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ORBIT MAINTENANCE STRATEGY

CÓDIGO AP

AR5

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11	1	01.06.88		
12	1	01.06.88		
13	1	01.06.88		
14	1	01.06.88		
15	1	01.06.88		

TABLE OF CONTENTS

1- INTRODUCTION

1.1- SCOPE

This document presents the response to the action proposed at the SSR CONCEPT REVIEW SYNTHESIS document, under the number S4.

1.2- THE RSS ORBIT PERTURBATIONS

The orbit of the RSS 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 the RSS.

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

Table 1. The main perturbations on the RSS orbital parameters.

The effects of these perturbations are analysed separately.

2- APPLICABLE AND REFERENCE DOCUMENTS

2.1- APPLICABLE DOCUMENTS

SSR SPACECRAFT SPECIFICATION, A-ETC-1001

REMOTE SENSING SATELLITE CONCEPT REVIEW. VOLUME II - SATELLITE, A-REV-1000.

SSR CONCEPT REVIEW SYNTHESIS, A-REV-0059

2.2- REFERENCE DOCUMENTS

JACCHIA, L. G. "Thermospheric Temperature, Density and Composition: New Models". Cambridge, MA, SAO 1977 (SAO Special Report n 375).

KOZAI, Y. "Effects of Solar Radiation Pressure on the motion of an Artificial Satellite". Cambridge, MA, SAO 1961 (SAO Special Report n 56).

MEDEIROS, V. M. "Análise de Missões: definição da geometria orbital de satélites artificiais". São José dos Campos, SP, INPE, Agosto 1983. (INPE-2483-TDL/141).

3- ORBIT PERTUBATION FORMULATIONS

3.1- SEMIMAJOR AXIS

The orbit decay due to the atmospheric drag can be linearized near the nominal orbit:

$$a = a_c + a_1 t$$

where a_c is the semi major axis just after a rising maneuver and a_1 is given by (Medeiros, 1983):

$$a_1 = -(\mu\ a_o)^{1/2}\ \rho\ C_D\ A/m$$

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

μ = 398600 km^3/s^2

a_o = 7017.89 km (nominal semimajor axis)

C_D = 3.8

A = 0.665 m^2 (configuration 2A)

m = 150 kg (at end of life)

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

$$T_\infty = 5.48\ \bar{F}^{0.8} + 101.8\ F^{0.4}$$

For the launching date, the solar flux F and the averaged solar flux $\bar{F}$ is supposed to be equal to 260 ($.10^{-22}$ W/m^2Hz) 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 in function of the geomagnetic activity.

$$\Delta_G T_\infty = 57.5^\circ \, Kp \, (1 + 0.027 \, e^{0.4 \, Kp}) \sin^4 \emptyset'$$

where $\emptyset'$ is the magnetic latitude:

$$\sin \emptyset' = 0.9792 \sin \emptyset + 0.2028 \cos \emptyset \cos(L - 291^\circ)$$

and L, $\emptyset$ are the longitude and latitude. Using a latitude equal to a half of the orbit inclination (41°) and a longitude such that cos(L-291)=1, it results $\emptyset' = 60.67^\circ$ and $\Delta_G T_\infty = 442^\circ K$ and, hence, $T_\infty = 1410 + 307 = 1717^\circ K$.

Adopting an exospheric temperature of 1800°K and with the 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. 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:

$$D = R_e \dot{\theta} \left\{ \left[1 - \left(\frac{a_o}{a_c} \right)^{3/2} \right] t + \frac{3}{4} \left(\frac{a_o}{a_c} \right)^{3/2} \left(\frac{a_1}{a_c} \right) t^2 \right\}$$

R_e = 6378.16 km (Earth radius)

$\dot{\theta}$ = 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

$$a_o = a_c + a_1 t_c/2$$

Resolving the above equations to a_c, it results that:

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

The semimajor axis maneuver increases the satellite altitude of

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

and the time between two consecutive maneuvers is 14.8 days.

To compute the total hydrazine consumption in the maneuvers, other than a 1800°K exospheric temperature must be used, as the strongly solar flare events does not happen as daily. The results for T_∞ = 1500°K are: a_i = 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. For the 1A satellite configuration, t_c = 26.6 days and the total hydrazine mass is 1.1 kg.

Figure 1 shows the spacecraft altitude, as function of the ground trace deviation, and the maneuver strategy.

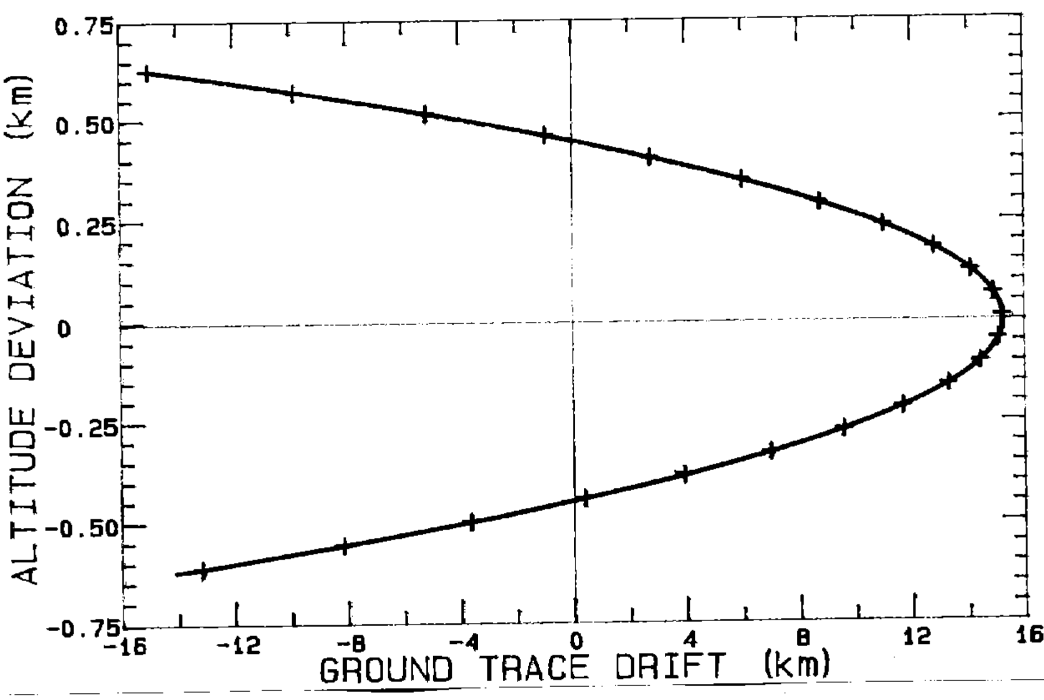

Fig 1. RSS orbit decay.

As the orbit decay rate a_i depends on the solar flux $F_{10.7}$ 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 is kept between the limits. It leads to use the minimum estimate for the solar flux in the computations of a_c. It is suggested to subtract from one to three solar flux standard deviations from its mean value, for example, in the a_1 calculation.

Considering a minimum solar flux estimate of 150 with quiet geomagnetic days, the resulting orbit decay is 4.1 m/day, and hence, Δa = .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.

3.2- 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 a circular initial orbit) due to the radiation pressure:

$$\delta e = a^2 F \left[\frac{1}{4} S(f) \cos 2E + \frac{1}{4} T(f) \sin 2E \right]_{E_1}^{E_2} + \int_{E_1}^{E_2} T(f)\, dE$$

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 anomaly at the entrance and exit from Earth shadow, respectively. F is the solar radiation force per unit of mass, divided by the Earth gravitational constant μ. The computations lead to:

$$\dot{e} = 8.\ 10^{-8}\ /\text{day}.$$

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

3.3- INCLINATION

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

$$\frac{di}{dt} = -\frac{3}{4}\frac{n_o^2}{n} \sin i \cos^4 \varepsilon/2 \sin 2(\alpha_o - \Omega)$$

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

$$\Omega - \alpha_o = 127.5^\circ$$

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

$$n_o = 0.98565^\circ/\text{day}$$

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

resulting

$$\frac{di}{dt} = -0.044^{\circ}/\text{year}.$$

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 on the semimajor axis, as to guarantee the repetition factor stability:

$$Q = \frac{\dot{M} + \dot{\omega}}{\dot{\theta} + \dot{\Omega}}$$

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

$$\dot{M} = n \left[1 + \frac{3}{2} J_2 \frac{R_e^2}{a^2} \frac{(1 - e^2)^{1/2}}{(1 - e^2)^2} \left(1 - \frac{3}{2} \sin^2 i \right) \right]$$

with J_2 being the 2^{nd} 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:

$$\dot{\omega} = \frac{3}{2} J_2 \frac{R_e^2}{a^2(1 - e^2)^2} \dot{M} \ (2 - 2.5 \sin^2 i)$$

$$\dot{\Omega} = - \frac{3}{2} J_2 \frac{R_e^2}{a^2(1 - e^2)^2} \dot{M} \cos i$$

As the orbit is near circular, neglecting the second order terms in J_2^2 and solving the Q expression for n, it has:

$$n = \frac{Q \dot{\theta}}{1 + A(4 \cos^2 i + Q \cos i - 1)}$$

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

with $A = \frac{3}{2} J_2 \frac{R_\bullet^2}{a^2}$

The above equations can be solved iteractively, as A is also a function of the semimajor axis a. Adopting an inclination of:

$$i_i = i_o - \frac{di}{dt}\frac{t_\bullet}{2} = 97.984^\circ$$

at the beggining of the total spacecraft lifetime $t_\bullet$ = 2 years, at the end of life the inclination will be:

$$i_\bullet = i_o + \frac{di}{dt}\frac{t_\bullet}{2} = 97.896^\circ$$

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

3.4- RIGHT ASCENSION OF THE ASCENDING NODE

To maintain the orbit with 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 confirmed 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:

$$\dot{\Omega} = -\frac{3}{2} J_2 \frac{R_e^2}{a^2(1 - e^2)^2} (\mu/a^3)^{1/2} \cos i$$

For a 630.73 km altitude orbit, the corresponding synchronous inclination is 97.84 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,

$$i = i_o + \frac{di}{dt} t$$

that causes a variation on the ascending node rate:

$$\dot{\Omega} = -\frac{3}{2} J_2 \frac{R_e^2}{a^2(1 - e^2)^2} (\mu/a^3)^{1/2} \left(\cos i_o + \sin i_o \frac{di}{dt} t \right)$$

for a circular orbit. After integration, it has:

$$\Omega = \Omega_o + \dot{\Omega}_o t - \dot{\Omega}_o \operatorname{tg} i_o \frac{di}{dt} \frac{t^2}{2}$$

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

$$H = \Omega - \alpha_o$$

where α_o is the Sun right ascension. In the nominal orbit, Ω and α_o precesse at same rate in the equator plane and, thus, it can be defined:

$$\Omega_o = \Omega_n + \Delta\Omega_o$$

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

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. $\Delta\Omega_o$ and $\Delta\dot{\Omega}_o$ should be calculated so as to guarantee the synchronism with Sun during the satellite lifetime. By substituing these equations into the equator crossing hour formulation, it has:

$$\Delta H = \Delta\Omega_o + \Delta\dot{\Omega}_o\, t - \dot{\Omega}_o\, tg\, i_o \frac{di}{dt} \frac{t^2}{2}$$

where ΔH is the variation on the equator crossing hour. Mission requirements have fixed the maximum allowable value for ΔH in $\pm$ 15 minutes (corresponding to an arc of $\pm$ 3.75°). This equation can be solved for by minimizing the maximum equator crossing time variation during the 2 years mission lifetime. Then

$$\left.\frac{d\Delta H}{dt}\right|_{t=t_e/2} = 0$$

$$\Delta H\Big|_{t=t_e/2} = -\Delta\Omega_o$$

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

$$\Delta\dot{\Omega}_o = \dot{\Omega}_o \, tg \, i_o \frac{di}{dt} \frac{t_*}{2}$$

Substituing the values i_o = 97.94°, di/dt = -0.044°/year and a_o = 7017.89 km yields:

$$\Delta\dot{\Omega}_o = 1.98°/\text{year, or}$$

$$\dot{\Omega}_o = 0.991°/\text{day.}$$

The orbital inclination that causes a drift of 0.991°/day is

$$i_o = 97.984°$$

Consequently, the satellite should be injected with a 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:

$$\Delta\Omega_o = - \Delta\dot{\Omega}_o \frac{t_*}{8} = - 0.495°$$

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

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

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. These manouvers should be commanded when the spacecraft is visible from the main ground station (Cuiaba). Nevertheless, the circularization manouvers should be commanded near the apogee position which can not be in visibility at that time and, then, the circularization will proceed by pre-programmed commands.

4- CONCLUSIONS

The orbit dacay 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 ($45 \cdot 10^{-22}$ W/m^2Hz) from its mean value ($198 \cdot 10^{-22}$ W/m^2Hz at the simulation epoch starting on 1/1/1981) and neglecting the contribution due to geomagnetic activity. It was adopted the same values for the aerodynamic and geometric properties of the spacecraft as shown in Section 3.1. The resulting semimajor axis and velocity increment were:

$$\Delta a = 340 \text{ m}$$

$$\Delta v = 0.183 \text{ m/s}.$$

The results of the integration are shown in Figure 2. 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 2) are only due to the increasing solar flux. In the integrations, the mean solar flux begins with the value 175 and ends with the value 206.

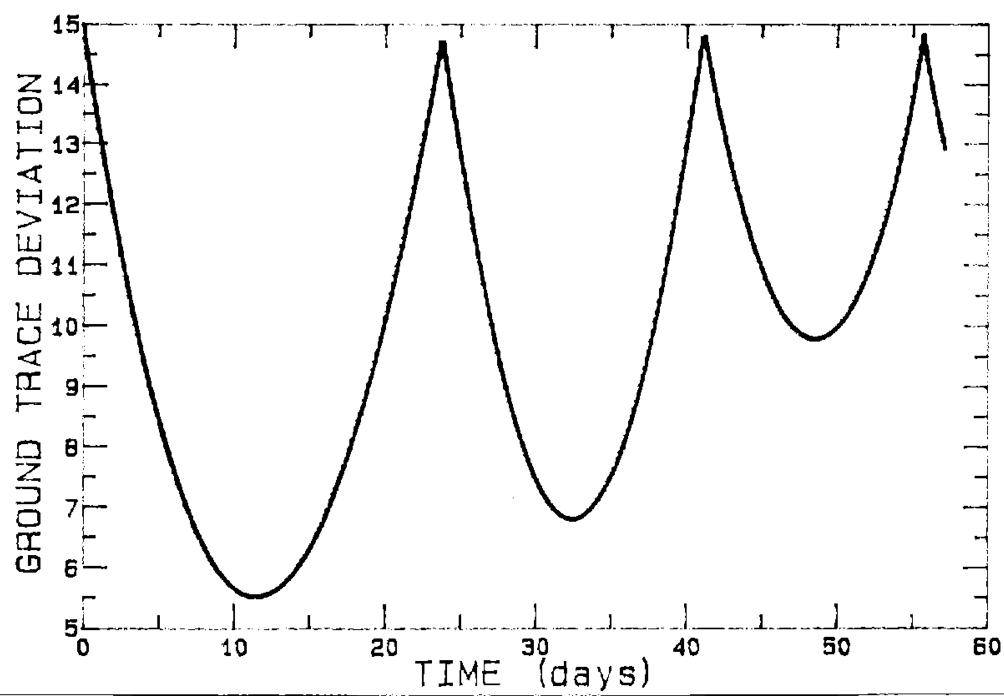

Fig. 2. Earth coverage maintenance.