ORBIT MAINTENANCE AND CONTROL OF "CBERS"

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Abstract

CBERS (China Brazil Earth Resources Satellite) marks the beginning of the operational remote sensing satellite era in Brazil. The mission objectives are directed towards the optimum and effective management of National natural resources. The other objective is to utilize the data from CBERS in conjunction with supplementary or complementary information from other resources for survey and management of important areas such as agriculture, geology and forestry. The nominal orbit for CBERS is repetitive, polar and near circular frozen orbit at an altitude of 770Km. The mission requirements imposed stringent accuracy on orbit computation system. Precise orbit maintenance is the mandatory mission requirement. Orbit maintenance maneuvers assures all nodes of 373 orbits in 26-day repeat ground track remain within a 20km equatorial longitude bandwidth. The other requirement is to maintain the local time within a small window of ±10 minutes. Orbit determination, maneuver execution, atmospheric prediction errors and third body perturbation effects limit overall targeting performance. The paper emphasizes the effects of the drag modeling errors due to solar activity especially the 27-day outlook of the 10.7cm solar flux, and the complexity in ground track maintenance in presence of high solar activity during the lifetime of CBERS. The paper further describes the maneuver strategy and the mathematical modeling applied for orbit maintenance essentially for keeping frozen orbit throughout the operational life. Our choice of appropriate force modeling and computation of precise decay rates is described. Performance was measured by predictions against real world satellites. This paper is the result of the orbit keeping maneuver analysis performed for CBERS satellite constellation and the algorithms presented are being implemented in the operations control center.

Key words: CBERS, orbit maintenance, frozen orbit.

Introduction

A Satellite, no matter how sophisticated, is of little use if its position in space is not known accurately. During initial phase, for satellite acquisition and commanding, the control center should know well in advance, with sufficient accuracy, the position of the spacecraft. Secondly for processing payload data of precision data products, the position and velocity should be estimated very precisely. The CBERS (China Brazil Earth Resources Satellite) operational orbit is near circular, near polar and frozen orbit. The ground resolution at the nadir is expected to be 19m. CBERS carries a CCD camera, an IR multi spectral scanner along with a WFI (Wide Field Imager) as primary payloads. The swath width is 113km and the inter-track distance at Equator is 107.4km. The local time at the descending node crossover is required to be 10:30 hours. The mission requirements demand precise orbit determination and orbit keeping. The ground track must be maintained in small dead band limits. It is necessary to have a precise trajectory modeling. Preliminary maneuver design determines approximate time and magnitude[1] of orbit corrections necessary for keeping the ground track under the required limits. The paper deals with orbit computation system followed by orbit keeping methodology and operational scenario. The software was validated using both simulated and live satellite data.

CBERS Operational orbit

The operational orbit of CBERS is given as:

Mean semi-major axis a = 7148.86km
Mean eccentricity e = 0.0011
Mean inclination i = 98.504°
Mean argument of perigee = 90°

The main characteristics of the operational orbit are as follows:

Local Sun Time: It corresponds to the angle between the meridian plane containing the mean sun and the meridian plane containing the satellite referring to the node.

Ground track repeatability: Every 26 days (cycle of observation) the satellite fly over the same ground track. The reference grid is to be controlled at the equator crossing. This grid includes 373 descending nodes resulting in 113km spacing. It is intended to keep two satellites phased to reduce cycle time from 26 to 13 days. Fig. 1 shows typical ground trace of CBERS.

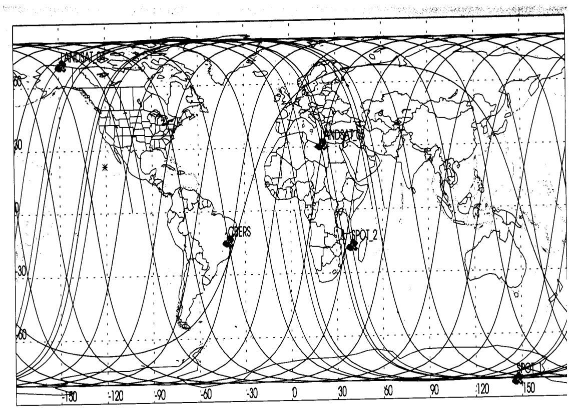

Fig. 1 - Ground trace of CBERS

Brief payload Description

The CBERS payload consists of CCD camera, IR-MSS and WFI (Wide Field Imager). The CCD camera acquires high quality image information in visible and near infrared spectral bands. The ground resolution in the processed imagery is better than 20m and spectral band registration is better than 0.3 pixels. The swath width is 113km. The IR-MSS has ground resolution of 80m. The swath width is 120km. The WFI ground resolution is 256m and swath is 885km.

Mission Requirements

The launch vehicle injects the spacecraft into a working orbit. Orbit maneuvers are to be executed to remove injection errors through the execution of 20N and 1N thrusters firing operation. The orbit is to be determined within ±1km and the semi-major axis to be estimated better than 50m. During station keeping phase, the local time is to be controlled within ±10 minutes. The frozen orbit is required to meet payload specifications. The ground track shall be maintained within ±10km. Due to drag, solar radiation pressure and other small forces, $da/dt \neq 0$ and therefore $d\omega/dt$ and de/dt which are functions of a can not realistically be kept zero, instead, the closed contour of $e \times \omega$ slowly spiral outwards[2].

Orbit acquisition

Orbit acquisition phase places the satellite from its working orbit to the desired targeted orbit. It includes in-plane orbit corrections and orbit inclination adjustment along with realization of frozen perigee. This includes the initial nominal ground track adjustment.

Trim Maneuver concept

Subsequent trim maneuvers will be used to counteract the effects of drag and other perturbations. A number of trim correction maneuvers is needed to maintain the desired orbit. The atmospheric density is strongly influenced by the intensity of solar activity[2] which has short term (27 days) and long-term (11 years) variations. During the operational life of CBERS the solar activity is expected to be around the maximum and therefore the modeling of atmospheric density gains paramount importance[3]. In this scenario ground track prediction is a challenging task in orbit maintenance.

Mean orbit elements

Orbit determination, ephemeris generation and long range predictions can be accomplished with mean elements. The osculating elements represent the true position and velocity but are poorly behaved over time as basis for prediction. On the other hand the mean elements do not represent the actual satellite position and velocity but are well behaved over time. Most of conversion techniques between both sets follow the methods of Kozai, Brouwer[4], or Merson[5], where short periodic perturbations are removed from osculating elements to yield the mean elements. Those analytical methods are fast and efficient computationally independent of prediction span. However accuracy is limited and inclusion of perturbations of dissipative nature are cumbersome. We adopted the Merson's method to generate the initial mean elements, and included in the analytical part the zonal harmonics perturbations (up to any order) and luni-solar perturbations. A numerical averaging technique[6] was used to evaluate small perturbations as the atmospheric drag. In this case, the short period oscillations can be accounted for by the process of variation of parameters in conjunction with a precise force model including atmospheric density computation. It was clearly demonstrated for orbit keeping purposes, especially for ground track maintenance, that accurate air drag modeling is essential along with geo-gravitational potential including higher order harmonics[7,8]. Table 1 reproduces one of our results already published elsewhere[6]:

Table 1- Satellite IRS-1C at 1997/04/11 00:00:00

Parameter	Merson's method	Brouwer' method	Averaged
a (km)	7195.130	7195.145	7195.118
e	0.00115	0.00116	0.001142
i (°)	98.6924	98.6924	98.6924
ω (°)	90.469	90.471	90.395
Ω (°)	177.137	177.137	177.138

Ground track variation

Accurate prediction of ground track is essential to control and maneuver spacing requirements. The actual spacing between successive ground tracks is[9]:

$$S = P_n(\omega_e - \dot{\Omega})$$

$$P_n = \left(\frac{2\pi}{\sqrt{\mu}}\right) a^{3/2}\left[1 - \frac{3}{2}J_2\left(\frac{R}{a}\right)^2 (4\cos^2 i - 1)\right]$$

$$\dot{\Omega} = -\frac{3}{2}J_2\frac{\sqrt{\mu}}{a^{3/2}}\left[\frac{R}{a(1-e^2)}\right]^2 \cos i$$

where P_n is the nodal period, ω_e is the Earth rotational rate, $\dot{\Omega}$ is the nodal precession rate, J_2 is the second zonal harmonic and μ is the Earth gravitational constant. For exact repeat ground track one has:

$$S_R = \frac{360D}{R}$$

where D and R are integer days and integer revolutions to repeat. The actual spacing between the successive ground tracks varies due to perturbations that change nodal period and nodal precession rate. When $S > S_R$ the actual ground track drifts west of the reference, whereas when $S < S_R$ the drift is eastwards. The other mission requirement is to minimize altitude variations; this can be done by means of a small value of eccentricity and further reduced by restricting the precession rate of argument of perigee. The governing equations[10] are:

$$\frac{d\omega}{dt} = \left(\frac{3nJ_2R^2}{a^2(1-e^2)^2}\right)(1 - \frac{5}{4}\sin^2 i) *$$
$$\left[1 + \frac{J_3}{2J_2}\frac{R}{a(1-e^2)}\left(\frac{\sin^2 i - e\cos^2 i}{\sin i}\right)\frac{\sin\omega}{e}\right]$$

$$\frac{de}{dt} = -\left(\frac{3nR^3}{2a^3}\frac{J_3 \sin i}{(1-e^2)^2}\right)\left(1 - \frac{5}{4}\sin^2 i\right)\cos\omega$$

where n is the mean motion, J_2 and J_3 are the zonal harmonics. It is seen from above equations that ω is to be 90° and e is to be suitably small to keep ω frozen at 90°. The e and ω move in counterclockwise on closed contours with the apsidal period. Thus the values of e and ω at the stable points will keep perigee location relatively constant and keeps the altitude constant for a given latitude[11]. Thus one concludes that it is required to control the eccentricity vector to satisfy the mission requirements.

Local time deviation

The lunar and solar gravitational attractions cause the inclination to vary periodically. The ground track targeting procedures absorbs the effects the predicted inclination variations by adjusting the mean semi-major axis to maintain the repeated ground track within the dead band limits. The effect of these perturbations can cause the variation in inclination and as result of this local time gets variations. It is computed as

$$H = \frac{T_{TE}}{2\pi}(\Omega - \Omega_H)\left|\frac{T_T}{2}\right|$$

$$\frac{d^2\Delta H}{dt^2} = \frac{T_{TE}}{2\pi}\omega_S\left[-\tan i\frac{di}{dt} + \frac{7}{2a}\frac{da}{dt}\right]$$

and can be reduced as

$$\frac{d^2\Delta H}{dt^2} = \frac{T_{TE}}{T_S}\left[-\tan i\frac{di}{dt} + \frac{7}{2a}\frac{da}{dt}\right]$$

where ω_s = 1.99099299x10^{-7} rad/s mean rotational rate of Sun around Earth. T_{TE} is mean period of solar day and T_S is the Sun period. It is planned to provide initial biasing for the inclination to do away with inclination corrections during the life time.

Ground Track deviation and prediction

The lower value of drag results in maximum westward deviation, which may be outside the desired box limits. If the drag effect is higher than predicted then the maximum westward deviation occurs much sooner with cross over of the dead band limit. The decay rate causes the satellite ground track to drift eastward. Assuming a constant decay rate, the accumulated change in satellite equatorial longitude after time t is

$$\Delta\lambda_d = \frac{3}{4}\omega_e \frac{C_D A}{m}\rho V t^2$$

where ω_e is earth's rotational rate, A is the cross sectional area of satellite, C_D is drag coefficient, m is the mass of the spacecraft, ρ is the density, and V is the velocity of the satellite. Periodic maneuvers are to be executed for drag makeup maneuvers to maintain the ground track within the required limits The accumulated change in equatorial longitude after time t is due to the combined effect of drag and compensating maneuver ΔV:

$$\Delta\lambda = -3\omega_e \frac{\Delta V}{V} t + \frac{3}{4}\omega_e \frac{C_D A}{m}\rho V t^2$$

The maximum maneuver magnitude ΔV_{Max} provides maximum maneuver spacing T_{Max}, while maintaining the ground track just inside the control width $\Delta\lambda_{Max.}$ This balance of drag maneuvers occurs at $T_{Max}/2$ when $d\Delta\lambda / dt \cong 0$; i.e.:

$$\Delta V_{Max} = \left[\frac{C_D A}{m}\frac{\rho V}{3}\frac{\Delta\lambda_{Max}}{\omega_e}\right]^{1/2}$$

$$T_{Max} = \frac{4m\,\Delta V_{Max}}{C_D A \rho V}$$

The offset for the semi-major axis is given by:

$$\delta a_o = \sqrt{-\frac{4a\,\dot{a}\,\Delta\lambda}{3\omega_e R}}$$

which provides the required biasing in a from the edge of the box. The prediction for subsequent maneuver is:

$$\Delta t = 2\left[-\frac{4a\,\Delta\lambda}{3\,\dot{a}\,\omega_e R}\right]^{1/2}$$

Here a is the current semi-major axis and a_o is the targeted semi-major axis, $\dot{a}$ is the decay rate described. The east and west boundary maneuvers can then be computed.

Inclination Adjustment and Local time

In view of an initial inclination biasing, local time increases initially and comes to its nominal value and further reduces and touches the other edge of the local time limits. Fig. 2 shows schematically the effect of adding such a bias.

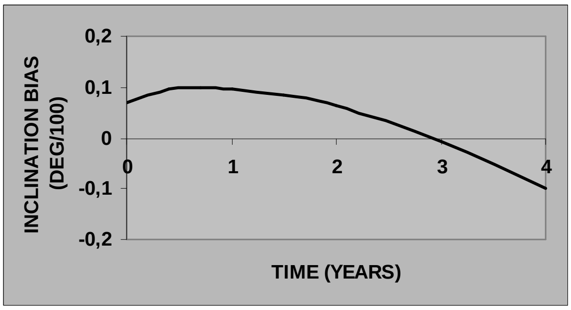

Fig. 2 – Variation of inclination bias with time

Inclination i reduces due to perturbations and as a result local time variation with respect to nominal value. Nevertheless an initial bias of 0.071745° in inclination shall ensure local time profile within the specified box for a span of 4 years.

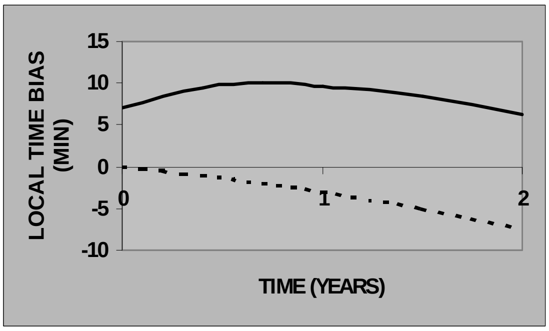

Fig. 3 – Local time variation with time. Upper curve for biased inclination, dashed curve for unbiased inclination

Generally speaking, the inclination rate is given by:

$$\frac{di}{dt} = -\frac{3n_s^2}{4n}\sin i \cos^4(i_s/2)\sin[2(\alpha_s - \Omega)]$$

where i_s is the obliquity of the ecliptic, n_s is the apparent angular rate of the Sun, n is the mean motion, α_s is the mean longitude.

Maneuver strategy

Once maneuvers are performed the semi-major axis and eccentricity vector (meaning magnitude of e and direction of ω) will return to their targeted nominal values[2]. The coordinate system for calculating the thrust is shown in Figure 4.

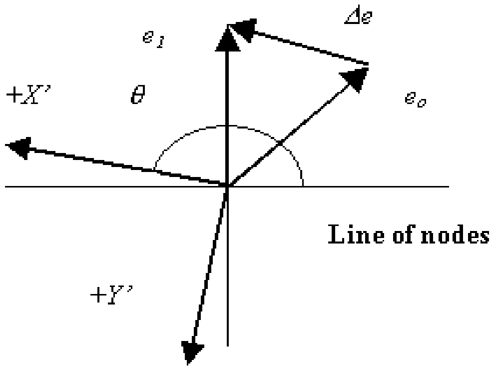

Fig.4 – Coordinates for maneuver

The coordinate system (X', Y') is defined for convenience such that $\Delta e_y = 0$ and $\Delta e_x = \Delta e$. The location of the X' axis is referenced to the line of nodes by θ which is a function of the initial desired value of e and ω. If the desired value of ω is taken as 90° for the frozen perigee then

$$\theta = \tan^{-1}\left(\frac{e_1 - e_o \sin\omega_o}{-e_o \cos\omega_o}\right)$$

where e_o is the initial eccentricity vector, Δe is the change in eccentricity vector, e_1 is the desired eccentricity vector, and θ is measured clockwise from ascending node. Table 1 (at the end) gives the location and direction of the thrust, where $\theta = f + \omega_o$, f is the true anomaly, ω_o is the initial argument of perigee, a_o is the initial semi-major axis, a_1 is the targeted semi-major axis, $\Delta a = a_1 - a_o$, e_o is the initial eccentricity, and e_1 is the targeted eccentricity. For illustration purposes, Table 2 gives a test case for satellite IRS-1C in which are stated the differences between the epoch and nominal orbit elements. In Table 3 one sees the resulting maneuver options as computed by the software. It yields several options listed in order of preference, as well as the ΔV , the TC station, and post-maneuver elements.

Table 2 – Initial conditions for IRS-1C test case

Parameter	Nominal mean elements	Epoch Merson's mean elements	Difference
a (km)	7195.332	7195.010	0.322
e	0.001140	0.001138	0.000002
i (°)	98.6911	98.7061	-0.0150
ω (°)	90.000	92.5988	-2.5988

Table 3 – Maneuver computed (osculating elements)

Parameter	Fist preference	Second preference	Second preference
Time (UTC)	97/09/21 05:23:30	97/09/21 21:05:24	97/09/22 05:02:30
a (km)	7204.278	7204.284	7204.274
e	0.001275	0.001238	0.001227
i (°)	98.6972	98.6972	98.6973
ω (°)	113.9351	114.7515	110.1329
ΔV (m/s)	0.1667	0.1667	0.1667
TC station	Bangalore	Bangalore	Bangalore

Summary of perturbations

The effects of the perturbations on the ground track are analyzed. To start with, the geopotential model[8] uses the truncated GEM-T_3 up to J_{29}. The luni-solar perturbation is included and the analytical form of planetary ephemeris is used. This perturbation causes periodic variations in inclination and affect the ground track. The atmospheric drag causes continuous decay in semi-major axis resulting in an eastward drift in the ground track. We adopted the MSIS-90 atmospheric density model that was considered one of the best models available. Nevertheless unpredictable solar flares and geomagnetic storms cause large changes in atmosphere, which gives rise to uncertainty in ground track prediction. The effects of low and high drag effects can be studied for extremes. The drag extremes are easily characterized using constant minimum and maximum values of solar flux $F_{10.7}$. Figure 5 shows the density variation with solar flux.

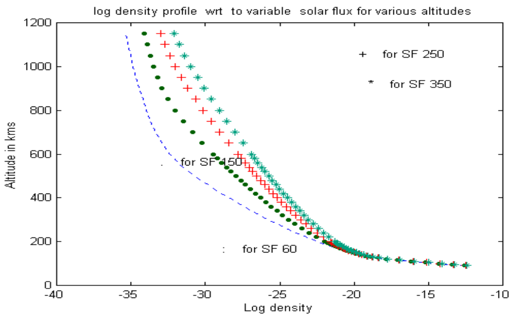

Fig. 5 - Density profile with varying solar flux

Concluding Remarks

CBERS (China Brazil Earth Resource Satellite) mission is the first operational remote sensing mission of Brazil. The orbit control and orbit keeping requirements are somewhat stringent. The ground track is to be controlled within dead-band limits of ±10km. During the operational life, the satellite shall experience high decay rates, due to the predicted peak of solar activity. The orbit maintenance methodology for CBERS is described. As far as the ground track maintenance and in turn the maneuver planning software is concerned one has to make a judicious choice of the orbit generator and the appropriate modeling of drag decay rates along with density model choice. The developed software was tested with live satellite data and also extensive simulations were carried out, including comparisons with numerical orbit generators. The analysis indicated the atmospheric modeling errors are the dominant for ground track prediction. The maneuver design study reflected thus the operational software system for orbit control and maintenance of CBERS realistically. Indeed, the study demonstrated the feasibility and applicability of the methodology adopted for precise ground track control.

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References

[1] Shapiro, B. et al. "Maintenance of exact Repeat ground track, The GEOSAT-ERM", *Advances in Astronautical Sciences,* 88, p.4301, 1988

[2] George Born.H et al. "Orbit Analysis for GEOSAT-ERM ", *Journal of Astronautical Sciences*, Vol. XXVI, no. 4, p. 425-446, Oct.-Dec. 1978

[3] Prasad .R "Review of air density models for orbit determination", *Artificial Satellites, Planetary Geodesy*, no. 22, Polish Academy of Sciences, 1994

[4] Brower D., "Solution of the problem of Artificial Satellite theory without Drag", *The Astronamical Journal*, no. 1274, p. 378, 1959

[5] Merson.R.H "The dynamical model of PROP ", *ERA*, TR 66255, Aug. 1966

[6] Prasad, R.; Kuga, H.K., "Analysis of averaged orbital elements for ground track maintenance". *49th International Astronautical Congress*, Paper IAF-98-A-2.03, 1998

[7] Prasad, R. et al. "Precise Orbit Determination for Oceansat mission", Proceedings of *International Symposium on Space Flight Dynamics*, CÉPADUÉS, Jun. 1995

[8] Tapley, B.D., et al. "Precision Orbit Determination for TOPEX/Poseidon", *Journal of Geophysical Research*, Vol. 96, Dec. 1994

[9] Bhat, R.S., "Topex/Poseidon Orbit Acquisition Maneuver design", AAS/AIAA *Astrodynamis Specialist conference*, AAS 91-514, Aug. 1991

[10] Cutting, E. et al. "Orbit Analysis for SEASAT–A", *Journal of Astronautical Sciences*, Vol. XXVI, no. 4, Oct.-Dec. 1978, p. 315-342

[11] Michieau, P. "Survey on Spot System orbit-keeping Exploitation", *ESA Symposium of Spacecraft Flight Dynamics*, Oct. 1991, ESA-SP-326

Table 1 – In plane maneuver thrusts

Element	Thrust location θ_1	Thrust location θ_2	Thrust magnitude ΔV_1	Thrust magnitude ΔV_2
a	any	$\theta_1 + 180°$	$\frac{1}{4}\frac{\Delta a}{a_o}\left(\frac{\mu}{a_o}\right)^{1/2}$	ΔV_1
e	$270°$ if $e_1 < e_o$ $90°$ if $e_o < e_1$	$\theta_1 + 180°$	$\frac{1}{4}\left[\frac{\mu}{a_o}(e_1^2 + e_o^2 - 2e_1 e_o)\right]^{1/2}$	$-\Delta V_1$
ω	$\tan^{-1}\left(\frac{1-\sin\omega_o}{-\cos\omega_o}\right)$	$\theta_1 + 180°$	$\frac{e_o}{4}\left[\frac{2\mu}{a_o}(1-\sin\omega_o\right]^{1/2}$	$-\Delta V_1$
$a+e+\omega$	$\tan^{-1}\left(\frac{1-\sin\omega_o}{-\cos\omega_o}\right)$	$\theta_1 + 180°$	$\frac{1}{4}\sqrt{\frac{\mu}{a_o}}\left[\frac{\Delta a}{a_o} + (e_1^2 + e_o^2 - 2e_1 e_o \sin\omega_o)^{1/2}\right]$	$-\Delta V_1$