FOUR-FORMATION IN-TRACK CONFIGURATION MAINTENANCE STRATEGY

Alain LAMY, Thierry COSTES

Centre National d'Etudes Spatiales (CNES)
18 avenue Edouard Belin, 31401 Toulouse Cedex 9, France
E-mail: Alain.Lamy@cnes.fr, Thierry.Costes@cnes.fr

ABSTRACT

The aim of this paper is to present the analysis conducted by CNES for the maintenance of a formation made of several LEO satellites (typically 4) in several planes (typically 2), 100 km or so apart from each other. The along-track separations between the satellites have to be controlled to within 15 km thanks to orbit correction maneuvers supposed to be performed every 2 weeks. The main difficulty is related to solar activity which is expected to be close to its maximum for the entire mission's lifespan. As a matter of fact, a high solar activity makes orbit prediction harder, and makes it impossible to keep the altitude of the formation constant. Thus, a specific relative maintenance strategy had to be devised in order to meet the mission's requirements.

The first part provides a few elements on the mission analysis process that has taken place. The method used for the evaluation of the maneuver frequency is detailed, based on the evaluation of the effects of atmospheric drag on the orbit. The second part is dedicated to the maintenance strategy that has been designed, and particularly to the computation of the reference orbits and of the velocity increments that enable the in-track inter-satellite distances to be maintained within the desired bounds. Finally a few simulation results are presented; they enable the performance of the maintenance strategy to be checked in a more realistic context.

Acronyms and notations

$a, \dot{a}$	a : semi major axis, $\dot{a}$: derivative with respect to time (also written da/dt)
AoL, α	(mean) Argument of latitude (i.e. ω+M, with ω: argument of perigee and M: mean anomaly), also called "satellite position" since mainly the arguments of latitude matter
ΔV	Velocity increment
LEO	Low Earth Orbit
N	Number of satellites in the formation
RAAN, Ω	Right Ascension of the Ascending Node

1. INTRODUCTION

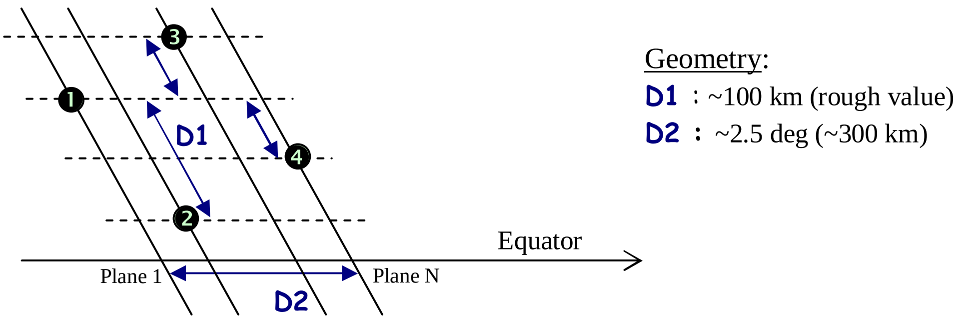
1.1 Formation description

We consider a formation composed of several satellites in several planes. The orbits are circular with altitudes of about 700 km, and nearly sun synchronous with local hours of ascending node close to 22h30. The satellites are rather widely separated as typical figures describing the geometry are the following:

– Along-track separation between any 2 satellites ~ 100 km,
– Maximum difference of RAAN between planes ~ 2.5 degrees (about 300 km at the Equator).

The typical formation configuration then looks as depicted in Figure 1.

Figure 1: Example of formation geometry



In the general case, the formation is composed of N (typically 4) satellites. The area/mass ratios are approximately the same for all the satellites (about 7.e-3 m^2/kg).

1.2 Objectives

The implementation is assumed to take place around 2012, that is when solar activity is expected to be close to its maximum (according to NOAA predictions [Ref. 2], cycle #24 high is expected between mid 2011 and mid 2012 depending on hypotheses considered).

One important aspect is the maintenance of the formation's geometry. In fact the cross-track separations do not have to be strictly controlled. The required performance is compatible with the natural evolution of the orbit planes, and adequate values for the initial inclinations enable the angle between the planes not to drift beyond the maximum limit allowed. The main purpose assigned to the maintenance process is the control of the along-track separations. They have to be controlled to within 15 km, which corresponds to maintaining the mean argument of latitude of each satellite to ±7.5 km relatively to some (to be defined) reference position.

Orbit maneuvers are performed regularly in order to maintain the configuration as described above. These maneuvers are tangential maneuvers only. The frequency of orbit corrections is desired to be less than one correction every 2 weeks (for each satellite, all the satellites nearly at the same time). This is mainly for operational reasons and in connection with mission programming scheduling.

2. MAINTENANCE ANALYSIS

2.1 Perturbation effects

The main perturbation effects that affect the geometry are the following:

Out-of-plane effects: The orbit RAANs evolve differently because the inclinations tend to drift at slightly different rates. As mentioned above, the differential effects are consistent with the mission's objectives. They are therefore not compensated for, and consequently will not be discussed in any more detail in this paper

Along-track effects: The semi major axes are expected to decrease by 15-20 meters per day on average, which affects the arguments of latitude of the satellites in a more or less predictable way. Differential effects between 2 satellites in 2 different planes stem from the slightly different values of the node local hours, i.e. from the slight difference in orientation of the orbit planes with respect to the Sun direction. The relative semi major axis decrease rate for 2 satellites in 2 different planes (assuming same area/mass ratios) is thus computed to be less than 1 m/day for an angle between the ascending nodes of 2.5 degrees, that is less than 2% of the "absolute" decrease rate.

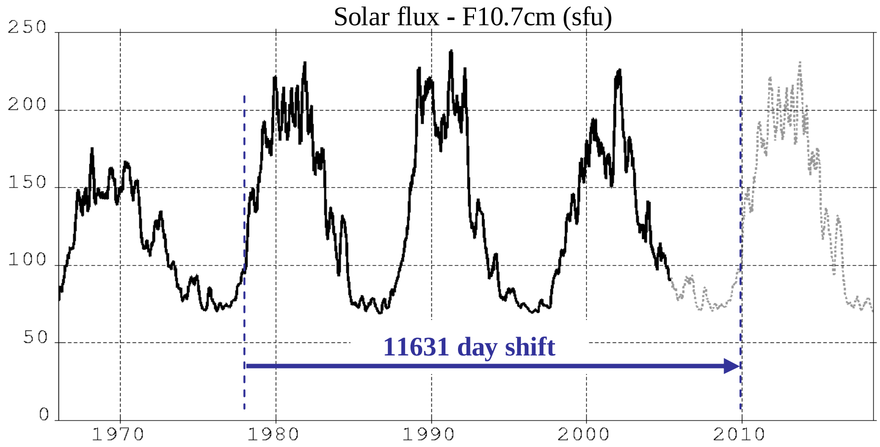
2.2 Prediction of atmospheric drag

Two aspects have to be considered when dealing with atmospheric predictions:

- the influence of mean solar activity level, which affects the mean maneuver frequency, hence the correction cost,
- the influence of short / mean term variations of solar and geomagnetic activity, which affects orbit prediction and maintenance accuracy.

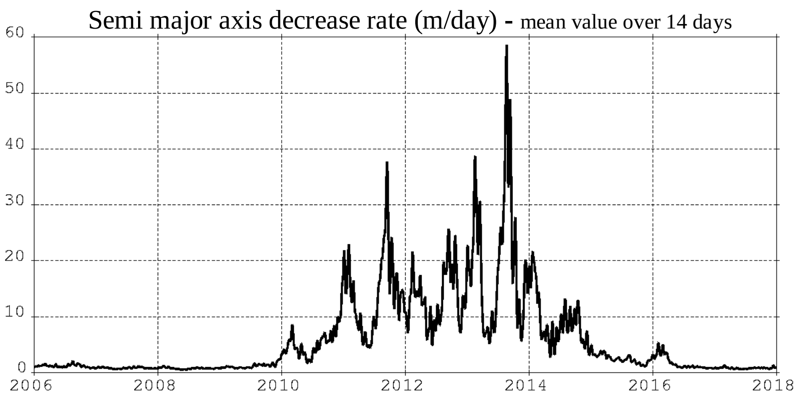
In order to evaluate both effects, solar and geomagnetic past real data have been used. The solar cycle used was in fact number 21 after a translation of as many days as required so as to fit the latest data available (see Figure 2). Such a prediction method may not be very accurate, but this is not a major problem since mainly the maximum level of solar activity (and its variations) will in fact matter.

Figure 2: Predicted solar activity as used in computations



The DTM78 density model has been used in all calculations in order to estimate the semi major axis decrease rate, although it is known to be a bit pessimistic. Depending on the mission's beginning of life, the mean value over 14 days is of the order of 15-20 m/day on average, as seen in Figure 3.

Figure 3: Estimated semi major axis decrease rate



2.3 Maintenance options

Two main options can be envisaged in order to maintain the difference in argument of latitude between any 2 satellites as constant as possible.

Option 1: The 1st option that springs to mind and which is also the simplest would be to control each satellite independently from the others, and to apply some control strategy close to what is generally done in most station keeping cases for the control of one isolated satellite. This implies to define some reference law for the argument of latitude of each satellite, all laws corresponding to the same mean motion (i.e. to the same mean semi major axis).

A straightforward evaluation consists in computing the theoretical maneuver frequency that corresponds to a given control window size (here ±5 km instead of the maximum allowed ±7.5 km). The computation is based on the formulas connecting da/dt and the maneuver frequency as described in Figure 4.

Figure 4: Rough estimation of maneuver frequency

$$T^2 = \frac{8\,w}{A} \qquad \text{where:} \quad A = \frac{3}{4}\frac{|\dot{a}|}{a}\,n$$

With:
T: time between 2 consecutive maneuvers
w: size of control window
n: mean motion,
a: semi major axis, $\dot{a}$ =d*a*/dt.
A: 2nd order coefficient of the polynomial that describes the evolution of the argument of latitude (assuming a constant value for da/dt).

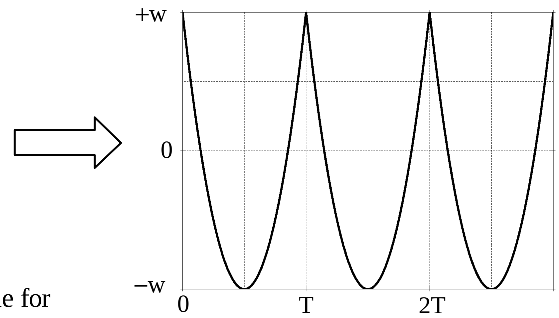
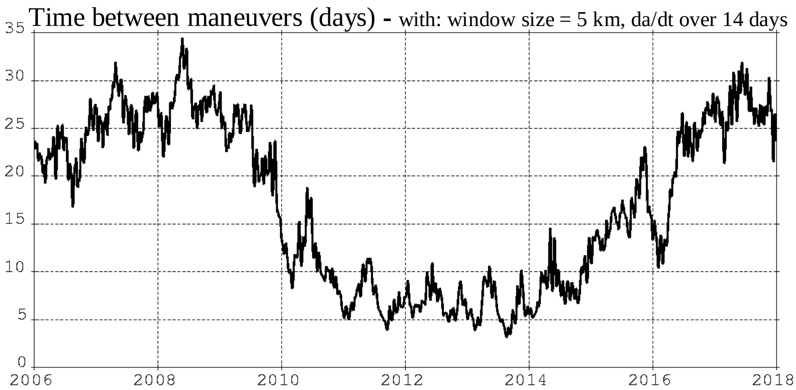


Figure 5 shows the duration (T) between 2 consecutive maneuvers resulting from using da/dt as evaluated in paragraph 2.2. We immediately see that T is smaller than expected, even without considering other uncertainties such as prediction or maneuver execution errors. It is even optimistic as the mean value of da/dt is underestimated for values of T less than 14 days. Consequently, this option could not be retained as incompatible with the requirements.

Figure 5: Theoretical maneuver frequency based on "true" solar activity



Option 2: This second option consists in allowing all the semi major axes to decrease gracefully. The maneuver frequency decreases too because the major part of the perturbation effects is not corrected; only the differential effects are. This option is studied in more detail in the next paragraphs.

2.4 Estimation of the required maneuver frequency (option 2: decreasing altitudes)

The maneuver frequency for option 2 is evaluated considering the 3 following main factors:

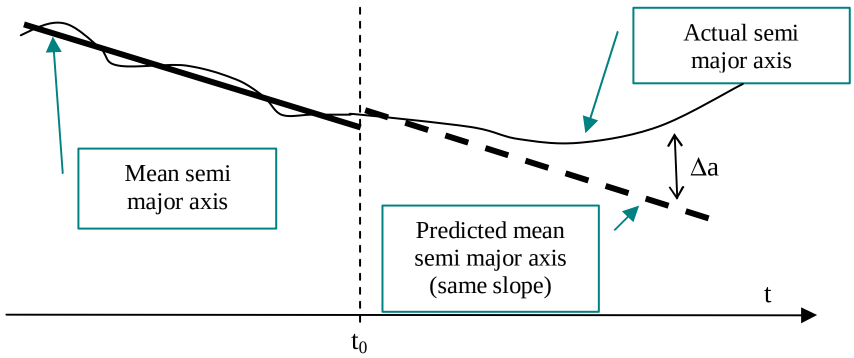
– The effect of mean differential atmospheric drag (1)

  This is identical to what was presented previously, save that $\dot{a}$ is now a relative quantity. It has been said that the difference of local hour between 2 planes leads to a difference of $\dot{a}$ of about 2% (with respect to the "absolute" decrease rate). The relative $\dot{a}$ with respect to the average decrease rate is then around 1%. Another 1% margin is added. This leads to 2% including uncertainties on area/mass ratio and attitude.

– The effect of prediction errors (2)

  Prediction errors exist because the predicted orbits on which the computation of the next maneuvers is based do not correspond to what will actually happen. The reason lies in the difficulty to predict solar activity accurately. The method considered to evaluate these prediction errors makes no use of any predicted data for the solar and geomagnetic activities but is based on the assumption that the mean value of the semi major axis decrease rate (or equivalently the mean value of the atmospheric density) will not change over the next 14 day period (see Figure 6).

Figure 6: Method used for the estimation of prediction errors



The computed values of Δa (error on semi major axis) are integrated in order to obtain the effect on the argument of latitude (which is of interest):

$$\Delta\alpha = \int_{t_0}^{t_0+D} K\ \Delta a(t)\ dt$$

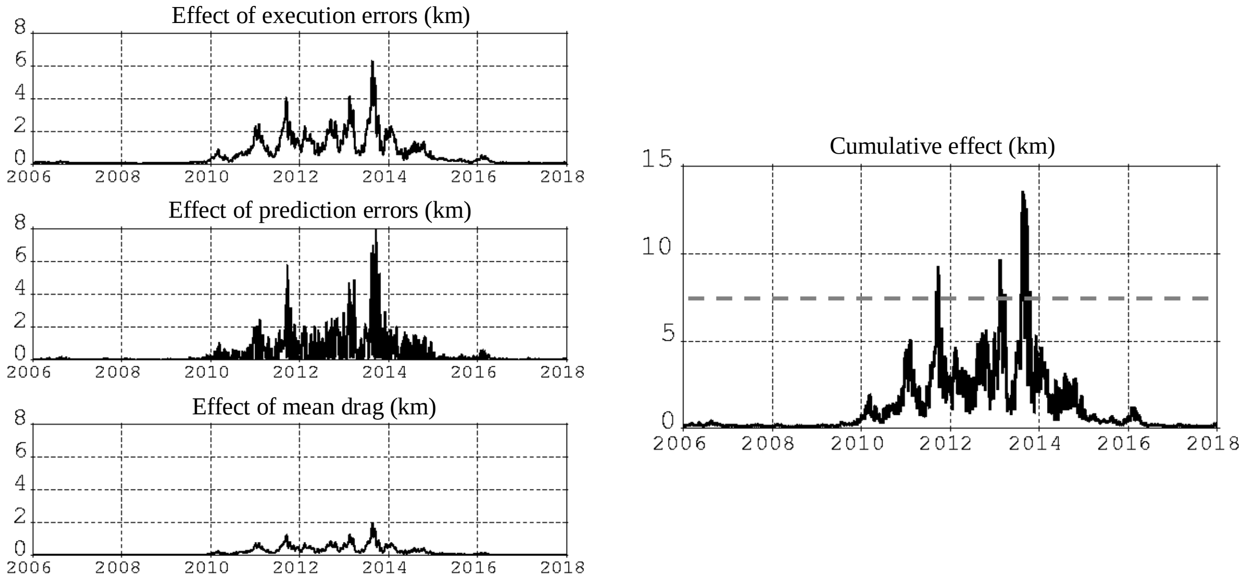
where K is approximately constant $(= \frac{\partial\dot{\alpha}}{\partial a})$

– The effect of maneuver execution errors (3)

  Hypotheses on maneuver execution errors are based on flight data obtained for other satellites with the same propulsion system characteristics. The maximum magnitude of the relative execution error appears to be between 5% (for a semi major axis change of more than 100m) and 20% (for a semi major axis change of less than 25m).

The 3 effects above (1, 2, 3) combine in the worst favorable situation. Figure 7 shows the resulting control accuracy.

Figure 7: Evaluation of relative control accuracy



Thus, the along-track errors are expected to remain below accepted limit (7.5 km) most of the time, that is except for very intense solar activity periods. It is considered as satisfactory since it might be possible to alter the maneuver frequency for short periods when solar activity happens to be too strong.

3. DETAILED (RELATIVE) MANEUVER STRATEGY

3.1 Summary of previous results and objectives

We have seen in the previous parts that the performance that can be expected is compatible with the requirements (most of the time). The question is now to devise an effective control strategy.
For this strategy to be consistent with the analysis as shown above, the effect of perturbations (hence the ΔV) should be minimized (see 2.4). The problem then comes down to determining the "best" reference satellite positions (i.e. the best reference mean arguments of latitude) that minimize the effect of perturbations, and to determining the maneuvers that enable the satellites to stay as close as possible to their respective reference positions.

The main points that have influenced the design of the maintenance strategy are the following:

- A constant solar/geomagnetic activity makes the semi major axes decrease at an almost constant rate, which yields a parabolic evolution versus time for the argument of latitude.
- The relative evolution of the argument of latitude of one satellite in one plane with respect to some other satellite in some other plane is still approximately parabolic as a function of time. For two identical satellites in two different planes, the "ideal" reference position would correspond to a fictitious satellite placed in the bisector plane, in which case the relative parabolic evolutions for the 2 satellites would be inverted.

The strategy that results from all these remarks is detailed in the next paragraphs.

3.2 Mathematical formulation

The problem could be written as follows:

Minimize: $\sum_k \max_{i=1..N} |\Delta V_i(t_k)|$ and minimize: $\max_{i,j,t} \left|\alpha_i(t) - \alpha_j(t) - \Delta\alpha_{i,j}^{nom})\right|$

where t_k stands for 'maneuver instant' (i.e. every 14 days), i and j are the satellite indices (1 to N), $\alpha_i(t)$ represents the mean argument of latitude of satellite i, and $\Delta\alpha_{i,j}^{nom}$ represents the nominal (and constant) difference of argument of latitude between the satellites i and j.

It is clear that this problem cannot be uniquely solved since the 2 above objectives are contradictory. But in our case station keeping maneuvers are small, so that minimizing the control cost is not of major importance.

If the problem is transformed into determining the best reference satellite positions, it becomes:

Minimize: $\sum_k \max_{i=1..N} |\Delta V_i(t_k)|$ and minimize: $\max_{i,t} \left|\alpha_i(t) - \alpha_i^{ref}(t)\right|$

where $\alpha_i^{ref}(t)$ represents the reference position of satellite i.

$\alpha_i^{ref}(t)$ is looked-for as: $\Delta\alpha_i^{nom} + \alpha^{ref}(t)$ where $\Delta\alpha_i^{nom}$ is a constant value that defines the nominal position of the satellite in the formation with respect to some arbitrary reference (e.g. one particular satellite), and $\alpha^{ref}(t)$ does not depend on any particular satellite.

The problem is then twofold:

- determine the reference mean AoL for the formation as a whole (i.e. $\alpha^{ref}(t)$) ,
- determine the expressions for the (tangential) ΔVs in order for the satellites to stay as close as possible to their respective reference positions.

3.3 Maneuver computation principles

The problem can be simplified further by imposing the expression for the ΔVs. The effect of a velocity change is modeled as consisting in a slope change only. The ΔVs as proposed change the slope at the time of the maneuver so that the predicted argument of latitude passes through some targeted point at the time of the next maneuver. This targeted point is deduced from the theoretical control window size derived from the (supposed constant) semi major axis decrease rate. This is illustrated in Figure 8.

Let A, B, C be the coefficients of the polynomial that describes the evolution of the argument of latitude relative to some (yet to be determined) reference:

$$P(t) = A\,t^2 + B\,t + C \qquad (t \le 0)$$

The instant of the looked-for maneuver is here t = 0 by convention.

The expression with the effect of the maneuver taken into account is then:

$$P^+(t) = A\,t^2 + B^+\,t + C \qquad (t \ge 0)$$

By definition, $P^+(t)$ is chosen such that:

$$P^+(T) = w = \frac{A\,T^2}{8}$$

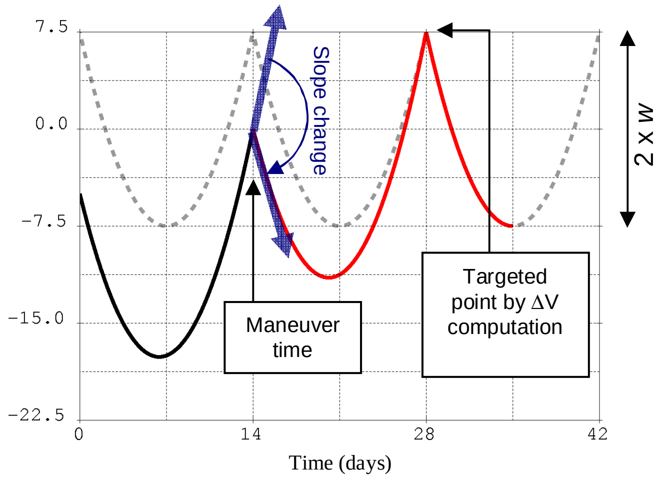
Hence:

$$B^+ = -\frac{7}{8} A\,T - \frac{C}{T} \qquad \text{(Eq. 3.3-1)}$$

The slope change at the time of the maneuver is then:

$$\Delta B = B^+ - B = -\frac{7}{8} A\,T - B - \frac{C}{T}$$

Figure 8: Illustration of the ΔV base strategy



The dashed line represents the ideal evolution (between –*w* and *w*, with *w*=7.5 km).

Initial conditions (black curve) are slightly non nominal. A maneuver (slope change) is performed at t=14 days.

After the slope change is applied, the situation is nominal again at t=28 days, that is for ordinate=+*w*.

If we now apply this method to the computation of the next 3 successive ΔVs (assumed evenly spaced, at respective times 0, T, 2*T), one gets:

Maneuver at t = 0 $\Rightarrow$ $\Delta B_1 = -\frac{7}{8} A\,T - B - \frac{C}{T}$

Maneuver at t = T $\Rightarrow$ $\Delta B_2 = -\frac{17}{8} A\,T + \frac{C}{T}$

Maneuver at t = 2T $\Rightarrow$ $\Delta B_3 = -2 A\,T$

Note: ΔB_3 is identical to the routine slope change that is needed to follow the nominal parabolas.

3.4 Determination of the reference arguments of latitude

We now consider all the satellites in the formation. The relative AoL for each satellite (after the constant offsets $\Delta\alpha_i^{nom}$ have been removed) are assumed to be defined by:

$$P_i(t) = (A_i - \overline{A})\,t^2 + (B_i - \overline{B})\,t + (C_i - \overline{C})$$

where $(\overline{A}, \overline{B}, \overline{C})$ defines the reference polynomial to be solved for.

The $(\overline{A}, \overline{B}, \overline{C})$ reference coefficients are determined so that the maximum ΔV computed over all the satellites is minimized at each of the next 3 maneuver dates: 0, T, 2T.

We obtain successively:

$$\min(\Delta B_{3,i}) = -\max(\Delta B_{3,i}) \qquad \text{(over all i in 1..N)} \qquad \Rightarrow \ \overline{A} = \frac{\min(A_i)+\max(A_i)}{2}$$

$$\min(\Delta B_{2,i}) = -\max(\Delta B_{2,i}) \qquad \text{(over all i in 1..N)} \qquad \Rightarrow \ \overline{C} = \frac{\min(K_i)+\max(K_i)}{2}$$

with: $K_i = -\frac{17}{8}(A_i - \overline{A})T^2 + C_i$

$$\min(\Delta B_{1,i}) = -\max(\Delta B_{1,i}) \qquad \text{(over all i in 1..N)} \qquad \Rightarrow \ \overline{B} = \frac{\min(L_i)+\max(L_i)}{2}$$

with: $L_i = \frac{7}{8}(A_i - \overline{A})T + B_i + \frac{(C_i - \overline{C})}{T}$

The knowledge of $(\overline{A},\overline{B},\overline{C})$ enables us to derive the successive values for the slope changes for each satellite, and particularly those corresponding to the next maneuver date (which mostly matter).

Remarks:

- The value found for $\overline{A}$ is not surprising. It means that the reference semi major axis decrease rate is exactly midst of the extremal rates (considering all the satellites).
- The above results were confirmed by linear programming simulation. The criterion to be minimized was chosen as: $\sum_{k=1..3} \max_{i=1..N} |\Delta B_i(t_k)|$, which is not exactly the same as above, but which appears to be equivalent.

3.5 Sample simulation cases

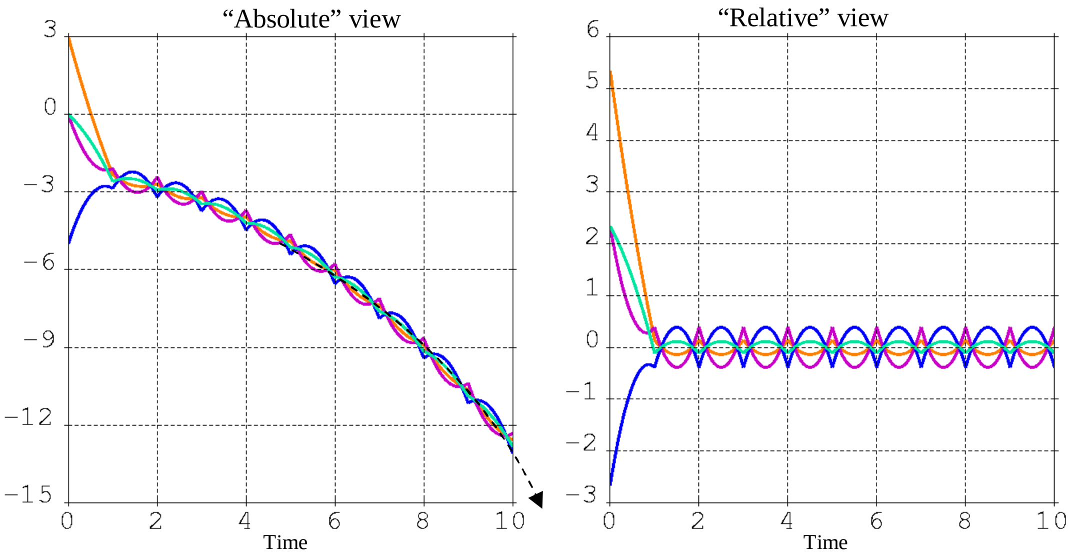
Two simulation cases of the application of the maneuver strategy as derived in the previous paragraph are given hereafter.

We start from arbitrary initial conditions defined by the N polynomials expressing the evolution of the (unbiased) argument of latitude for each satellite i (i in 1..N). The algorithm determines the successive slope changes for each satellite at each maneuver instant kT (k=0,1,2...). T is given the value 1. Units are here arbitrary.

Case 1: The polynomial models are defined at t=0 by the matrix $\begin{bmatrix} 3 & 0 & 1 \\ 0 & 1 & 3 \\ -5 & -1 & -3.2 \\ 0 & 0 & -1 \end{bmatrix}$ in which the i^th row contains the coefficients of order 0, 1, 2 respectively for satellite number i.

The right hand plot (see Figure 9) represents the evolutions relative to the reference polynomial (previously denoted by $(\overline{A},\overline{B},\overline{C})$). We see that the steady state is rapidly reached (after the 3^rd maneuver), and that the curves are well balanced around 0. The left hand plot represents the "absolute" evolutions. This second graph clearly exhibits the reference polynomial (in dashed line).

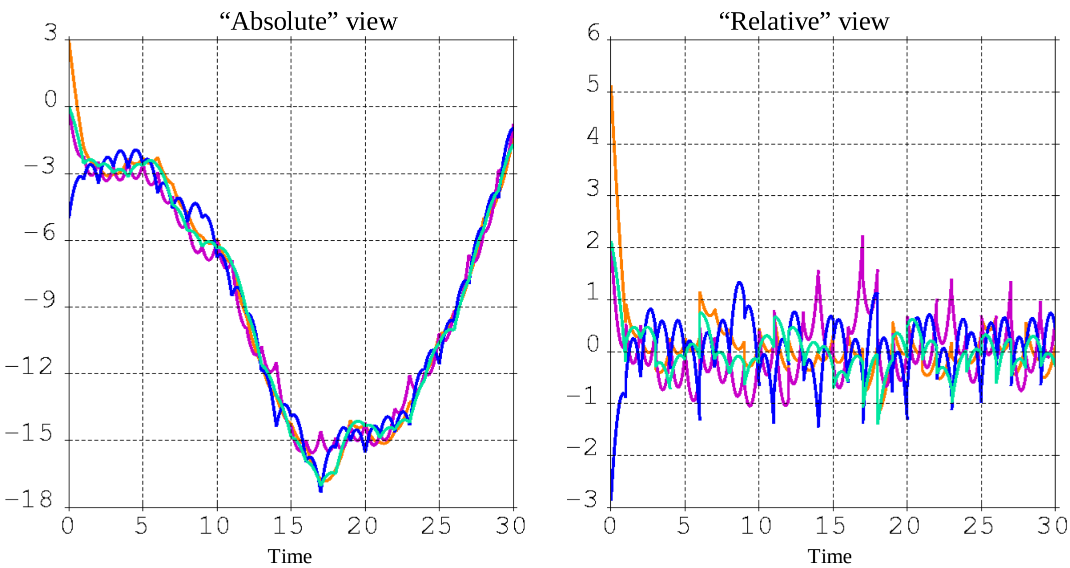
Figure 9: Strategy simulation - case 1



Case 2: For this second case, random errors are added to each slope change. The relative errors are distributed according to a Gaussian law (with mean=0 and standard deviation=10%). The initial conditions are also slightly modified (the coefficient -3.2 becomes -3) so that the nominal reference curve would have been a straight line if no random errors had been added.

Figure 10 shows the effect of errors on the reference trajectory. We can see how the reference trajectory adjusts continuously (left view) so as to reduce the effects of errors as much as possible. The right view shows how these errors tend to increase the effective window size. Still, the curves remain more or less balanced around 0.

Figure 10: Strategy simulation - case 2



3.6 Strategy enhancements

It might be objected that the expression of the slope change is imposed whereas it should theoretically be the result of the optimization process.

Different variants have been proposed. They all consist in varying the expression giving the slope change. Equation 3.3-1 is then replaced by one among the following:

$$B^{+} = -\frac{7}{8} A T - \frac{C}{T} \qquad \text{(nominal formula)}$$

$$B^{+} = 2\,AT - \sqrt{8\,A\,(AT^{2} + C)} \qquad \text{(minimum value over [0,T] = – value at T)}$$

$$B^{+} = 0 \qquad \text{(cancel slope)}$$

$$B^{+} = -\,AT - \frac{C}{T} \qquad \text{(value at T = 0)}$$

The algorithm is then modified as follows:

- compute $\overline{A}$ and $\overline{C}$ as before (or possibly by using a slightly modified formula for $\overline{C}$),
- Depending on the value of $C_i - \overline{C}$ (only), choose among the formulas above where $\overline{A}$ and $\overline{C}$ are to be replaced by $A_i - \overline{A}$ and $C_i - \overline{C}$ respectively,
- $\overline{B}$ is deduced so that the maximal et minimal slope changes have same amplitudes.

A few algorithms have been tested but none has seemed to show up as being really superior.

3.7 Practical implementation

The implementation of the algorithms as described in 3.4 is fairly straightforward except for a few subtleties. What is most important is the determination of the degree 2 polynomials that represent the evolutions of the mean arguments of latitude. The algorithm that has been chosen is basically the following:

☞ Step 1: Data selection

The calculations are based on past determined orbit data, from the date of the previous maneuver to the date of the one being computed. A few days of propagated orbit data may also be added to extend the data arc to about the duration between 2 maneuvers. Doing so, the algorithm is not too much dependent on the accuracy of the propagated relative orbits (that may more or less strongly depend on the drag coefficient as determined by the latest orbit determination process).

☞ Step 2: Data fitting

In order to eliminate a few perturbation effects that affect all the satellites at the same time, an intermediate reference is built as the usual average argument of latitude computed over a subset of satellites (nominally all of them but not necessarily). The differences of AoL are then computed for each satellite and sent as input to the fitting process. This fitting is simply a weighted least-squares computation that generates the N looked-for polynomials. Exponential weighting is possible although it doesn’t seem to be a mandatory option; it is mainly present as an additional degree of freedom that could be used to improve the performance later on if needed.

☞ Step 3: Maneuver computation

Maneuver computation is performed exactly as explained in paragraphs 3.3 and 3.4. The tangential ΔVs are derived from the slope changes by using the simple formula: $\Delta V = -\frac{a}{3}\Delta slope$, where a is the mean semi major axis (whose determination does not need to be extremely accurate).

4. OPERATIONAL CONCERNS

Solutions exist in order to handle some possibly occurring critical situations. Here is a brief description:

– Partial execution of maneuvers

This situation could be that of several maneuvers not being executed (or not being uploaded) correctly. It could be handled in 2 different ways. One way is to enable the calculation to be partially redone by using some previously stored data. Another possibility is to redo the computation by using predicted data only (as it is not considered possible to use data obtained before the latest maneuver). Reference satellites would be those whose maneuver has not been performed correctly and maneuvers would be computed for the other satellites only.

– One satellite temporarily out of formation but still controlled

This situation would consist in having one satellite temporarily off the formation, while the N-1 remaining ones would have to be controlled normally. The solution would then be to nominally use the algorithms for these N-1 satellites and to compute a maneuver for the Nth one with a reference AoL derived from the computation done for the other N-1 satellites.

5. EVALUATION OF THE MAINTENANCE ACCURACY

5.1 Objectives and hypotheses

The objective is here to assess the performance of the maintenance strategy. That is firstly to check that the control accuracy meets the mission's requirements and secondly (and secondarily) that the performance is compatible with the estimate obtained during the mission analysis phase.

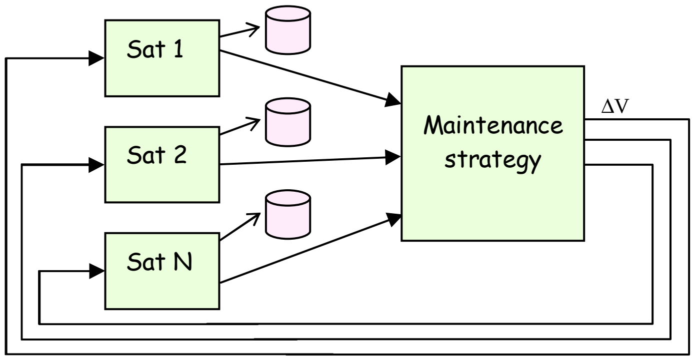
The algorithms are implemented in a simulation environment illustrated in Figure 11.

Orbit propagation for the N satellites takes account of the main usual forces that perturb the orbits: Earth, Sun, Moon gravitation, solar pressure, and of course atmospheric drag simulated by using actual solar and geomagnetic data (the same as in paragraph 2.2) along with the DTM78 density model.

Data from the N orbit propagators are sent to the maintenance software. The data actually sent consist of the (mean) orbit parameters ("mean" in the sense: with osculating terms coming from J2-J6 analytically removed). The lengths of the data arcs are equal to the time between 2 maneuvers minus small margins: after the previous maneuver (0.5 day) and before the one being computed (1 day). Thus, no predicted data are used in the computation, otherwise than predicted by the own method of the maintenance strategy.

The algorithm used is the basic one (as described in paragraph 3.4). Variants are not considered.

Figure 11: Simulation environment



Sat1, Sat2... SatN: N orbit propagators that compute the orbits considering the maneuvers received from the maintenance strategy process.

Maintenance strategy: software (written using *scilab* tool) that computes the N maneuvers, using the orbit data coming from the N satellites (from the previous maneuver and up to the one to be computed).

5.2 Description of the simulation case

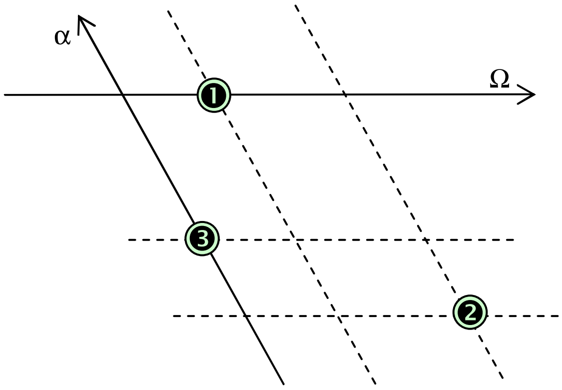
This simulation case aims at the maintenance of a formation of 3 satellites in 3 orbit planes. The orbits, formation geometry and other satellite characteristics are compatible with the hypotheses given in paragraph 1.

The nominal configuration is as depicted in Figure 12. Nominal value for the area/mass ratio is 8.e-3 m^2/kg. Variations to this ratio are applied: up to 1% with respect to average value (see Figure 12 for details).

Simulation is performed over more than 2 years starting from mid 2012, that is when solar activity is strong. Maneuvers occur every 14 days. Maneuver execution errors are drawn at random as follows:

- $\Delta V_{actual} = \Delta V_{nominal}\,(1+\varepsilon)$; ε follows a zero-mean Gaussian distribution with standard deviation 0.066.
- ΔV_{actual} is truncated to the nearest 0.5 mm/s.

Figure 12: Geometry for the simulated case and specific hypotheses



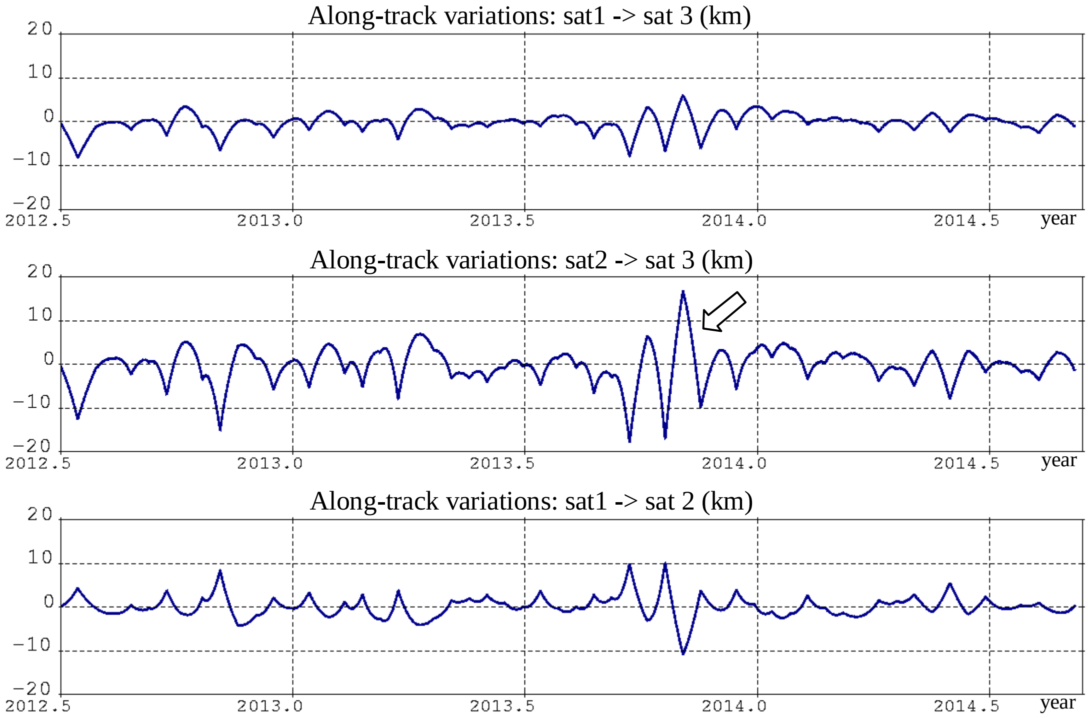
Satellite number	Nominal relative argument of latitude (deg)	Initial RAAN (deg)	Drag coefficient -
1	0 (reference)	+1	1 (nominal value)
2	-1.2	+2.5	1.01 (+1%)
3	-0.8	0 (reference)	0.99 (-1%)

NB: The values as chosen for the drag coefficients tend to increase the atmospheric drag differential effect.

5.3 Simulation results

The main simulation results are presented in Figure 13. They represent the variations of along-track distance between any 2 satellites: 1-2, 2-3, 1-3.

Figure 13: Maintenance simulation – variations of along-track distance



Variations of along-track inter-satellite distances should be less than 15 km, and indeed they are nearly all the time. The effect of the strong solar activity is clearly visible at the end of year 2013 (as indicated by the arrow). The rapid variations appear to be connected to solar activity prediction, not very accurate in this case.

The evolution of the semi major axes is also shown (Figure 14). As expected, all the semi major axes decrease at the same rate and follow solar activity.

Figure 14: Maintenance simulation – relative variation of semi major axes

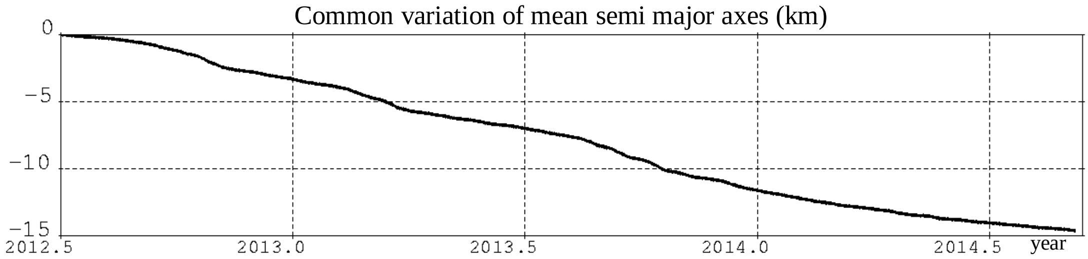
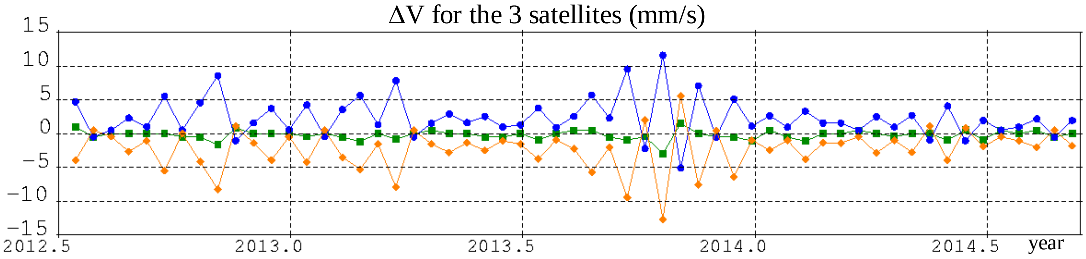


Figure 15 shows the ΔVs for the 3 satellites. The ΔV amplitudes are consistent with the expected relative semi major drift. The saw-tooth shaped ups and downs at the end of 2014 are another consequence of the rapid variations of solar activity at this period of time.

Figure 15: Maintenance simulation – maneuvers



6. CONCLUSION

The control strategy that has been designed and presented in this paper is satisfying and well suited to our maintenance problem. First of all, the accuracy requirements concerning the separations between the satellites are met, which is of course what matters most. In addition, the simulation results enabled us to validate the studies made during the mission analysis phase, whereas these studies made no particular reference as to how the maintenance strategy should be implemented.

A few assets of the maintenance strategy can be listed. Firstly it is simple enough, which is of interest when considering the risks for potential software anomalies. One second point is that it is not too much dependent on the ground segment orbit determination and prediction accuracy, as it nominally makes use of past data only. This gives us add-on guarantee that the simulation results that have been obtained are representative of the final performance. Finally, it has been shown how the algorithms could be implemented, should unexpected contingency situations occur.

Although satisfactory enough, a few improvements could probably be conceivable.

One aspect that could certainly be improved is the prediction method, although in practice tuning options should enable us to optimize the algorithm's behavior as required (e.g. extension of data arcs, data weighting).

One second aspect that could potentially be improved is related to the ΔV computation algorithm. Some limited variations have been experimented, but without significant visible change on the performance. A more global (thereby less empirical) approach to the definition of the maintenance strategy could probably have been envisaged. Yet the benefits that could have been obtained are questionable when considering that the uncertainties seem to mainly originate in the solar activity prediction errors.

References

1. Jean-Pierre Carrou et al., *SpaceFlight Dynamics - part I (1995 edition)*, CNES, Cépadues Editions
2. Douglas Biesecker and the Solar Cycle 24 Panel, *The Solar Cycle 24 Consensus Prediction (slides), http://www.sec.noaa.gov/SolarCycle/SC24/index.html*