos(L - 291°) \quad (5)$$

and L, ϕ are the longitude and latitude. Using a mean latitude equal to one half of the orbit inclination (41°) and a longitude such that cos(L–291°)=1 (in order to maximize the exospheric temperature for safety purposes), it results ϕ'= 60°.67 and $\Delta_G T_i$ = 442°K and, hence, T_i = 1410 + 307 = 1717°K.

Adopting an exospheric temperature of 1800°K and with the given orbit altitude, it can be found from Jacchia (table 11) that $\rho = 1.66 \; 10^{-12}$ kg/m^3 and, finally, $a_1 = -128$ m/day.

The orbit decay is related with the repetition factor. When the semimajor axis is greater than the nominal value (639.73 km), the orbital period is greater than the nominal one. The satellite ground trace then presents a left motion relative to the nominal ground trace (figure 1). As the semimajor axis decreases due to drag, the motion ceases (at the nominal altitude) and then starts a right drift. The maximum drift imposed by mission requirements is 15 km at equator. The strategy is to make an orbit maneuver to increase the semimajor axis (and to nullify the eccentricity) every time the ground trace reaches the maximum deviation at the right of the nominal trace. The drift is given by:

$$D = R_e \theta \, \{[1-(a_0/a_c)^{1/2}] \, t + [(3/4)(a_0/a_c)^{3/2}(a_1/a_c)] \, t^2\} \quad (6)$$

R_e = 6378.16 km (Earth radius)

θ = 360°.98565/day

After a time interval of $t_c/2$ where t_c is the time between two semimajor axis maneuvers, the satellite altitude is equal to the nominal value and

$$a_0 = a_c + a_1 t_c/2 \quad (7)$$

Solving the above equations for a_c, it results that:

$$a_c = 7018.84 \text{ km}$$

The semimajor axis maneuver increases the satellite altitude by

$$\Delta a = 2\,(a_c - a_0) = 1.89 \text{ km}$$

and the time between two consecutive maneuvers is 14.8 days.

To compute the total hydrazine consumption in the maneuvers, other than 1800°K exospheric temperature must be used, as the strongly solar flare events do not occur every day. The results for more conservative values such as T_i = 1500°K are: a_1 = 56.4 m/day, Δa = 1.26 km and t_c = 22.3 days. The hydrazine consumption is 47 g per maneuver and 1.55 kg during the total spacecraft lifetime.

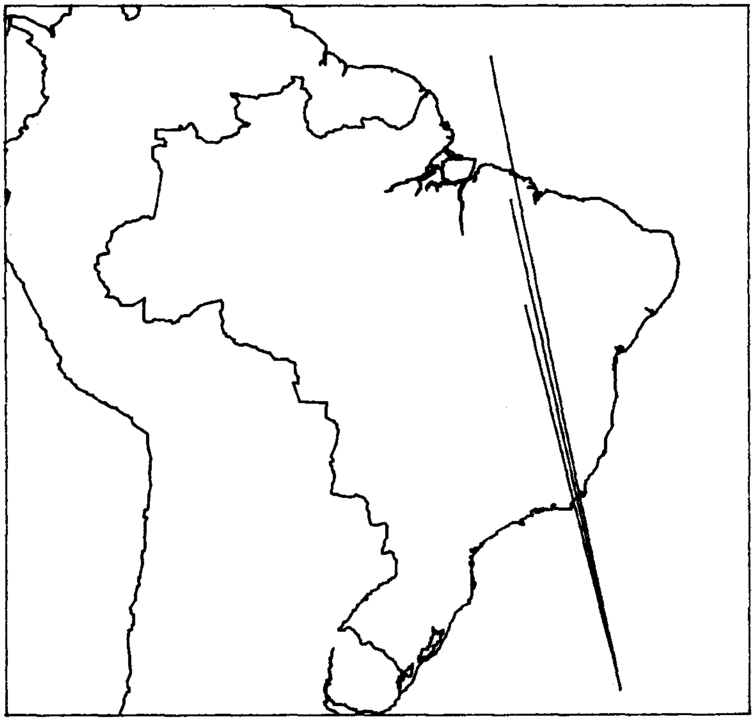

Fig. 1. The ground trace drift as function of the orbital altitude.

Figure 2 shows the spacecraft altitude relative to the nominal value, as a function of the ground trace deviation, and the maneuver strategy.

As the orbit decay rate a_1 depends on the solar flux F which is almost unpredictable, the altitude after the maneuver, a_c, must be calculated in a way to guarantee that the maximum ground trace deviation from its nominal sub–satellite trajectory is kept between the limits. If one uses a value for the solar flux greater than the real one (resulting a high atmospheric density), the real orbit decay would be smaller than the predicted one and, hence, it is possible that the ground trace exceeds the upper limit (Figure 3). On the other hand, if the predicted solar flux is smaller than the real one, the real orbit decay would be greater and the ground trace reaches the lower limit before the time between maneuvers has ended. Nevertheless, from the mission requirement impositions, the first result is more crictical than the second. It leads to use a minimum estimate for the solar flux in the computations of a_c, in order to avoid the ground trace to exceed the limits. It is suggested to subtract one to three solar flux standard deviations from its mean value, for example, in the a_1 calculation.

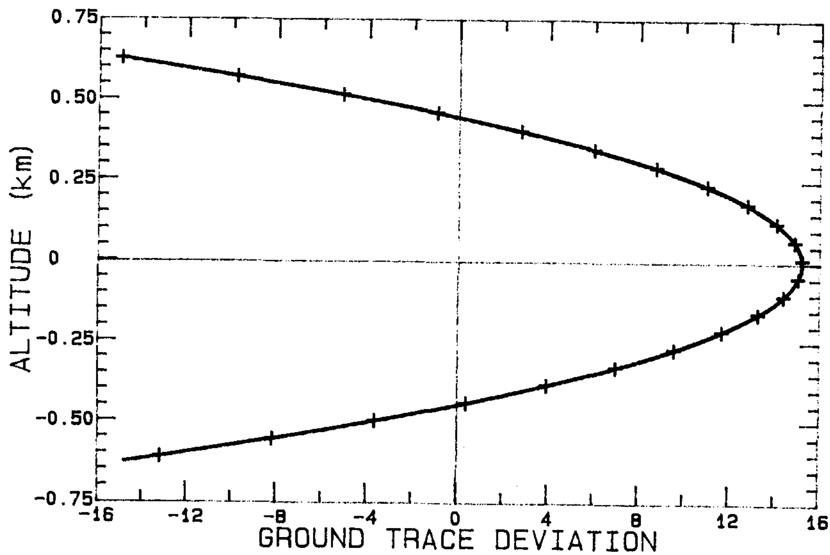

Fig 2. RSS orbit decay.

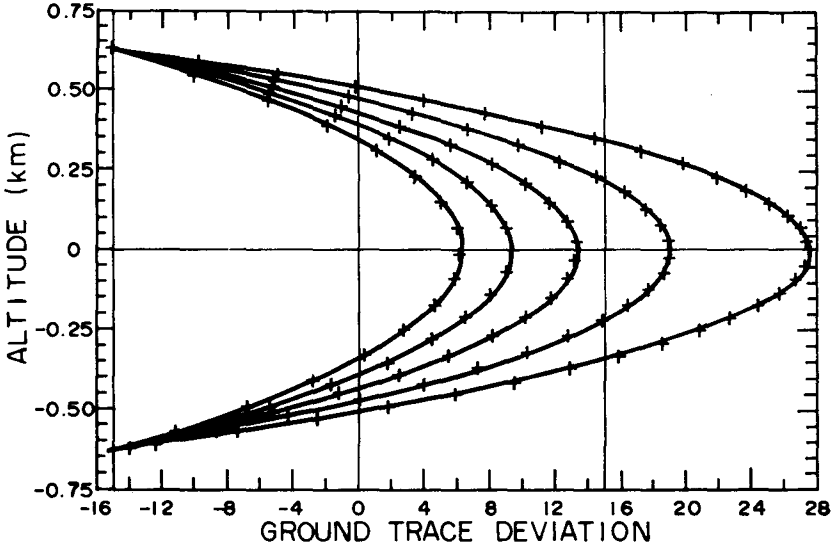

Fig. 3. Ground trace deviation for several semimajor axis decay rate.

For example, considering a minimum solar flux estimate of 150 with quiet geomagnetic days, the resulting orbit decay is 4.1 m/day, and hence, Δa = 0.34 km and t_c = 82.6 days. Nevertheless, if the actual solar flux is 240 (with strong solar storms), a new orbit correction will be necessary after 5.3 days.

2.2. ECCENTRICITY

The orbital eccentricity is affected by the solar radiation pressure and, to a lesser extent, by the aerodynamic drag. Kozai (1961) developed an expression to compute the eccentricity variation in one orbit (supposing an initially circular orbit) due to the radiation pressure:

$$\delta_e = a^2 \, F_S \, \left[(1/4) \; S(f) \; \cos 2E + (1/4) \; T(f) \; \sin \; 2E\right]_{E1}^{E2} + \int_{E1}^{E2} T(f) \; dE \qquad (8)$$

where S(f) and T(f) are trigonometric functions of the true anomaly f, and E is the eccentric anomaly. E_1 and E_2 are the eccentric anomalies at the entrance and exit from Earth shadow, respectively. F_s is the solar radiation force per unit of mass, divided by the Earth gravitational constant μ. The computations lead to:

$$e = 8 \times 10^{-8}/\text{day},$$

a value too small to be considered. The long period term of the eccentricity variation due to drag will always be decreasing and, thus, can also be neglected. Gravitational forces do not introduce secular variation on the eccentricity.

2.3. INCLINATION

The orbit inclination presents a short time variation due to the gravitational force of the Moon. Secular variation in the inclination is more influenced by the gravitational force of the Sun:

$$di/dt = -(3n_0^2/4n) \sin i \cos^4 (\varepsilon/2) \sin 2(\alpha_0 - \Omega) \qquad (9)$$

as introduced by Kozai. n_0 and n are the mean motion of the Sun (relative to Earth) and satellite, respectively. ε is the ecliptic inclination and $\alpha_0 - \Omega$ is related to the equator crossing hour, which is almost constant:

$$\alpha_0 - \Omega = -127^\circ.5$$

for H = 20:30 hs ascending node. The Sun and satellite mean motion are given by:

$$n_0 = 0^\circ.98565/\text{day}$$

$$n = (\mu/a_0^3)^{1/2} = 5310^\circ/\text{day},$$

resulting

$$di/dt = -0^\circ.044/\text{year}$$

This variation will affect the repetition factor $Q = 14.75$, fixed by mission requirements. As the repetition factor is a function of the semimajor axis and inclination, an inclination motion can be compensated for by an appropriate correction of the semimajor axis, so as to guarantee the repetition factor stability:

$$Q = (\dot{M} + \dot{\omega}) / (\dot{\theta} + \dot{\Omega}) \quad (10)$$

where $\dot{M}$ is the mean anomaly time derivative:

$$\dot{M} = n \{1 + [3J_2R^2_e(1-e^2)^{1/2} (1 - (3/2) \sin^2 i)]/[2a^2 (1-e^2)^2]\} \quad (11)$$

with J_2 being the second order zonal gravitational coefficient and R_e the equatorial Earth radius. The perigee argument and the right ascension of the ascending node time derivatives are:

$$\dot{\omega} = - [J_2R^2_e\dot{M} (2 - 2.5 \sin^2 i)] / [a^2(1 - e^2)^2] \quad (12)$$

$$\dot{\Omega} = - [3 J_2R^2_e\dot{M} \cos i] / [2 a^2(1 - e^2)^2] \quad (13)$$

As the orbit is near circular, neglecting the second order terms in J_2 and solving the Q expression for n, it has:

$$n = (Q \dot{\theta}) / [1 + A (4 \cos^2 i + Q \cos i - 1)] \quad (14)$$

and

$$a = (\mu/n^2)^{1/3} \quad (15)$$

with

$$A = (3 J_2 R^2_e) / (2 a^2) \quad (16)$$

The equations 14, 15 and 16 can be solved iteratively, as A is also a function of the semimajor axis *a*. Adopting an inclination of:

$$i_i = i_0 - (di/dt) (t_e/2) = 97^\circ.984$$

at the begining of the total spacecraft lifetime t_e = 2 years, at the end of life, the inclination will be:

$$i_e = i_0 + (di/dt)\,(t_e/2) = 97^{\circ}.896$$

The corresponding semimajor axis for these two inclinations are: a_i = 7017.965 km (639.80 km altitude) and a_e = 7017.815 km (639.65 km altitude). As the difference between these two values is less than the semimajor axis increment during a maneuver, the effect on the altitude due to the variation in the inclination can be included in the semimajor axis rising maneuver.

2.4. RIGHT ASCENSION OF THE ASCENDING NODE

To maintain the orbit in a synchronism with the sun motion in the equator plane, both inclination and altitude should be controlled during the satellite lifetime. As only semimajor axis maneuvers are intended to be made, it should be verified if the inclination variation does not modify the ascending node motion and, consequently, looses the syncronism.

In a sun–synchronous orbit, the right ascension time variation is:

$$\dot{\Omega} = -\,[3\,J_2 R_e^2 (\mu/a^3)^{1/2} \cos i]\,/\,[2\,a^2(1-e^2)^2] \qquad (17)$$

For a 639.73 km altitude orbit, the corresponding synchronous inclination is 97.94 degrees. This value does not remain constant during the satellite lifetime: due to the gravitational forces of Sun and Moon, the inclination presents a linear term with time,

$$i = i_0 + (di/dt)\,t \qquad (18)$$

that causes a variation on the ascending node rate:

$$\dot{\Omega} = -\,\{3\,J_2 R_e^2 (\mu/a^3)^{1/2}[\cos i_0 - \sin i_0\,(di/dt)\,t]\}\,/\,[2\,a^2(1-e^2)^2] \qquad (19)$$

for a circular orbit. After integration, it has:

$$\Omega = \Omega_0 + \dot{\Omega}_0\,t - \dot{\Omega}_0\,\mathrm{tg}\,i_0\,(di/dt)\,(t^2/2) \qquad (20)$$

where Ω_0 and $\dot{\Omega}_0$ are respectively the ascending node and its time derivative at t = 0. The equator crossing hour is given by:

$$H = \Omega - \alpha_0$$

where α_0 is the Sun's right ascension. In the nominal orbit, both Ω and α_0 precess at the same rate in the equator plane and, thus, it can be defined:

$$\Omega_0 = \Omega_n + \Delta\Omega_0$$

$$\dot{\Omega}_0 = \dot{\Omega}_n + \Delta\dot{\Omega}_0$$

where Ω_n and $\dot{\Omega}_n$ are the right ascension of the ascending node and its time derivative, supposing no variation in the inclination. $\Delta\Omega_0$ and $\Delta\dot{\Omega}_0$ should be calculated so as to guarantee the synchronism with Sun during the satellite lifetime. By substituting these equations into the equator crossing hour formulation, it has:

$$\Delta H = \Delta\Omega_0 + \Delta\dot{\Omega}_0\, t - \dot{\Omega}_0 \operatorname{tg} i_0\, (di/dt)\, (t^2/2) \qquad (21)$$

where ΔH is the variation on the equator crossing hour. Mission requirements have fixed the maximum allowable value for ΔH in +/−15 minutes (corresponding to an arc of +/−3°.75). This equation can be solved for by minimizing the maximum equator crossing time variation during the 2 years mission lifetime. Then:

$$d\,\Delta H/dt = 0 \qquad \text{for} \quad t = t_e/2$$

$$\Delta H = -\,\Delta\Omega_0 \qquad \text{for} \quad t = t_e/2$$

with t_e = 2 years. The first condition leads to:

$$\Delta\dot{\Omega}_0 = \dot{\Omega}_0 \operatorname{tg} i_0\, (di/dt)\, (t_e/2) \qquad (22)$$

Substituting the values i_0 = 97°.94, di/dt = − 0°.044 /year and a_0 = 7017.89 km yields:

$$\Delta\dot{\Omega}_0 = 1^\circ.98/\text{year}, \quad \text{or}$$

$\dot{\Omega} = 0°.991/\text{day}.$

The orbital inclination that causes a drift of 0°.991/day is

$i_0 = 97°.984$

Consequently, the satellite should be injected with an initial inclination of 97°.984. After one year, the inclination decreases to its nominal value of 97.94 degrees and, at the end of life, the inclination will be 97°.896.

From the second condition, it results:

$$\Delta\Omega_0 = -\Delta\dot{\Omega}_0 (t_e/8) = -0°.495$$

which corresponds to a 1 minute and 59 seconds of advance at the launch nominal time. With this procedure, the maximum deviation of the equator crossing hour from its nominal value due only to the change in the orbit inclination will be

$$\Delta H_{max} = 2\,\Delta\Omega_0 = 3.96 \text{ minutes.}$$

It can be concluded that only semimajor axis combined with eccentricity correction manouvers will be necessary to keep the repetibility of the orbit ground trace.

3. CONCLUSIONS

The orbit decay and semimajor axis manouvers were simulated in a numerical example. The effects of the air drag, solar radiation pressure, Sun and Moon gravitational forces and Earth geopotential (J_2) were taken into account in the computations. The orbit was numerically integrated and the ground trace deviation from its nominal value at the equator crossing on the ascending node was monitored. The velocity increment for the manouvers was computed by subtracting one solar flux standard deviation ($45.\ 10^{-22}$ W/m^{2}Hz) from its mean value ($198.\ 10^{-22}$ W/m^{2}Hz at the simulation epoch – starting on 1/1/1981) and neglecting the contribution due to geomagnetic activity. The same values for the aerodynamic and geometric properties of the spacecraft, as shown in Section 2.1, were adopted. The resulting semimajor axis and velocity increment were:

$\Delta a = 340$ m

$\Delta v = 0.183$ m/s.

The results of the integration are shown in figure 4. The manouvers are computed whenever the ground trace deviation reaches its upper value, +15 km. As all the velocity increments have the same magnitude for all the manouvers, the differences between the minimum ground trace deviations (figure 4) are only due to the increasing solar flux. In fact, during simulations, the observed mean solar flux has begun with the value 175 and ended with the value 206.

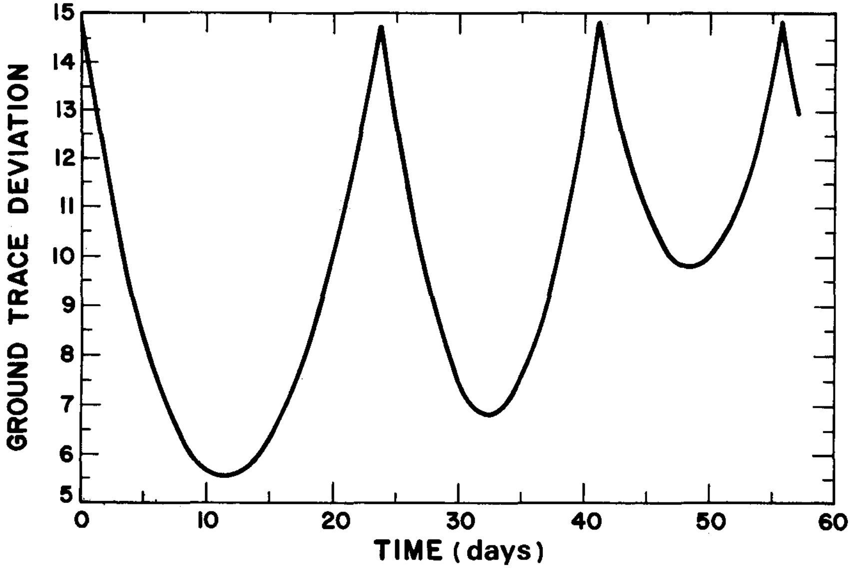

Fig. 4. Earth coverage maintenance.

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