MINISTÉRIO DA CIÊNCIA E TECNOLOGIA
INSTITUTO DE PESQUISAS ESPACIAIS

AUTORIZAÇÃO PARA PUBLICAÇÃO
AUTHORIZATION FOR PUBLICATION

PALAVRAS CHAVES/KEY WORDS
ASSISTED TRACKING
ORBIT DETERMINATION
REAL TIME

AUTORIZADA POR/AUTHORIZED BY
Múcio Roberto Dias
Diretor da ETE

AUTOR RESPONSAVEL / RESPONSIBLE AUTHOR
Hélio Koiti Kuga

DISTRIBUIÇÃO/DISTRIBUTION
- [ ] INTERNA / INTERNAL
- [x] EXTERNA / EXTERNAL
- [ ] RESTRITA / RESTRICTED

REVISADA POR / REVISED BY

CDU/UDC: 521.3

DATA / DATE: Outubro/1990

PUBLICAÇÃO Nº / PUBLICATION NO: INPE-5157-PRE/1636

ORIGEM / ORIGIN: DEM

PROJETO / PROJECT: A76221

Nº DE PAG. / NO OF PAGES: 13

ULTIMA PAG. / LAST PAGE: 12

VERSÃO / VERSION:

Nº DE MAPAS / NO OF MAPS:

TÍTULO/TITLE

REAL TIME ASSISTED TRACKING FOR LOW EARTH ORBIT SATELLITES

AUTORES/AUTHORSHIP

Hélio Koiti Kuga
Valdemir Carrara

RESUMO-NOTAS/ABSTRACT-NOTES

The problem of assisting the tracking antenna during a satellite pass over the tracking ground station is addressed in this work. The assisted tracking provides the antenna pointing angles so as to allow AOS (acquisition of signal), and to avoid or minimize loss of communication with a reduced computation time. It estimates two orbit elements: the mean anomaly and the right ascension of the ascending node. The other keplerian elements are considered constant. Whenever the communication with the satellite is established (locked), the assisted tracking process transforms the measured input tracking angles in an estimate of the mean anomaly and right ascension. The process was simulated in a digital computer for different tracking situations. It was supposed that in a given instant, the signal was lost. The error between the predicted values of the pointing angles and the real ones was then monitored. The error remained bellow 1 degree for any situation, assuring that the communication could be promptly reestablished whenever the signal is recovered.

OBSERVAÇÕES/REMARKS

Presented at the 1st Brazilian Symposium of Aerospace Technology, 27-31 August, 1990, ITA, São José dos Campos, Brazil.

REAL TIME ASSISTED TRACKING FOR LOW EARTH ORBIT SATELLITES

Helio Koiti Kuga
Valdemir Carrara

INPE - Instituto de Pesquisas Espaciais
Department of Mechanics - DEM
CP 515 - São José dos Campos - SP
CEP 12201 - Brazil - BR - FAX 0123-218743

ABSTRACT: The problem of assisting the tracking antenna during a satellite pass over the tracking ground station is addressed in this work. The assisted tracking provides the antenna pointing angles so as to allow AOS (acquisition of signal), and to avoid or minimize loss of communication with a reduced computation time. It estimates two orbit elements: the mean anomaly and the right ascension of the ascending node. The other keplerian elements are considered constant. Whenever the communication with the satellite is established (locked), the assisted tracking process transforms the measured input tracking angles in an estimate of the mean anomaly and right ascension. The process was simulated in a digital computer for different tracking situations. It was supposed that in a given instant, the signal was lost. The error between the predicted values of the pointing angles and the real ones was then monitored. The error remained bellow 1 degree for any situation, assuring that the communication could be promptly reestablished whenever the signal is recovered.

1 - Introduction

The problem of assisting the tracking antenna during a satellite pass over the tracking ground station is addressed in this work. Assisted tracking does not deal with a full real-time orbit determination, but assumes some of the orbital elements previously known and identifies the others (Kuga and Carrara, 1988; Aguirre and Medeiros, 1988). It aims to provide the antenna pointing angles to allow AOS (acquisition of signal), and to avoid or minimize loss of communication with a reduced computation time. The main goal is to estimate timewise the mean anomaly and right ascension of the ascending node, by using the measured antenna pointing angles (azimuth and elevation). A typical application of the assisted tracking is during a satellite zenith pass. Depending on the angle between the spacecraft and the antenna polar axis, the azimuth tracking velocity becomes larger than the tracking gimbal mechanism upper limit. The result is normally a Loss Of Signal (LOS) from the satellite, frequently unrecoverable over the rest of the pass. The LOS may occur due to several factors. A zenith pass would provoke a dead lock of the gimble mechanism of antenna, if not predicted, and the corresponding delay, where the azimuthal gimble axis should rotate 180° within a reduced time, would result in LOS. Some procedures to avoid the zenith pass include a hardware tilt mechanism in the antenna polar axis. However, some antennas are not provided by such mechanisms and, besides that, there are other sources of communication loss not solved by tilt strategies, as an unfavourable spacecraft attitude relative to the ground antenna. It is also usual to employ a table containing predicted azimuth and elevation angles previously stored in the ground station computer before the satellite pass. In case of signal loss, the antenna controler is switched to follow the table data. The inconvenience of this procedure lies in the fact that there are errors in the computed values due to both orbit estimation and propagation and, consequently, the recovery of the signal is not guaranteed. Momentaneous interferences, such as sun pointing, may as well mislead the antenna and the recovery may be not instantaneous. Besides, some satellites may carry on-board antennas which have regions of blind field of views, such that, depending on the geometry between the satellite attitude and the ground antenna, some silent zones may happen.

Simulations and comparisons were conducted to typical passes of a satellite, and under conditions extremely more severe than those encountered in the real situation. The process was simulated in a digital computer, for different tracking situations. It was supposed that in a given instant the signal was lost. The error between the predicted values of the pointing angles and the real ones was then monitored. The error remained bellow 1 degree for any situation, assuring that the communication could be promptly reestablished whenever the signal were recovered. Finally, some final comments are drawn, to furnish useful information about the algorithm performance.

2 - Assisted tracking philosophy

The assisted tracking process consists of two phases: i) partial orbit determination; ii) prediction of the pointing angles. In the first, only two orbit elements are estimated from the antenna angular measurements. In the second, from the orbit estimated in the first phase, prediction of the antenna pointing angles is generated. In order to accomplish the real-time constraints, the algorithms should be fast and robust to yield reliable estimates with minimum computer burden (memory and CPU).

The major component of the pointing errors is along the direction of the satellite velocity (along track). The assisted tracking proposes then to estimate the mean anomaly M (correcting the along track error) and the right ascension of the ascending node Ω. The reasons for the choice of Ω and M as the elements to be estimated are natural when the geometry of the orbital errors with respect to the station is analysed. The ascending node indicates approximately the azimuthal position of antenna and the mean anomaly locates the satellite along the orbit path. Errors in Ω affect the azimuth of antenna in the horizon acquisition, and errors in M affects the satellite location within the orbital trajectory. This last error corresponds to the time interval in which the satellite is delayed or is advanced with relation to the predicted time. Besides, errors in the perigee argument ω are compensated for an increase in the true anomaly f because, roughly speaking, one estimates firstly the sum ω+f and after M. This modelling is valid for elliptical orbits as well, assuming a known eccentricity. This fact is well-known because the major orbital errors are in the along-track component and not in the orbital ellipse characterization. The other keplerian elements (semimajor axis, eccentricity and inclination) are considered constant, and given by the full orbit determination procedures (batch processing in the mission control center).

Azimuth and elevation angles, measured usually by the antenna, are then used to refine the orbit partially when available. When a LOS occurs, the last refined orbit estimate is used to initialize a simplified procedure to predict the orbit and the antenna pointing angles (azimuth and elevation). This allows the antenna to position continuously and adequately in order to reacquire the satellite signal for subsequent instants.

The estimation of all orbit elements implies a statistical estimation process which increases the computational overhead to forbidden levels, once the computer should also simultaneously be executing other not less important tasks. The idea of partial orbit determination (estimation of only some orbital elements) arised from a succeeded experiment in the Cuiaba (brazilian) reception station, for the tracking of Landsat satellites (Aguirre and Medeiros, 1988). There, it is assumed a circular orbit to update the orbit elements (right ascension of ascending node) and f (true anomaly). This procedure for the partial orbit estimation and pointing angles prediction was named "Assisted Tracking."

In this work, it is proposed to estimate the right ascension of ascending node Ω and the mean anomaly M, accounting also for elliptical orbits. The algorithms assume:

- slow variation of the unestimated orbit elements,
- simplified dynamical model, and
- availability of angular data.

For a pass of a low satellite, around 15 minutes (well less than the orbital period of 100 minutes),the unestimated elements (semimajor axis a, eccentricity e, inclination i, and perigee argument ω) vary slowly and may be considered constant. A simplified dynamical model is needed to obey the real-time constraints. The Brouwer model (Brouwer, 1959) is used to compute the variation of elements to be estimated.

One algorithm of deterministic feature is proposed, where a simplified averaging method is applied to the two-body model. The whole assisted tracking block diagram is shown in Figure 1. Whenever the communication with the satellite is established (locked), the orbit determination process transforms the measured input tracking angles in an estimate of the mean anomaly and right ascension. It also updates the value of these elements to the time of the last measured data. A new value is then calculated and the result is stored to be employed in the next iteration or by the orbit propagation. In case of communication failure (loss of signal - LOS), the process control commands the orbit propagation module to generate the tracking pointing angles. These data are sent to the antenna control mechanism, which points the antenna according to these tracking angles.

3 - Partial orbit determination phase

The assisted tracking problem is simpler than the full orbit determination, because only two out of six elements are to be estimated. The deterministic method to estimate the orbit elements with fast variation, the mean anomaly and, in a slower rate, the right ascension of ascending node, consists of a process of arithmetic averaging of the measured values. The measurements of azimuth and elevation of the tracking antenna are transformed, by means of this process, into measurements of the ascending node and the mean anomaly. Afterwards, both mean values are updated. The transformation requires, even though approximately, values of the ascending node and the mean anomaly in a given instant to startup the process.

The first step consists of obtaining a first guess for the magnitude of the position vector between the satellite and the tracking station (range). Once the direction of this vector is known timely through the azimuth (A) and elevation (h), one can use the relation:

$$\underline{r} = \underline{\rho} + \underline{R} \tag{1}$$

to obtain the magnitude of the range vector $\underline{\rho}$. In this relation, $\underline{r}$ is the position vector of the satellite and $\underline{R}$ is the position vector of the station, given in the geocentric system (Escobal, 1965). Although one has approximately the vector $\underline{r}$, the method uses only the magnitude of the vector as a first approximation. On the other hand, the range vector, in the geocentric system, is obtained through a rotation from the topocentric system:

$$\begin{aligned}\underline{\rho} &= \rho\, R(\psi,\lambda)\, (-\cos A \cos h\, \underline{i} + \sin A \cos h\, \underline{j} + \sin h\, \underline{k}) \\ &= \rho\, (c_x \underline{i} + c_y \underline{j} + c_z \underline{k})\end{aligned} \tag{2}$$

where c_x, c_y and c_z are the direction cossines of the direction station-satellite in the geocentric system, and $\underline{i}$, $\underline{j}$, and $\underline{k}$ are unit vectors in the same system. The station coordinates are defined by means of the longitude λ, geodetic latitude ψ, and height H. The rotation matrix which transforms the topocentric into geocentric coordinates is given by:

$$R(\psi,\lambda) = \begin{bmatrix} \sin\psi \cos\lambda & -\sin\lambda & \cos\psi \cos\lambda \\ \sin\psi \sin\lambda & \cos\lambda & \cos\psi \sin\lambda \\ -\cos\psi & 0 & \sin\psi \end{bmatrix} \tag{3}$$

Solving for ρ one gets:

$$\rho = (r^2 - R^2 + B^2)^{1/2} - B \tag{4}$$

$$B = x_e\, c_x + y_e\, c_y + z_e\, c_z \tag{5}$$

where x_e, y_e and z_e are the components of the station position in the geocentric system.

The next step is to retrieve the position vector of the satellite:

$$\underline{r} = (x_e + \rho\, c_x)\, \underline{i} + (y_e + \rho\, c_y)\, \underline{j} + (z_e + \rho\, c_z)\, \underline{k} \tag{6}$$

The geocentric latitude ψ_s and longitude λ_s of the satellite may now be obtained by:

$$\sin\psi_s = (z_e + \rho\, c_z) / r \tag{7}$$

$$\tan \lambda_s = (y_e + \rho\, c_y) \;/\; (x_e - \rho\, c_x) \tag{8}$$

The right ascension of ascending node Ω is related to the longitude of the satellite through:

$$\Omega = \theta_g + \lambda_s - \lambda_o \tag{9}$$

if the satellite is in the ascending part of orbit and

$$\Omega = \theta_g + \lambda_s - (\pi - \lambda_o) \tag{10}$$

otherwise. In this, θ_g is the Greenwich sidereal time, and λ_o is computed by:

$$\sin \lambda_o = \tan \psi_s \cot\text{an}\ i \tag{11}$$

where i is the orbit inclination.

The mean anomaly M is obtained from the true anomaly f:

$$f = \tan^{-1}[\tan \psi_s \;/\; (\cos \lambda_o \ \sin i)] - \omega \tag{12}$$

with ω being the perigee argument. In this equation, λ_o should be replaced by its complement $(\pi - \lambda_o)$ if the satellite is descending. The true anomaly is used to obtain the eccentric anomaly E through:

$$\sin E = r \sin f \;/\; [\ a\ (1-e^2)^{1/2}\ ] \tag{13}$$

$$\cos E = (\ r \cos f + a\, e) \;/\; a \tag{14}$$

where a is the semimajor axis and e is the eccentricity. Finally, the Kepler equation furnishes:

$$M = E - e \sin E \tag{15}$$

which yields the mean anomaly M in the desired time.

The new values of Ω and M will produce the magnitude r of the satellite geocentric position, slightly different from the first assumed value. The process can notwithstanding be iterated until a suitable accuracy is accomplished. Some tests indicated that a second iteration already gives an accuracy better than 0.1% in the value of r, computed by:

$$r = a\ (1 - e \cos E) \tag{16}$$

Although the values of Ω and M are mathematically exact, they have imbedded the errors in the azimuth and elevation measurements. A method of averaging or statistics shall therefore be used to refine the knowledge of Ω and M, and to filter out an eventual locally bad measurement. It was adopted, as an averaging method, a simple arithmetic mean, based upon the following thoughts:

- more complex methods demand, almost always, a greater processing time, which one is trying to reduce;

- during the time span of a pass over the station, one can consider that Ω and the mean mean motion of the satellite do not modify, mainly because the estimated elements are mean and not osculating.

Consider then Ω_{ti} as the ascending node computed through the transformation of the angular measurements, azimuth A and elevation h, associated to the time t_i. Consider as well that Ω_i represents the mean estimate of Ω at time t_i. By definition of the arithmetic mean, one has

$$\Omega_i = [\ \Omega_{ti} + (i-1)\ \Omega_{i-1}\ ]\ /\ i \tag{17}$$

which gives the new mean as a function of the old mean Ω_{i-1}.

In the case of the mean anomaly M, its fast variation with time does not allow that an averaging be computed before translating the observations to the same instant. Let then M_{ti} be the mean anomaly at time t_i, obtained through the measurements A and h. Consider yet the mean anomaly estimated in the previous time t_{i-1}, M_{i-1}. Within this time interval, the mean anomaly must be propagated analytically:

$$\bar{M}_i = M_{i-1} + \dot{M}\ (t_i - t_{i-1}) \tag{18}$$

where $\dot{M}$ is the time variation rate of the mean anomaly. The new estimate is given by:

$$M_i = [\ M_{ti} + (i-1)\ \bar{M}_i\ ]\ /\ i \tag{19}$$

The propagation, nevertheless, introduces another source of error through the time interval between measurements $(t_i - t_{i-1})$, and through the value of M, which depends on the accuracy which the semimajor axis is known. In other words, the mean semimajor axis is assumed to be nearly equal to the osculating one.

Finally, the time variation rate $\dot{M}$ up to the J_2 term is computed by (Brouwer, 1959):

$$\dot{M} = n\ \{1 + 1.5\ J_2\ (R_e/a)^2\ [1-1.5\ \sin^2 i]/(1-e^2)^{3/2}\} \tag{20}$$

where J_2 is the second zonal harmonic coefficient, a,e,i are the semimajor axis, eccentricity and inclination respectively, R_e is the earth equatorial radius, and n the mean orbital motion given by:

$$n = (u/a^3)^{1/2} \tag{21}$$

with u being the geogravitational constant ($3.986\ 10^{14}\ m^3/s^2$).

4 - Prediction phase

In this phase, the last updated estimate of orbit elements is used to generate the antenna pointing angles. From the known elements a,e,i, the rate of variation in the other elements, Ω, ω, M is computed (Brouwer, 1959) up to the J_4 term. The knowledge of the rates allows the prediction of the orbit along time, at a sampling rate pre-established (around 1 Hz). At these times, a transformation of coordinates from the geocentric to the topocentric system (Escobal, 1965) generates the predicted pointing angles of azimuth and elevation. Basically, Equations 1 and 2 are used in this. To provide a smooth movement of the antenna, the predicted angles are linearly interpolated to produce intermediate angles time spaced by 0.1s.

5 - Simulations

The reference orbit was simulated based upon the following conditions:

epoch: 1989/01/31 09:58:00 UTC	
semimajor axis	$a = 7180$ km
eccentricity	$e = 0.0436$
inclination	$i = 23.82°$
ascending node	$\Omega = 240.14°$
perigee argument	$\omega = 15.47°$
mean anomaly	$M = 350.56°$

The perturbations considered were the geopotential up to the 6th zonal harmonic and up to the 4th tesseral/sectoral harmonic, and the atmospheric drag using the Jacchia (1971) model with moderate solar activity. Figure 2 shows a sketch of the geometry of several passes over the Cuiaba and Alcantara tracking stations for a minimum elevation of 5°. The passes number 2,6,7,8 and 9 were selected for the tests of the assisted tracking algorithms. Table 1 shows information of the passes.

Table 1
Passes geometry

station	duration of pass (min.)	pass	maximum elevation (°)
	7:30	2 asc.	15.9
	9:47	6 desc.	12.3
Alcantara	15:25	7 desc.	36.8
	15:57	8 desc.	46.8
Cuiaba	16:18	8 desc.	41.2
	16:58	9 desc.	60.4

The errors in the azimuth and elevation measurements were simulated considering random and biased components. For the random errors σ the following model to generate gaussian noise was adopted:

Azimuth:

$$\sigma_A = C_1 + C_2 / \cos h + C_3 \rho / \cos h \tag{22}$$

$$C_1 = 0.05°, \ C_2 = 0.01, \ C_3 = 10^{-6}$$

Elevation:

$$\sigma_h = C_1 + C_2 \operatorname{cotan} h + C_3 \rho \sin h \tag{23}$$

$$C_1 = 0.05°, \ C_2 = 0.001, \ C_3 = 0.01$$

where h is the elevation and ρ is the range. The component C_1 basically represents an error of timing of the order of 0.1 s. The 2nd and 3rd parcels may become meaningful when the passes are zenithal or azimuthal (h = 90° or h = 0°). For the bias errors b the following model was adopted:

Azimuth:

$$b_A = C_1 + C_2 \,/\, \cos h + C_3 \,\rho \,/\, \cos h \tag{24}$$

$$C_1 = 0.05°,\ C_2 = 0.01,\ C_3 = 10^{-6}$$

Elevation:

$$b_h = C_1 + C_2 \,/\, \sin h + C_3 \,\mathrm{cotan}\, h \tag{25}$$

$$C_1 = 0.05°,\ C_2 = 0.001,\ C_3 = 0.01$$

One can notice that the biases may be of the order of the random errors. The situation degrades for high and low elevations of the satellite. Thus, the simulation conditions can be considered more severe than those encountered in real life. Table 2 shows the maximum random and bias errors for the simulations performed.

Table 2
Simulated random and bias errors

Station	pass	maximum bias A(°)	maximum bias h(°)	maximum noise A(°)	maximum noise h(°)
Alcantara	2 asc.	0.06	0.18	-0.21	-0.44
	6 desc.	0.06	0.18	-0.21	-0.48
	7 desc.	0.06	0.17	-0.22	0.39
	8 desc.	0.07	0.18	-0.22	-0.37
Cuiaba	8 desc.	0.06	0.17	-0.22	-0.43
	9 desc.	0.07	0.17	-0.22	-0.37

In order to simulate the observations, the following coordinates of the tracking stations were assumed as exact:

Alcantara: 44.26°W, 2.18°S, 39m
Cuiaba: 56.07°W, 15.53°S, 277m

For the assisted tracking algorithms, the following coordinates were taken:

Alcantara: 44.2°W, 2.1°S, 30m
Cuiaba: 56° W, 15.5°S, 270m

The errors in coordinates have the effect of adding a new component of bias on purpose. The magnitude of errors are of order of 5km in each direction.

The orbit elements of each pass over the station were purposely corrupted with errors of 3km in position and 3m/s in velocity (5.2km and 5.2m/s RMS). The results of the simulation were condensed in Table 3. The column determination contains information about the phase of processing of angular measurements and refinement of orbit elements, in terms of amount of measurements and CPU time. The sampling rate used corresponds to measurements of azimuth and elevation at each second (1 Hz). The subcolumn Δt shows the time span in minutes from the acquisition (elevation $h=5°$). So, Δt=01:00 min. means that 60 pairs of angular measurements were processed in the determination phase. The subcolumn CPU shows the CPU time in minutes expended by the VAX 780 computer to process these measurements and refine the orbit.

The column prediction contains information about the phase of generation of the antenna pointing angles. The refined estimates in the determination phase are used as initial conditions to propagate the orbit analytically through the method of Brouwer (1959), and to generate at 1 Hz rate the azimuth and elevation pointing angles which the antenna should follow. In this phase, the subcolumn Δt represents the total time interval in minutes in which the prediction of the pointing angles were performed. The subcolumn CPU represents the CPU time in minutes of the VAX 780 computer to yield these predictions. For each pass, simulated were 3 situations of loss of signal, at the begining, in the middle, and at the end.

The column ε_{max}(° RMS) represents the maximum RMS committed in the prediction phase:

$$\varepsilon_{max} = \max \{ \ \varepsilon = (\varepsilon_A^2 + \varepsilon_h^2)^{1/2} \ \} \tag{26}$$

where ε_A and ε_h are the errors in azimuth and elevation when one compares the predicted angles with the simulated angles, without errors in the orbit elements, station coordinates, and measurements.

Table 3
Simulation results

station	pass	determination Δt	determination CPU	prediction Δt	prediction CPU	ε_{max} (°)
Alcantara	2 asc.	01:00	00:01	06:30	00:05	0.90
		03:30	00:03	04:00	00:03	0.65
		06:00	00:06	01:30	00:01	0.52
	6 desc.	01:00	00:01	08:47	00:07	0.32
		03:30	00:03	06:17	00:05	0.25
		06:00	00:06	03:47	00:03	0.23
	7 desc.	05:00	00:05	10:25	00:08	0.40
		10:00	00:10	05:25	00:04	0.28
		15:00	00:15	00:25	00:01	0.17
	8 desc.	05:00	00:05	10:57	00:09	0.50
		10:00	00:10	05:57	00:05	0.32
		15:00	00:15	00:57	00:01	0.20
Cuiaba	8 desc.	05:00	00:05	11:18	00:09	0.52
		10:00	00:10	06:18	00:05	0.36
		15:00	00:15	01:18	00:01	0.22
	9 desc.	05:00	00:05	11:58	00:10	0.88
		10:00	00:09	06:58	01:05	0.45
		15:00	00:15	01:58	00:01	0.25

Figure 3 illustrates the behaviour of the RMS error ε along time, during the phases of determination and prediction, for a pass chosen arbitrarily (pass 2 asc. over Alcantara). In the measurements simulation, a numerical orbit propagator was used. In the prediction phase, an analytical propagator for the estimated mean orbit elements was used. The difference between the osculating and mean elements is the basic error which influences the prediction. It is natural therefore that, in some situations, the error is large in the begining of propagation (prediction) and decreases afterwards, once the osculating elements are cyclic in time. Nevertheless, in all the tests carried out, the maximum error ε_{max} (0.9°) seems to be under control, because the antenna has a lock angle of around 1° at 3 Db.

6 - Final comments

Simulations were realized with the aim of testing the algorithm of the assisted tracking, and of furnishing information about the performance of the system. In terms of CPU time, the process is very fast and causes negligible computational overhead, so as to freed the station computer to other tasks. The communication interface between the antenna controller and the computer has also been asserted to cause little problem in general.

References

Aguirre, J. L. B.; Medeiros, V. M. A satellite assisted tracking system with real time accomodation for maximal pointing accuracy. XVI Congress of the International Society for Photogrammetry and Remote Sensing, Kyoto, Japan, 1-10 July 1988. Proceedings, Vol. II, p.1.

Brouwer, D. Solution of the problem of artificial satellite theory without drag. The Astronomical Journal, 64(9), Nov. 1959.

Escobal, P.R. Methods of orbit determination. NY, John Wiley, 1965.

Jacchia, L.G. Revised static models of the thermosphere and exosphere with empirical temperature profile. Cambridge, MA, Smithsonian Astrophysical Observatory, 1971 (SAO Special Report 332).

Kuga, H. K.; Carrara, V. Algoritmos de rastreio assistido para a estaçao terrena. São José dos Campos, INPE, Sep. 1988. (INPE-MECB/SS-A-ETD-0046)

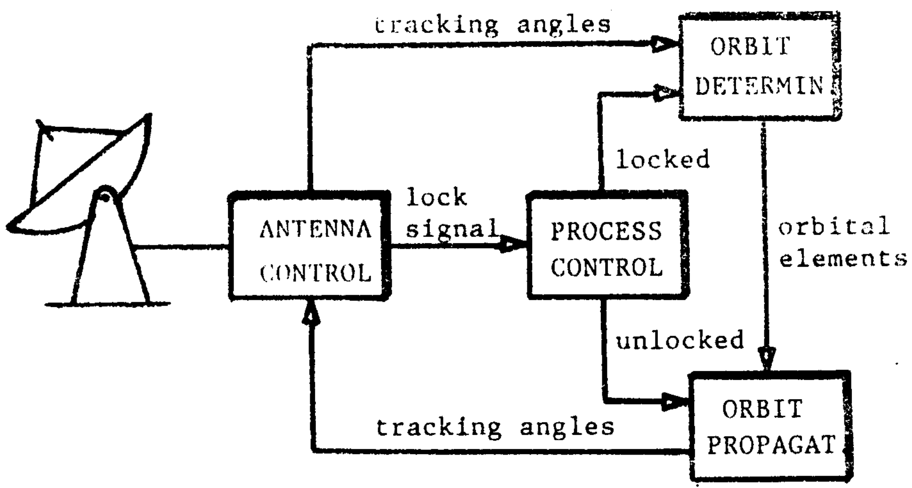

Fig. 1 - Assisted tracking scheme

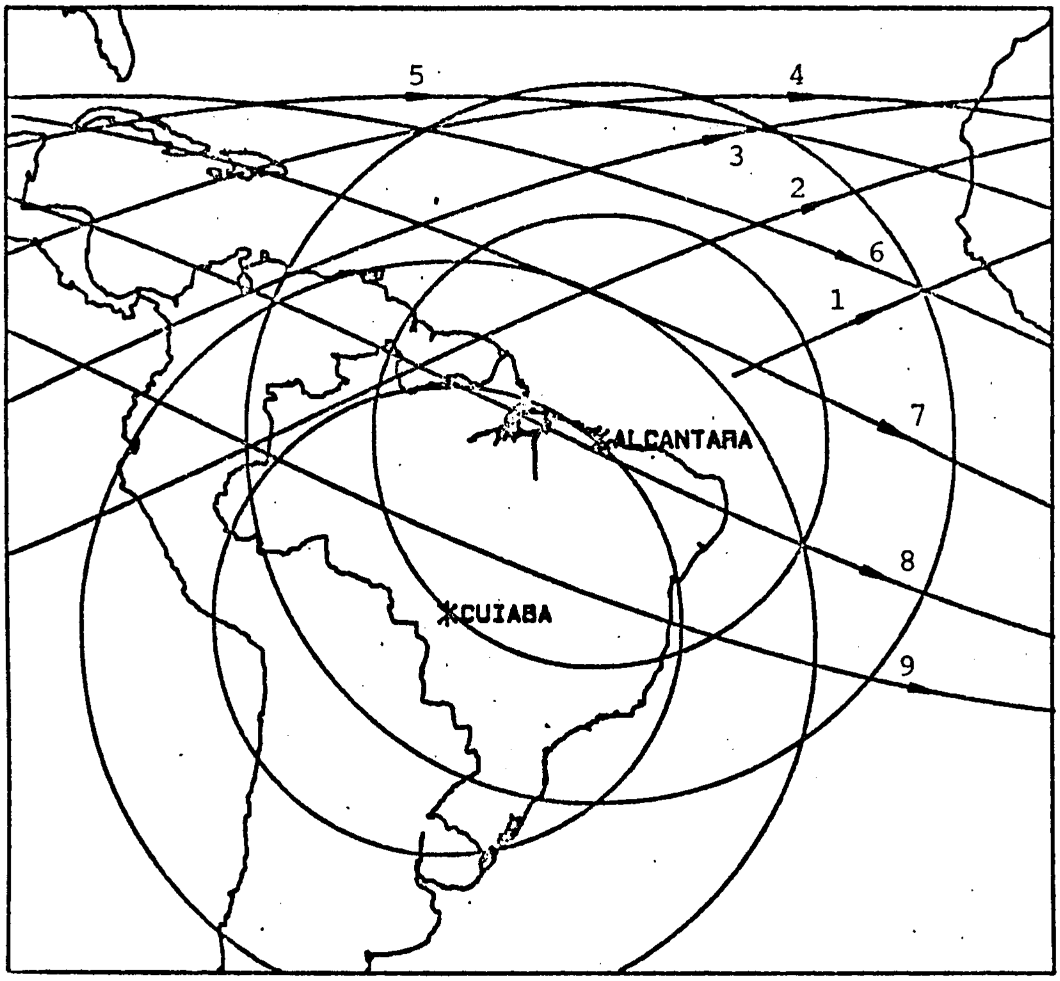

Fig. 2 - Geometry of the orbital passes over the stations

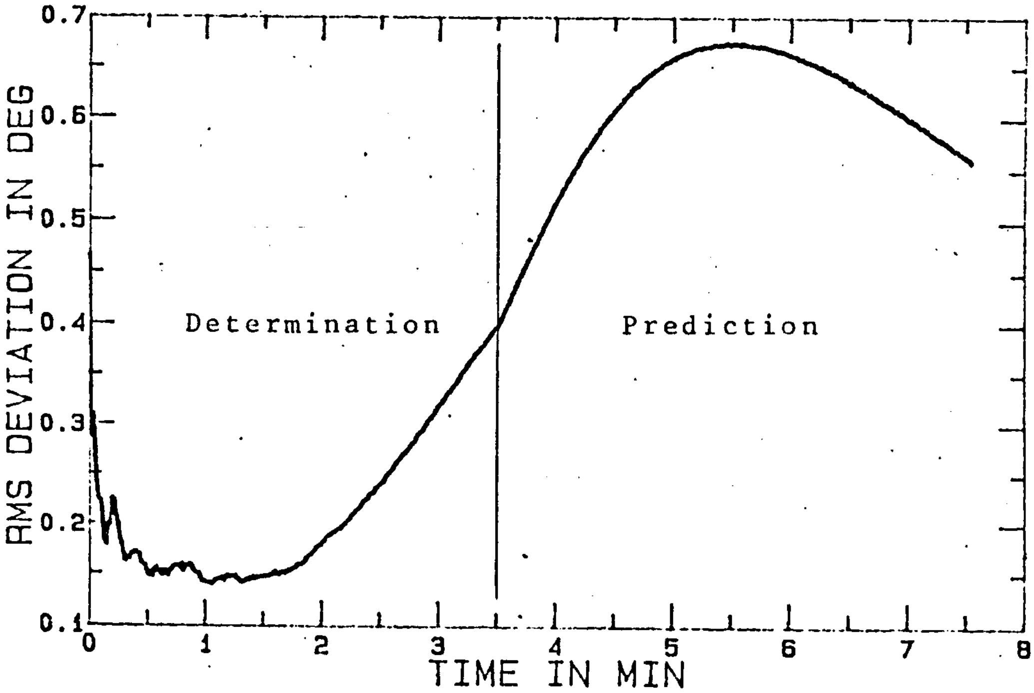

Fig. 3 - Simulation results