COLOCATION STRATEGY AND COLLISION AVOIDANCE FOR THE GEOSTATIONARY SATELLITES AT 19 DEGREES WEST

M. C. Eckstein, C. K. Rajasingh, P. Blumer

DLR GSOC, 8031 Oberpfaffenhofen, Germany

Abstract

When the World Administrative Radio Conference 1977 decided to place several satellites in the same tolerance window of $\pm 0.1^0$ width around 19^0 West, the collision risk for such colocated satellites was disregarded or estimated to be negligible. Recent studies based on extensive simulations showed, however, a substantial collision hazard unless coordinated strategies are applied for station keeping. The paper presents briefly the results of these investigations and a method how an imminent close encounter, caused by some failure inspite of the precautions taken by the coordinated strategy, can be avoided by special maneuvers without seriously disturbing the routine station keeping maneuver schedule. Finally, the present situation of the satellites TDF-1, OLYMPUS,TVSAT-2 and TDF-2, all of them positioned at 19^0 West, is briefly reported.

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1. Introduction

The World Administrative Radio Conference 1977 (WARC 77) assigned the longitude slot at 19^0 West to several geostationary satellites which would share a tolerance window of $\pm 0.1^0$ in longitude and latitude. At that time the chance of a collision or other interferences between such colocated satellites was estimated to be remote in view of the fact that the window represents a box about 140 km wide, 140 km high and roughly 70 km deep.

However, more recent investigations based on extensive simulations of the orbital motion revealed that the collision hazard between colocated geostationary satellites is *quite substantial* (up to about 1 approach closer than 50 m per year) unless their orbit control is properly coordinated (Ref. 1, 2, 3). The reasons for this unexpected result can be seen in the following considerations:

- Unlike gas molecules in a closed box, the satellites are optimally controlled along orbits which minimize the overall fuel consumption as well as the operational effort on the ground.
- During coast arcs, the colocated satellites are influenced by almost identical forces because they are, in general, similar in size, weight and shape.
- The satellites stay in the assigned window for long times (7 years or more).

The independent optimization of the orbital motion of the individual satellites with similar cost functions and constraints will obviously lead to similar solutions, i. e., all satellites will be guided along almost identical trajectories. Deviations will be mainly due to individual control errors. Hence, at every moment, the random distribution of satellite positions will be restricted to a volume much smaller than the entire control box represented by the tolerance window, and the collision hazard turns out correspondingly higher.

It was proposed to leave the station keeping strategies unchanged but to perform special *collision avoidance maneuvers* whenever a close approach of any 2 satellites is imminent. Such policy is reasonable as long as close approaches are rare events. It is noted, however, that due to the limited orbit prediction accuracy even a distance of 1 km must be considered very dangerous and would require an avoidance maneuver. As such distances occur very frequently (about every month according to the simulations of Ref. 3) for uncoordinated station keeping, the only solution is to modify the individual station keeping strategies in such a way that the satellite positions remain sufficiently separated for the nominal motion, and additionally provide a collision avoidance procedure for contingency cases.

In the following sections the main features of station keeping strategies normally used for single satellites are outlined. The next step is the explanation of *coordinated* strategies which separate the satellites sharing the same window and the assessment of the collision risk relative to uncoordinated station keeping. The subsequent section describes the task to determine *collision avoidance maneuvers* for the cases where two satellites get on a collision course because of some control failure. Finally, the actual situation of the satellites TDF-1, OLYMPUS, TVSAT-2 and TDF-2 to be located at 19^0 West is reported.

2. Station Keeping Strategy for Single Satellites

If a single satellite is to be kept within a prescribed angular region around its nominal position inspite of the continuously acting perturbations, orbit correction maneuvers have to be performed from time to time. This is done by means of an on-board propulsion system which is, in general, capable of producing thrusts in North/South

It is operationally convenient to perform the correction maneuvers periodically, as a sequence of repeating *correction cycles*, preferably with periods equal to an integer number of weeks so as to plan the maneuvers always on the same weekday.

The maneuver times and the velocity increments are computed on the basis of the latest orbit determination and a set of *target orbital elements* which have to be reached at prescribed times, for instance, at the end of each cycle. The target orbital elements depend on time and are defined for each correction cycle by the station keeping strategy in terms of

- *mean drift rate*
- *mean longitude offset* from the window center
- *eccentricity vector components*

$$e_x = e\cos(\Omega+\omega)$$
$$e_y = e\sin(\Omega+\omega)$$

- *inclination vector components*

$$i_x = i\cos\Omega$$
$$i_y = i\sin\Omega$$

with the objective to minimize the fuel required for the entire mission. In these definitions, the symbols e, i, Ω, ω denote classical orbital elements with the usual meaning.

The triaxiality of the Earth's potential causes an acceleration in longitude. The solar radiation pressure has the effect that the eccentricity vector (e_x, e_y) moves along a natural eccentricity circle, completing one revolution per year. Both perturbations must be compensated by means of properly scheduled E/W maneuvers. The fuel optimal control is to reverse the longitude drift rate in each cycle and simultaneously keep the eccentricity vector on a control circle which normally has to be smaller than the natural eccentricity circle in order not to exceed the tolerance window. This is done by the *sun pointing perigee control strategy* which essentially keeps the eccentricity vector on the control circle and in the direction to the sun. Accordingly the target eccentricity vector is defined such that in the middle of each correction cycle the direction of the eccentricity vector (longitude of perigee) coincides with the right ascension of the sun.

The luni-solar gravitation causes a secular variation of the inclination vector (i_x, i_y) which has to be compensated by N/S corrections. The target inclination vector for each cycle is defined such that the controlled overall motion of the inclination vector during the satellite life time is symmetric to the origin of the inclination vector plane.

N/S maneuvers are generally much larger than E/W maneuvers. Their yearly velocity increment amounts to about 50 m/s in contrast to only 1 m/s to 5 m/s for the E/W control.

The design of the station keeping strategy must include the effects of orbit prediction errors and maneuver execution errors. The cross coupling errors of the large N/S maneuvers on the E/W motion enforce scheduling the E/W maneuvers shortly after each N/S correction in order to minimize their effects in longitude. Nevertheless, the cross coupling errors represent the major error source in station keeping.

A station keeping error analysis for TVSAT-1 (Ref. 4) resulted in the following linear errors (3σ) after a 14-day prediction including all error sources:

- 0.41 km (radial component)
- 18.34 km (tangential component)
- 0.92 km (out-of-plane component)

showing that the along-track errors are by a factor 20 larger than the errors of the other components. This result, which is supported by simulations, is based on orbit determination with range and angular measurements from a single S-band station after 2 days tracking and maneuver errors of 2 % in magnitude. The cross coupling of the N/S maneuvers was assumed to be 0.87 % and 0.36 % into the radial resp. tangential direction (3σ).

The predominance of the along-track errors is striking and plays a crucial part also in the design of coordinated strategies for colocated satellites.

3. Separation of Colocated Satellites by Coordinated Strategies

A separation of the individual satellite motions can be achieved by offsets of the target orbits and synchronisation of the maneuvers. The latter means that the maneuvers of all satellites should be performed on the same day and in the same sequence, for instance, an East maneuver followed by a West maneuver. In this way the relative motion will remain separated if only the target orbits exclude a close approach, because the accelerations acting on the individual satellites during the coast phases between the maneuvers are almost identical. An exception is the acceleration by solar pressure which depends on the individual area/mass ratios which are slightly different. However, such differential effects can be included as a minor contribution to the orbit control error budget which is to be considered anyway.

3.1 Longitude Separation

The most obvious possibility of separation seems to be an offset in the target longitude as indicated in Fig. 1.

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19.1                   19.0                    18.9
|----------------------|-----------------------|
                                    18.95
|---------deadband Sat.1-----------|
          19.05
           |---------deadband Sat.2-----------|
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Fig. 1 : Longitude Separation by 0.05°.

Such separation entails shorter longitude deadbands and, hence, a smaller eccentricity control circle with the consequence of more fuel consumption for each satellite.

It is emphasized here that an increased fuel requirement is a common consequence of any coordinated colocation strategy. It is the price to be paid for the additional constraint of excluding or reducing the collision risk.

However, the main difficulty of longitude separation is the fact that the longitude suffers from the largest control errors. Simulations have shown (Ref. 2) that about 0.05° longitude separation is required to substantially reduce the risk of close approaches, which makes the method impractical for more than 2 colocated satellites in a $\pm 0.1°$ window.

As the radial and tangential error components are much smaller than the along-track errors, a separation in latitude or in radial direction appears to be more promising. Unfortunately, one cannot assign different latitudes or different radial distances to the individual satellites because this would need continuous thrust against the main term of the Earth's gravity with excessively high fuel requirement.

However, it is possible to achieve a *periodic* separation of the radial and/or out-of-plane components by offsets in the eccentricity vectors and/or inclination vectors between any 2 satellites. Both components are then sinusoidal functions of time with the period of one sidereal day.

The eccentricity vector offset additionally provides a periodic longitude separation with twice the amplitude of the radial separation. As a result, an offset $\Delta\vec{e}$ with respect to some reference orbit leads to a relative motion of the satellite along a coplanar ellipse in retrograde direction around the reference position. The minimum distance is $r_{\min} = R_{geo}\,|\Delta\vec{e}\,|$, where R_{geo} is the geostationary radius.

Since offsets can be applied in any direction of the eccentricity vector plane, it is possible to accommodate more satellites than in only one dimension provided by the longitude separation. Fig. 2 illustrates the eccentricity vector offsets of 4 colocated satellites. The reference orbit in this example is the exact geostationary one. The absolute orbits are shown on the left side, the relative motion on the right side. As the offsets are equal in size, all satellites move on the same relative ellipse with different phase. Of course, some margin has to be left within the window limits for the additional longitude control deadband common to the whole group of satellites. Note that the *mean longitude* of all participating satellites is nominally identical to that of the reference orbit.

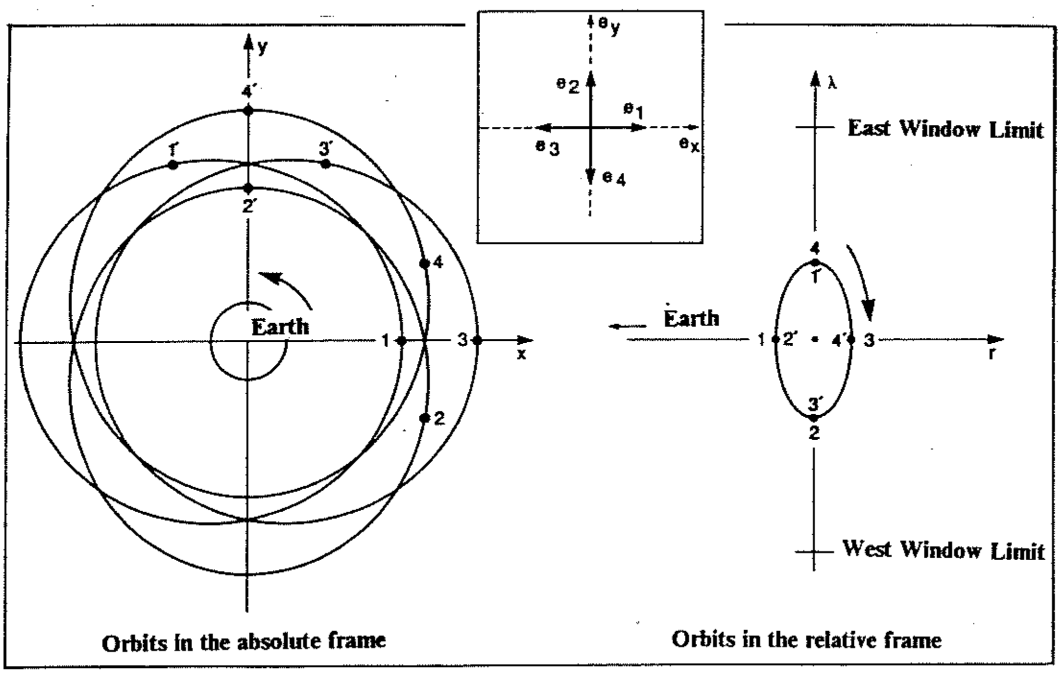

Fig. 2 : Orbits of 4 colocated geostationary satellites separated by eccentricity vector offsets.

An eccentricity vector separation strategy of this type was proposed in Ref. 5. However, the errors in longitude, which may be as large as the nominal separation, shift the individual ellipses at random in longitude. This creates trajectory crossings and may still lead to close encounters whenever the radial geocentric distances coincide, which happens twice a day.

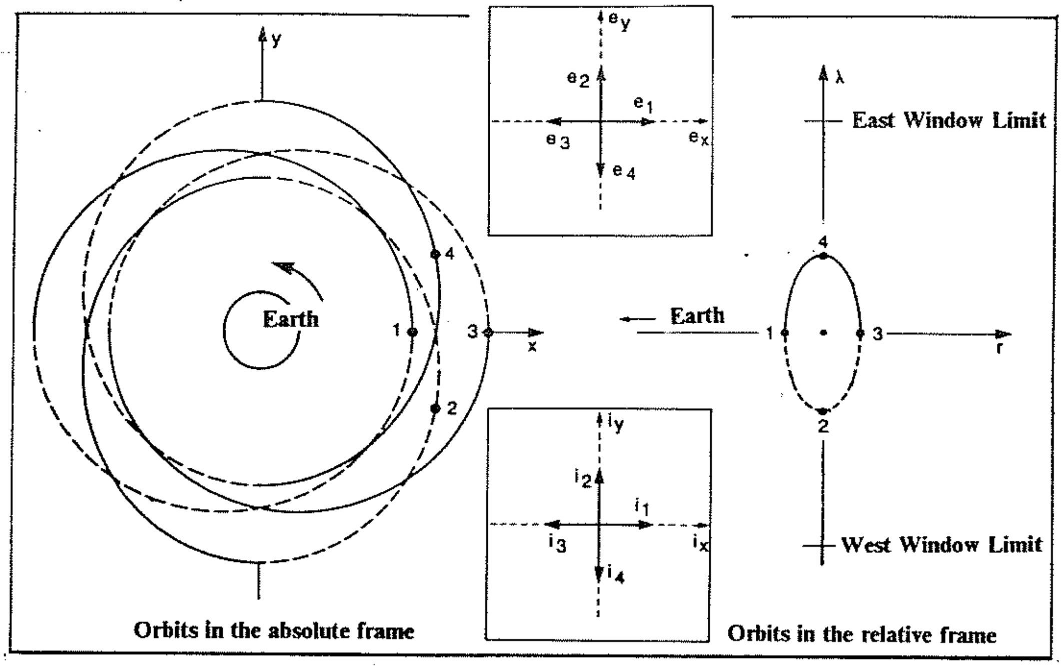

Fig. 3 : Orbits of 4 colocated satellites separated by combined offsets of eccentricity and inclination vectors.

3.3 Combined Eccentricity and Inclination Vector Separation

The collision probability can be reduced even further by a combination of eccentricity vector offsets and inclination vector offsets. Proper phasing of the resulting radial and out-of-plane oscillations leads to an inclined elliptic relative motion with a sufficiently large out-of-plane component whenever the radial component vanishes and vice versa. The optimal choice of vector offsets for 4 satellites is shown in Fig. 3 where the individual inclination vector offsets are parallel to the corresponding eccentricity vector offsets. The out-of-plane component of the relative position is maximum when the radial component is zero and vice versa. The relative motion of each satellite around the reference position (or around each other) is an ellipse with the minor axis along the radial direction and the major axis perpendicular to it, inclined with respect to the equator by an angle

$$\gamma = arctan\left(\frac{|\Delta\vec{i}\,|}{2\,|\Delta\vec{e}\,|}\right)$$

The separation is accomplished by the radial and out-of-plane components alone even if the longitude component vanishes because of some error. The minimum distance in this case is nominally (Ref. 4)

$$r_{\min} = R_{geo}\,\min(|\Delta\vec{e}\,|,\ |\Delta\vec{i}\,|)$$

and the errors in longitude can no longer lead to a close encounter.

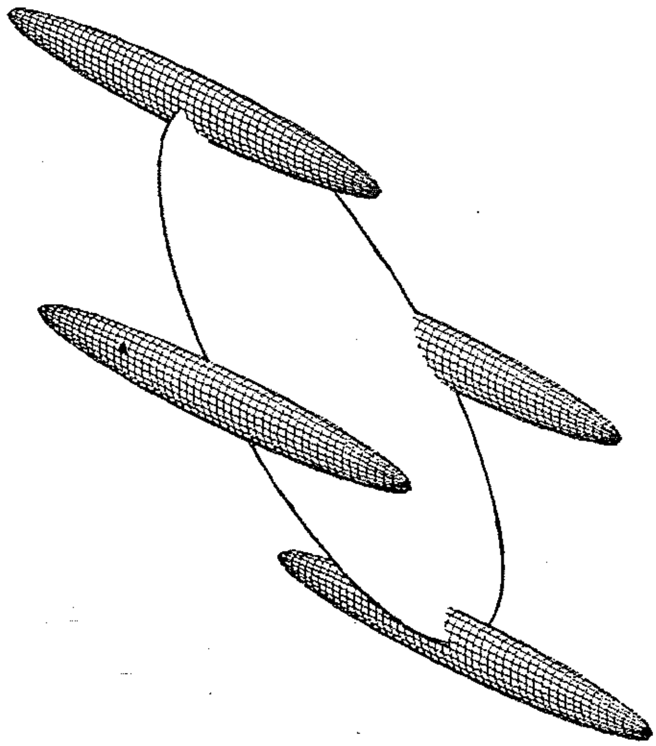

Fig. 4 : Error ellipsoids of 4 colocated satellites moving relatively to a common reference orbit.

A perspective view of the relative motion of 4 satellites around the common reference orbit according to the combined eccentricity / inclination vector separation is presented in Fig. 4. The long stretched ellipsoids indicate the control error distribution with the largest axis in the longitude direction. They never touch each other because the *relative* orbit is inclined with respect to the major axis of the error ellipsoids.

In order to verify such expectations, the motion of 4 satellites, controlled according to the mentioned separation strategy, was simulated on the computer by means of the station keeping simulation program SKSIM (Ref. 6) which also considers errors of orbit determination and thrust execution by a Monte Carlo method. Many simulations comprising 980 years of station keeping for each satellite were carried out and statistically evaluated with respect to the yearly minimum distances. The relevant spacecraft data of TDF-1, OLYMPUS, TVSAT-2 and TDF-2, which are to share the 19^0 West position, have been taken as input.

The simulations have been carried out for 2 cases:

1. Using the individual strategies as planned *without* regard of the other satellites
2. Using the *coordinated* strategy outlined above, with vector offsets $|\Delta\vec{e}| = 0.00015$ and $|\Delta\vec{i}| = 0.015^0$ and a cycle period of 2 weeks for both, E/W and N/S corrections.

ones without violating the common window.

The main results are listed in Table 1. An *event* is defined here as a close encounter below the threshold distance between any pair of the 4 satellites. The numbers in the last 3 columns demonstrate the drastic superiority of the coordinated strategy. It was therefore proposed for the satellites to be positioned at this longitude.

Station Keeping Strategy	Relative Distance Threshold (m)	Probability for Events %	Mean Event Rate per Year	Expected Time between Events (Years)
Uncoordinated	50	0.79	1.55	0.6
	100	0.99	4.74	0.2
	200	1.00		
Coordinated as proposed	50	0.00	0.00	infinite *
	1000	2.10	0.02	46.4 *
	2000	5.40	0.06	18.0 *
	5000	38.00	0.48	2.1

* The number of events found for the coordinated strategy below 5000 m threshold distance is too low for statistical significance.

Table 1. Close Approach Characteristics of the 4 Satellites Controlled by the Uncoordinated and the Proposed Coordinated Station Keeping Strategies.

3.4 Generalization to n Satellites

The separation by parallel offsets of the eccentricity vectors and inclination vectors can be generalized to more than 4 satellites. In the case of 4 satellites the end points of the vector offsets form a square as large as possible without violating the tolerance window. For n satellites up to n=6 the optimal constellation is a regular polygon because it provides the largest offsets for a given maximum eccentricity resp. inclination (Ref. 2). The offset of a seventh member would have to be placed in the center of the hexagon, and more complex constellations will have to be applied for more than 7 members (Ref. 2). Of course, the nominal minimum distances decrease as the number of colocated satellites increases.

4. Collision Avoidance Maneuvers

The aforementioned coordinated colocation strategy provides safe separation as long as the expected error budget is not abnormally exceeded. However, in a contingency case, i. e., a maneuver failure, the separation is no longer guaranteed. The procedure in such cases would be the following:

1. check the non-nominal trajectory with respect to close encounters with any other participating satellite
2. compute special maneuvers which bring the satellite back to its nominal orbit with minimum fuel expenditure

participating satellite

4. in case of a close approach below some given threshold distance r_{thresh}, modify the maneuvers such that the deviation from the target orbit is minimized under the constraint that $r_{\min} > r_{thresh}$

The first task can be solved as soon as the non-nominal orbit has been determined and requires an exchange of orbit data between the concerned control centers in order to evaluate the relative distances.

The second task can be accomplished by the same algorithm which is also applied for maneuver calculation of regular station keeping. At least 2 E/W maneuvers and 1 N/S maneuver are needed to reach the target orbit. There is, in general, a finite number of solutions with maneuver times differing by an integer number of half days.

The third task again requires an exchange of orbit data in order to evaluate the relative distances. If the collision risk avoidance constraint is not violated, the transfer with the least fuel consumption can be selected.

4.1 Case of Close Approach

If the collision risk avoidance constraint is active, the fourth task has to be executed. This requires the solution of an optimization problem which can be formulated in the following way:

- *Cost function* :

$$F = C_0 \sum_{i=1}^{N} |\Delta V_i| + C_1 \sum_{k=1}^{3} \Delta x_{Tk}{}^2 + C_2 \sum_{k=1}^{3} \Delta \dot{x}_{Tk}{}^2$$

where C_0, C_1, C_2 are weighting factors and Δx_{Tk} , $\Delta \dot{x}_{Tk}$ the components of the position and velocity vector deviations from the target state.

- *Inequality constraints* :

$$r_{\min} - r_{thresh} \geq 0$$

$$\begin{aligned} \lambda_{\min} + 0.1^0 &\geq 0 \\ \lambda_{\max} - 0.1^0 &\leq 0 \\ \beta_{\min} + 0.1^0 &\geq 0 \\ \beta_{\max} - 0.1^0 &\leq 0 \end{aligned}$$

where λ, β are the longitude and latitude deviations from the window center. The first inequality constraint avoids the collision risk, the remaining ones the window violation.

The value chosen for the threshold distance depends on the orbit prediction accuracy which can be achieved independently for each participating satellite.

Corresponding to the multiple solutions in the unconstrained transfer problem, there will be a finite number of local minima of the cost function. In order to find all local minima and to select the global one, the available time span for maneuvers is split in intervals of half days. The optimization is then performed for maneuvers in every possible combination of the selected half-day intervals.

The number of half-day intervals available for the first collision avoidance maneuver is, of course, limited to the time *before* the expected encounter. The optimization problem is simplified by formulating the equations of motion in terms of *relative coordinates* between 2 satellites, one of which is moving nominally according to the coordinated strategy. The coordinate system is centered in the nominal satellite and oriented in radial, tangential and normal direction. This allows a linearization of the equations of relative motion and a simplified transformation from the relative state to relative orbital elements.

In order to perform the optimization, a software was developed in Ref. 7 which contains the parameter optimization module SLLSQP (Ref. 8). The latter demands additional subroutines for the evaluation of the cost function, the constraints and the associated derivatives with respect to the variables to be optimized. The details of this software, the code and a numerical example are given in Ref. 7.

4.2 Numerical Example of Collision Avoidance

The aforementioned software is applied whenever the relative distance between the two satellites falls short of the threshold distance r_{tresh}. The Figures 5 to 7 illustrate the results of a numerical example which is based on the following assumptions :

- A two-week correction cycle of station keeping is chosen.
- Every satellite performs two E/W maneuvers at the beginning of the cycle.
- At cycle end a required relative state has to be reached.
- Both satellites are colocated by eccentricity vector separation only and their motion is assumed to be coplanar.
- Satellite A moves along its nominal trajectory, represented by the origin of the relative coordinates, whereas the other satellite B causes a collision hazard because it deviates from its nominal path.
- The threshold distance $r_{tresh} = 1$ km.

Fig. 5 shows the *nominal relative motion* of satellite B with respect to A without any abnormal error. The minimum relative distance amounts to about.8 km.

In Fig 6, a small maneuver execution error of 0.01 m/s occurring during the performance of the E/W maneuvers of satellite B leads to a *collision hazard* after 8.17 days, as the minimum relative distance is only 0.9 km. The error is detected within the first half of the cycle by orbit determination.

The sequence of *collision avoidance maneuvers* consists of one East and one West maneuver within the first three half days after error detection, but before the encounter. The additional velocity increment is about 0.123 m/s and corresponds to usual station keeping fuel expenditure. Satellite B is brought back to its original target elements without approaching the other satellite A closer than 7 km throughout the entire motion and without any window violation as illustrated in Fig. 7.

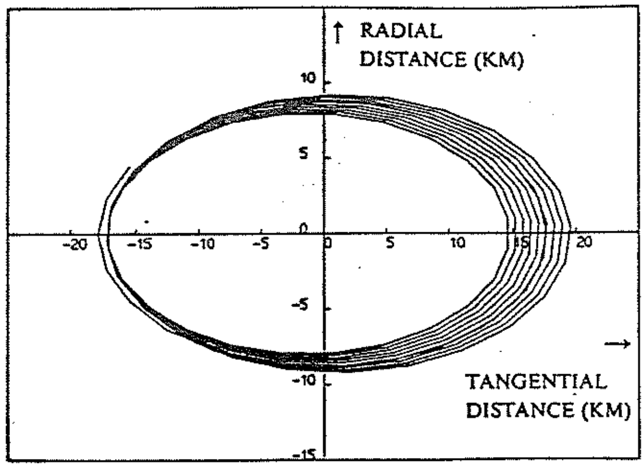

Fig. 5 : Nominal relative motion of satellite B around satellite A.

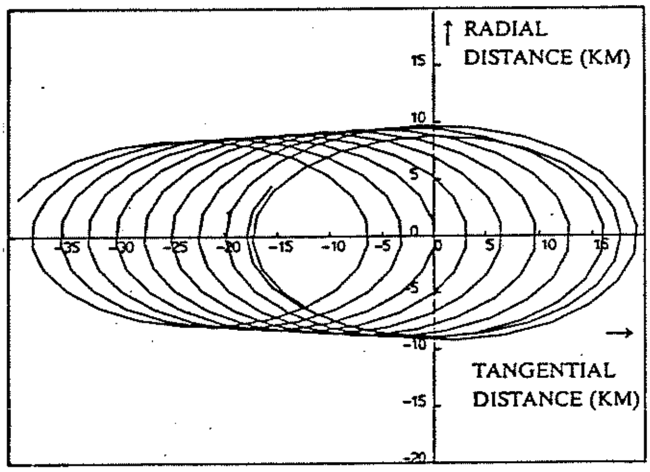

Fig. 6 : Collision hazard caused by a close approach due to a maneuver execution error of satellite B.

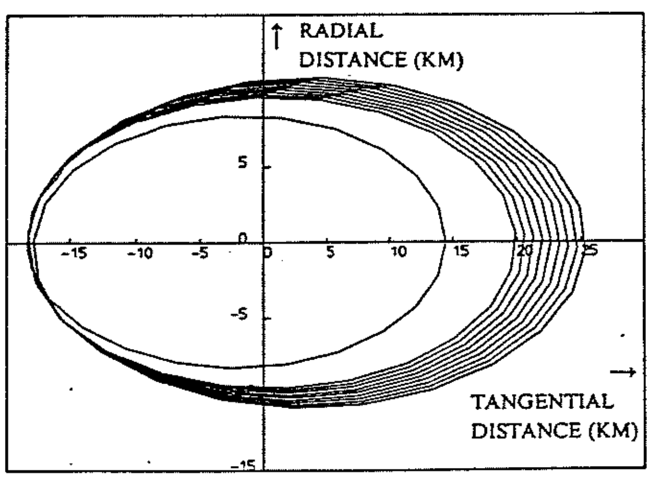

Fig. 7 : Relative motion of satellite B around satellite A with application of a collision avoidance maneuver.

5. The Present Situation at 19 Degrees West

Confronted with the prospect that the colocation of 4 satellites at 19^0 West will be a reality by 1990, the agencies CNES, ESA-ESOC and DLR-GSOC which are in charge of controlling TDF-1, TDF-2, OLYMPUS, TVSAT-2, organized a number of meetings on this subject. They agreed on the following points:

- A coordinated station keeping strategy will have to be applied as soon as the next satellite joins TDF-1 which has been maintaining the position 19^0 West since November 1988.
- The orbit maneuvers should, as far as possible, be synchronized with the maneuver schedule of TDF-1.
- The target orbits should be separated according to the combined eccentricity and inclination vector separation method as outlined above.
- Orbit data are to be exchanged between the three agencies after every orbit determination.
- The parameters and the format of the "Geostationary Longitude 19 deg West Colocation Interface Telex" was defined.

A resolution was drafted informing the respective satellite owners on the consequences of coordinated station keeping.

Furthermore, a tracking campaign was initiated in order to assess the attainable orbit determination / prediction accuracies and to calibrate the different tracking systems.

A major obstacle against realizing the envisaged coordinated station keeping strategy turned out to be the limited number of admissible N/S maneuvers for OLYMPUS during its life time, which prevents it from following the maneuver schedule required for the other participating satellites.

While studies were going on to overcome this difficulty, the German *Bundespost* announced that TVSAT-2 would be positioned at $19.2^0 \pm 0.1^0$ West instead of $19.0^0 \pm 0.1^0$ West in order to avoid any consequences of colocation.

With this decision in mind, and after the launch of OLYMPUS, CNES and ESA-ESOC agreed on a preliminary strategy which separates the orbits of the 2 remaining satellites on an "ad hoc" basis: The inclination vectors are restricted each to one half of the inclination vector plane, and the eccentricity vectors are selected at every E/W correction so as to separate the relative orbit crossing points in radial direction. The maneuvers are not synchronized but shifted one week between TDF-1 and OLYMPUS, so as to involve the orbit correction of only a single satellite in each orbit prediction. The orbit and maneuver planning data are exchanged as planned.

This preliminary colocation strategy retains some features of the proposed combination of eccentricity and inclination vector offsets. It is not fuel optimal but largely works without the need of strict coordination regarding target orbits and maneuver schedules. It is planned to use it until the launch of TDF-2 when the colocation of a third satellite has to be considered.

6. References

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