# DESIGN AND OPERATIONAL IMPLEMENTATION OF SWARM ORBIT INSERTION PHASE AND ROUTINE ORBIT CONTROL

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***Abstract:*** *Swarm is a three spacecraft ESA Earth Explorer mission which is to be launched in the first half of 2013. The spacecraft will be launched together onboard a Rockot-Briz launcher into a common orbit with inclination 87.55 degrees, altitude 490 km and frozen eccentricity. The spacecraft are then manoeuvred into their operational constellation during the first three months after launch. This orbit insertion phase would consist of around 150 manoeuvres per spacecraft. After the orbit insertion the routine operations phase starts. This paper presents the selection of the final operational orbit considering the injection orbit, the orbit insertion phase plan, and the routine phase strategy with some simulations demonstrating its robustness.*

***Keywords:*** *Swarm, orbit control, orbit insertion, routine operations, manoeuvre optimization.*

## 1. Introduction

The Swarm operational constellation consists of a lower pair (Swarm A/B) and a higher spacecraft (Swarm C). The lower pair fly in a side by side formation with their ascending nodes separated by 1.4 degrees in right ascension of ascending node (RAAN) and with an initial altitude of around 460 km. The lower pair should decay to an altitude of 300 km after four years and must be maintained such that their node crossing times are within 10 seconds of each other. The higher spacecraft has a targeted inclination of 88 degrees, 0.6 degrees greater than the lower pair and an altitude of 530 km. The expected life of the mission is 4 years, long enough so that the lower pair and the higher spacecraft have a local time separation of 9 hours.

The satellites use a cold gas propulsion system with pairs of 50 mN orbit control thrusters in the along and cross track directions. This low thrust requires that each spacecraft must perform around 150 manoeuvres of around 20 minutes duration to reach its operational orbit. The manoeuvres simultaneously change the semi major axis and inclination and correct eccentricity dispersions from the launcher injection. The manoeuvres are commanded in advance and inserted in the on-board timeline as batches of up to three days duration, twice per orbit manoeuvres along with associated slews.

The design of the orbit insertion phase has been performed in order to simplify these operations as much as possible. The spacecraft are manoeuvred one at a time with a single batch of manoeuvres being commanded per week. The manoeuvre schedule has been implemented such that manoeuvre batch failures due to spacecraft problems can be corrected for by a replan of the remaining sequence without the need for a rapid response.

The commanding in advance of large numbers of manoeuvres is new for the Flight Dynamics team and this has meant significant software changes in the orbit control and telecommand generation areas. The approach followed has been to re-use some flexible orbit control software already in use for planetary missions.

During the routine phase, orbit control of the lower pair is required to maintain the constellation, with manoeuvre opportunities in regular slots to simplify operations. By selecting the spacecraft to be manoeuvred, the constellation can be maintained whilst decreasing or increasing the rate of altitude loss of the pair in order to help satisfy the requirement that the spacecraft reach 300 km altitude after 4 years. The required delta-v, manoeuvre frequency and rate of orbit decay depend on the difference in spacecraft ballistic coefficient and the atmospheric density experienced. A simulation tool has been developed to test the sensitivity of the orbit control implementation to these aspects and to develop and test the implemented approach.

## 2. Orbit injection

Due to the low thrust of the orbit control thrusters (a pair of 50 mN) it is necessary to perform several manoeuvres to move the spacecraft from its injected orbit to the operational ones. These manoeuvres are optimized and commanded in batches of several days. Each of these batches can move the inclination, the semi major axis, or both. Inclination change pure manoeuvres can be performed purely doing a manoeuvre centred on the ascending or descending node. Semi major axis manoeuvres must perform two manoeuvres per orbit at opposite points in the orbit to avoid eccentricity build up. Combined manoeuvres must also perform two burns per orbit, around the ascending and descending nodes to avoid eccentricity build up. The chosen manoeuvre duration is 20 minutes (which corresponds to about a 7% efficiency loss for the inclination change compared to impulsive manoeuvres) in order to have enough time for the slews, but the exact magnitude, which can be smaller, is optimized for each batch.

### 2.1. Selection of initial orbit

The launcher injection altitude can be considered fixed to an altitude of 490 km, and the inclination is 87.55 degrees. From mission analysis work done by industry it is known that a direct change of the RAAN to achieve the 1.4 degree difference between Swarm A/B would require more than the total available fuel and thus a phase of several weeks with different RAAN precession rates is required. With these constraints it was necessary to determine how to optimally place the three spacecraft into their constellation such that Swarm A/B and Swarm C are separated by 0.6 degrees inclination and Swarm A and B have an initial altitude such that they reach 300 km after 4 years. Since the injection altitude cannot be adjusted, the target orbits of the spacecraft was optimised in order to meet the mission requirements.

The RAAN precession rate depends on the inclination and the altitude. The stronger dependency is on the inclination and thus the RAAN separation is mainly achieved by a temporary inclination difference.

The rate of decay of the orbits is highly unpredictable due to the limited accuracy of solar activity predictions but it has become clear that since the launch was delayed from 2010 (before the solar maximum) until 2013 without adjusting the launcher injection altitude and because of the low solar cycle currently being experienced the satellites will probably take longer than 4 years to get to 300 km.

During the orbit insertion phase the spacecraft are manoeuvred from their injection orbit to their constellation orbits. These constellation orbits must satisfy requirements outlined in the introduction but the orbits are not absolutely fixed.

The higher satellite is not expected to perform manoeuvres in routine operations with the exception of collision avoidance. Therefore the propellant consumption during the orbit insertion phase should be balanced so that the maximum amount of propellant is left on the lower pair.

To increase the decay rate over the mission lifetime the initial orbit of the lower pair should be selected to be as low as possible whilst keeping enough propellant for the full routine phase. This is achieved by performing as much of the inclination separation as possible using the upper spacecraft whilst raising its altitude to the required 530 km. This way the lower pair spend less fuel on inclination change and more on altitude reduction.

The fuel tanks were planned not to be filled to full capacity in order to comply with the maximum launch mass. Additional fuel on board the spacecraft would allow Swarm C a larger share of the inclination split, thereby saving fuel spent on inclination change for Swarm A and B in addition to the extra fuel loaded on board. This saving could be used to perform routine orbit control for an extended mission or help to reduce the altitude to 300 km in 4 years.

Figure 1 shows the trade off of altitudes, changes in inclination and propellant usage for each spacecraft. The left blue shaded area is forbidden because there the inclination difference between Swarm A and B is too small to ensure that the 1.4 degrees of separation in ascending node will occur in the duration of the orbit insertion phase. The green shaded areas on top are also forbidden depending on fuel loaded on board and the maximum delta-v the spacecraft can perform. The examples of 50 m/s, 55 m/s and 60 m/s are shown as illustration.

The figure shows the delta-v cost of the orbit insertion phase for Swarm C (curves labelled 'SWC alt 530km' and 'SWC alt 520 km') and Swarm A and B (curves labelled 'SWAB alt: 450km …' ) plotted as a function of the amount of inclination change to be performed by Swarm A/B (Swarm C would perform 0.6 degrees minus this value).

For a given maximum delta-v available for Swarm C the point where the 'SWC alt 530km' line crosses into the green forbidden area gives the optimum inclination change for Swarm A/B and Swarm C. The cost of reaching the various altitudes for Swarm A and B can then be read for that 'SWC DV max'. The cost of the orbit insertion phase for Swarm A and B is reduced by performing more inclination change on Swarm C. It is seen how almost 10 m/s of DV is saved on Swarm A and B by adding 10 m/s to Swarm C orbit insertion phase budget. This is roughly the same as the extra cost to go from 490 km to 460 km for Swarm A and B.

A balance of the inclination change of +0.4 degrees for Swarm C and -0.2 for Swarm A and B was chosen. This was possible because the fuel to be loaded on board was increased from 99 kg to 106 kg (the equivalent to 82.9 m/s of delta-v budget). Swarm A and B are to be lowered to about 460 km. It would be possible to go lower, but this decision was arrived at in order to ensure more fuel was available for the routine phase. An inclination of 87.55 degrees at launch was chosen so that the final inclination of Swarm C would be 87.95, close to the targeted 88 degrees, leaving a margin to subsume any nominal launcher dispersion. Following this strategy, in which the final altitude of Swarm A/B is not the injected one, all the batches are performed correcting simultaneously inclination and semi major axis, and hence two manoeuvres per orbit.

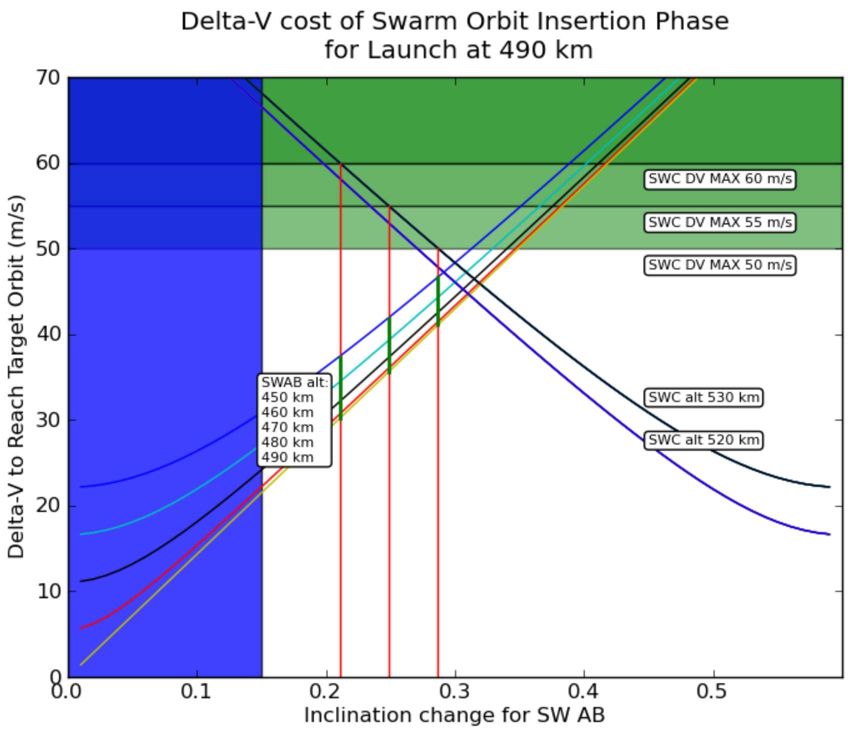


**Figure 1. Mission design diagram for the launch at 490 km. Blue and green shadowed areas are forbidden regions, and the different curves represent different altitudes for Swarm A/B/C.**

Figure 2 shows the estimated loss in altitude for the current predictions of solar cycle 24 activity at the 5%, 50% and 95% (predictions obtained from [1]) for an average area of 1.1 $m^2$, and for an area 20% higher, to account for potential depointing by the attitude control. As seen in the figures Swarm A/B will most likely remain above 300 km long after the 4 years mission lifetime. Either the initial orbit should be lower or the mission should be longer. Prudence dictated that the later was the course chosen and this is the reason for the choice of about 460 km altitude for Swarm A and B. The extra fuel from filling the tanks is therefore available to perform more station keeping.

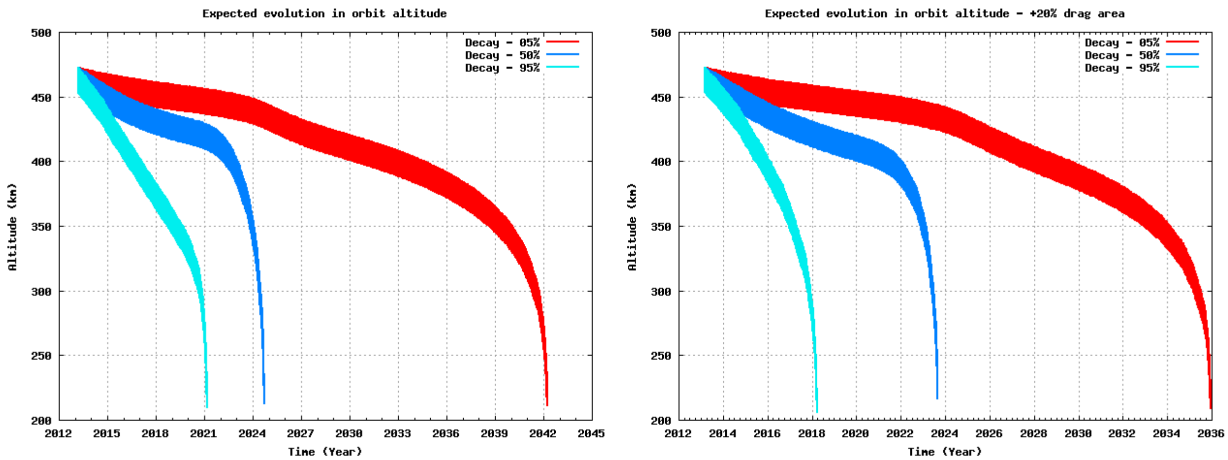


**Figure 2. Expected evolution of the altitude of the lower pair considering the solar activity predictions of the 5%, 50% and 95%. On the left with a nominal average drag area of 1.1 m$^2$, on the right, with an area 20% higher.**

### 2.2. Orbit insertion implementation

During the orbit insertion phase (also known as commissioning phase) the spacecraft are manoeuvred from the injected to their operational orbits. The nominal duration of this phase is limited to 90 days although around 150 manoeuvres per spacecraft are needed. The following constraints have been used for the baseline strategy:

- The target for completion of the orbit insertion phase is launch + 13 weeks
- Manoeuvre only one spacecraft at the same time and avoid weekends.
- Alternate the manoeuvres between a satellite of the lower pair (A/B) and the upper satellite (C). This gives plenty of time to prepare already the next optimization of a satellite whilst another one is manoeuvring as the optimizations of the lower pair and the upper spacecraft are independent from each other.
- The maximum number of manoeuvres to be commanded in a single batch is 100.
- One week needs to be manoeuvre free for instrument calibration/commissioning.

Table 1 shows a summary of the injection and operational orbital elements for each spacecraft. Here the Swarm A/B altitude is refined to 462 km based on the orbit insertion strategy explained below.

**Table 1. Injection and operational orbital elements**

| | **Injection** | **Swarm A** | **Swarm B** | **Swarm C** |
|---|---|---|---|---|
| Semi major axis | $R_{Earth}$+490 km | $R_{Earth}$+462 km | $R_{Earth}$+462 km | $R_{Earth}$+530 km |
| Eccentricity $e_x$ | 0.0005 | 0.0005 | 0.0005 | 0.0005 |
| Eccentricity $e_y$ | 0.0014 | 0.0014 | 0.0014 | 0.0014 |
| Inclination | 87.55 deg | 87.35 deg | 87.35 deg | 87.95 deg |
| Initial RAAN | $\Omega_0$ such that LTAN=14:30 | $\Omega_0$ | $\Omega_0$ - 1.4 deg | $\Omega_0$ |
| Argument of latitude | - | $ALA_A$ | $ALA_A+\Delta$ $\Delta$ = 4...10s | - |

Due to the large amount of manoeuvres required for each spacecraft they are grouped in batches of up to three days that are optimized and commanded together. During these days the satellites perform two 20-minutes burns per orbit (one at ascending and one at descending nodes) which change simultaneously the semi major axis and the inclination. Full control of the eccentricity can be done by moving a small delta-v amount between the ascending and descending node manoeuvres and by optimizing the manoeuvre direction to get a radial component. Launcher dispersions of the eccentricity in the order of 0.001 in any direction can be corrected with less than 0.5 m/s additional delta-v.

The overall timeline of the orbit insertion phase is:

- Test phase, to perform small test manoeuvres with all Swarm satellites.
- First, to perform the manoeuvres for Swarm B. This will change its inclination (-0.2 degrees in relation to injection) and semi major axis (to 462 km of altitude) to the operational one. These manoeuvres will start a relative drift to Swarm A in terms of RAAN and argument of latitude.
- Second, to perform the manoeuvres for Swarm C. This will change its inclination (+0.4 degrees in relation to injection) and semi major axis (to 530 km of altitude) to the operational one.
- Third, to perform the manoeuvres for Swarm A. This will change its inclination and semi major axis to be the same as Swarm B. The manoeuvres have to be properly balanced and phased so that the RAAN relative drift is stopped at 1.4 degrees, and the argument of latitude relative drift is stopped when the spacecraft are closed.
- Fourth, to perform a fine tuning of the relative argument of latitude of the lower pair.

The third phase starts before the end of the second phase, but always considering that only one spacecraft can be manoeuvred per week.

The strategy is illustrated in Fig. 3, which consists of three parts. The upper part of the figure gives the nominal evolution of the altitude separation between Swarm A/B during the orbit acquisition phase assuming an altitude lowering of 28 km. The plot contains a horizontal dashed line at 2 km which just indicates that below and above different scales are applied to make the much smaller altitude changes at the beginning and at the end visible. The major middle part of the figure shows the evolution of the accumulated delta-v over time for each of the three Swarm satellites. The given delta-v does not include slews or attitude maintenance. The lower part gives a condensed timeline showing where manoeuvres are performed. Each manoeuvre batch is represented as a block, filled with the colour of the satellite performing the manoeuvre. The manoeuvre blocks are shown in two rows depending on the purpose of the manoeuvre. Manoeuvres spanning only the first row are pure in-plane manoeuvres and change only semi major axis and eccentricity and therefore altitude and orbit period. However almost all manoeuvres span both rows as they are combined manoeuvres changing also the inclination.

This eases the separation strategy between the satellites as dedicated manoeuvres for an adjustment of the along track drift are only needed at the very end. Already after the test manoeuvres the altitude differences between each satellite pair is slightly increased and the along-track separation increases rapidly. During the major relative RAAN precession phase, i.e. between the first long Swarm A and the last long Swarm B manoeuvre batches, the Swarm A/B altitude separation is always above 10 km. Due to the period difference Swarm A performs 5 additional orbital revolutions. The number of additional revolutions puts an indirect constraint on the altitude lowering (if at the same time the ratio between Swarm A/B relative along track and RAAN drift is fixed). One further revolution corresponds to 5.6 km altitude lowering. Due to this constraint the Swarm A/B altitude is refined to 462 km.

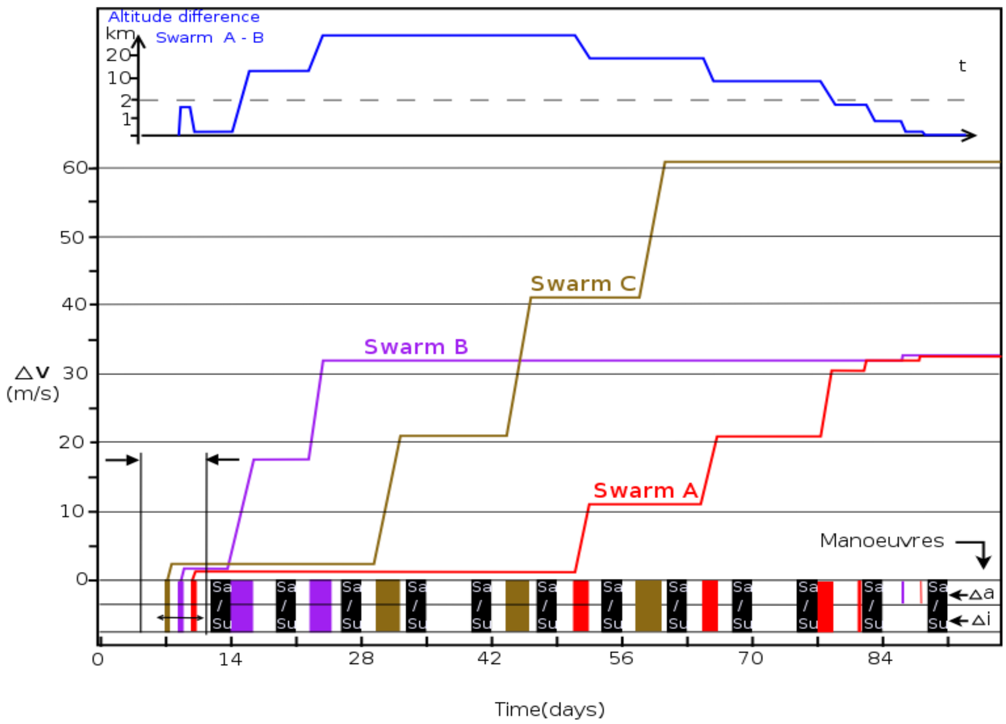


**Figure 3. Orbit insertion strategy.**

The manoeuvres have been planned such that the flight dynamics operations can be performed during normal working hours. The typical work plan for most of the main batches is:

- Friday morning: Orbit determination of the satellite to be manoeuvred next week followed by manoeuvre optimization. Manoeuvre start times are defined by optimizing the argument of latitude.
- Friday afternoon: Orbit determination of the satellite which manoeuvred this week with calibration of manoeuvre batch and update of station predictions for the weekend.
- Monday morning: Final orbit determination of satellite to be manoeuvred. Refinement of manoeuvre start times based on latest prediction of argument of latitude crossing times.
- Monday afternoon: Generation of manoeuvre commands and delivery to Flight Control Team
- Tuesday morning: Uplink of manoeuvre commands by Flight Control Team
- Tuesday afternoon – Friday morning: Execution and monitoring of manoeuvre batch

The distribution of batches has been performed in order to be able to compensate any batch failure during any of the phases. To simplify the re-planning it is important that all Swarm A/B manoeuvre batches are planned with the same ratio of inclination to semi major axis change. Thus any contingency affects the precession rate of the nodes by the same fraction as the along track drift rates. This means that the relative along track position of the satellites is recovered simultaneously by the re-planned manoeuvres.

For contingency planning, a failure of a batch in each of the phases has a different effect on the overall timeline and different approaches can be done to mitigate it:

- A failure in the first phase can be fixed by a proper replanning of Swarm A manoeuvres. This means addition of a third Swarm A manoeuvre batch, redistribution of the delta-v and a delay of the Swarm B manoeuvres.
- A failure in the second phase can easily be fixed, as Swarm C is independent of the other two spacecraft. This means, that if a batch fails for Swarm C and cannot be recovered without interfering with Swarm B manoeuvres, it can be performed after Swarm B manoeuvres, slightly increasing the orbit insertion phase length.
- A failure in the third phase is the most critical, as there is a strong drift in both RAAN and argument of latitude between Swarm A/B. To leave enough time for Swarm B recovery, the RAAN relative precession stop delta-v has been split into three batches which are performed only every second week. Any failure of the first two batches can be recovered by adding a batch in the free week and rebalancing of the delta-v between the remaining batches. A delay of the last batch leads to a slightly larger RAAN separation and a replanning of the fourth phase with an along track drift back phase.
- A failure in the fourth phase should be treated with care as now Swarm A/B are manoeuvred close together. The manoeuvres in this phase are progressively smaller to perform the end phase of the along track drift stop. The sequence is designed such that in case of a manoeuvre abortion there is either enough time to react or the satellites cross over with sufficient altitude separation.

## 3. Routine phase

The routine phase starts once the satellites are in their intended orbits. During this phase the higher satellite will not be manoeuvred, while the lower pair will perform regular manoeuvres to keep the difference in the crossing times of the ascending node (*dt*) between 4 and 10 seconds. These manoeuvres will also be used to systematically decrease or increase the altitude of both satellites in order to better fit the end of mission altitude of 300 km after 4 years, and to keep their eccentricity as close as possible. Considering current solar activity predictions for the solar cycle 24, the manoeuvres will probably be used to decrease the altitude. Some additional manoeuvres could also be considered in order to purely decrease or increase the altitude of the pair.

It is desirable to perform the manoeuvres at fixed weekly intervals. Depending on the spacecraft in flight behaviour and the solar activity evolution a higher frequency of manoeuvre opportunities may be necessary.

The 10 seconds maximum differences come from scientific requirements. The 4 seconds minimum difference is set in order to avoid collision risks around the two points close to the poles where the orbital planes intersect. In this sense, if the trailing spacecraft would enter a safe mode (increasing its mean drag area), it would start losing altitude more quickly and advancing towards the leading one. This 4 seconds difference would make sure that about 500 m separation is built up before the satellites cross over. Prerequisite for this strategy is that 1) the eccentricity vectors of Swarm A/B are sufficiently close together during the whole routine phase such that the temporal variation of the altitude difference during the eccentricity cycle is negligible compared to the altitude change till crossover, and 2) that both satellites have a maximum semi major axis separation well below 500 m. The second point will easily be fulfilled by the own routine strategy. A separation of 500 m in semi major axis will be equivalent to a relative drift of 9.5 seconds per day, so the radial separation will be much lower. The first point is fulfilled by a proper placement in the orbit of the along track manoeuvres.

The strategy to keep the lower pair will be to let them drift until one of the thresholds is about to be hit (*dt* reaches 4 or 10 seconds), and then perform a manoeuvre to keep them inside the thresholds maximizing the time until the next manoeuvre is required. The required manoeuvre will be optimized considering the predicted evolution of *dt* for the next months using the latest solar activity predictions.

The main source of the *dt* evolution of the lower pair is the differential acceleration of the satellites due to differential air drag. This acceleration is determined by:

- Differences in the ballistic coefficients caused by mass differences.
- Differences in the ballistic coefficients due to different pointing performances and resulting effective air drag areas.
- The amount of differential drag experienced due to the sub-solar bulge as the orbital planes of the spacecraft are separated by 1.4 degrees. This effect is dependent on the amount of solar activity and on the local time of the ascending node.

### 3.1. Simulation setup

A simulation tool has been developed in order to check the effectiveness of the routine phase strategy. From a starting date it propagates in a weekly basis the orbits of both satellites, and when one of the *dt* boundaries is going to be violated, it performs a manoeuvre optimization for that week. The "real" propagation is done using the real solar activity records, and the optimization is done using a predicted solar activity, generated following an equivalent procedure to the routine generation of predicted solar activity daily done at ESOC by the software *Prediction of Flux and Ap* (PDFLAP) for short-term forecasts [2].

The manoeuvre optimization works differently depending on the difference of the ballistic coefficient of both spacecraft (the assumed evolution of *dt*). If the difference is large, the *dt* evolution will be parabolic; in this case, the optimization will seek to maximize the time until the next manoeuvre is required (maximizing the deadband usage, but using an optimization band smaller than the real deadband, to account for uncertainties and inaccuracies). If the difference is close to zero, *dt* will not have such a clear pattern and the evolution will be mainly driven by the drag difference due to the different orbital planes. It is still to be seen with real data how accurate we are able to predict this difference in drag. The current approach for this case would be to target a specific distance after several days (in particular, a point close to the middle of the deadband after 100 days), but this may change during the operational life of Swarm.

No manoeuvre errors or inaccuracies in the orbit knowledge are simulated. The first would be mitigated by reducing the optimization band, at the cost of an increase in the frequency of manoeuvres. An error in the orbit knowledge can have an important impact depending on the relative error of the semi major axis estimation. The orbit determination will be done using the on-board GPS receiver. The experience of the Flight Dynamics team using the GPS data from GOCE satellite is that we are able to obtain an accuracy around 1-2 meters in semi major axis. Considering a relative error in semi major axis of around 2 meters between Swarm A/B, this is equivalent to an error in period of 2.4 ms, which accounts for 37 ms per day and 0.36 seconds per week. This can add up to the uncertainty of the future solar activity (which is higher), and can have a more significant impact when the ballistic coefficient of both satellites is close, and thence the initial error accumulates over several weeks. The final difference in ballistic coefficient will only be seen when the operations start, then the orbit control will be adapted accordingly.

The spacecraft properties and the list of the models used in the simulation are summarized in Tab. 2.

**Table 2. Spacecraft properties and models used**

| | |
|---|---|
| Total mass after commissioning | 438.0 kg |
| Average nominal drag area | 1.1 $m^2$ |
| Drag coefficient | 2.2 |
| Average nominal solar radiation pressure area | 8.0 $m^2$ |
| Solar radiation pressure coefficient | 1.3 |
| Atmospheric model | MSIS-00 |
| Gravity field | JGM-3 (50x50) |

Twelve different simulations (cases 1Ah, 1Al, 1Bh, 1Bl, 1Ch, 1Cl, 2Ah, 2Al, 2Bh, 2Bl, 2Ch and 2Cl) have been performed with the combinations of the following configurations:

- In relation to the satellite ballistic coefficients:
    1. The satellites have the same initial mass and their Attitude and Orbit Control System (AOCS) follows a perfect attitude law.
    2. One satellite has a perfect AOCS pointing (area of 1.1 $m^2$) and the other a permanent 2 degree offset (accounting for an area of 1.379 $m^2$).
- In relation to the solar activity. Different periods of time have been taken from solar cycle 23. Real data of solar cycle 23 has been preferred over predicted data of solar cycle 24 in order to have real variations and hence, real uncertainties in the prediction. The periods of time have been taken considering which are the predictions for the 5%, 50% and 95% of the solar activity from the expected beginning of the routine phase (June 2013). The chosen periods are (Figure 4):
    A. High solar activity: F10.7 (mean=182, stddev=41), Ap (mean=13, stddev=14). From 2001/01/08 to 2002/01/08.
    B. Medium solar activity: F10.7 (mean=117, stddev=29), Ap (mean=18, stddev=18). From 2003/06/30 to 2004/06/30.
    C. Low solar activity: F10.7 (mean=73, stddev=6), Ap (mean=7, stddev=6). From 2007/01/08 to 2008/01/08.
- In relation to the initial altitude (or the phase of the mission):
    h. Initial altitude at 462 km, as at the beginning of the operational life.
    l. Altitude at 350 km, as close to the end of the operational life.

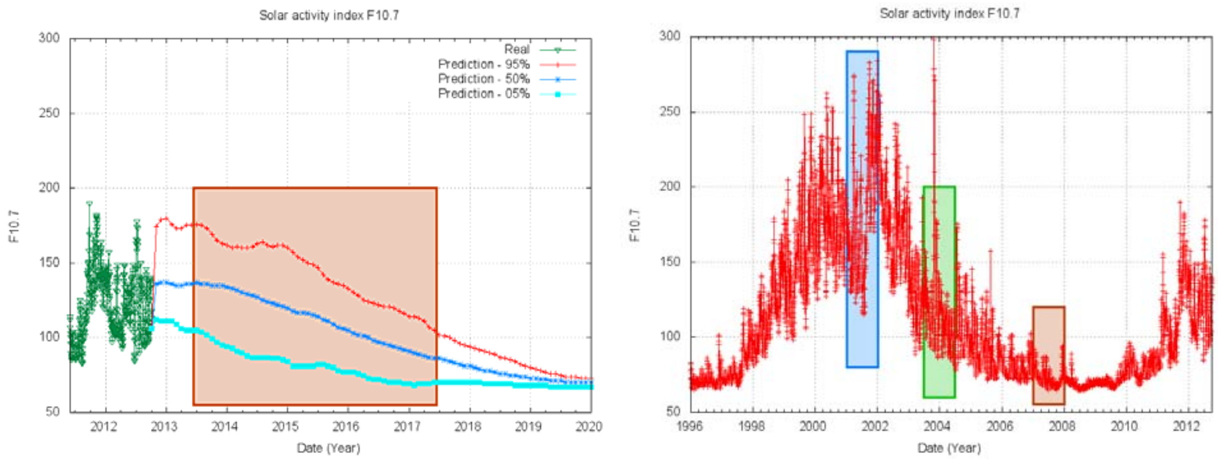


**Figure 4. *Left*: Predicted solar activity from 2012 to 2020. The shadowed area is the expected 4 years life from the beginning of the routine phase (June 2013). *Right*: Solar activity from 1996 to 2012. Shadowed areas are the chosen time span for the simulations, which are representative of the 95%, 50% and 5% of the solar activity predictions for the life of the mission. Blue is for high solar activity (95%), green for medium (50%), and brown for low (5%).**

### 3.2. Simulation results

Figures 5 and 6 show the evolution of *dt* for the 1 year simulation for each of the (h) and (l) cases. Table 3 shows a comparison between all the cases. Interesting to see that with a similar ballistic coefficient between both spacecraft (cases 1x) the frequency and size of the manoeuvres are quite similar independently of the solar activity. The effect of the depointing bias of 2 degrees in one of the spacecraft (cases 2x) amplifies the effect of the solar activity levels.

The cases 2Ah, 2Al, 2Bl and 2Cl required more than one manoeuvre opportunity per week in order to keep *dt* between 4 and 10 seconds due to the high differential drag between both spacecraft. For these cases the simulation was run with two manoeuvre slots per week. Note that this does not mean that the performed manoeuvre frequency is more than once per week since the additional manoeuvre opportunity is only performed when required, usually due to worse than normal drag predictions. Also, it may be possible to mitigate the situation by inserting sub-optimal manoeuvre cycles to ensure the next manoeuvre can make full use of the deadband when performing an optimal cycle.

For the 462 km cases, the 0 degree depointing of both AOCS means that both satellites have the same ballistic coefficient, which allows long cycles requiring only 2-3 manoeuvres per year. With the 2 degree depointing of one of the AOCS the manoeuvres are far more frequent. In the case of the high solar activity, two weekly manoeuvre opportunities are required in order to keep the spacecraft in the 4-10 seconds deadband. The medium and low solar activity cases worked fine obtaining a manoeuvre every 2 and 5 weeks respectively.

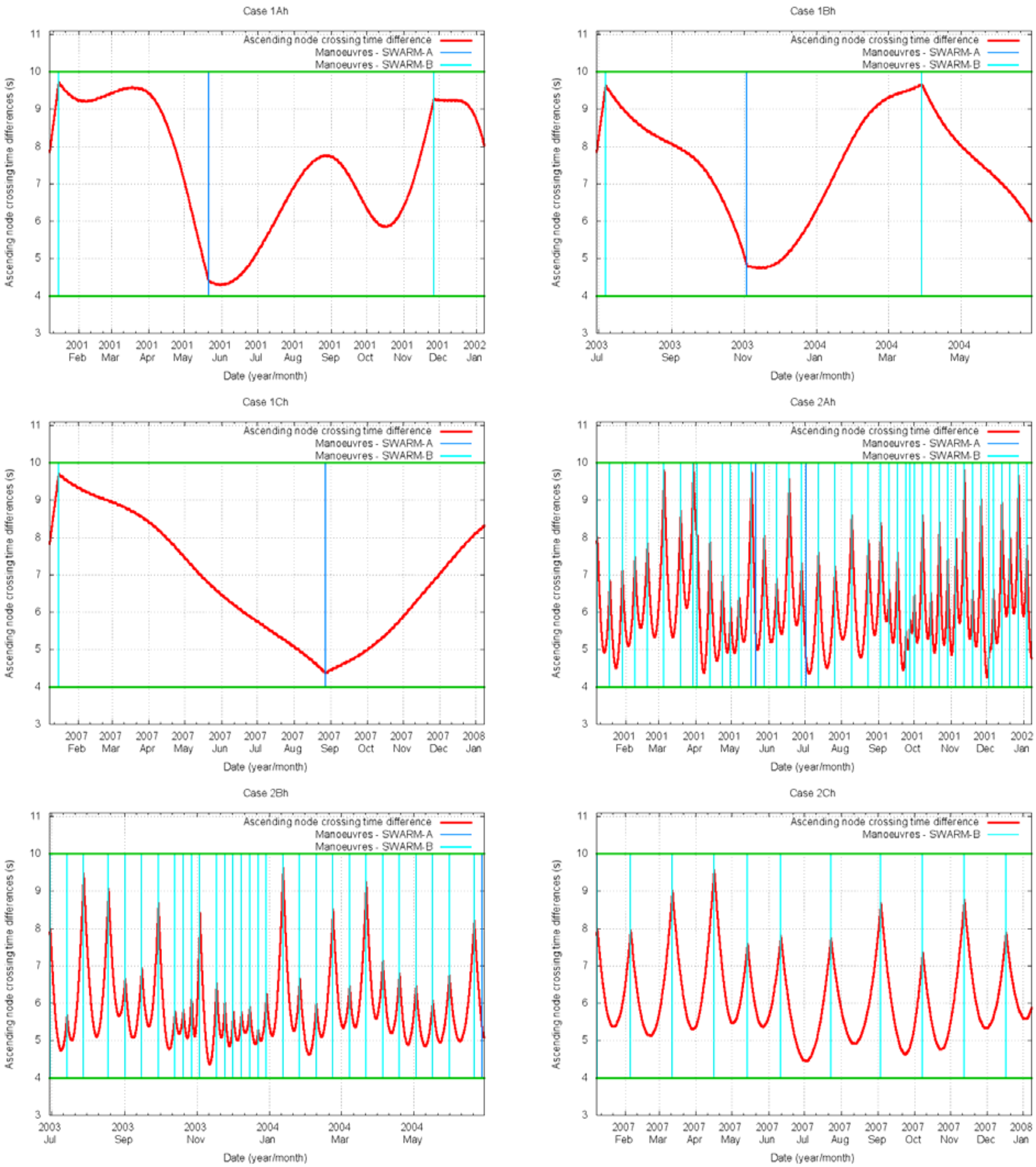


**Figure 5. Results of the simulations starting at 462 km in terms of *dt* evolution as a function of time. Vertical lines are manoeuvres. From top to bottom and left to right, cases 1Ah, 1Bh, 1Ch, 2Ah, 2Bh and 2Ch.**

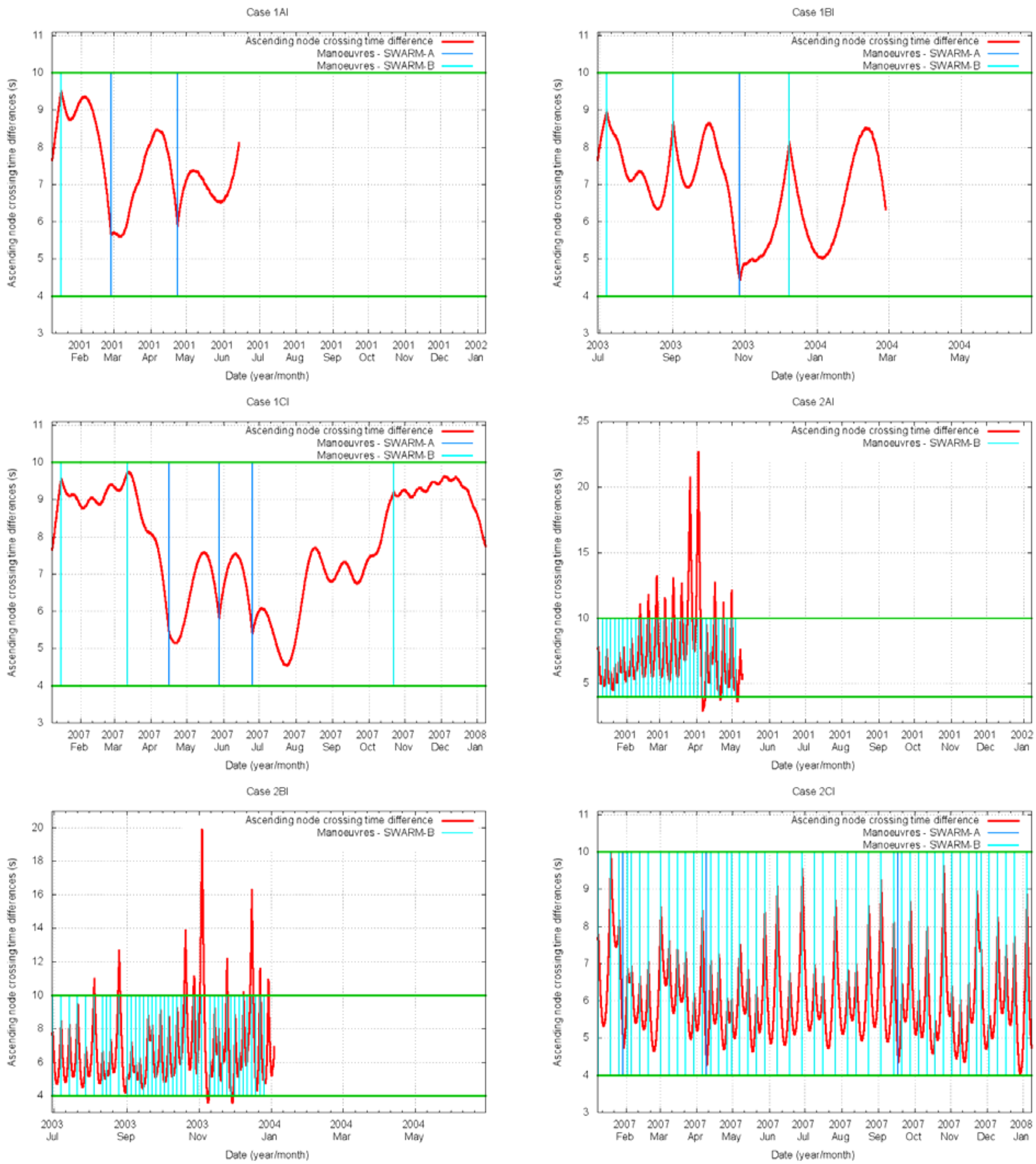


**Figure 6. Results of the simulations starting at 350 km in terms of *dt* evolution as a function of time. Vertical lines are manoeuvres. From top to bottom and left to right, cases 1Al, 1Bl, 1Cl, 2Al, 2Bl and 2Cl.**

**Table 3. Results of the simulation**

| Case | Total delta-v (m/s) | Manoeuvres | Days | Mean cycle duration (days) |
|---|---|---|---|---|
| 1Ah | 0.017 | 3 | 373 | 124 |
| 1Bh | 0.014 | 3 | 373 | 124 |
| 1Ch | 0.009 | 2 | 373 | 187 |
| 2Ah | 2.861 | 45 | 373 | 8.3 |
| 2Bh | 1.010 | 31 | 373 | 12 |
| 2Ch | 0.232 | 11 | 373 | 34 |
| 1Al | 0.039 | 3 | 158 | 53 |
| 1Bl | 0.057 | 4 | 188 | 47 |
| 1Cl | 0.052 | 6 | 373 | 62 |
| 2Al | 5.387 | 34 | 122 | 3.6 |
| 2Bl | 5.564 | 42 | 188 | 4.5 |
| 2Cl | 3.256 | 52 | 373 | 7.2 |

For the 350 km cases, the simulations stop when the spacecraft reach 300 km and the reentry is imminent. The 0 degree depointing difference still provides good results, with just once per two months manoeuvres. For the case of the 2 degree depointing differences, two manoeuvre opportunities per week are required in all solar activity cases. For low solar activity the *dt* thresholds are properly met and maintained, but for the high and medium solar activity cases there are some deadband violations. The violations in the medium solar activity case come mainly from uncertainties in the drag. At this altitude, for some extreme cases the difference of propagation using the predicted solar activity and the "real" one reached a difference of up to 10 seconds after one week, which means that the prediction capability is greatly reduced. Besides, both the medium and high solar activity cases would require more than two manoeuvre opportunities per week in order to keep the *dt* in margins.

For the medium/high solar activity cases, the uncertainty due to drag at low altitudes becomes too high, and the prediction capability is reduced drastically. This means that if the ballistic coefficient difference is also high a close monitoring of the *dt* evolution and several manoeuvre opportunities per week would be required. Besides, the need of delta-v at 350 km when there are significant differences in the ballistic coefficient is quite large, so it may become too expensive (depending on the propellant left on-board) to keep the same spacecraft configuration.

Having several manoeuvres per week would not interfere with the gathering of scientific data. The longest manoeuvres obtained in the simulation (for 350 km, 2 degree depointing and high solar activity) are in the order of 20-30 minutes, if there is a need for slewing the spacecraft before and afterwards, as a worst case, a full orbit could be required for this. Considering the maximum rate of manoeuvres required this would hardly had any impact in the availability of operational scientific measurement time.

Following the last inputs for the total amount of propellant loaded on the spacecraft (106 kg), the distribution of inclination change and the desired initial altitude for Swarm A/B the budget of the routine phase is 13.0 m/s. As seen in Tab. 3, even in high solar activity with the 2 degree depointing error, only 2.8 m/s are required in a full year. As the altitude keeps decreasing the yearly delta-v need will tend to increase, but also the solar activity will decrease yearly, as we are past the solar cycle maximum. Considering this, the delta-v budget should be more than the required for the mission life, providing a high margin for unexpected problems during routine phase, and the possibility to increase the mission life if the lower pair has not reentered the atmosphere after the 4 years life.

There is a large difference in the required delta-v and number of manoeuvres when comparing the 0 and the 2 degree pointing difference cases. This means that in the event that the ballistic coefficients of the lower pair are found to be different, it may be desirable to bias the attitude of one spacecraft to balance this difference and so reduce the frequency and size of the manoeuvres performed.

## 4. Summary

Swarm mission is expected to be launched in the first half of 2013. The orbit acquisition phase will last for 3 months, and has been designed considering the low thrust capabilities of its thrusters. The manoeuvres will be optimized and commanded in batches, and the strategy has been prepared to minimize the effect of potential failures during the execution of one of the batches.

The routine phase strategy has also been tested under two different AOCS configurations (perfect nominal alignment for both lower pair spacecraft, and a 2 degree depointing bias in one of them), under different solar activity conditions considering the expected 5%, 50% and 95% solar activity and at different altitudes, considering the beginning and end of life altitudes. The required delta-v in all cases is below the budget, and the strategy has been proved to work in all cases at the beginning of life conditions. At the end of life conditions, and depending on the solar activity level and the difference in the ballistic coefficients, two or even three manoeuvre opportunities per week may be necessary. Additionally, even at the beginning of the mission a high difference in ballistic coefficient and high solar activity would mean that two manoeuvre opportunities per week would also be required to ensure that the requirements can be met. Note that this does not mean that all these opportunities would be used, the higher frequency of possible manoeuvres are necessary to account for the uncertainties in drag predictions at the time of the manoeuvre optimisation. Besides, depending on the difference on ballistic coefficient between both spacecraft it may be desirable to bias the attitude of one of them to keep the difference as small as possible.

## 5. References

[1] NOAA National Weather Service, Space Weather Prediction Center, http://www.swpc.noaa.gov, Accessed October 2012.

[2] Mugellesi-Dow, R.; Kerridge, D.J.; Clark, T.D.D. and Thomson, A.W.P. "SOLMAG: an operational system for prediction of solar and geomagnetic indices.", Proceedings of the First European Conference on Space Debris, pp. 373 – 376, Darmstadt, Germany, 1993.