MISSION ANALYSIS AND ORBIT CONTROL OF INTERFEROMETRIC WHEEL FORMATION FLYING

Jean FOURCADE

Centre National d'Etudes Spatiales (CNES),18 avenue Edouard Belin, F31401 Toulouse Cedex 9, France
E-mail:Jean.Fourcade@cnes.fr

ABSTRACT

Flying satellite in formation requires maintaining the specific relative geometry of the spacecraft with high precision. This requirement raises new problem of orbit control.

This paper presents the results of the mission analysis of a low Earth observation system, the interferometric wheel, patented by CNES. This wheel is made up of three receiving spacecraft, which follow an emitting Earth observation radar satellite.

The first part of this paper presents trades off which were performed to choose orbital elements of the formation flying which fulfils all constraints.

The second part presents orbit positioning strategies including reconfiguration of the wheel to change its size.

The last part describes the station keeping of the formation. Two kinds of constraints are imposed by the interferometric system : a constraint on the distance between the wheel and the radar satellite, and constraints on the distance between the wheel satellites. The first constraint is fulfilled with a classical chemical station keeping strategy. The second one is fulfilled using pure passive actuators. Due to the high stability of the relative eccentricity of the formation, only the relative semi major axis had to be controlled. Differential drag due to differential attitude motion was used to control relative altitude. An autonomous orbit controller was developed and tested. The final accuracy is a relative station keeping better than few meters for a wheel size of one kilometer.

1. INTRODUCTION

A formation flying is composed of several satellites close to each other and on quasi-similar orbits. The principal advantage of a formation flying is to allow flying over the same area at the same moment with different point of view. Such a configuration allows precise interferometric or correlation measurements.

The problematic aspect of formation flying is the relative station keeping. This point is more important than for classical satellite because the short distances between satellites lead to a more accurate control. Some studies on this subject have already been made (see [1], [2]).

2. THE INTERFEORMETRIC WHEEL

The interferometric wheel is composed of three micro satellites following a main radar satellite. The three satellites are located on a triangle and turn on a wheel. They act as receivers of the main satellite transmitted signal. The first objective is the interferometric experiment. Data will be used to determine a precise terrestrial topography and a precise coasting sea current model. Data combination allows accurate measurements impossible to obtain with one satellite.

The emitting satellite has not been chosen but several choices are possible. Some early studies have been performed with ALOS satellite which orbit is a phased and heliosynchronic orbit located at 700 km of altitude and with ENVISAT which orbit is also a phased and heliosynchronic orbit (see [3]). In the last missions analysis (see [4]) the main satellite is TerraSAR-L.

The distance between the main satellite and the wheel depends of the choice of the main satellite. The wheel can be in front of or behind the main satellite. The range is between 30 to 150 km. The size of the wheel depends on the SAR frequency. The semi-major axis of the wheel varies from 1 km to 15 km.

3. INTERFEROMETRIC WHEEL GEOMETRIE

The three satellites of the wheel must have the same orbit period, which means same semi major axis than the main satellite. The three satellites of the wheel have same mean nodal elongation (ω+M) and are shifted on eccentricity vector. Two configurations have been studied. Figure 1 represents initial conditions of the first configuration. The three micro satellites eccentricity vectors are uniformly distributed on a circle around the frozen eccentricity and whose radius is computed to fulfill the right size depending on the SAR frequency. We use the q,s,w orbital local frame of the main satellite (q : radial, w : perpendicular to the orbit plane) to study the relative motions of the three satellites of the wheel.

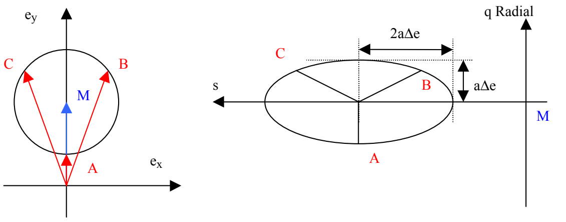

Fig 1. Eccentricity vector. First configuration

In this configuration the three satellites are shifted of 120 degrees. This configuration allows to optimise the baseline value, which is the vertical range between two satellites. Choosing the right couple of satellite allow to have a vertical baseline which variation is no greater than 7%. This allows to performed interferometric measurement anywhere along the orbit.

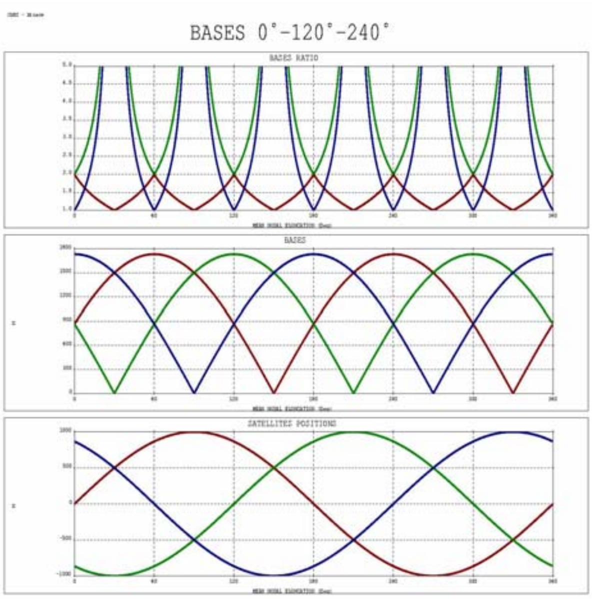

Fig 2. Baseline for first configuration

Figure 2 shows the baseline ratio, baselines values and motion of each micro-satellite relative to the centre S of the formation.

A second configuration called the Two-Scale Cartwheel has been optimised to have two baselines instead of one to combine interferometric measurement.

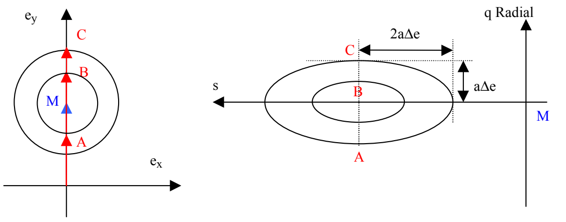

Fig 3. Eccentricity vector. Second configuration

Figure 3 represents initial conditions of the second configuration and Figure 4 baseline values.

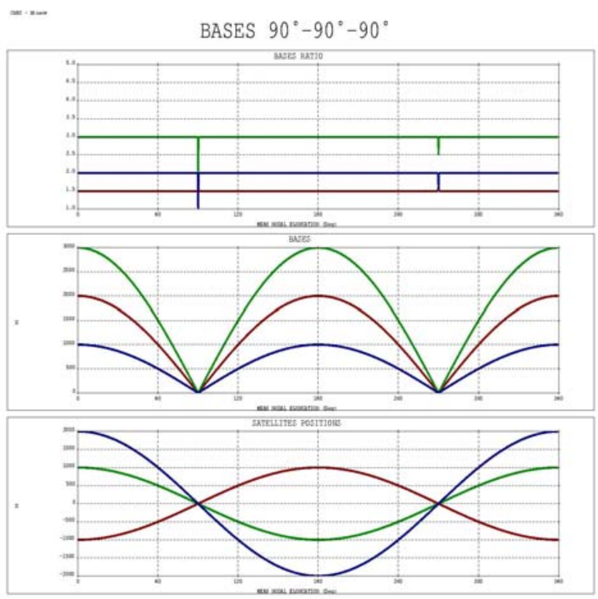

Fig 4. Baseline for second configuration

This time, baselines are not constant but there are two baselines which ratio can be chosen to any given value. This configuration does not allow performing interferometric measurement anywhere along the orbit. The interferometric measurement can be realized only at two given anomaly along the orbit where baseline is maximum. One may think that this configuration will not allowed covering all latitude. In fact, the natural motion of the eccentricity vector which turns around the frozen eccentricity of the main satellite allow covering all latitude in 55 days as showed on Figure 5.

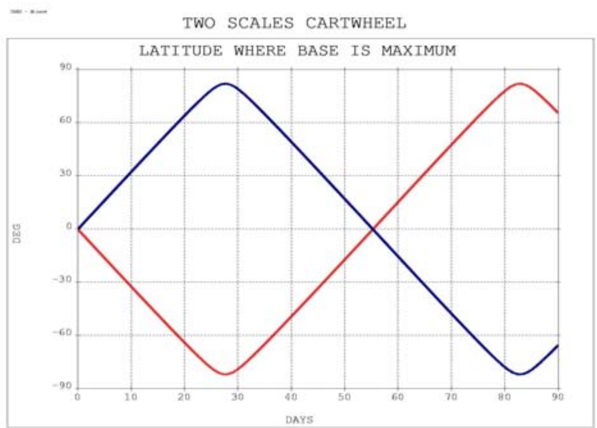

Figure 5. Latitude where baseline is maximum

4. POSITIONING STRATEGY

The positioning strategies which have been studied was based on the assumption that all satellite of the wheel are launched simultaneously on the same launcher directly to the same orbit (behind for instance) of the main satellite. There is no major difficulty for the positioning. Strategies consist to create drift orbits (different semi-major axis and same eccentricity) and then target the same semi-major axis as the main with the right shift in eccentricity. If the size of the wheel is not too big, we can target the right value of perigee with the drift orbit so that the positioning consist only in three manoeuvres (A,B and C see Figure 6).

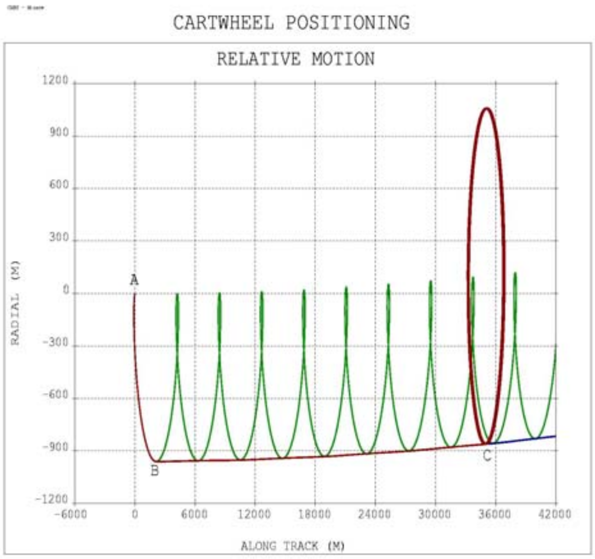

Figure 6. Positioning of the standard Cartwheel

The cost of the positioning for this figure is 1 m/s.

In case of a bigger wheel, it will be too fuel consumptive to target directly the right perigee. On more manoeuvre is required as showed on figure 7 (A,B,C and D).

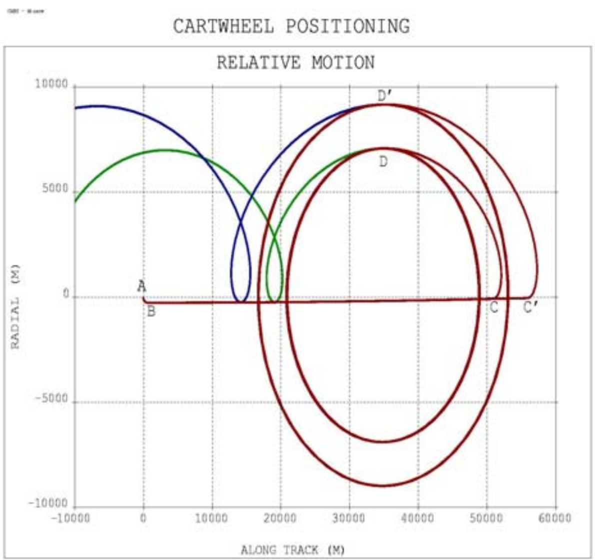

Figure 7. Positioning of the Two-Scale Cartwheel

Figure 7 is the positioning of the Two-Scale Cartwheel. The fuel consumption is 5 m/s.

On figures 6 and 7, the nominal trajectories are plotted in red. Green and blue trajectories are trajectories in case of engines failures. It can be demonstrated that for chosen wheel size, there are no collision risks.

Figure 8 shows reconfiguration of the wheel to change its size.

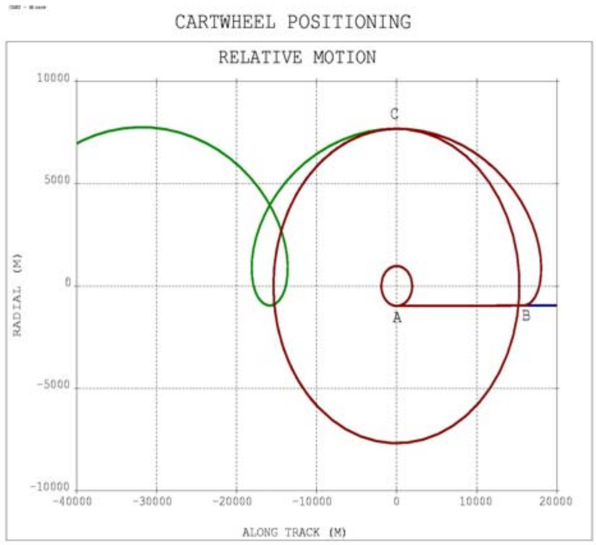

Figure 8 Reconfiguration strategy

There are three manoeuvres (A,B and C). As all satellites take the same way to go from A to C, the angle shift is not modified during the reconfiguration. For instance, if the satellite were shifted with an angle of 120 degrees on the small wheel, there will be shifted of 120 degrees on the big wheel.

5. STATION KEEPING

There is two kind of window orbit control for the interferometric wheel station keeping.

The first window concerns the distance between the wheel and the main satellite. This window is defined so that all the satellite of the wheel must be inside two constraints :

- The first one L_1 is to avoid a collision risk with the main satellite,

- The second one L_2 is due to a maximum pointing angle above which the radar measurements will be too much degraded (see figure 2).

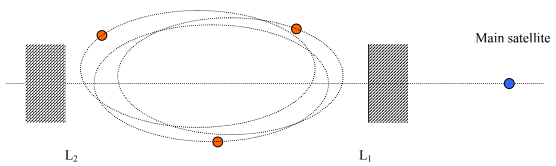

Fig 9. Constraint with the main

The second window concerns the relative positions of each satellite of the wheel. This window is express as a constraint on the difference of the position for each couple of satellite of the wheel. This difference of anomaly should fit between +/- Y horizontally (e. g. along the s axis) which value is determined on performances of the radar system. There is no constraint along the radius component.

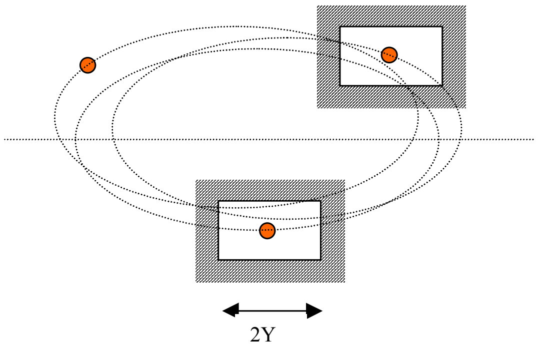

Fig 10. Constraints between Wheel Satellites

These two constraints can easily be transformed into constraints on the mean nodal elongation of each satellite of the wheel.

The first window leads to a constraint on the nodal elongation of each satellite of the wheel relative to the nodal elongation of the main satellite. The mean anomaly must be between $\Delta\alpha_1$ and $\Delta\alpha_2$ (see figure 11) :

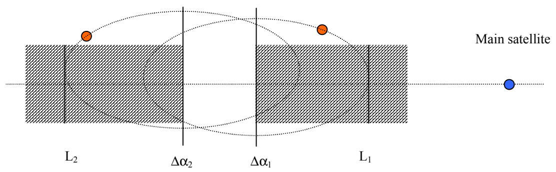

Fig 11. First window

The second window implies a constraint on the difference of nodal elongation between each couple of satellite of the wheel (see figure 12). This window implies also a constraint on the eccentricity vector of each satellite. However this constraints was not taken into account in the control strategy as simulation have proven that eccentricity vectors remain stable (see simulations paragraph).

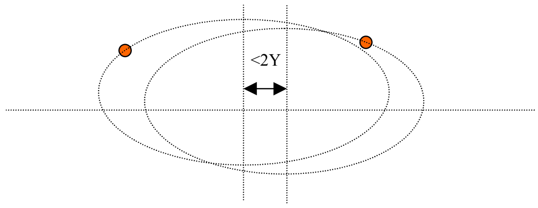

Fig 12. Second window

The orbital perturbation studies have already been presented in [3] and [4]. Perturbations that have to be taken into account for the station keeping are secular effects, which affect the geometry of the wheel. The main effect is coming from the atmospheric drag.

6. ORBITAL PERTURBATION STUDY

The most important effect of atmospheric drag is on the semi major axis. It implies a relatively strong secular drift because of the low orbit altitude of the formation, around 5 km of decrease during one month for a mean solar activity. This involves heliosynchronism property loss, and also an orbit period drift. As the mater of fact, there is an important variation between each satellite of the wheel and the main satellite as they have different surface over mass ratio. The relative drift observed on the semi major axis of each satellite of the wheel is not really high but the wheel is very sensitive because of the short distances between the 3 micro satellites. Figure 13 shows that this motion is not secular but periodic.

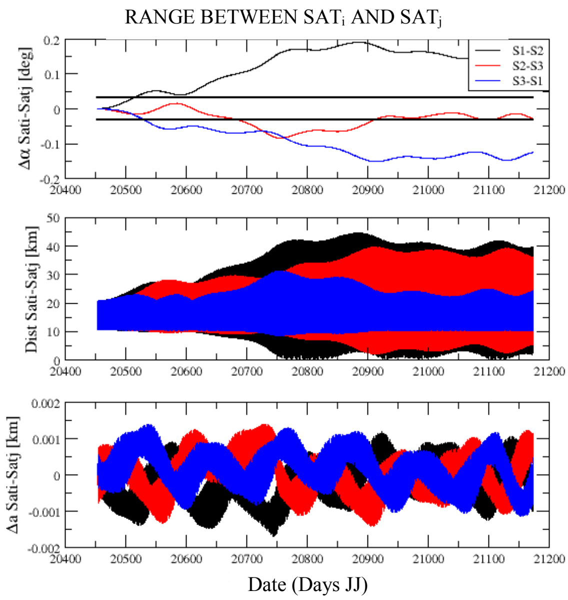

Fig 13. Perturbations effects

7. STATION KEEPING CONTROL STRATEGY

The control strategy of the wheel consists in maneuvers along the velocity vector to control the semi major axis and then the mean nodal elongation. There is no control of the eccentricity.

Two kinds of maneuvers are performed. The first kind consists of chemical maneuvers to control the center of the wheel relatively to the main satellite. These are classical maneuvers and are not described in this paper.

The second kind of maneuvers consists to control the mean nodal elongation between each satellite of the wheel. For that purpose a pure passive actuator is used. Differential drag due to a differential attitude allows controlling the formation even in low solar activity. This method has already been used for satellite constellation station keeping (see [5]).

The algorithm of the autonomous orbit controller is the following :

Let us note $\alpha=\omega+M$ the mean nodal elongation. The wheel configuration is so that each satellite has same nodal elongation value. The control strategy consists thus to minimize delta between nodal elongation.

Let us note α_c and a_c the nodal elongation and semi major axis of the satellite that will be considered as the reference satellite. Let us note α and a the nodal elongation and semi major axis of the slave satellite that will be controlled relatively to the reference satellite. The reference satellite has a fixed attitude, and attitude of the slave satellite is controlled. Considering that the relative motion of semi-major axis, due to the drag, is linear, one can write :

$$\Delta\ddot{\alpha} + 3\frac{n_c}{a_c}\Delta\dot{a} = 0$$

Where $\Delta\alpha = \alpha - \alpha_c$, $\Delta a = a - a_c$ and n_c is the mean motion of the reference satellite.

Changing the attitude of the slave satellite (or the attitude of its solar panel) changes the cross section of the satellite and introduces differential drag. We can therefore write :

$$\Delta\dot{a} = K\frac{\Delta S}{M}$$

where K is a coefficient depending on solar activity. K is supposed to be constant for a short period (days or weeks). ΔS is the differential cross section attitude, which depends of differential attitude and M is the satellite mass. Let us note φ_0 the fixed attitude of the reference satellite, φ the attitude of the slave satellite and S_p the satellite surface aligned to the path motion (perpendicular to the cross section).

Then, one can write :

$$\Delta\dot{a} = K\frac{\Delta S}{M} = K\frac{S_P}{M}(\varphi_0 - \varphi)$$

The value of φ_0 is chosen so that the salve satellite can increase or decrease the drag relatively to the reference satellite (the value taken in the simulation is 15 degrees).

Using state feedback control, attitude φ of the salve satellite is computed as follow :

$$\varphi = \varphi_0 + K_1\Delta\dot{\alpha} + K_2\Delta\alpha$$

where K_1 and K_2 are constant gains. The close loop equation is then :

$$\Delta\ddot{\alpha} + 3\frac{n_c}{a_c}\frac{K}{M}S_pK_1\Delta\dot{\alpha} + 3\frac{n_c}{a_c}\frac{K}{M}S_pK_2\Delta\alpha = 0$$

For a given value of dumping factor ξ and close loop pulsation ω, we get the value of gains :

$$K_1 = \frac{4\pi\xi a_c M}{3n_c K S_P T}$$

$$K_2 = \frac{4\pi^2 a_c M}{3n_c K S_P T^2}$$

where T is the period corresponding to ω ($\omega = \frac{2\pi}{T}$)

As it is relatively difficult to compute the value of $\Delta\dot{\alpha}$ onboard, the following equation due to Kepler motion :

$$\Delta\dot{\alpha} = -\frac{3}{2}\frac{n_c}{a_c}\Delta a$$

is used to get φ. Thus φ is computed onboard with the following formula :

$$\varphi = \varphi_0 - \frac{3}{2}\frac{n_c}{a_c}K_1\Delta a + K_2\Delta\alpha$$

The attitude is change once per orbit and is computed with average orbital parameters.

The figure 14 shows a simulation over a period of one month. The following values have been taken for this simulation :

Orbital elements of the main satellite (TerraSAR-L orbit) : Semi-major axis : 7007.137 km, inclination : 97.93 degrees, Eccentricity : frozen

Wheel size : 1 km ($a\Delta e$). Solar panel surface (Sp) : 2 m^2.

Cross section : 1 m^2. Low solar activity (flux 100).

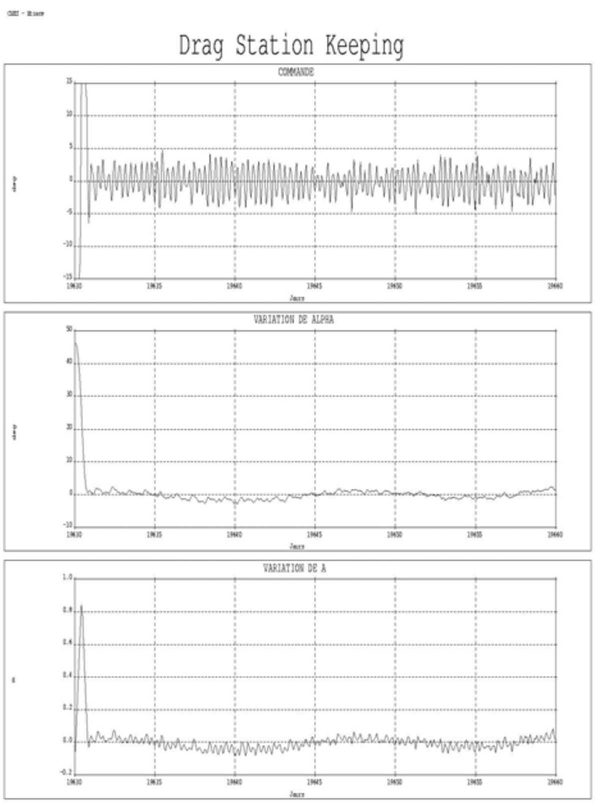

Figure 14 : Drag station Keeping simulation

The initial condition for this simulation was : same semi-major axis for the two satellite (reference and salve) but with a shift of mean nodal elongation of 45 meters. Top of the figure 14 is the attitude motion of the slave satellite (φ angle in degrees), medium figure is the delta in mean nodal elongation (converted in meters) and bottom figure is the delta in semi major axis.

We can see that, at the beginning, the command goes up to the limit (15 degrees) to compensate as fast as possible the shift in mean nodal elongation. This leads to a Δa of 0.8 m. It takes about 65000 sec to converge. After the convergence, the command oscillates between plus and minus 5 degrees. The maximum delta value in mean nodal elongation is 2.5 m.

Gains were computed with ξ=0.8 and T=100000 sec. Simulations show that ξ =0.60 and T=65000 sec which values are quite close to the ones given by the simple analytic model. It is possible to modify gains to obtain a given value for ξ and T . The value of T cannot be to reduce to much avoid high command value (φ), especially in case of low solar activity. 65000 sec is a good optimization.

8. REFERENCES

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