CBERS ORBIT ACQUISITION AND PERTURBATION ANALYSIS

Rajendra Prasad POLAMRAJU#, Hélio Koiti KUGA

Instituto Nacional de Pesquisas Espaciais - INPE
CP 515 - São José dos Campos,São Paulo, Brazil
#Visiting Researcher, e-mail: prprasad@dem.inpe.br

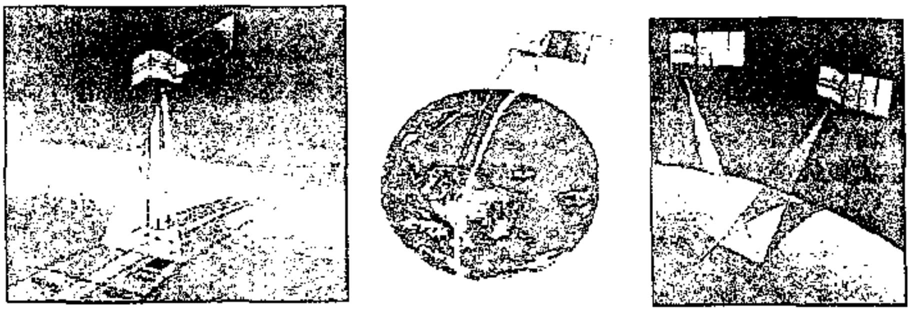
ABSTRACT - *CBERS (China Brazil Earth Resources Satellite) was injected into the near polar, near circular, Sun-synchronous orbit at a mean altitude of 735 km on 14th of October 1999 at 3: 28:32 UTC. The satellite carries a CCD camera, IR Multi Spectral Scanner (MSS) along with Wide Field Imager (WFI) as primary payloads. The location accuracy of the processed imagery is expected to be better than 500 meters. The satellite crosses the equator on a descending orbital node at approximately 10:30 a.m. on each pass. The spacecraft completes just over 14 orbits per day covering the entire Earth every 26 days. To achieve the maximum benefit from the payload data, mission requirements dictate the orbit to be frozen and an exact repeat mission, this in turn requires the orbit to be computed precisely. The immediate post launch maneuver design is to fulfil the primary objective of removing injection dispersions and realize the operational orbit with specified reference ground track pattern with minimum acquisition time. The initial dispersions of the launch vehicle are corrected and the satellite is maneuvered into its station-keeping window after a series of planned maneuvers. The paper briefly describes orbit acquisition methodology, essentially to correct the size, shape and orientation of orbit due to injection errors. In this paper emphasis is more on the execution of in-plane maneuvers to satisfy operational constraints with optimum fuel consumption. The Maneuver strategy for local time maintenance and if desired to correct the node and inclination is also addressed. The mission requirement is to accomplish near circular, near polar and frozen orbit with desired path locking. The paper presents briefly the perturbation analysis carried out along with some sample set of results using both live satellite and simulated data.*

1 - INTRODUCTION

CBERS (China Brazil Earth Resources Satellite) is the premiere operational remote sensing Brazilian satellite. 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. One of the problems that confronts mission designers for the artificial satellites are the selection of the initial conditions for orbits that are approximately periodic. A possible way of choosing initial conditions for periodic orbits for satellites is by computing frozen orbits. "Frozen" means that the eccentricity and perigee are kept fixed values reducing changes in orbit altitude. "Exact Repeat" means that the satellite overflies the same ground track after a fixed period of time [Born 78]. The mission demands precise ground track maintenance within ±10km. The nominal orbit is repetitive, circular and nears polar at a nominal mean altitude of 770-km [Kuga 96]. Initial biasing in the inclination for the injection orbit was planned, so as to reduce the out of plane corrections during the first years of operation. The mission requirements are to maintain the frozen, exact repeat mission. Preliminary maneuver design determines approximate time and magnitude of orbit corrections necessary to realize the targeted orbit and subsequently keeping the ground track under the required limits. The major deviation that reflects the orbit maintenance is the variation of nodal period due to atmospheric drag. The solar activity is in its maximum during the first years of CBERS-1 mission. One of the challenging tasks is to estimate precisely the decay rates in the presence of varying solar activity [Pras 89] at the current values. The paper describes the methods employed for orbit acquisition and perturbation analysis for precise orbit computation along with some test results.

2 - PAYLOAD DESCRIPTION

Infrared Multi Spectral Scanner (IR-MSS) operates in 4 spectral bands and images a 120km swath with a resolution of 80m. In 26 days one obtains a complete Earth coverage that can be correlated with the images of the CCD camera. Wide Field Imager (WFI) has a ground swath of 890km, which provides a synoptic view with resolution of 260m. The CCD camera provides images of a 113km wide strip with 20m spatial resolution. Fig. 1a, 1c depicts the swath and 1-b describes CBERS orbit.



(a) (b) (c)

Fig. 1 : CBERS operational orbit and Swath

The operational orbit was designed to maximize the usefulness of the data from the experimenters. Due to the wide range of payloads the "necessity" compromise ruled the selection of the orbit. The operational orbit of CBERS [Pras 98] is given in Table 1 and depicted in Fig. 1b.

Table 1 – Nominal orbit parameters of CBERS

Parameter		Notation	Value
Mean semi-major axis	(km)	a	7148.86
Mean inclination	(°)	i	98.504
Mean eccentricity	( )	e	0.0011
Mean arg. Of perigee	(°)	ω	90.0
Nodal period	(s)	T_n	6020.4
Number of orbits		No	373

3 - FROZEN ORBIT DESCRIPTION

The accuracy of ground imagery improves if a constant orbital altitude is maintained. Of course perfect altitude control can not be achieved because the Earth's surface has a different ellipticity from the satellite orbit. Also the satellite is constantly dipping and yawing in response to local gravity anomalies [Guin 91] that is underneath. Even with these problems controlling the orbital eccentricity and argument of perigee can reduce the altitude variations [Pras 95]. The Earth's non-spherical gravity field causes a precession of perigee for all Earth satellites. Perigee rate is negative for an inclination of this magnitude and moving against orbital motion. This can cause orbital altitude over given latitude to change from one orbit to next, which is not a desirable feature for precise observations [Tang 87]. Thus a fixed perigee location is desirable to stabilize orbit altitudes. The perigee rate would approach zero when the perturbing forces due to the even and odd harmonics of the Earth's gravity field cancel each other. This occurs when $\omega = \pm 90^{\circ}$ or $\omega = 1 - 5(\sin^2 i)/4 = 0$ viz., $i = 63.4^{\circ}$ (critical inclination). The best choice for Sun- synchronous missions places perigee over North Pole, minimizing altitude variation over the Northern Hemisphere. Canonical equations reveal another surprising property: If eccentricity is of a near small value and perigee is close to 90° the two parameters shall oscillate slowly about a "phase point". e and ω rotate counterclockwise within closed contours, with error magnitude dependent on the initial orbit conditions[Cutn 78]. Atmospheric drag degrades the orbit altitude decreasing the semi-major axis a. This affects eccentricity and perigee, since the 3 parameters are coupled together in Earth's non-spherical gravity field. Thus the drag causes $e \times \omega$ plot to spiral slowly outwards eventually loosing stability [Shap 88]. This is prevented by periodic drag make up maneuvers.

3.1 - Ground track variation:

Accurate prediction of ground track is essential to meet control and maneuver spacing requirements. The actual spacing between successive ground tracks is given as [Bhat 91]

$$S = P_n(\omega_e - \dot{\Omega}) \tag{3.1}$$

$$P_n = \frac{2\pi}{\sqrt{\mu}} a^{3/2} \left[1 - \frac{3}{2} J_2 \left(\frac{R}{a} \right)^2 (4\cos^2 i - 1) \right] \tag{3.2}$$

$$\dot{\Omega} = -\frac{3}{2} J_2 \frac{\sqrt{\mu}}{a^{3/2}} \left[\frac{R}{a(1-e^2)} \right]^2 \cos i \tag{3.3}$$

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. The primary factors which affect the track spacing are atmospheric drag, maneuver errors and uncertainty in knowledge of mean orbital period at the time of the maneuver [Bhat 98]. All these factors affect the nodal period of the orbit and hence the repeat track spacing.

4 – PERTURBATION ANALYSIS

A precise ephemeris generation is a pre-requisite for accurate knowledge of state of the satellite.

4.1 - Geo-potential perturbation

The accuracy with which the orbit is to be computed imposed high demand on the applied orbit computation methodology and the used tracking system. To meet the orbit requirements of the mission the conservative and non-conservative forces on the satellite to be modeled with extreme accuracy. Therefore it is almost mandatory to use the most up dated models in the orbit computations. For the gravity fields recent models have been made available. Recent analysis indicated the higher expectations are put on JGM –2-variance –covariance matrix. This provides better knowledge of the coefficients. The Earth's gravity field is modeled as a spherical harmonic expansion of potential U, given by:

$$U = \frac{\mu}{r} + \frac{\mu}{r} \sum_n \sum_m \left(\frac{R_\oplus}{r} \right)^n P_{nm} \sin \phi \, (C_{nm} \cos m\lambda + S_{nm} \sin m\lambda) \quad (4.1)$$

Where C_{nm} and S_{nm} are the coefficients of harmonics, μ is the Earth's gravitational constant, and P_{nm} is the Legendre polynomial. Geo-potential models have become more and more accurate [Smit 94]. Contemporary *50x50* gravity models like JGM-2 and GEM-T 3 are available that can yield accuracy of cm. The simulation software uses JGM-2. A gravity model comparison was carried out.

4.2 - Luni-solar gravitational perturbation

Third body gravitational attraction is included and the analytical form of planetary ephemeris is used. These perturbations affect the ground track through periodic variations in the orbit inclination. The deviation of inclination from reference value affects the ground trace as the equatorial crossing slowly deviates due to variation in inclination and the nodal period, which is also a function of inclination.

4.3 - Atmospheric drag perturbation

As indicated earlier the atmospheric drag is the main perturbation to be taken into account for ground track control. Atmospheric drag causes continuous decay in the orbit semi-major axis, resulting in an eastward drift in the satellite ground track. The decay is a function of the satellite physical parameters and atmospheric density. Tight ground track control within a small box imposes the estimation of accurate decay rates and, in turns an accurate density model. The atmospheric drag acceleration $\vec{F}$ is given by

$$\begin{aligned} \vec{F} &= -\left(\frac{C_D A}{2m} \right) \rho V_r \vec{V}_r = -B \rho V_r \vec{V}_r \\ B &= \left(\frac{C_D A}{m} \right) \end{aligned} \quad (4.2)$$

Where C_D is the drag coefficient, A is the satellite area, m is the mass of the satellite, ρ is the atmospheric density, and $\vec{V}_r$ is the relative velocity of satellite to the ambient atmosphere. The rate of change of semi-major axis due to atmospheric drag and longitude drift rate is calculated by:

$$\begin{aligned} \left(\frac{da}{dt} \right) &= - \left[\frac{2a^2}{\mu} \right] B \rho V_r \vec{V}_r \vec{V} \\ \dot{\lambda} &= \left[\frac{\tau_e - \tau}{\tau} \right] \left(\omega_e - \dot{\Omega} \right) \end{aligned} \quad (4.3)$$

where $\vec{V}$ is the satellite's velocity, τ is an actual nodal period of the satellite and τ_e is the nodal period of exact repeat mission, ω_e is the Earth's rotational rate and Ω is the longitude of ascending node. The primary actuator upon the period is drag. However there will be a second order variation due to inclination which is obtained by differentiating the nodal period [Bruc 94].

$$\left(\frac{dP_n}{di} \right) = \left[\frac{12 \pi J_2 R^2}{\sqrt{\mu a}} \right] \sin 2i \quad (4.4)$$

Solar flares and geomagnetic storms cause large unpredictable changes in the atmosphere. This results in larger than nominal predicted decay rate, which in turn gives rise to an uncertainty in ground track prediction. The decay rate for a satellite in a circular orbit is essentially given by:

$$\dot{a} = \left[\frac{\rho A C_D \sqrt{\mu a}}{m}\right]\left(1-\frac{\omega_e \cos i}{n}\right)^2 \tag{4.5}$$

where n is the mean motion, ω_e is the Earth's rotation rate, i is the inclination, a is the semi-major axis and other terms are defined. Two contributing factors for atmospheric density variation are, Solar cycle which induces large variation over long period ≈11 years, which is to certain extent predictable and Geomagnetic storms which are of short-term duration. Geomagnetic storms can cause significant disturbance in the density computation. The inaccuracy in the mean semi-major axis can show variable behavior in the ground track drift. Modeling of air density models is the demanding task.

4.4 Air density models

The shape of the temperature profile and the exosphere temperature T_∞ form the core of the current density model for upper atmosphere. The temperature is a function of solar activity and the seasonal, latitudinal variation, semi annual variation. With increasing technical and experimental efforts, in situ Mass Spectrometer measurements led to the new generation of thermosphere empirical models [Pras 94]. The most accurate and efficient model is that of MSIS. The formulation is

$$\rho = 1.6603d-24 \ \sum M_i \ n_i(z) \tag{4.6}$$

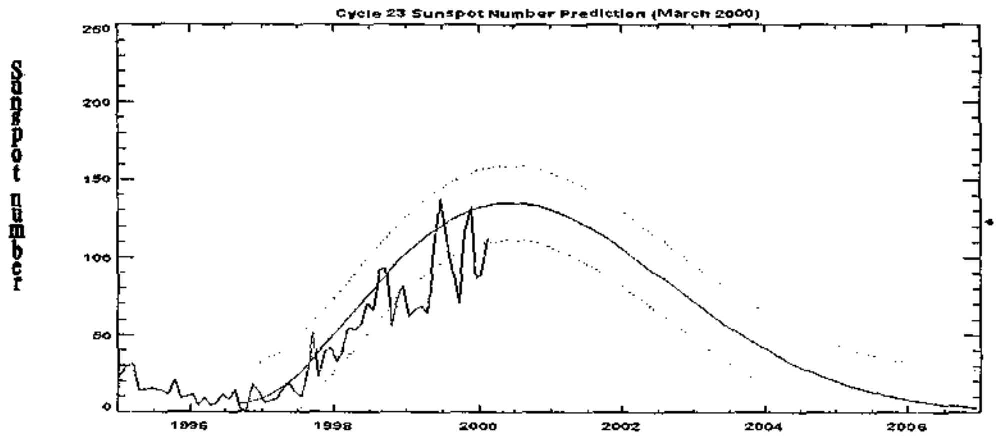
where M_i is the molecular mass and

$$n_i(Z) = A_i \exp\left[G_i(L) - 1\right] * f_i(Z) \tag{4.7}$$

where $A_i \exp(G_I(L) - 1)$ represent individual gas densities computation at 120 km and

$$f_i(Z) = \left(\frac{1-a}{1-ae^{\sigma\varsigma}}\right)^{(1+\alpha_1+\gamma_1)} * e^{(-\sigma\gamma_1\varsigma)} \tag{4.8}$$

MSIS-90 is used for density computation to model atmospheric density since it incorporates both solar and geomagnetic activity. The drag coefficient is derived empirically.



Time period (1996 to 2006)

FIG. 2 : Predicted variation of Sunspot number during 23rd Cycle.

The above figure indicates the maximum solar activity occurs during the initial phase of CBERS.

5 – LAUNCH AND INITIAL ORBIT ADJUST

Table 2- Launch orbit and dispersions

Parameter	Injected orbit Crude Estimate	Nominal Targeted Orbit	Absolute Difference
Semi-major axis a (km)	7113.745	7148.867	35.122
Eccentricity e	0.0011	0.001	0.0001
Inclination i (°)	98.546	98.504	0.042
Nodal period (s)	5981.506	6025.237	43.731

Errors in inclination of ±0.1° have a small effect and no corrections are planned

6 - MANEUVER DESIGN PROCEDURE

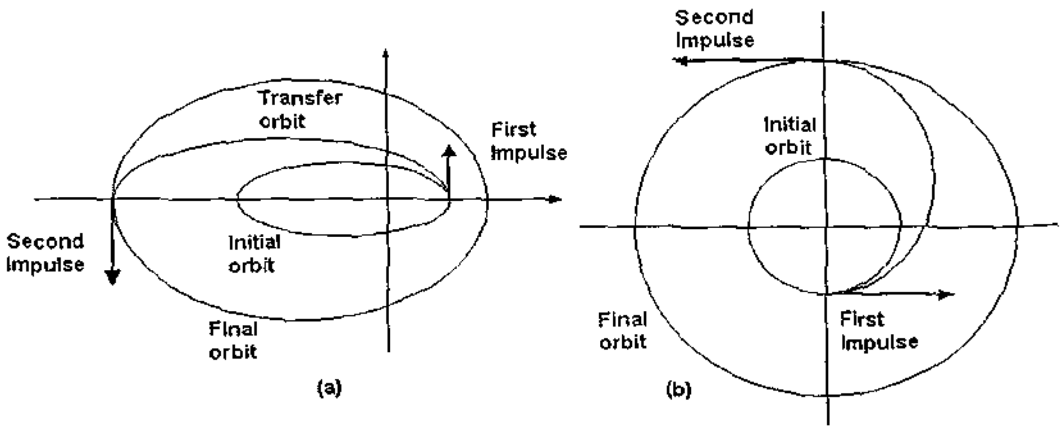
The orbit acquisition is executed according to the following equations. Table 3 shows the equations needed for in-plane orbit corrections [Fran 98].

Table 3 - Mathematical development: Evaluation of ΔV for in-plane orbit maneuvers

Ele Ment	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 $e_o<e_1$	$\theta_1+180°$	$\frac{1}{4}\left[\frac{\mu}{a_o}(e_1^2+e_o^2-2e_1e_o)\right]^{1/2}$	$-\Delta V_1$
ω	$tan^{-1}\left(\frac{1-sin\omega_o}{-cos\,\omega_o}\right)$	$\theta_1+180°$	$\frac{e_o}{4}\left[\frac{2\mu}{a_o}(1-sin\omega_o)\right]^{1/2}$	$-\Delta V_1$
$a+e+\omega$	$tan^{-1}\left(\frac{e_1-sin\omega_o}{-e_o cos\,\omega_o}\right)$	$\theta_1+180°$	$\frac{1}{4}\sqrt{\frac{\mu}{a_o}}\left[\frac{\Delta a}{a_o}+(e_1^2+e_o^2-2e_1e_o sin\omega_o)^{1/2}\right]$	$\frac{1}{4}\sqrt{\frac{\mu}{a_o}}\left[\frac{\Delta a}{a_o}-(e_1^2+e_o^2-2e_1e_o sin\omega_o)^{1/2}\right]$

Here μ is Earth's gravitational constant, $\theta = f+\omega_o$, a_o is initial semi-major axis, f is the true anomaly, a_1 is the targeted semi-major axis, ω_o is the initial argument of perigee, $\Delta a = a_1 - a_o$. The thrust must be made when the satellite is at 0° &180° in XY system or at 90° & 270° relative to the line of nodes [Mich 91].

Fig. 3 : – Transfer between two orbits: a) maneuver to change semi-major axis between elliptical orbits; b) Hohmann transfer between two circular orbits



7 – CBERS ORBIT ACQUISITION

Orbit acquisition of CBERS was completed on 9th of November, 99. The targeted orbit was realized through five in-plane orbit maneuvers. The CCD, IRMSS, WFI payloads were successfully commissioned in 411th revolution. Following figures presents the summary of mean orbital parameters for first few months of the orbit acquisition and maintenance phase.

Fig-4 (a)

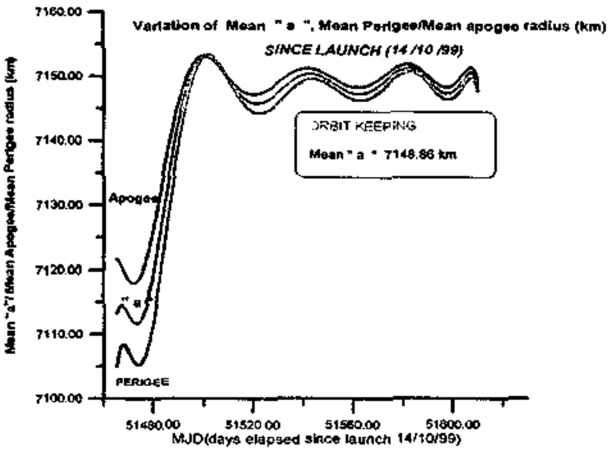


Fig-4 (b)

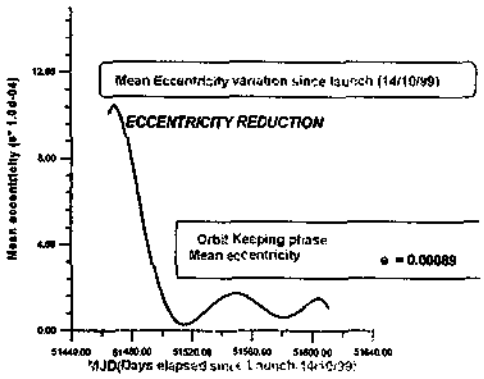


Fig-4 (c)

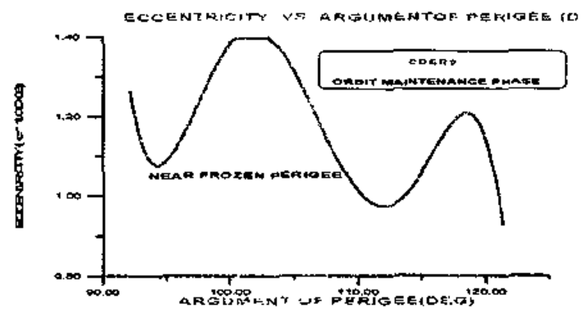
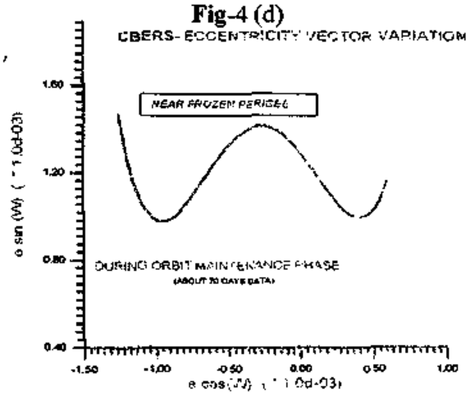


Fig-4 (d)



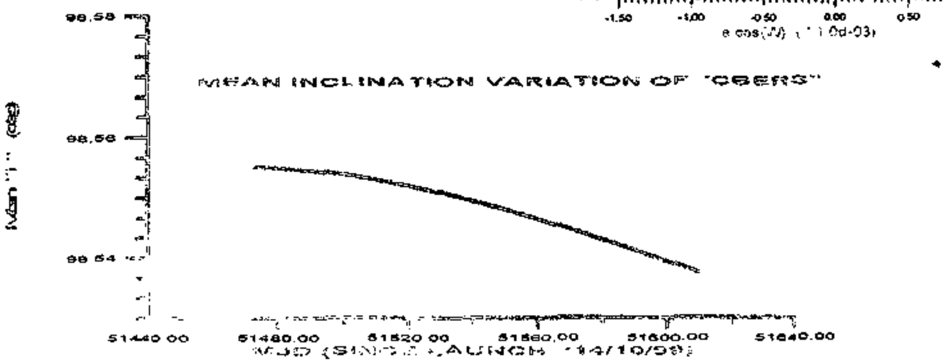


Fig-4 (e)

8 – ORBIT MANEUVER EXECUTION ANALYSIS

In-plane maneuvers: The analyses indicate the errors in each of orbital parameters to be corrected independently or as one of the combinations. In Table 4 the errors of the elements are given, corresponding ΔV and fuel budget is also given. Here the assumption is that the I_{SP}=200 (specific impulse of the fuel) and the maneuver is executed with two impulses. It can be seen from the results that the combined maneuver to execute the correction for all the orbital elements is the optimal solution. One can summarize from the results that the fuel spent for a correction is 69g as against 247g, 215g and 335g for e, ω and $a+e+\omega$ respectively. The ΔV also is significantly less. The ΔV was 0.104m/s for executing a correction as against 0.373m/s, 0.325m/s and 0.506m/s as against the orbit corrections for e, ω and $a+e+\omega$ respectively. Another interesting observation was that the error in ω despite its direction yields the same fuel consumption since ΔV is the same.

Table 4 - Simulation of CBERS for orbit maintenance

Parameter Error	Thrust location θ_1 and θ_2		ΔV_1 and ΔV_2 (m/s)		Firing Duration Δt_1 and Δt_2 (s)		Fuel consumption Δm_1 and Δm_2 (kg)	
$\delta a = 200\text{m}$	90°	270°	0.05225	0.05225	2.56	2.56	0.0346	0.0346
$\delta e = 0.0001$	1.5°	181.5°	0.18667	-0.1866	9.17	9.17	0.1237	0.1237
$\delta\omega = 5^o$	177.5°	357.5°	0.17914	-0.17914	8.80	8.80	0.1187	0.1187
$\delta\omega = -5^o$	177.5°	357.5°	0.17914	-0.17914	8.80	8.80	0.1187	0.1187
$\delta a = 200\text{m} +$ $\delta e = 0.0001$	90°	270°	0.2389	0.1344	11.70	6.60	0.1580	0.8900
$\delta a = 200\text{m} +$ $\delta\omega = 5^o$	177°	357°	0.2313	0.1269	11.3	6.23	0.153	0.084
$\delta a = 200\text{m} +$ $\delta e = 0.0001 +$ $\delta\omega = 5^o$	49°	229.9°	0.3050	0.2008	14.9	9.86	0.202	0.133

8.1 CBERS Orbit acquisition summary

Table-5 provides the summary of the results of orbit acquisition software.

Table-5 Results of orbit acquisition

Impulse time							"a" correction (km)	ΔV(m/s)	Fuel consumed (kg)	Firing Duration (s)
1999	11	3	16	8	18	909	1.9990	1.05159	0.69	51.256
1999	11	5	03	29	42	164	14.536	7.64376	5.05540	372.718
1999	11	6	13	54	37	288	6.846	6.93071	4.59378	338.935
1999	11	8	3	35	29	640	9.826	5.1446	3.4098	251.948

The above corrections include a small residual "e" correction also.

9 – "CBERS" ANALYSIS

The local time at the time of injection is 10:33:49am. The local time gradually increases, as the inclination realized is marginally more than the targeted. In this scenario the local time slowly peaks up and touches around 10:40am after about 10 months and reverses and touches the nominal in about almost 2 years time. This goes down further, that is, the local time further decreases and touches 10:20am in another year and half. So the out-of-plane maneuver need not be executed throughout the operational life of CBERS under normal circumstances. The local time variation is described in the Fig. 5.

9.1 - Local Time Variation

The solar and lunar gravitational attraction causes the secular perturbation to the satellite inclination. The orbit inclination decreases at the rate of 0.041°/year. This secular drift in inclination causes drift in local time at the descending node. The local time (in degrees) is defined as $LT = \alpha_S - \Omega$ where α_S is the right ascension of the Sun and Ω is the right ascension of descending node. As the orbit is Sun-synchronous the local time remains constant. The solar lunar perturbations cause secular perturbations in inclination and local time changes (Prasad, 1995). Fig-5 describes the behavior of Local time over the length of period in the present scenario of CBERS.

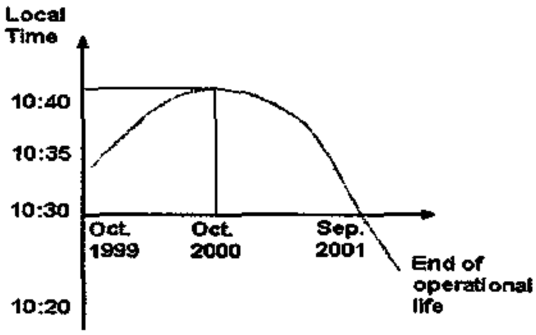


Fig. 5 – Local time variation with time

9.2 - ΔV requirement:

Here the orbit correction is basically needed to correct the semi-major axis, which has generally a large deviation from targeted value of about 40-km lower. Hence it is necessary to execute in-plane maneuver by employing the two impulses which can be distributed over time in split-up maneuvers.

Table 6 – CBERS orbit acquisition case study

Parameter Error	Thrust location θ_1 and θ_2		ΔV_1 and ΔV_2 (m/s)		Firing Duration Δt_1 and Δt_2 (s)		Fuel consumption Δm_1 and Δm_2 (kg)	
$\delta a = 39.59$km	40°	220°	10.340	10.340	507	507	6.85	6.85
$\delta\omega = 50^o$	155°	335°	1.7987	1.7987	88.3	88.3	1.19	1.19
$\delta a = 39.59\text{km} +$ $\delta e = 0.0001 +$ $\delta\omega = 50^o$	157°	337°	12.108	8.57	594.5	420.8	8.025	5.681

One orbit maneuver strategy could be, for instance, to wait till perigee reaches the desired limit in a natural way and execute the maneuver, which can reduce the fuel expenditure in case mission constraints permit.

9.3 - Case study of IRS (Indian Remote Sensing Satellite)

The case study carried out here corresponds to IRS orbit. IRS is an operational remote sensing mission. The targeted orbit is 817km, circular and a frozen orbit. The inclination is 98.591°. The local time at the descending node crossover is 10:30am. An analysis is carried out using live satellite data. The dispersions in the injected orbit were 7.740km, 0.0004 and 29° in semi-major axis, eccentricity and argument of perigee respectively.

Table 7 – IRS orbit acquisition case study

Parameter Error	Thrust location θ_1 and θ_2		ΔV_1 and ΔV_2 (m/s)		Firing Duration Δt_1 and Δt_2 (s)		Fuel consumption Δm_1 and Δm_2 (kg)	
$\delta a = 7.74$km	119°	299°	2.0048	2.0048	99.23	99.23	1.32	1.32
$\delta e = 0.0004$	4.71°	184.71°	0.670	-0.670	33.17	33.17	0.442	0.442
$\delta\omega = -29^o$	14.5°	194.5°	1.3984	-1.3984	69.22	69.22	0.926	0.926
$\delta a = 7.74$km + $\delta e = 0.0004$ + $\delta\omega = -29^o$	346°	166.69°	3.396	0.6136	168	30.3	2.25	0.406

Some of the constraints that may impact the maneuver and are considered in the design of the software are:

- Visibility constraint: The mission requirement is to execute the Orbit Maneuver (OM) during the radio visibility of the station configured for TTC (telemetry tracking and commanding)
- Orbit Determination: Whenever a maneuver is executed it is almost necessary to execute next burn or subsequent maneuver after establishing the orbit or after precise orbit determination (Tapl 1994). Normally for precise orbit determination, the tracking arc needed is around 18 hours of tracking data. Then such time is needed for orbit determination along with the maneuver realization and assessment for re-calibration of the thruster efficiency.
- Orbit information exchange: The mission requirement was to provide the orbit information exchange with other network stations in loop, well in advance for TTC purposes. This calls for OM planning time.
- The mission requirement takes into consideration the "preparation time" needed for keeping some of the heaters to be switched on before OM execute and also the 3 axis pointing stability along with sub-system requirements connected with the firing, or OM execution during the ascending node pass (night pass).

10.0 - CONCLUDING REMARKS

CBERS is the premiere operational remote sensing mission of Brazil. The mission requirements on orbit computation orbit acquisition and maintenance were stringent. Methodology for orbit acquisition and ground track maintenance was evolved. Extensive perturbation analysis was carried out. The software was designed, developed and tested for operational use. The software was validated using both the live satellite data viz. IRS, ERS, TOPEX and that of simulated data. The results are quite encouraging and comparable with that of the operational missions. The software is set up for operational use in the CBERS mission.

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