# SIMBOL-X: A FORMATION FLYING MISSION ON HEO FOR EXPLORING THE UNIVERSE

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## ABSTRACT

SIMBOL-X is a high energy new generation telescope covering by a single instrument a continuous energy range starting at classical X-rays and extending to hard X-rays, i.e. from 0.5 to 80 keV. It is using in this field a focalizing payload which until now was used for energy below 10 keV only, via the construction of a telescope distributed on two satellites flying in formation. SIMBOL-X permits a gain of two orders of magnitude in sensibility and spatial resolution in comparison to state of the art hard X-rays instruments.

The mirror satellite will be in free flight on a high elliptical orbit and will target the object to observe very precisely, thus focusing the hard X-ray emission thanks to this mirror module.

At the focal point area which is situated 20 meters behind the mirror satellite, the detector satellite maintains its position on a forced orbit thanks to a radio link with the mirror satellite and a lateral displacement sensor using a beam emitted onboard the mirror satellite. This configuration is said "formation flying".

The location of the detector satellite shall be very finely tuned as it carries the focal plane of this distributed telescope.

To provide science measurements, the Simbol-X orbit has been chosen High elliptic (HEO), which means elliptical orbit with a high perigee altitude. Preliminary studies where made with an orbit with an altitude of the perigee of 44000km and altitude of the apogee of 253000km. The orbit was seven days ground track repeated in order to maintain a perigee pass over the Malindi ground station to download scientific telemetry. But as studies went on, difficulties in mass budget, link budget, perigee maintenance and formation flying maintenance were raised. This was mainly due to the vicinity of the Moon and its disturbing effect on the satellites' orbits. Alternative orbits have been proposed in order to demonstrate the feasibility of the mission.

The problematic of bringing the two satellites from their injection orbit to their operational orbit 20 m apart from each other and then maintain this configuration is very challenging. It requires theoretical development of the relative motion between two satellites in high eccentric orbit with large differential disturbance on the two bodies.

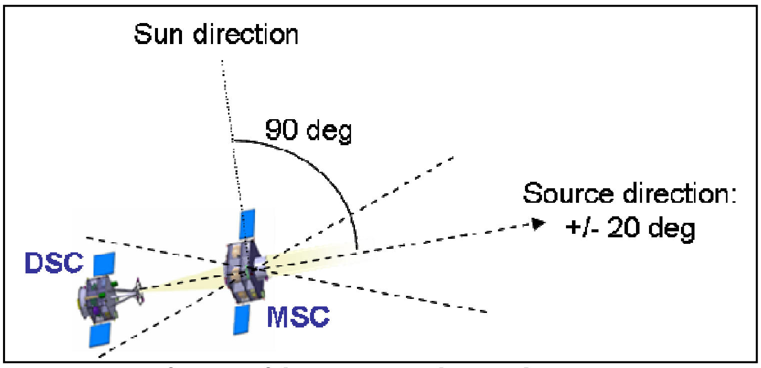
This paper will present the mission analysis for the Simbol-X satellites with the complex problematic of doing formation flying in high elliptic orbit.

## ACRONYMNS AND NOTATIONS

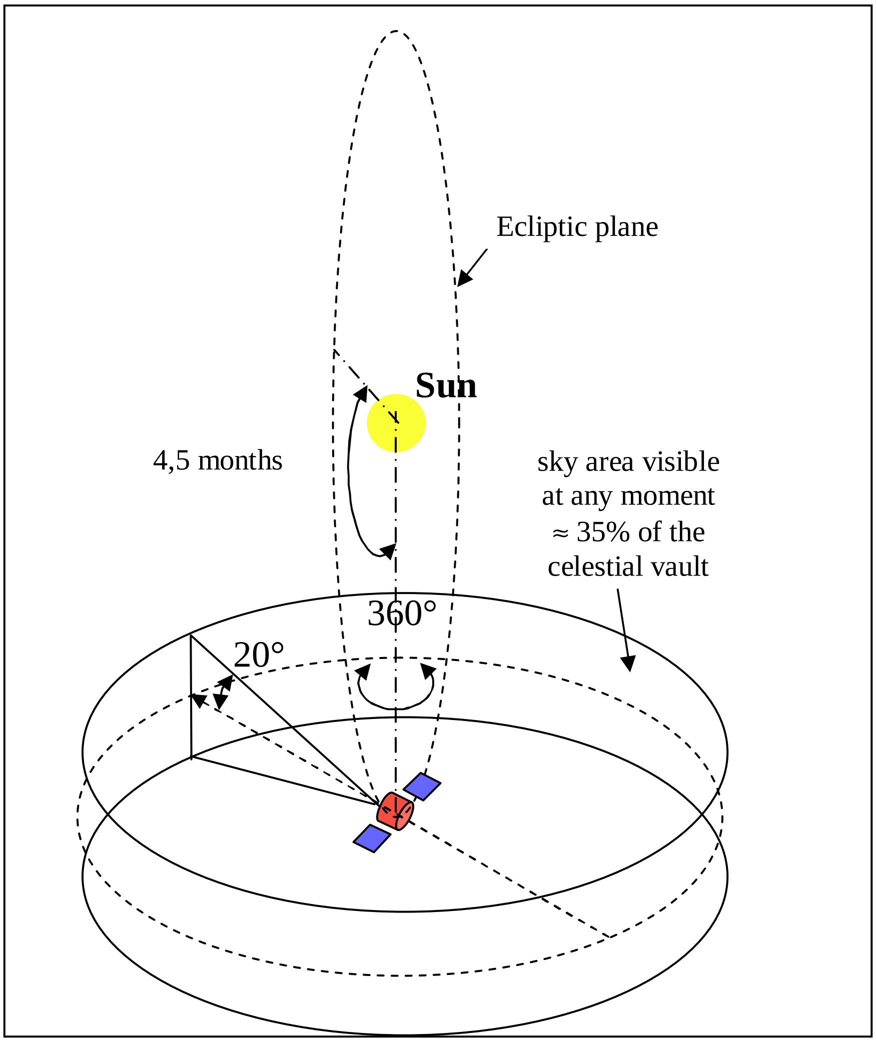
| | |
|---|---|
| CAM | Collision Avoidance Manoeuvre |
| DSC | Detector Satellite |
| HEO | High Elliptic Orbit |
| IAR | Integer Ambiguity Resolution |
| ISD | Inter Satellite Distance |
| ISL | Inter Satellite Link |
| LOS | Line Of Sight |
| MSC | Mirror Satellite |
| RAAN | Right Ascension of Ascending Node |
| $\Delta V$ | Velocity increment |
| $\omega$ | Argument of perigee |

# 1 MISSION OVERVIEW

The Mirror satellite, carrying the mirror of the telescope, flies on its natural orbit. The Detector satellite, carrying the focal plane assembly, controls its position with respect to the Mirror satellite. The Line Of Sight of the telescope is the line joining the center of the detector to the center of the mirror. It corresponds to the observation direction, which shall be within a cone of semi-aperture of 20 degrees that can rotate in a plane perpendicular to the satellites/Sun axis.




*Definition of the LOS w.r.t. the Sun direction*




*Evolution over the year of the SIMBOL-X observable zone*

This pointing allows 1000 targets during the 2.5 years of nominal mission, and 500 additional targets during mission extension (2 more years). The pointing is similar to the one of XMM and Integral.

# 2 CHOICE OF THE OPERATIONAL ORBIT

## 2.1 CONSTRAINTS

A major constraint is that science can only be made at an altitude higher than 73000 km. It is the estimated minimum value for the altitude in order not to be disturbed by the Van Allen outer radiation belt. The availability for science is a function of the altitude of the perigee and apogee of the orbit.

Then, the hydrazine budget to maintain the formation grows significantly when the altitude of the satellites is less than 15000 km. One solution would be to break the formation at each orbit when approaching the perigee, but this solution was rejected, because of the complexity of the organisation of the operations associated, and of the collision avoidance issues. Thus, the perigee altitude should be above 15000 km: the simulations show that the 1-year budget for formation flying is 2.5 m/s for a 20000 km perigee altitude, 3 m/s for a 15000 km perigee altitude, and more thant 4 m/s for a 10000 km perigee altitude. 1 m/s represents more or less 1.5 kg of cold gas.

Another requirement is that the orbit should be phased with the Malindi ground station (Longitude 40.19°W, Latitude 2.99°S) for TM/TC convenience. The studies made show that the TM/TC coverage is satisfying if the difference between the perigee longitude and the longitude of the ground station does not exceed 30 degrees, even if the perigee latitude grows up to 40 degrees. We will see later the impact of this constraint on the station-keeping strategy.

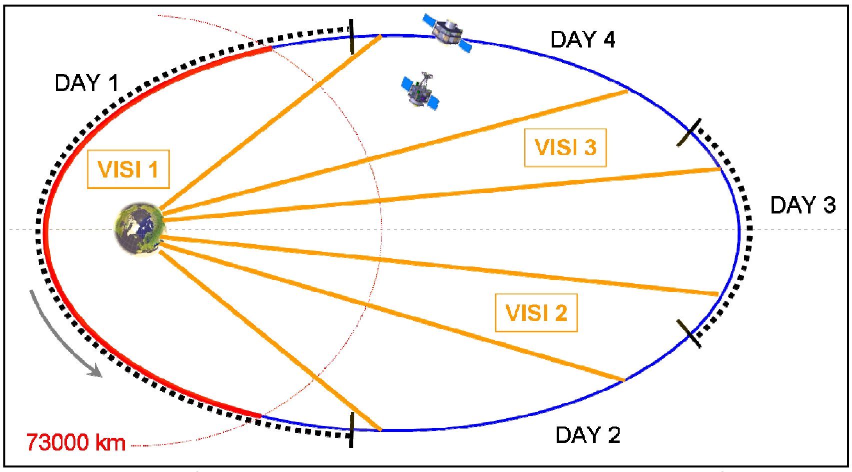
Then, the perigee and apogee have obviously maximum values depending mainly on the following characteristics: the launcher capacity in terms of mass is of course limited. The SOYUZ capacity for the chosen transfer orbit is around 2300 kg. Thus, the perigee altitude is to be chosen between 15000 km and 20000 km. There is also a limitation on the platforms total ΔV capacity for LEOP manoeuvres, station-keeping and end-of-life manoeuvres. For a perigee altitude raising at 15000 km, the ΔV LEOP budget is between 275 and 302 m/s. For 20000 km, the ΔV LEOP budget is between 347 and 380 m/s.

And finally, the link budget is significantly improved by a decrease of the perigee altitude.

## 2.2 ORBITAL PARAMETERS

To ensure the maximum availability for science, the orbit is chosen high elliptic (HEO). Then, in order to maximise the availability for science (the objective was 90%), the orbit initially chosen had a repeat cycle of 7 sideral days, but this kind of orbits are not compatible with link and mass budget. This is the reason why the altitude of the perigee has been decreased compared to the initial values given in the abstract, and the repeat cycle is now 4 sideral days.

The main advantages of the 4-day orbit are: The maximum perigee altitude, which will be reached after 5 years of free evolution of the orbit, is close to 36000 km, whereas it is close to 70000 km for a 7-day orbit. An altitude of 36000 km is compatible with the link budget. Then, for a 4-day orbit, the minimum perigee altitude is 15000 km, which is compatible with the cold gas budget for formation flying. Another advantage is that the perigee pass over Malindi is very stable –with the hypothesis taken in the previous paragraph- through the lifetime of the satellites: only a few correction manoeuvres during the mission are necessary. On each orbit, there are 3 passes over the ground station: the first one, around the perigee pass, lasts about 29 hours for a 10 degrees minimum elevation.



*Representation of the 4-day orbit with the 73000 km altitude limit for science and the visibilities from a given ground station*

Indeed, the altitude of the satellites is crossing the geostationnary altitude, so the satellites catches up the Earth rotation, and a visibility duration lasting more than a day is obtained. There are also two other passes lasting both about 11 hours, just before and after the apogee pass. During the operational phase, and out of contingency cases, communications are established once at each Malindi pass, that is to say 3 times per orbit, for 2.5 hours on average (4.5 hours on average for the perigee pass). Ranging and correlation between onboard time and universal time are performed during each communication session.

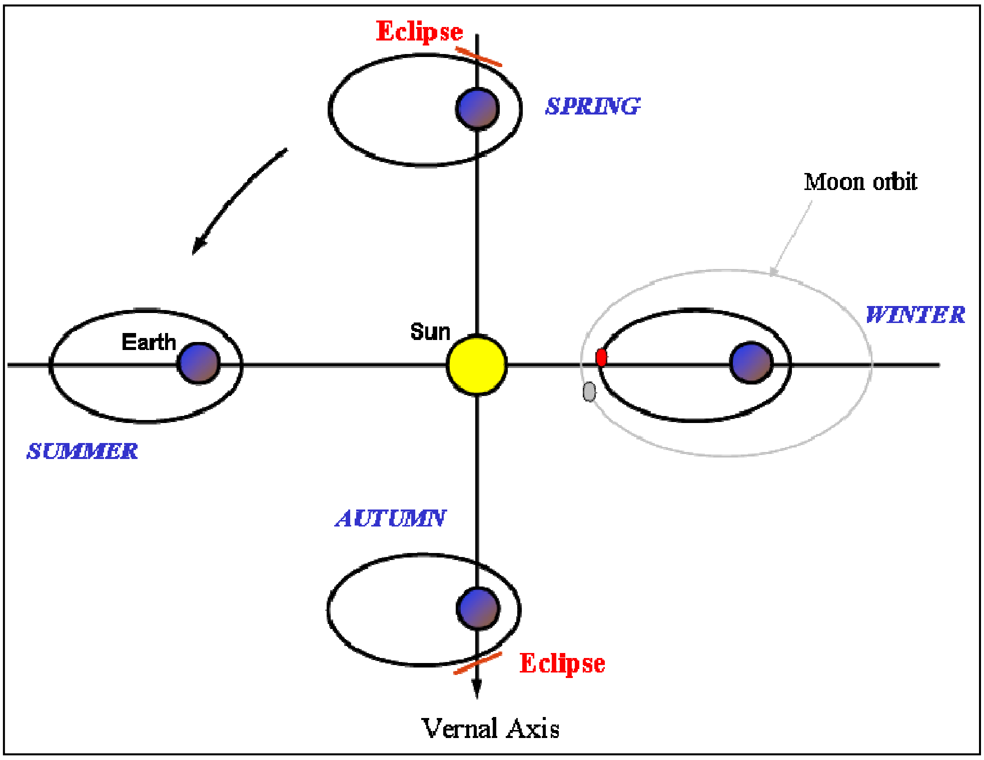
Another characteristics of this orbit is that we have of course a much lower hydrazine cost for station acquisition and maintenance, and a gain of performance of the launcher since the target orbit is of lower energy. The principal drawback of the 4-day orbit for SIMBOL-X mission is that the cycling ratio, i.e. the time spent at an altitude higher than 73000km, which is directly linked to the availability for science, is 3.3 days per orbit, which represents 83%, instead of 90% with the initial 7-day orbit.

This reduction of the availability for science has been accepted by the scientific community participating in the SIMBOL-X project. So the chosen orbit has an initial perigee altitude of 20000 km, and an initial apogee altitude of 179738 km, in order to obtain a 4-sideral day period. Thus, the semi-major axis is 106247 km, and the initial eccentricity is 0.7517281. The initial inclination is induced by the inclination of the launch pad: between 5.2 and 7.2 degrees.

## 2.3 ORBIT ORIENTATION

For HEO orbits, the orientation of the apsidal line remains nearly constant, even if the argument of perigee and the right ascension of the ascending node are not constant. Indeed, the main perturbation on the apsidal line orientation is the J2 perturbation but for HEO, the regression of the line of nodes is very small.

The choice of the orbit orientation will actually have an influence on the eclipse periods and on the evolution of the perigee and apogee altitudes. We can choose the appropriate orientation to take advantage of the combined effect of the Sun, Moon and Earth attraction. With the apogee toward the sun, the perturbations will lead to an increase of the perigee altitude and a decrease of the apogee altitude. Any other orientation of the apsidal line may lead to a decrease of the altitude of the perigee, which could be dramatic when the satellites are on their transfer orbit. Thus, the orbit at the beginning of the mission is chosen with the line of nodes perpendicular to the Earth-Sun direction at vernal equinox (which means RAAN = 90 deg, constant value all over the year in a Keplerian motion) and the apsidal line is in the same direction, the perigee being the same point than the ascending node (which means argument of perigee = 0 deg, constant value all over the year in a Keplerian motion).




*Impact of the orbit orientation (RAAN=90 deg; ω=0 deg; low inclination) on the eclipse periods*

## 2.4 INFLUENCE OF THE MOON

The Moon orbit is located in a plane with an inclination of 5 degrees w.r.t. the ecliptic. The Moon is rotating around the Earth at the mean altitude of 384 400 kilometres. Its eccentricity is 0.0549. The apogee altitude is 405 503 km and the perigee altitude is 363 296 km. The orbital period of the Moon is 27.32 days.

The Moon effect is more important at the apogee of the orbit, where the satellite is at the closest point to the Moon. The constraint is to maintain the perigee altitude, especially on the transfer orbit, because the perigee alttitude is relatively low (300 km), which means maintaining the velocity at the apogee. The worst effect on the altitude of the perigee is when the velocity at the apogee decreases. The problem of the minimisation of the Moon effect is not easy to solve: an analytical model is not accurate, and the optimisation has to be made with a numerical model. Nevertheless, the influence of the Moon on the maintenance budget is negligible compared to the impact on the station acquisition. However, a good conjunction for the station acquisition budget means a good conjunction for the station keeping budget.

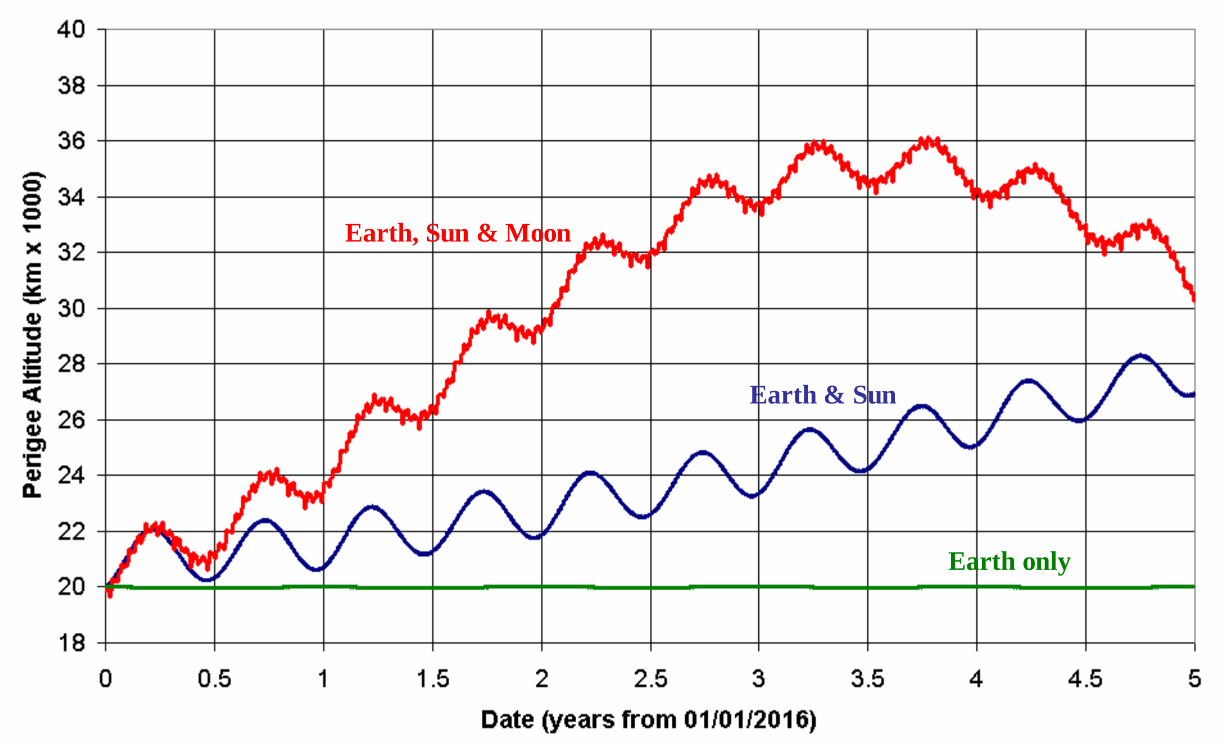
## 2.5 LONG TERM EVOLUTION OF THE ORBIT

The main perturbations on the orbit are the attraction of the Earth, the Moon, the Sun and the solar radiation pressure. The other perturbations can be neglected.

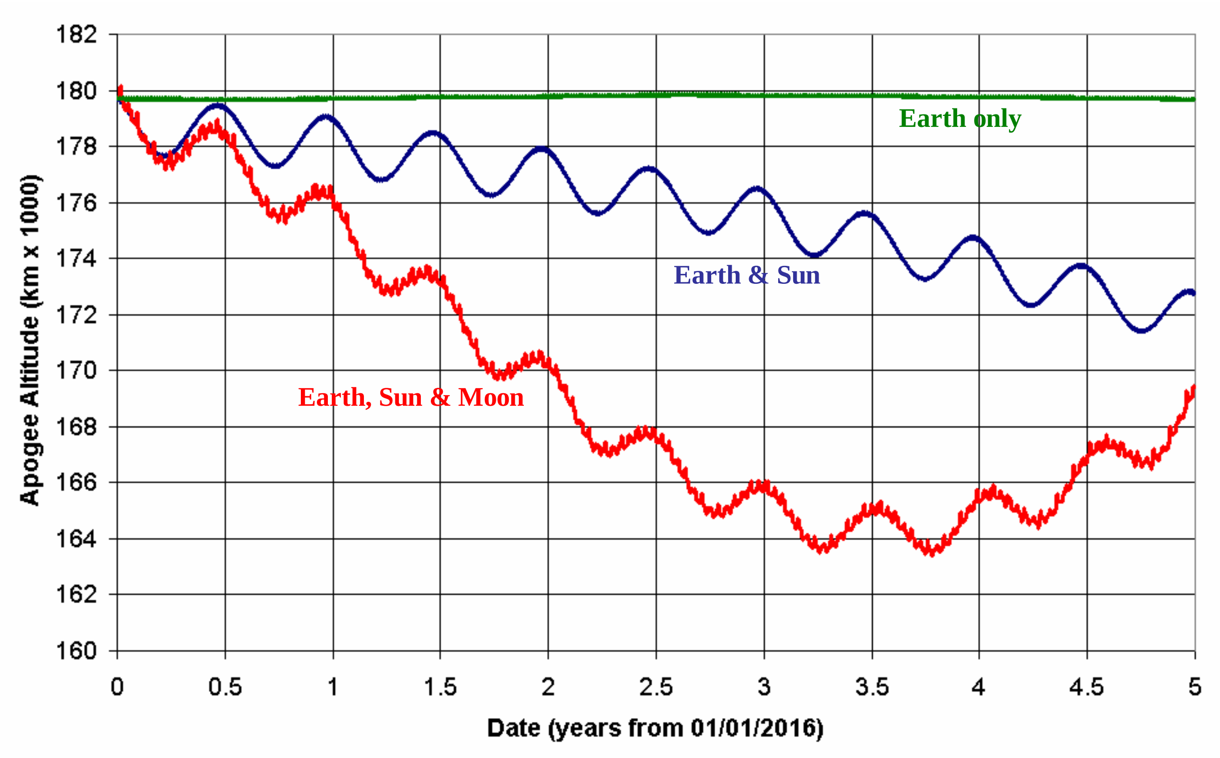
### 2.5.1 Natural evolution of the orbit parameters

The nominal mission is set to 2.5 years, and the additional program has a duration of another 2 years, so the evolution of the orbital parameters is studied for a duration of 5 years. Since no constraint on these parameters is given by the mission, their natural evolution during this extended lifetime is studied. During this period of time, the inclination and the other angular parameters of the orbit will have a high variation. In particular, the inclination will increase up to 40 degrees.

During the first 3 years of the mission, the perigee will increase, and the apogee will decrease. The sinusoidal effect of six month period is due to the Sun. A14-day periodic effect due to the Moon can also be observed. For a five-year mission, the perigee altitude will increase up to 35000 km and the apogee altitude will decrease down to 164000 km, but the semi-major axis remains nearly constant.




*Influence of the gravitational potential of the Sun and Moon*
*on the long-term natural evolution of the perigee altitude*




*Influence of the gravitational potential of the Sun and Moon on the long-term natural evolution of the apogee altitude*

The previous plots show the evolution of the apogee and perigee altitudes, assuming the operational life begins in 2016. However, for any other date for the beginning of the operational life, we notice the same long term evolution for all the orbital parameters and with nearly the same magnitude. Only short period terms differ because there are dependent of the launch date, since they are linked to the position of the Moon at the time of the launch.

Since we are dealing with 2 satellites -the MSC and the DSC-, there might be some differences in the long-term free (i.e. without manoeuvre) evolution of their orbital parameters. The main perturbing forces are the attraction of the Sun and the Moon. These forces are derived from a potential, so they are independent of the characteristics of the satellites. The other perturbing force that must be taken into account is the solar radiation pressure. If we consider that the two satellites have the same ratio for Surface / Mass, the evolution is the same for the MSC and for the DSC. If this ratio is different, the evolution will be slightly different as we will see later, but the global pattern of the orbital parameters evolution will be the same for the two satellites.

We have considered here the free evolution of the satellites, but we have to keep in mind that station keeping will be necessary to fulfil the requirement of downloading data at the perigee over Malindi ground station. This will be developped in chapter 5.

### 2.5.2 Sun Eclipses duration

The orbit at the beginning of the mission is chosen with the line of nodes perpendicular to the Earth-Sun direction at vernal equinox, and the apsidal line is in the same direction. As a consequence , the eclipses take place in the station-keeping part of the orbit (i.e. close to the perigee), so they do not perturbate the mission. There are two periods of Sun eclipse per years: one in spring and one in winter. The maximum duration of the total eclipse is 190 min, but penumbra can have duration of 270 min. The natural evolution of the RAAN will increase the duration of the eclipses during the mission.

# 3 LAUNCH

## 3.1 LAUNCH CHARACTERISTICS

The satellites are launched stacked by Soyuz from the Kourou launch pad in French Guyana. The launch pad is located at 52.8° W, 5.2°N. The satellites are then separated from each other when injected on a transfer orbit. The ground manages the operations to have them reach their final scientific orbit thanks to their own propulsion subsystem, as well as the closing manoeuvres to bring them at the nominal distance from each other. Assuming that no inclination correction will be performed, the launcher performance for a 300km x 180000km separation orbit is about 2330 kg, which fits the Simbol-X satellites mass budget.

## 3.2 LAUNCH WINDOW CONSTRAINTS

For a launch onto a Highly Eccentric Orbit, the injection orbit must be such that its perigee does not re-entry into the Earth's atmosphere during the first orbits. Such dipping motion of the perigee height is due to luni-solar (Moon and Sun) perturbations and depends on the orientation of the orbit at launch time and on the Moon position. The following parameters are fundamental for the evolution of the orbit: the right ascension of the ascending node, the position of the Moon on its orbit and the launch day in the year for the initial effect of the Sun. The argument of the perigee will also have an impact. The right ascension of the ascending node and the argument of the perigee have been chosen previously, taking system constraints into account.

For platform constraint, the first apogees after separation must not be in eclipse. For a launch in spring or autumn, the eclipses are located just before or after the perigee of the orbit, so this constraint has no impact on the launch window. The inertial orientation of the orbit is achieved by choosing the launch time in the day, depending on the launch pad and on the launch duration. These parameters are not yet determined. If the Moon is in the vicinity of the transfer orbit or of the apogee of an intermediate orbit, it can disturb the transfer orbit or change the intermediate orbit. Variations in the perigee of an intermediate orbit can have serious impacts: a decrease could cause an atmospheric re-entry: this corresponds to a 2- to 3-day period every month. As the Moon effect is important on the perigee it is preferable to launch in a period such that the Moon effect will increase the perigee and during which there is no risk of Sun eclipse by the Moon.

# 4 ORBIT RAISING STRATEGY

## 4.1 LEOP SCENARIO

The satellites are separated one from the other at the time of injection on the transfer orbit. The on-orbit position of both satellites is the same at the instant of separation, but the semi-major axes are slightly different because of the small $\Delta V$ (order of magnitude: 1 m/s) produced by the launcher between the 2 satellites at separation. A phasing strategy has been studied (cf 4.2), and also a coordinated strategy (nominal strategy, cf 4.3).

## 4.2 PHASING STRATEGY

The aim of the study is to determine the total $\Delta V$ for SIMBOL-X orbit raising including the manoeuvres necessary to phase the two vehicles. Preliminary results are given as examples for a given launch date. The simplified phasing strategy proposed does not depend on this date but a new run of the corresponding software has simply to be performed for each different launch date.

As we will see below, the phasing conditions used in this work are not SIMBOL-X ones. In fact, we use here the same conditions as in a Lambert's problem (equality of the orbital parameters or equivalently, equality of the positions and velocities) instead of SIMBOL-X conditions related to the distance between the satellites.

We assume here that the manoeuvres are impulsive and we compute the parameters of these manoeuvres, i.e. the date of application, modulus and direction, in order to reach the desired final values of the perigee and apogee altitudes for the two vehicles, taking into account perturbing forces: the terrestrial potential, the influence of the Moon and the Sun, the Solar pressure, the drag force. No correction of the inclination is performed. Another objective of the study is to meet simplified phasing conditions: the orbital parameters of the two vehicles must be the same after the last manoeuvre, taking into account again perturbing forces in the computation.

In order to fulfill the above conditions, the following strategy is used:

In a first phase, the DSC is put on an intermediate orbit with the same apogee altitude than the initial transfer orbit but with a perigee altitude equal to 2000 km. This is for avoiding re-entry in the atmosphere before the end of the phasing. So, we have to determine the amplitude, direction and date of the manoeuvre ΔV1 in order to reach the desired orbital parameters (perigee and apogee altitudes) at a given target date, while minimizing the amplitude of the manoeuvre ΔV1. Note that the solution values depend on this target date because final parameters are osculating ones and because perturbing forces are taken into account.

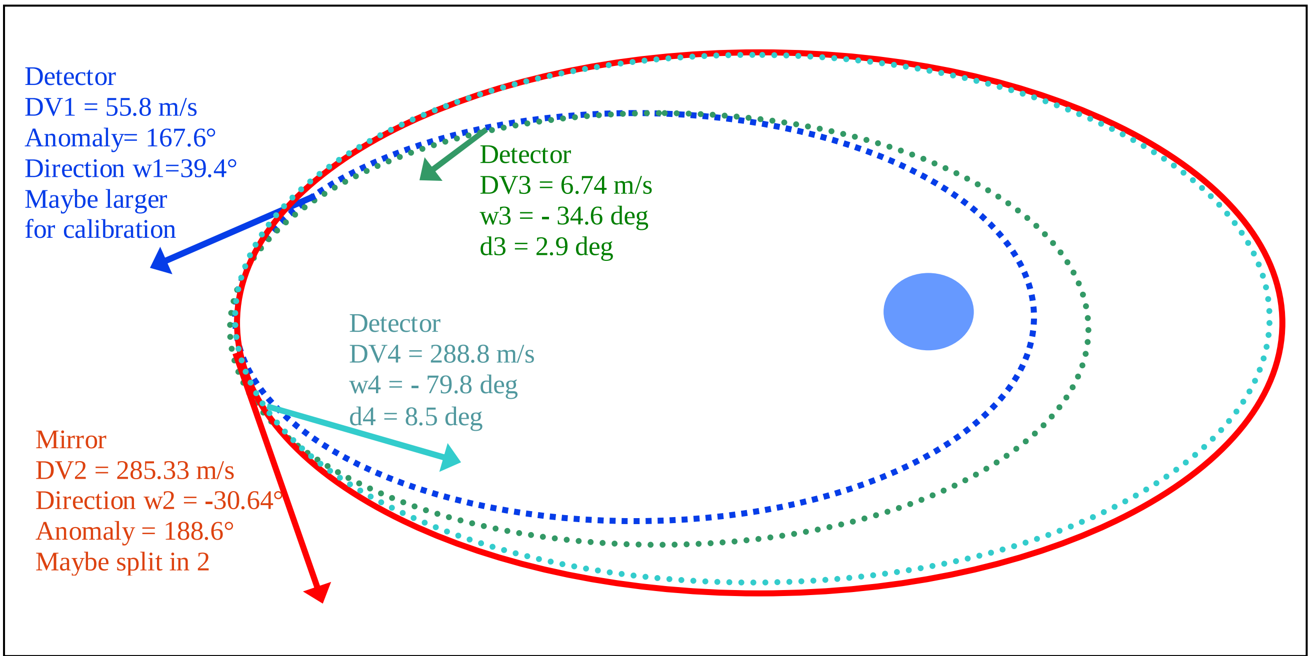
In a second phase, the MSC is put on the final orbit using a near apogee manoeuvre as in the first phase. Then, we have two mathematical programming problems to solve for phases 1 and 2, each one with three unknowns and two constraints. The cost function is simply ΔV1 for the first phase (respectively ΔV2 for the second one) and the two constraints force each satellite to reach the desired final perigee and apogee altitudes at the corresponding given target date. These problems are solved by means of mathematical programming software called NLPQL1, the tool used for extrapolation being here PSIMU. Let us notice that we have to provide NLPQL1 with the partial derivatives of the constraints with respect to the unknown parameters. Due to the numerical computation of the constraints by means of PSIMU, a finite difference scheme is used for computing these partial derivatives.

As a third phase, rhe DSC performs two phasing manoeuvres to satisfy the simplified phasing conditions. To determine the parameters of these two manoeuvres, we begin by computing the solutions of a succession of Lambert’s problems.

Let us compute the Keplerian synodic period of these two orbits, denoted by Ts:

$$T_S = \frac{T_i T_f}{T_f - T_i}$$

Then, for each date on a time interval between a grid between ΔV2 and Δt2 + Ts, and for each duration of the phasing less or equal to the period of the intermediate orbit of the DSC, we compute the solution of a Lambert’s problem with less than one revolution. Thus, for each date and each duration, we obtain the parameters of the two Keplerian phasing manoeuvres. Then, we simply choose the initial date and the duration of the phasing that make the sum of norms for the two ΔVs minimum. These values, together with the characteristics of the computed Keplerian manoeuvres are used as an initial guess for a NLPQL1 optimization that has to take into account the perturbing forces. Thus, NLPQL1 has to solve a mathematical programming problem with eight unknowns and six constraints using again a finite difference scheme for computing the partial derivatives of the constraints with respect to the unknowns.




*Preliminary strategy obtained for t0 = 01/01/2012*

The main problem of this phasing strategy is that the satellites are most of the time on orbits with different periods. The consequence is that the orbital planes have different evolutions (nodal drift and drift in inclination), and, depending on the date considered, an important correction of the out-of-plane parameters could be necessary, which could be expensive in terms of hydrazine consumption. Making-up for this angular difference will also increase the station acquisition duration.

For these reasons, a coordinated strategy has been chosen as the nominal one.

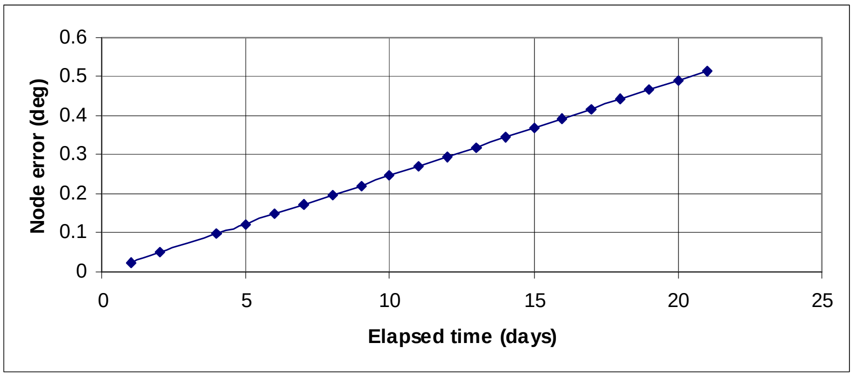
## 4.3 COORDINATED STRATEGY

### 4.3.1 Characteristics of the nominal strategy

The main advantages of the coordinated strategy are that the nominal duration of the LEOP is reduced, together with the hydrazine consumption, and the drift of the orbital planes remains very limited. The main drawbacks of such a strategy are that the anticollision has to be managed carefully, and that back-up strategies have to be studied: for each manoeuvre, there is a back-up phasing strategy to prepare, with a specific set of initial conditions. The nominal strategy is to increase the perigee altitudes of both satellites using almost simultaneous manoeuvres (4 to 5 manoeuvres for each satellite, for a total LEOP duration of 20 days). The manoeuvre execution is programmed during the pass that takes place just before the apogee. The next pass, which takes place the day after the apogee, is used as a back-up window in case of non execution of the nominal manoeuvre on one of the satellites.

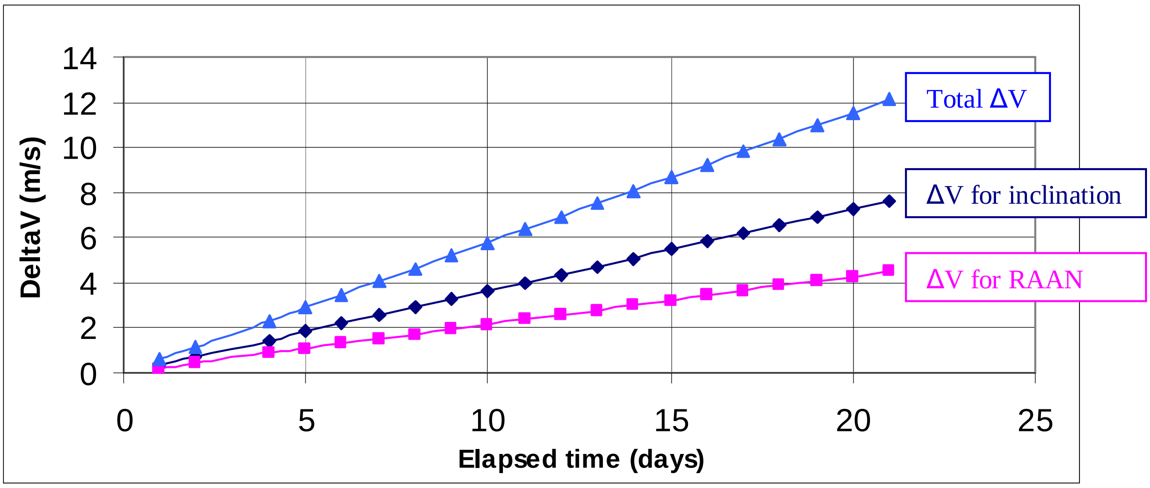
### 4.3.2 Back-up strategies

In case of failure on one of the satellites, i.e. non execution of the manoeuvre at the scheduled orbit, neither during the pass just before the apogee pass, nor during the one following the apogee pass, a back-up strategy is used. It consists of a phasing rendez-vous: the healthy satellite is quickly brought on the operational orbit, and a rendez-vous is performed with the other one after isolation and recovery of the anomaly, using the healthy satellite as a target. This strategy has an over-cost in terms of duration of the LEOP. The worst case is close to 45 days, to be confirmed by coming studies. There is also an associated over-cost in terms of hydrazine consumption, related to the orbital plane relative drift, which depends on the date. Indeed, in case of anomaly during station acquisition, the satellite which is on a lower altitude moves faster (in terms of mean anomaly) and then it becomes necessary to wait until the slowest satellite has made a complete turn and reached again a position just behind the fastest satellite to raise its semi-major axis and stop this drift. This difference in semi-major axis will create a node error function of the drift duration that must be corrected. That implies using more hydrazine that in the nominal case.




*Node error between the 2 satellites as a function of the drift duration*

The worst case is when a satellite remains on the transfer orbit and the other has reached the operational orbit. Then the semi-major axis and inclination difference between the two satellites will create a nodal drift. Moreover, as the perturbations are different between the injection orbit and the final orbit, a drift in inclination will also be created. Assuming that the inclination in the HEO has a secular drift of about 9.3 degrees in 200 days whereas the inclination of the satellite in the transfer orbit is nearly constant we will have a drift in inclination during this period and a nodal drift. Then, an additional manoeuvre to correct inclination and node will be necessary. To correct those two parameters, two manoeuvres will be necessary. This strategy is costless than a coupled manoeuvre at the orbital position of 45 degrees. The inclination should be corrected at the apogee, which on SIMBOL-X orbit corresponds to the descending node. The node error should be corrected at the orbital position of 90 degrees , i.e. 1 day before or after. To limit the drift, the two manoeuvres should be performed as quickly as possible: the orbit determination and manoeuvre telecommand should be calculated and sent within 1 day around the apogee.

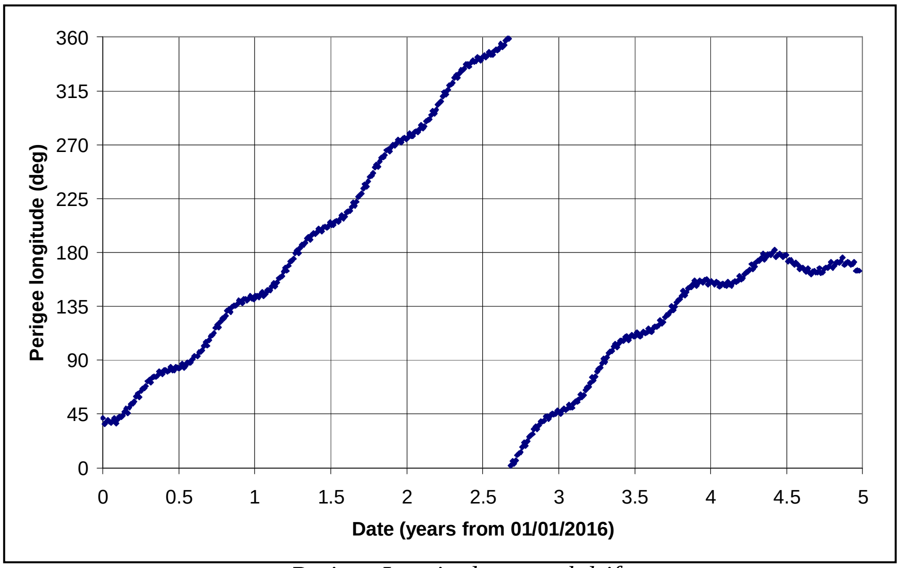



*Manoeuvre amplitude necessary to correct the drift*

For instance, if a satellite has reached the HEO and the other waits 3 weeks before manoeuvring , two additional manoeuvres (than the apogee manoeuvres) will then be needed to correct the inclination and Ω with an over consumption of **13 m/s** more than in the nominal case.

## 5 STATION KEEPING

The only requirement is to maintain the orbit phased with the MALINDI ground station. If the orbit is not maintained, the large variation of the eccentricity and of the inclination creates a drift in longitude of the perigee of the orbit. The variation on the RAAN is nearly compensated by the variations of the argument of perigee. The drift in longitude of the perigee is shown below, for a launch in 01/01/2016. Since it depends on the Moon position, the drift would be different for a different launch date.




*Perigee Longitude natural drift*

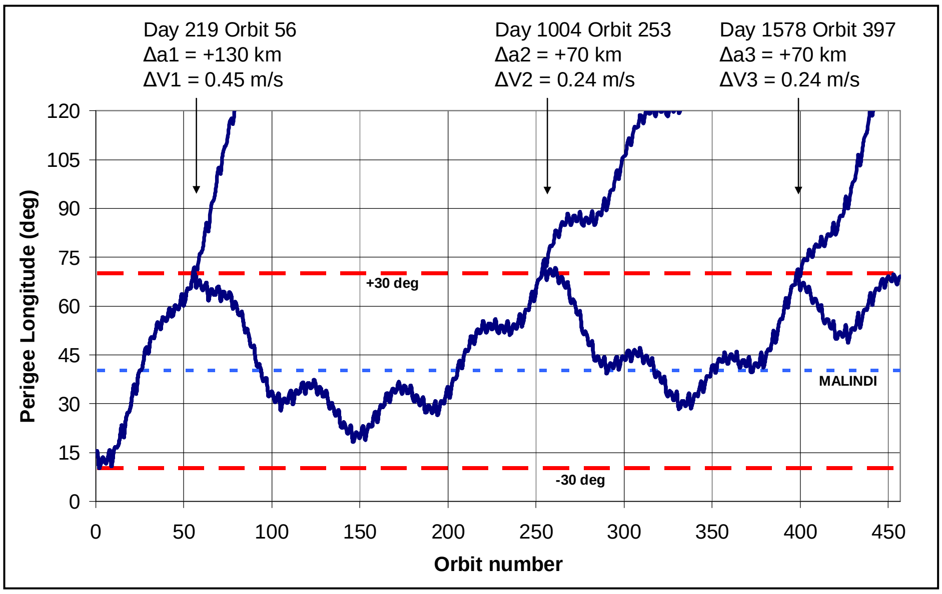
Thus, the perigee should be maintained to compensate for the mid-term perturbations. Otherwise the longitude of the perigee will drift and will no longer be phased with the ground station.

Long periods of time with no manoeuvre can be obtained by choosing the right semi-major axis at the beginning of the mission. This is obtained by a bias on the final semi-major axis (w.r.t. to the theoretical one). Then the change in orbital period will compensate the Moon influence.

The effect of the perigee maintenance manoeuvre will produce a drift in longitude of the satellite. The frequency of the manoeuvres depends on the margin allocated around the ground station longitude. But the ΔV budget for maintenance is independent from this margin and only depends on the orbit. A strategy that consists in doing a semi-major axis raise at the perigee can be applied: at the perigee, the velocity is : $V_{PER} = 5145$ m/s and at the apogee: $V_{APO} = 729$ m/s. With a manoeuvre in the velocity direction we obtain (from Gauss equation): $\Delta a = \frac{2}{n\sqrt{(1-e^2)}}\left[(1+e\cos v)\Delta V\right]$ with the mean motion n = 1.8230E-5 rad/s

Due to the high velocity at the perigee we can correct a large magnitude of Semi-major axis during the mission without much consumption. With a few manoeuvres during the mission we can achieve the maintenance of the perigee passage to -/+30 degrees around the Malindi ground station. Depending on the launch period and on the initial position of the Moon, the number of manoeuvres will be different: a bad conjunction will lead to 6 to 9 manoeuvres for a 3 year mission (one every 6.5 months). Then a total $\Delta a$ = 800 km will be necessary ($\Delta V$ = 2.75 m/s). The number of manoeuvres may be less important depending on the epoch (and then of the conjunction between the Moon and the satellite position). Moreover, by choosing the appropriate semi-major axis and perigee longitude at the beginning of the mission, in most cases, only 3 to 4 manoeuvres will be necessary.

For example, for a mission beginning on the 01/01/2016, we obtain the following pattern, only 3 manoeuvres are necessary to maintain the perigee longitude within the specifications:




*Strategy for perigee longitude maintenance (from 01/01/2016)*

# 6 FORMATION FLYING

## 6.1 CHARACTERISTICS

### 6.1.1 Formation acquisition steps

Satellite modes are being defined to describe the different steps of the formation acquisition, in the beginning of life, or in case of formation flying interruption. In the **Free Flying** mode, the satellites are controlled independently, the Inter Satellite Distance is 30 km or more, and there is no inter-satellite link. In the **Secured Free Flying** mode, the ISD is monitored through the radio frequency link. A Collision Avoidance Manoeuvre is automatically calculated onboard if necessary. Attitude manoeuvres are also performed for Integer Ambiguity resolution. During these first phases, rendez-vous manœuvres are computed on ground, in order to reduce the ISD down to 500 meters. The manoeuvres are performed close to the apogee, using classical orbit determination information coupled with RF metrology information, which precision is about 1 meter. In the **Formation Acquisition** mode, the fine RF metrology (1 degree, 1 cm) is available after IAR. The formation is automatically controlled by the onboard Guidance, Navigation and Control subsystem, using the hydrazine thrusters as actuators. Only the ground-satellite link with the DSC is used: the MSC TM/TC is performed through the Inter Satellite Link. In the **Coarse Formation** mode, the ISD is reduced down to 20 meters. The necessary translation manoeuvres are computed from ground. And finally, the **Fine Formation** mode is the science mode. The formation is controlled via the optic sensor, and the cold gas thrusters. The definition of the phases is in progress. More detailed information on each sub-phase, and the associated manoeuvre strategy should be available soon.

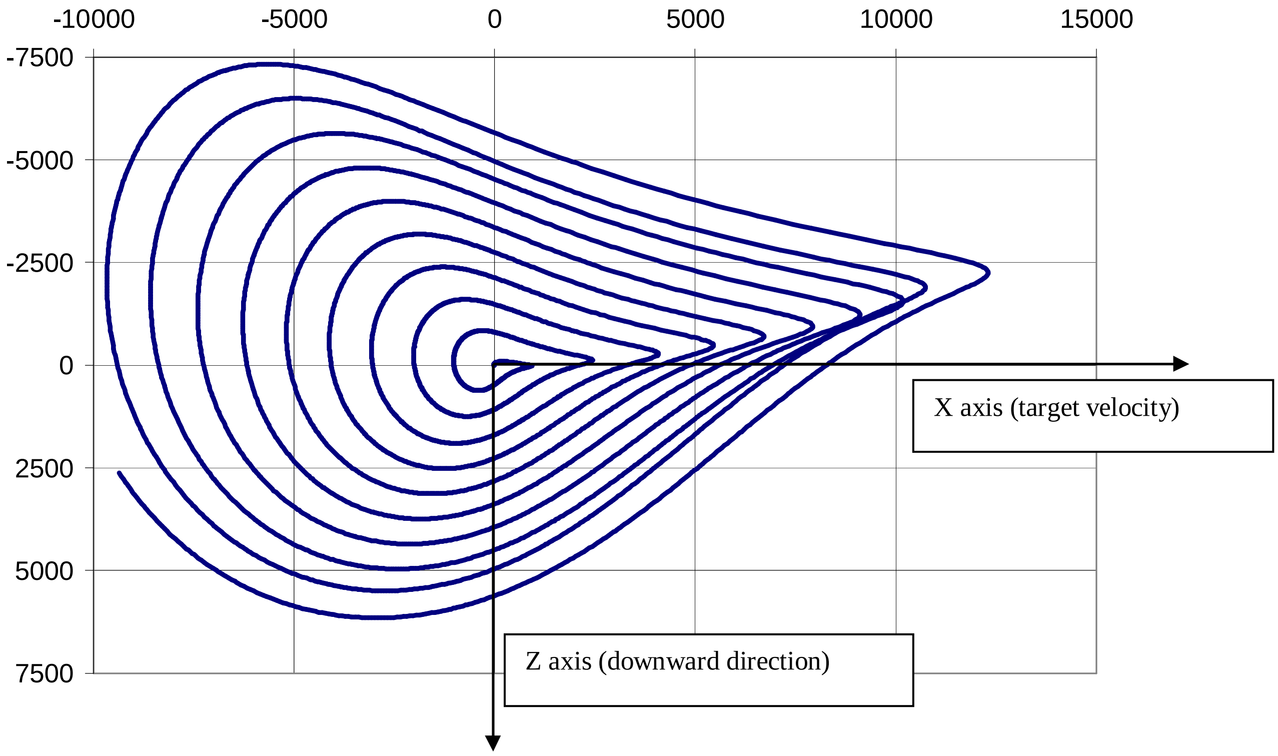
### 6.1.2 Formation flying principles

The MSC is in free flight, whereas the DSC maintains its position on a forced orbit relative to the MSC. The relative position control and the attitude control are performed by cold gas thrusters on the whole orbit. The system is in the Fine Formation Mode.

## 6.2 SIMULATIONS & ANALYTICAL METHODS

We need to analyse the free evolution of the relative orbit of each satellite. A local orbital reference frame tied to the target is used. Since the orbits are eccentric, the classical Clohessy-Wiltshire system of equations cannot be used. Lawden equations can be used. But the main drawback of Lawden's equations is that they do not stand for the effect of perturbations. Indeed, in the SIMBOL-X case, the perturbations due to the differential solar pressure affecting each satellite can not be neglected. A method is being studied, which takes advantage of different representations of relative motions, the Cartesian coordinates and the differences of orbital elements [1], [2], [3]. The results for a particular case - the non-perturbed linear motion- are already available, and the work to take into account the solar pressure is in progress. Assuming that the ratio between the drag coefficients (Cp.S/m) of the satellites is 2.3, the simulations show clearly the impact of the solar pressure on the relative motion of the satellites.

Trajectory control is needed for the DSC to compensate for the perturbing forces. The first force that shall be taken into account is the differential solar pressure, which depends on the angle between the sun and the orbital plan, and on the delta of Surface/Mass ratio between both satellites. This perturbing force increases when the delta of Surface/Mass ratio increases. The other force that we have to deal with is the gravity gradient. It depends on the orientation of the formation (LOS) w.r.t. the orbital plane; on the altitude of the formation. The gravity gradient increases when the altitude of the formation decreases. It also depends on the distance between the satellites. The gravity gradient increases when the ISD increases.




*Example of relative motion of the DSC in the MSC local orbital reference frame (the distances are expressed in meters)*

The inter-satellite direction is driven by scientific requirements: it can be in plane or out-of-plane. Thus, the initial conditions of the study can be very different.The work on the formation flying issues is in progress. We hope to have interesting results to present in the coming months and years.

References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