GENERATION OF AN OPTIMUM TARGET TRAJECTORY FOR THE TERRASAR-X REPEAT OBSERVATION SATELLITE

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ABSTRACT

TerraSAR-X (TS-X) is a new generation, high resolution synthetic aperture radar (SAR) satellite for scientific and commercial applications. The launch of the satellite into a sun-synchronous dusk-dawn orbit with an 11-day repeat period is planned on top of a Russian DNEPR-1 rocket for 2006. The requirement for the TS-X orbit is that the spacecraft shall fly over the same regions following identical flight paths relative to ground. The most important orbit control requirement is to maintain the satellite osculating orbit within a maximum absolute distance of 250 m from a target orbit. For dedicated operations this prescribed maximum distance is lowered to 10 m.

This paper addresses the design and computation of the target path, the so-called reference orbit. The high accuracy orbit control requirements pose tight constraints on the generation of the target trajectory. Due to the high non-linearity of the problem, former strategies based solely on analytical calculus are not accurate enough. Instead the combination of numerical perturbation analysis and sequential optimization is shown to be more effective. It enables automation of the analysis and can be easily applied to all Earth remote sensing satellites.

1. INTRODUCTION

SAR interferometry is the basic principle for the TS-X mission. This technique is based on the stereoscopic effect that is obtained by matching two SAR images obtained from two slightly different orbits, [1]. This off-set creates an interferometric baseline (IB). The TS-X IB is 500 m. For dedicated interferometric operations the baseline is reduced to 20 m, [2]. One way to achieve this is to prescribe a certain trajectory (relative to an Earth-fixed rotating coordinate system) for the spacecraft to follow together with specific limitations on the satellite motion relative to the reference orbit, [3].

In contrast to present day missions (like Envisat at 800 km and ERS at 700 km altitude), that are controlled to a mean reference orbit within typical deadbands of ±1 km, TS-X will be controlled at an altitude of 500 km, facing highly dynamic disturbance forces, inside a tube-shaped boundary defined around the osculating reference orbit with a diameter that equals IB, [3].

Since a conservative mission approach with a state-of-the-art mono-propellant propulsion system is the baseline, an autonomous on-board orbit control is ruled out. Instead a ground-based approach was chosen and will be implemented for TS-X. This is challenging and gives great importance to the design of the target trajectory. In order to reduce the amount of orbit maintenance maneuvers and therefore increase the mission lifetime, an appropriate strategy is developed in the following.

2. ORBIT REQUIREMENTS AND BASIC PROPERTIES

The driving requirements for the TS-X orbit are

- an exact 11 day repeat cycle for the ground-track,
- Sun-synchronicity,
- frozen orbit at about 500 km altitude,
- a mean local time of 18 h at the ascending node

Average ground-track repeatability can be achieved by having an orbital period which is a rational fraction of a day. The value selected for the draconic period, P, (time between two ascending node passages) is 11/167 days resulting in a repetition cycle of 11 days and 167 orbits in the repeat, [2]. If only a two-body potential is considered (J_0, J_1), the period of an elliptical orbit depends only on the size of the semi-major axis, a_{J1}, this provides a preliminary estimation of

$$a_{J_1} = \left(\frac{P}{2\pi} \sqrt{GM_{\oplus}} \right)^{\frac{2}{3}}, \qquad (1)$$

with $GM_{\oplus}$ as the Earth's gravitational coefficient. If we expand the geo-potential to the J_2 zonal term (assuming an Earth's mass distribution that is symmetric with respect to the axis of rotation), and neglect the eccentricity (assuming a circular orbit), then

$$a_{J_2} = a_{J_1} + \frac{1}{J_2 GM_{\oplus}} \left(\frac{4\dot{\Omega} a_{J_1}^3}{3R_{\oplus}} \right)^2 - \frac{J_2 R_{\oplus}^2}{a_{J_1}}, \qquad (2)$$

where $R_{\oplus}$ is the Earth equatorial radius and

$$\dot{\Omega} = \frac{2\pi}{\text{year}} = -\frac{3}{2}\sqrt{GM_{\oplus}} J_2 \frac{R_{\oplus}^2}{a_{J_2}^{3.5}} \cos i_{J_2}\,, \tag{3}$$

is the regression of the right ascension of the ascending node [4], whose size is imposed by the sun-synchronicity requirement. Under our preliminary geo-potential (J_2) assumption the time-derivative of Ω, (3), depends only on the size of semi-major axis, (1)-(3), and on the inclination, i_{J2}, thus

$$i_{J_2} = \arccos\left(-\frac{2}{3}\frac{\dot{\Omega} a_{J_2}^{3.5}}{\sqrt{GM_{\oplus}} J_2 R_{\oplus}^2}\right). \tag{4}$$

Assuming that a given point, S (e.g. separation point), is part of the reference orbit at a given time, T_S (e.g. separation time), the remaining elements can be easily calculated by means of spherical geometry and basic assumptions, [5]. Given the latitude, φ, and the flight direction, *sign* (positive to the North),

$$u = \arctan\left(\frac{\sin\varphi/\sin i}{sign \cdot \sqrt{1-(\sin\varphi/\sin i)^2}}\right), \tag{5}$$

is the argument of latitude for the given point. The flight time from S to the next ascending node (A.N.) can be expressed by a fraction of the orbital period. Based on the Earth rotation, the longitude shift of the A.N. is given by

$$\Delta\lambda_N = P\cdot\left(1-\frac{u}{2\pi}\right)\cdot\left(-\frac{2\pi}{86400}\right). \tag{6}$$

Again by means of spherical geometry, the longitude difference, $\Delta\lambda$, between A.N. and S is determined as

$$\Delta\lambda = \arctan\left(\frac{\cos i \cdot \sin u/\cos\varphi}{\cos u/\cos\varphi}\right). \tag{7}$$

Given the longitude of S, λ (in hours), the node crossing longitude can be calculated by

$$\lambda_N = \lambda - \Delta\lambda + \Delta\lambda_N\,. \tag{8}$$

Finally, for a given date, the right ascension of the A.N. results from

$$\Omega = \mathrm{GMST}(t_{UT1}) + \lambda_N\,, \tag{9}$$

where GMST is the Greenwich mean sidereal time and t_{UT1} is the mean solar time. Table 1 shows a preliminary set of mean orbital elements calculated by equations (1)-(9) for a launch date at 2006/04/06 and a separation point $S = (\varphi,\lambda)$ = (-0.14°, 52.66°).

Table 1. Preliminary mean elements

Mean elements at first A.N.	J_2	J_4 *SGP4 elements*
Semi-major axis [km], a	6883.510	6892.950
Eccentricity, e	0	0.0000001
Inclination [°], i	97.4220	97.4464
Right ascension A.N.[°], Ω	104.2750	104.2749
Argument of perigee [°], ω	-	180.0270
Mean anomaly [°], M	360.00	180.0965

As soon as the zonal terms up to J_4 are included, the analytical formulation is better replaced by a Simplified General Perturbations 4 (SGP4) numerical propagator, [6]. The J_2-elements calculated so far are used as input mean elements to the SGP4 drag-free propagation (the air drag is zero). In particular, the analytical expression for the semi-major axis and inclination corrections, (2)-(3), is replaced by the following numerical formulation. Latitude, φ, and longitude, λ, are assumed to be linear functions of a and i:

$$\varphi = g(a,i) \qquad \lambda = f(a,i). \tag{10}$$

The differentiation of both functions, (10), and the integration over an entire repeat-period from the first A.N. time, T_0, to the end of the cycle, $T_f=T_0$+11days, bring the following linear system:

$$\begin{Bmatrix}\Delta\lambda \\ \Delta\varphi\end{Bmatrix} = \begin{bmatrix}\partial f/\partial a & \partial f/\partial i \\ \partial g/\partial a & \partial g/\partial i\end{bmatrix}\begin{Bmatrix}\Delta a \\ \Delta i\end{Bmatrix}. \tag{11}$$

Where Δ represents the operator difference over a cycle (e.g. $\Delta\lambda=\lambda(T_f)-\lambda(T_0)$). The partial derivatives w.r.t. a and i are calculated by numerical difference using the SGP4 propagator. The formulation of a simplified-Newton iterative algorithm [5], provided that the Jacoby-matrix is not singular, gives the corrections for semi-major axis and inclination that satisfy the Sun-synchronicity constraint. Table 1 shows again the SGP4 derived mean elements at the A.N..

3. REFERENCE ORBIT FINE ADJUSTMENT

The reference orbit is a trade-off. First of all it shall be as realistic as possible, because we want the spacecraft's actual orbit to be as close as possible to the target. On the other hand the reference orbit shall be as simple as possible, because we want a completely periodic orbit (completely periodic relative to an Earth-fixed coordinate system), [1]. The outcome is that the theoretical reference orbit corresponds to an orbit affected only by the gravitational force (120 order, 120 degree) from a real rotating Earth (the Earth motion around its axis is not assumed to be uniform and precession, nutation, polar motion and UCT/UT1 corrections are modeled, [6]).

3.1 GRACE Earth Gravitation Potential

The GRACE Gravity Model 01 (GGM01) was released on July 21, 2003. This model was estimated with 111 days of in-flight data (K-band, attitude and accelerometer data) gathered during the commissioning phase of the Gravity Recovery And Climate Experiment (GRACE) mission, which was launched on March 17, 2002. This model is between 10 to 50 times more accurate than all previous Earth gravity models at the long and medium wavelengths. This improvement has been possible by the measurement of the inter-satellite range-rate which is itself very sensitive to the Earth gravity field. In the resulting gravity model, GGM01S, much more detail is clearly evident in the Earth's geophysical features, [7]. The GGM01S field was estimated to degree and order 120, and is here implemented in order to model accurately the TerraSAR-X reference orbit.

3.2 Sun-synchronous Orbit Refinement

To fulfill the requirements on a precise modeled Sun-synchronous osculating repeat orbit, the semi-major axis and the inclination estimated in the previous chapter by means of two-line-elements have to be carefully adjusted. As already mentioned, the osculating longitude and latitude have to be the same at begin and end of a cycle. In order to calculate the appropriate corrections for a and i, the numerical strategy based on (10)-(11) is applied. The orbit is now propagated numerically by the integration of the equations of motion using GGM01S as gravity model.

Table 2. Osculating elements after Sun-synchronicity and Frozen orbit adjustment with a 60x60 geo-potential

Osculating elements at A.N.	*Sun-synch.*	*Frozen orbit*
Semi-major axis [km], a	6892.94381	6892.94465
Eccentricity, e	0.001180	0.001370
Inclination [°], i	97.440434	97.440124
Right ascension A.N. [°], Ω	104.274891	104.274891
Argument of perigee [°], ω	65.962352	67.975723
Mean anomaly [°], M	294.161136	292.169756

As an example, Table 2 shows the osculating Keplerian elements at the A.N. in case of a 60x60 gravity field model. The reference orbit is mapped into true-of-date coordinates (TOD), thus referred to the true equator and equinox of date.

3.3 Frozen Orbit Adjustment

The non-circularity of the orbit has been almost neglected so far. The gravitational field of the Earth is such that the distance between the Earth center and the spacecraft inevitably varies with several kilometers over one orbit revolution. In order to have that also altitudes are equal for consecutive passes over the same area, a necessary and sufficient condition is that the secular perturbation of e and ω are zero. This is generally known as the "frozen orbit problem". An algorithm to compute a frozen orbit as stable as possible over a long term basis has been derived by [8]. Accordingly to the proposed technique, our preliminary orbit is propagated for an integer number of complete repeat cycles (e.g. 10, triggered by seasonal periods), taking into account all perturbing forces except air drag. Once the osculating eccentricity vector is defined as $e_x=e\cdot\cos\omega$, $e_y=e\cdot\sin\omega$, the mean eccentricity vector is computed over each cycle and follows the 'circle' shown in Figure 1. The 'center' of the 'circle' represents the preliminary frozen eccentricity (-0.0000039, 0.00123). The 'diameter' of the 'circle' can be reduced iteratively, by adding an appropriate increment to the initial osculating eccentricity vector. The process is iterated until no further improvements are obtained. Figure 2 shows the achieved significant reduction of the movement of the eccentricity vector around the new frozen eccentricity center (0.00000022, 0.00125).

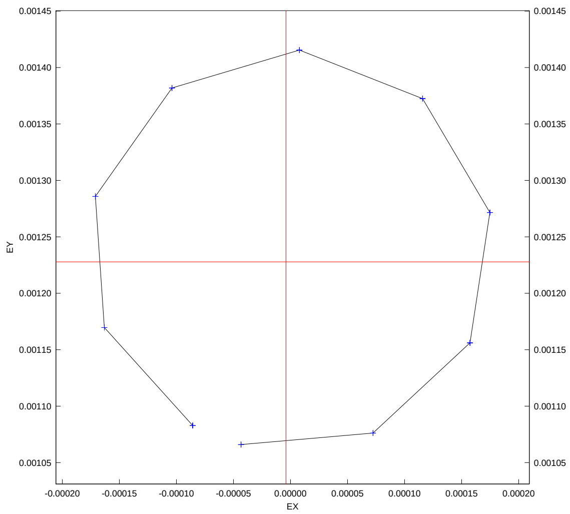

Fig. 1. Eccentricity vector over 10 cycles (first iteration)

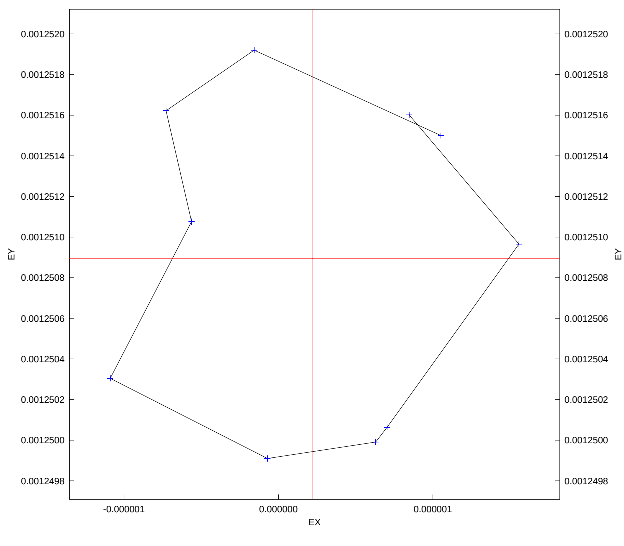

Fig. 2. Eccentricity vector over 10 cycles (last iteration)

4. REFERENCE ORBIT OPTIMIZATION

The reference orbit that has been calculated so far by classical orbit perturbation analysis does not satisfy completely the TS-X requirements. The reference orbit is the target of the ground-in-the-loop orbit control system and should not show at any time discontinuities, or at least large discontinuities with respect to the orbit control deadband (nominal: ± 250 m, occasional: ± 10 m). At this stage of the analysis the reference orbit is characterized by a discontinuity at the end of each 11 days-cycle. In particular the difference between the osculating state vectors at the initial and final point of each cycle in the WGS-84 frame is not negligible. Table 3 shows the so-called repeat-cycle discontinuity for three reference orbits generated by the method described in the previous sections.

Table 3. Repeat-cycle discontinuity for three preliminary reference orbits (A.N.).

WGS-84	*20x20*	*40x40*	*60x60*
ΔX [m]	233.856	65.255	75.890
ΔY [m]	-283.922	-82.740	-95.377
ΔZ [m]	-2802.130	-361.680	-500.838
ΔV_X [m/s]	-2.386	-0.319	-0.437
ΔV_Y [m/s]	-2.050	-0.264	-0.367
ΔV_Z [m/s]	-5.430	-5.757	-5.776

For each reference orbit a different degree and order of the gravity field has been adopted (20, 40 and 60). A further increase of the geo-potential resolution does not show significant changes in the reference orbit, although slows down considerably the computation. This limitation and the large discontinuities shown in table 3 force the implementation of a new strategy able to

- match exactly the state vector in Earth-fixed coordinates at the beginning and end of a cycle (11 days) in order to avoid artificial errors in the orbit control system,
- use the most accurate gravity field (120 order and degree) at an affordable computational cost in order to create a target trajectory as close as possible to reality.

4.1 Optimization problem

The problem we want to solve is highly non linear. A very small change of the reference orbit initial elements causes a large variation of the final elements 11 days later. A suitable approach extensively used in interplanetary trajectories planning is based on numerical optimization techniques. Our trajectory design problem can be generally posed as:

Find the "virtual maneuver" times and sizes to minimize "fuel consumption" (Δv) for an Earth-, sun-synchronous, dusk-dawn orbit trajectory starting at a specific time T_0 with an initial osculating state vector $\mathbf{x}_0$ and ending at time T_0+11 days with the same osculating vector $\mathbf{x}_0$.

The optimization problem as stated has some important features. First, it involves discontinuous controls, since the impulsive virtual maneuvers are represented by jumps in the velocity. Secondly, all intermediate maneuver times shall be included among the optimization parameters. This would require further reformulation of the dynamical model to capture the influence of these parameters on the solution at a given optimization iteration. Finally, the overall problem under consideration has some important long term properties. The TerraSAR-X mission lifetime is 5 years and, as shown in the previous section, the "passive" eccentricity characteristics are given by an appropriate choice of the initial elements e and ω. This means that the initial guess solution for our optimization strategy will be provided by the approach described in the previous sections and that artificial maneuvers must be avoided at the beginning of the cycle (T_0) in order not to degrade the long-term stability.

Problem Formulation

Next, we discuss the formulation of the posed optimization problem. The evolution of the spacecraft is described by a set of six ordinary differential equations

$$\mathbf{x}' = f(t, \mathbf{x}), \tag{12}$$

where $\mathbf{x}=(\mathbf{x}^p;\mathbf{x}^v)\in\mathfrak{R}^6$ contains both positions ($\mathbf{x}^p$) and velocities ($\mathbf{x}^v$). The model of Equation (12) is described in [6], [9]. A more general and probably too ambitious approach would require the model itself to be included among the optimization parameters, since the reference orbit model is a trade-off. In this paper several resolutions of the geo-potential are investigated. The equations of motion are solved simultaneously on each interval between two maneuvers. Let the maneuvers $M_1, M_2, \ldots, M_n$ take place at times T_i, i=1,2,…,n and let $\mathbf{x}_i(t)$, $t\in[T_{i-1},T_i]$ be the solution of Equation (12) on the interval $[T_{i-1},T_i]$, (see Figure 3). A particular case is $\mathbf{x}_{n+1}(t)$, $t\in[T_n,T_f]$ that is the solution of (12) in the interval between the last maneuver (M_n at time T_n) and the end of the repeat cycle T_f. The dimension of the dynamical system is thus $N_x=6n$. Position continuity constraints are imposed at each maneuver, that is,

$$\mathbf{x}_i^p(T_i) = \mathbf{x}_{i+1}^p(T_i), \qquad i = 1,2,\ldots,n-1. \tag{13}$$

The final state vector is forced to match exactly the initial state vector, that is,

$$\begin{aligned} \mathbf{x}_{n+1}^p(T_f) &= \mathbf{x}_1^p(T_0) \\ \mathbf{x}_{n+1}^v(T_f) &= \mathbf{x}_1^v(T_0) \end{aligned}. \tag{14}$$

Additional constraints dictate that the first maneuver does not change the initial elements, and that the order of maneuvers is respected,

$$T_{i-1} < T_i < T_{i+1}, \qquad i = 1,2,...,n-1. \tag{15}$$

With a cost function defined as some measure of the velocity discontinuities

$$\Delta \mathbf{v}_i = \mathbf{x}_{i+1}^v(T_i) - \mathbf{x}_i^v(T_i), \qquad i = 1,2,...,n-1, \tag{16}$$

the optimization problem becomes

$$\min_{T_i, \mathbf{x}_i, \Delta \mathbf{v}_i} C(\Delta \mathbf{v}_i), \tag{17}$$

subject to the constraints in Equations (13)-(16).

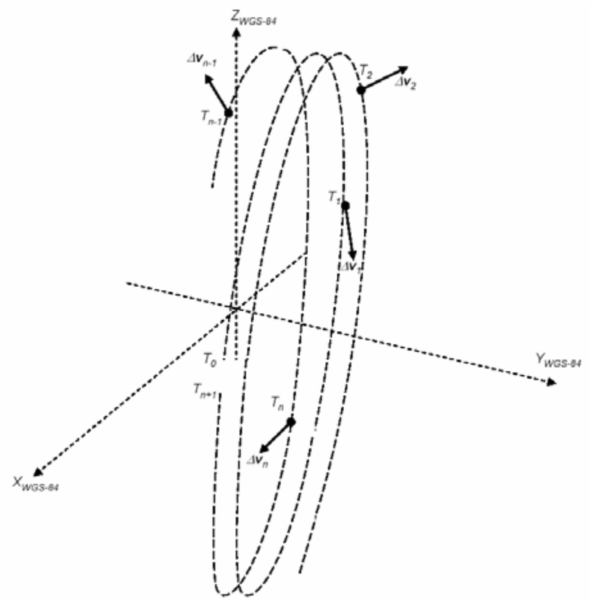

Fig. 3. Target trajectory (167 revolutions). Maneuvers take place at times T_i=1,2,…,n.

An appropriate cost function for the optimization problem has to be chosen. We consider the following two cost functions:

$$C_1(\Delta \mathbf{v}) = \sum_{i=1}^{n} \|\Delta \mathbf{v}_i\|^2 \qquad C_2(\Delta \mathbf{v}) = \sum_{i=1}^{n} \|\Delta \mathbf{v}_i\|. \tag{18}$$

While the second of these is the most meaningful, as it measures the total sum of the introduced discontinuities, such a cost function is non-differentiable whenever one of the maneuvers vanishes. The first cost function C_1, on the other hand, is differentiable everywhere, thus it is more appropriate for the optimizer. Due to the fact that, in some particular cases, decreasing C_1 may actually lead to increases in C_2, the optimization process should be carefully monitored.

TOMP Software

TOMP (Trajectory Optimization by Mathematical Programming) is a software package designed to control and optimize a general class of dynamic systems, developed by the Institut für Dynamik der Flugsysteme, DLR (German Aerospace Center), Oberpfaffenhofen [10]. We make use of TOMP to find the solution of the general constrained nonlinear optimization problem adopting the method of sequential quadratic programming, that has been shown to be very efficient in terms of function evaluations and computation time, [11]. Here we describe the basic algorithm used in TOMP. We consider the general nonlinear programming problem

$$(\text{NLP}): \qquad \min_{\mathbf{x} \in \Re^n} f(\mathbf{x}), \tag{19}$$

subject to

$$\begin{aligned} &g_j(\mathbf{x}) = 0, && j = 1,...,m_e \\ &g_j(\mathbf{x}) > 0, && j = m_e + 1,...,m, \\ &\mathbf{x}_l \le \mathbf{x} \le \mathbf{x}_u \end{aligned} \tag{20}$$

for a local minimum, where the problem functions are assumed to be continuously differentiable and to have no specific structure. Problem NLP is solved iteratively; starting with a given vector of parameters $\mathbf{x}^0$, the $(k+1)^{st}$ iterate $\mathbf{x}^{k+1}$ is obtained from $\mathbf{x}^k$ by the step

$$\mathbf{x}^{k+1} = \mathbf{x}^k + \alpha^k \mathbf{d}^k, \tag{21}$$

where $\mathbf{d}^k$ is the search direction within the k^{th} step and α^k is the step length. The search direction is determined by a quadratic programming sub-problem, which is formulated by a quadratic approximation of the Lagrange function of problem NLP and a linear approximation of the constraints g_j. The general quadratic programming problem is solved by the formulation of an equivalent linear least squares formulation that we omit for simplicity; this problem has been extensively treated by Lawson and Hanson, [12].

4.2 Numerical Results

Initial Guess

The initialization of the optimization process is given by the method described in the previous sections. Sun-synchronous and Frozen orbit adjustments are iteratively implemented until no further improvement is achieved for the reference orbit. At each iterate the order and degree of the geo-potential is increased yielding the results shown in table 3. While the 20x20 gravity field does not provide a good enough initial guess, the 40x40 choice seems to be the most appropriate in terms of computational cost and reference orbit quality. The following initial condition is defined by the guess reference orbit in true-of-date coordinates at the A.N.:

$$x_1^p(T_0)=\{-1698.74795\ 6676.67724\quad 0.00000\}^t\,km$$
$$x_1^v(T_0)=\{\ 0.95716509\ 0.23357008\ 7.54428117\}^t\,km/s$$

Computational Solution

We present complete results for the case in which a 120x120 geo-potential (goal of the analysis) is used as force model for (12). We allow for n=2 maneuvers at times T_1=T_0+11/3 days and T_2=T_1+11/3 days. After 2 iterations the first cost function has a value of C_1=4.4331·10^{-3} m^2/s^2 and the constraint vector is not acceptable:

$$\mathbf{x}_1^p(T_0)-\mathbf{x}_{n+1}^p(T_{n+1})=\{\ 282.8\ \ -361.6\ \ -3251.4\}^t\,m$$
$$\mathbf{x}_1^v(T_0)-\mathbf{x}_{n+1}^v(T_{n+1})=\{-2.7708-2.3812-0.0056\}^t\,m/s$$

After 40 iterations, the optimization was interrupted providing as minimal cost function C_1=1.7160·10^{-3} m^2/s^2 and as constraints:

$$\mathbf{x}_1^p(T_0)-\mathbf{x}_{n+1}^p(T_{n+1})=\{-2.05\quad 2.42\quad 24.29\}^t\cdot 10^{-4}\ m$$
$$\mathbf{x}_1^v(T_0)-\mathbf{x}_{n+1}^v(T_{n+1})=\{2.066\ \ 1.784\ \ 0.001\}^t\cdot 10^{-6}\ m/s$$

The optimal solution vector is expressed in the height or radial (H), tangential or along-track (L) and normal or cross-track (C) co-moving orbital frame for a better physical comprehension

$$\Delta\mathbf{v}_1=\{\quad 0.4463\quad 3.7087\quad 36.6113\}^t\cdot 10^{-3}\ m/s$$
$$\Delta\mathbf{v}_2=\{-0.4477\ -0.1423\ -19.0127\}^t\cdot 10^{-3}\ m/s$$

As the maneuvers are uniformly distributed over the repeat-cycle, the solution shows a peculiar symmetry. Basically, the maneuvers have similar magnitude but opposite versus. In fact the difference between the two consecutive velocity corrections compensates for the end-cycle discontinuity of the guess reference orbit. The locations (long.[°],lat.[°],alt.[km]) of the maneuvers are (148.046,-59.545,532.423)&(-66.630,59.120,517.549).

5. CONCLUSION AND FUTURE WORK

The computation of the reference orbit for a repeat observation satellite has been analyzed, with special emphasis on the upcoming TerraSAR-X mission. The problem is highly nonlinear because of the complexity of the dynamic model (Earth gravity field resolution up to 120x120) and the fact that a large parameter space is investigated (we target an entire surface by requiring the exact match of the state vector after 11 days). The presented method treats the trajectory planning problem as an optimization problem and consists of two main steps. First of all an initial approximate solution is necessary. A good initial guess is found by applying an iterative passive eccentricity control algorithm in order to obtain a frozen orbit as stable as possible over a long-term basis (years). Secondly a non-linear constrained optimization problem is defined and solved by the formulation of sequential linear least-squares sub-problems. A more general and ambitious optimization problem may be formulated, that joins together the autonomous ground-in-the-loop orbit control system and the target trajectory generation. The inclusion of the reference orbit dynamic model among the optimization parameters could lead to an overall minimization of the maneuvers required to satisfy the control requirements.

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