# FEASIBILITY OF METEOSAT THIRD GENERATION COLLOCATION USING A SINGLE S-BAND TRACKING STATION AND LARGE INCLINATION MANEUVERS

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***Abstract:*** *The Meteosat Third Generation (MTG) mission requires two 3-axis stabilized satellites, one implementing the imaging instrument (MTG-I) and another the sounding instrument (MTG-S), to be operated in orbit simultaneously in collocation around the 0º geostationary position.*
*Satellites collocation, even if requiring more complex operations, may allow operating both satellites using a single S-band station, with an important reduction of the ground support required for Flight Dynamics operations, and consequently of the number of needed ground stations and of the operational workload. Further reduction of the operational workload, together with a potential benefit in terms of availability of the scientific data, could be achieved if the collocation strategy is implemented through very large inclination maneuvers executed yearly.*
*The paper will present in detail the analyses performed to assess the flight dynamics feasibility of the proposed operational strategy and will address its potential benefit; at the same time the major constraints at system level, which currently may prevent the implementation of the described strategy, are presented.*

***Keywords:*** *Collocation, Orbit Determination, Maneuver Execution Errors.*

## 1. Introduction

The Meteosat Third Generation (MTG) observation missions comprise a Full Disk High Spectral resolution Imagery (FDHSI) mission, a High spatial Resolution Fast Imagery (HRFI) mission, a Lightning Imagery (LI) mission, an Infra-Red Sounding (IRS) mission and a Ultra-violet, Visible and Near-infrared (UVN) sounding mission. The UVN mission is implemented with the GMES Sentinel-4 instrument. Separate 3-axis stabilized satellites will be carrying the imaging (MTG-I) and the sounding (MTG-S) missions. Due to frequency allocations constraints, there is a need for collocation of the two satellites around the 0º geostationary position.

On one side the collocation of two satellites requires slightly more complex operations to ensure proper separation between the satellites, while maintaining both within the dedicated longitude and latitude window (+/-0.1 degrees and +/-0.5 degrees respectively), on another the very small angular separation of two satellites may allow to track both using a single S-band station. That may result then in an important reduction of the ground support required for Flight Dynamics operations, with consequent reduction of the number of needed ground stations and of the operational workload.

Further reduction of the operational workload can be achieved if the collocation strategy is implemented through very large inclination maneuvers executed yearly (instead of monthly, as normally done for telecom GEO satellites). Moreover, a remarkable benefit on the availability of the scientific data is also caused by the reduced number of interruption, being a large part of the outage caused by an inclination maneuver independent from the size of the maneuver itself. Maximization of the availability of quality image data to the end users is a key objective of EUMETSAT missions.

Several analyses have been performed by the EUMETSAT Flight Dynamics team in order to assess if a one-station strategy is really feasible for two collocated satellites and if the proposed collocation strategy can be safely implemented:

1) first of all, the orbit determination accuracy that can be achieved using tracking data from a single S-band station was assessed; both routine and post maneuver cases were considered;
2) then, a second analysis was performed to assess if a single ground station can continuously observe both satellites at the same time, in consideration of the expected relatively large difference in inclination of the two orbits (up to 1 degree);
3) finally it was analyzed the feasibility of the assumed collocation strategy, based on a single very large inclination maneuver per year, as the impact of the execution error both in longitude and eccentricity may make impossible to implement a safe collocation scheme.

It shall be kept in mind that these analyses are focused on the feasibility only from a Flight Dynamics point of view; several constraints at system level are present, which prevent the direct implementation of the suggested strategy, which, therefore, cannot be considered as a baseline for the implementation of MTG operations.

## 2. Single Station Orbit Determination

The primary goal of this analysis (described in detail in [1]) is to evaluate how accurately it is possible to estimate the orbit of a GEO satellite using ranging data from a single station in case no maneuver is present in the determination arc. Moreover, cases where a maneuver is present in the determination are also considered to evaluate how quickly the orbit determination process converges back to nominal performances in case ranging data from only one station are used.

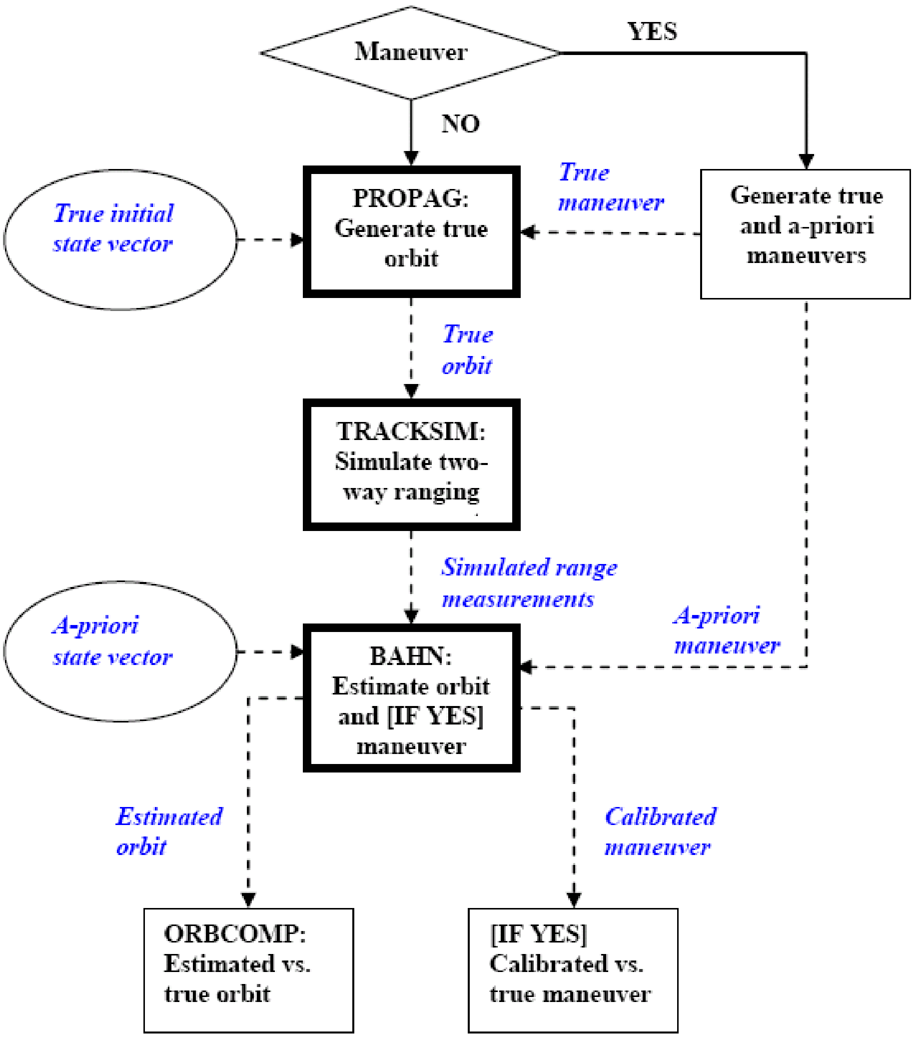


**Figure 1. Orbit Determination Simulations Flow-Chart**

The analysis was performed using the operational Flight Dynamics software used for LEO satellites operations in EUMETSAT (EPSFDF, see [7]). The following steps have been performed;

1) [if applicable] generation of a true maneuver (from an a-priori maneuver plus expected execution error);
2) generation of a true orbit with full perturbation model (from a true state vector) and [if applicable] through the true maneuver;
3) generation of simulated two-way range measurements based on the true orbit and with realistic measurement errors;
4) orbit determination (from an a-priori state vector different than the true state vector) and [if applicable] maneuver calibration (starting from the a-priori maneuver);
5) comparison of determined orbit with true orbit and [if applicable] of calibrated maneuver with true maneuver.

For all the cases analyzed 7 days are considered as simulation span. The block diagram in Fig. 1 summarizes the performed steps together with the main data flow.

For each simulation scenario only one “run” is performed. Therefore statistical reliability of the results is not ensured. Nevertheless the results provided are believed in most of the cases to be quite representative of the real expected behavior. The detailed description of the process is presented hereafter first for the maneuver free cases and then for the cases with maneuver; furthermore, cases with auxiliary data are presented.

### 2.1. True Orbit Generation

The module PROPAG of EPSFDF is used to generate the true orbit by numerical propagation over 7 days from a pre-defined initial state vector and considering the highest accuracy available perturbation models:

- Earth gravity model: GM 36x36
- Sun, Moon gravity
- J2/Moon gravity interaction
- Solid Tides
- Solar Radiation Pressure (constant area model)

The perturbation models used ensure that this true orbit represents with sufficient accuracy a real satellite evolution; in other terms, the difference with the trajectory the satellite would follow from the given initial state vector is much smaller than the typical orbit determination accuracy for a GEO satellite. As initial state vector, a geostationary orbit at 0.0 degrees longitude is considered.

### 2.2. Simulated Measurements Generation

The module TRACKSIM of EPSFDF is used to generate simulated two-way ranging data on the seven days of orbit arc; these data are generated by accurate computation of the flight time (in the J2000 inertial frame) from a selected station to the satellite and back to the station.

The coordinates of the stations considered in the analysis, the same used operationally for MSG operations, are provided in the Tab. 1.

**Table 1. Stations Coordinates**

| Station | $\lambda$ [°] | $\varphi$ [°] | h [m] |
|---|---|---|---|
| Usingen | 8.482 | 50.333 | 427.6 |
| Maspalomas | 344.367 | 27.763 | 191.4 |

To generate realistic simulated measurements, several sources of errors, summarized in Tab. 2, have been considered; each contributor is classified either as a high frequency noise or as a daily variation error or as a long term bias.

**Table 2. Measurement Errors**

| Error type | Error [m] |
|---|---|
| High frequency noise | |
| Ground station noise | 3 RMS |
| S-Band transponder noise | 1.5 RMS |
| Clock accuracy | Negligible |
| Daily variation errors | |
| Clock synchronization | Negligible |
| Solid tides | Negligible |
| Ionosphere delay | 3 P2P |
| Troposphere delay | 1 P2P |
| Ground station delay | Negligible |
| S-Band transponder delay | Negligible |
| Long term biases | |
| Station coordinates | 2 |
| Ionosphere delay | 2 |
| Troposphere delay | 1 |
| Ground station delay | 1 |
| S-Band transponder delay | 1 |

The total contribution (both as RMS sum and ABS sum) is presented in Tab. 3. RMS sum values shall be used for high frequency noise and daily variation errors and ABS sum values for long term biases.

**Table 3. Measurement Errors Total Contribution**

| | RMS sum [m] | ABS sum [m] |
|---|---|---|
| High frequency noise | 3.4 | 4.5 |
| Daily variation errors [P2P] | 3.2 | 4 |
| Long term biases | 3.3 | 7 |

The long term variation error is included as a fix bias in the estimation process, as described in Section 2.3. That approach permits to analyze different values of the error without having to regenerate the simulated measurements.

With the available SW it is not possible however to define errors at daily frequency; to generate these errors the approach of removing from the data generation a correction with daily oscillation close to the expected error amplitude is taken. As the ionosphere correction presents amplitude very close to the expected one of the daily error, its contribution is removed from the data generation and an artificial bias is introduced to compensate for its mean value.

The following 4 sets of measurements are generated:

- M1: Ranging from prime station (Usingen) at 3h frequency with 3.4m noise.
- M2: Ranging from prime station (Usingen) at 30min frequency with 3.4m noise.
- M3: Ranging from prime station (Usingen) at 15min frequency with 3.4m noise.
- M4: Ranging from secondary station (Maspalomas) at 15min frequency with 3.4m noise.

### 2.3. Routine Orbit Determination

The module BAHN of EPSFDF, implementing a batch least square dynamical filter, is used to determine the orbit.
The same measurement model used for the measurement simulation (in 2.2) is applied with no errors and the dynamical model is reduced with respect to the true orbit generation (in 2.1):

- Earth gravity model: GM 6x6
- Sun, Moon gravity
- Solar Radiation Pressure (constant area model)

To ensure that the performances of the orbit determination process are representative, the initial knowledge of the system parameters is set to values similar to what is expected in real operations:

- a-priori state vector differing from initial state vector of the true orbit (in 2.1) by around 1000m in the along-track direction and around 300 meters in the other two directions;
- initial knowledge of the solar radiation coefficient CR differing from the one used in the true orbit generation by around 5%;
- 7m error in the initial knowledge of the overall measurement bias, according to the level of accuracy of the bias measurements and correction presented above (in 2.2).

The following parameters can then be estimated:

- Initial state vector
- Solar radiation coefficient CR

The measurement range bias cannot be estimated due to the insufficient observability and has therefore to be kept fix.

The following 5 orbit determination scenarios are presented:

- S1, S2, S3: Ranging from prime station (Usingen) at 3h, 30min and 15min frequency (M1, M2, M3 sets) with 3.4m noise, to analyze the impact of the data frequency on the solution
- S4: Ranging from prime station (Usingen) at 15min frequency (M3 set) with 3.4m noise and reduced bias (3m), to analyze the impact of the bias knowledge on the solution
- S5: Ranging from prime and secondary stations (Usingen and Maspalomas) at 15m frequency with 3.4m noise (M3 and M4 sets), to provide a reference value for the achievable accuracy; in this case the ranging bias of the secondary station is estimated

The accuracy achieved in the 5 cases is computed comparing the true orbit with the estimated orbit (using the module ORBCOMP of EPSFDF); the obtained results, in terms of maximum error in the entire determination arc, are presented in Tab. 4.

**Table 4. Orbit Determination Accuracy**

| | Radial [m] | Along-track [m] | Cross-track [m] |
|---|---|---|---|
| S1 | ~370 | ~1030 | ~2870 |
| S2 | ~320 | ~850 | ~2520 |
| S3 | ~180 | ~560 | ~1400 |
| S4 | ~170 | ~440 | ~1380 |
| S5 | ~2 | ~200 | ~5 |

The following can be observed:

- high frequency data (15 minutes) are required to satisfactorily estimate the orbit with only one station (S3), even if a quite large cross-track error remains;

- the reduction of the bias error does not improve the solution in a remarkable manner, probably due to the impact of the daily variation errors;
- the addition of a second station improves significantly the solution, as expected.

### 2.4. Orbit Determination across Maneuver

Three scenarios of orbit determination with maneuver are presented, depending on the selected maneuver:

1) SM1: a-priori longitude maneuver of 0.11 m/s; true maneuver considering an execution error of 10% in the in-thrust direction and 1% in the other two directions; ranging from prime station (Usingen) at15min frequency
2) SM2: a-priori inclination maneuver of 4.0 m/s; true maneuver considering an execution error of 1% in the three directions, ranging from prime station (Usingen) at15min frequency
3) SM3: as SM2 plus ranging from secondary station (Maspalomas) at15min frequency

These two maneuvers represent respectively the typical inclination and longitude control maneuver required monthly for a GEO satellite; the level of errors considered are the one that can be expected with calibrated thrusters, assuming thrusting system optimized for big maneuvers (large in-thrust error for small maneuvers).

The following maneuver scale factors are then estimated:

- Only in-thrust component for the longitude control maneuver
- All three components for the inclination control maneuver

For each scenario the end of the determination arc is set a fixed time after the maneuver (6h, 12h, 24h), to evaluate how the determination accuracy after the maneuver evolves in time. The obtained results, in terms of maximum error in the post maneuver arc, are presented in Tab. 5.

**Table 5. Post Maneuver Orbit Determination Accuracy**

| | T after mano [h] | Radial [m] | Along-track [m] | Cross-track [m] |
|---|---|---|---|---|
| SM1 | 6 | ~120 | ~1150 | ~910 |
| | 12 | ~250 | ~730 | ~2010 |
| | 24 | ~920 | ~2220 | ~7160 |
| SM2 | 6 | ~1640 | ~5880 | ~8690 |
| | 12 | ~200 | ~860 | ~1500 |
| | 24 | ~100 | ~410 | ~790 |
| SM3 | 6 | ~8 | ~200 | ~25 |
| | 12 | ~5 | ~160 | ~20 |
| | 24 | ~2 | ~100 | ~5 |

The following can be observed:

- the estimation of only one component provide acceptable results only in the short period; in practice, the maneuver parameter is not really observable and the propagation error is returned; when the arc get larger, the estimation seems not to work satisfactory;
- the estimation of three components provide acceptable results already after 12 hours; before, the data are insufficient to perform a satisfactory estimation;
- the addition of a second station improves significantly the solution, as expected.

For small maneuvers it seems possible to conceive a strategy where, in the first 12 hours only one component is estimated and afterwards all three components. For big maneuvers however, the impact of the propagation of the execution error in the first 12 hours may be unacceptable; in this case the addition of a second station to mitigate that effect would be required.

### 2.4. Orbit Determination with Auxiliary Data

The usage of auxiliary data from several sources can be considered to improve the orbit accuracy obtained with single station ranging. Some simulations have been run to analyze the benefit of the following techniques, using S3 as reference scenario:

1) SA1: addition of image geolocation data into the processing; image geolocation data provide an enhanced knowledge in longitude and latitude, thanks to the observation of the differential displacement of land-marks in an image (see [5]); these data can therefore be modeled as state vector observables; 4 observables per day with a random error of 200m (expected performances of the image processing) are considered
2) SA2: addition of angular measurements; azimuth-elevation measurements from the same antenna performing the ranging are ingested at 5 minutes frequency; a noise of 0.01 degrees (compatible with modern mono-pulse antenna) in both components is assumed
3) SA3: addition of telescope measurements; right ascension and declination measurements from an high accurate space telescope (several location used) during a night are considered; these measurements are converted into high accuracy azimuth-elevation measurements at one hour frequency; a noise of 0.003 degrees in both components, compatible with standard wide-field telescopes (as Starbrook telescope in Cyprus, see [6]), is assumed
4) SA4: addition of an a-priori covariance of 200 meters in the initial state vector (set around 200 meters apart from true initial state vector), equivalent to the expected performances of the single station system enhanced with auxiliary data

The ranging bias can now be estimated, thanks to the availability of independent data.
The obtained results are presented in Tab. 6.

**Table 6. Orbit Determination Accuracy with Auxiliary Data**

| | Radial [m] | Along-track [m] | Cross-track [m] |
|---|---|---|---|
| S3 [ref] | ~180 | ~560 | ~1400 |
| SA1 | ~40 | ~170 | ~290 |
| SA2 | ~30 | ~250 | ~250 |
| SA3 | ~60 | ~250 | ~450 |
| SA4 | ~60 | ~460 | ~490 |

The following can be observed:

- the addition of auxiliary data permits to enhance significantly the orbit determination accuracy achieved by the single station ranging alone, above all in the cross-track direction;
- the usage of a-priori covariance permits to maintain an acceptable solution, even if the degradation is evident; regular addition of auxiliary data would be required.

## 3. Tracking of Two Collocated Satellites from a Single Ground Station

The goal of this analysis (described in detail in [2]) is first of all to analyze the relative angular separation of two collocated MTG satellites as seen from the primary ground station (Usingen) and then to define a strategy permitting to track both satellite using a single S-band antenna.

Orbit inclination and eccentricity are the principal causes of the apparent daily motion of a geostationary satellite. The inclination of the orbit produces a “eight” figure centered on the equator, as seen by an observer at the Earth centre. The half height $h_i$ (in North/South angle) and half width $w_i$ (in East/West angle) of the figure of "eight" due to an inclination $i$ are given by Eq. 1 (see [8])

$$\begin{aligned} h_i &= \pm i \\ \tan w_i &\cong \tan^2(i/2) \end{aligned} \tag{1}$$

The eccentricity $e$ causes an oscillation of amplitude $2e$ in East-West. For the MTG satellites, the eccentricity value is very small, causing oscillation of the order of few hundredths of degrees; the non-zero inclination effect dominates and eccentricity effects are therefore neglected in this analysis.

For the two collocated MTG satellites the inclination is bounded to ±0.5 degrees; therefore, if no coordinated inclination control strategy is implemented, the maximum inclination difference between the two satellites can get up to 1.0 degree; at the same time, the maximum longitude difference can get up to 0.2 degrees, being the longitude window of ±0.1 degree; all that translate in an angular separation as seen from the primary ground station of around 1.1 degrees (larger than the value observed from the earth center as the station is closer to the geostationary ring). For the S-band antenna used for MSG operations (taken as reference for this analysis) the beam width is of 0.947 degrees, so insufficient to track both satellites in this configuration.

Let's assume that coordinated inclination control is implemented for both satellites, with six months of time de-correlation between the inclination evolutions (depicted in Fig. 2, A, B, C, D being separated by around 3 months and $C_1$-$C_2$ being the effect of inclination control maneuver). In this case, when the inclination of a satellite is maximum, the inclination of the other collocation satellite is minimum (points A and C); the case of both inclinations close to the maximum is never observed.

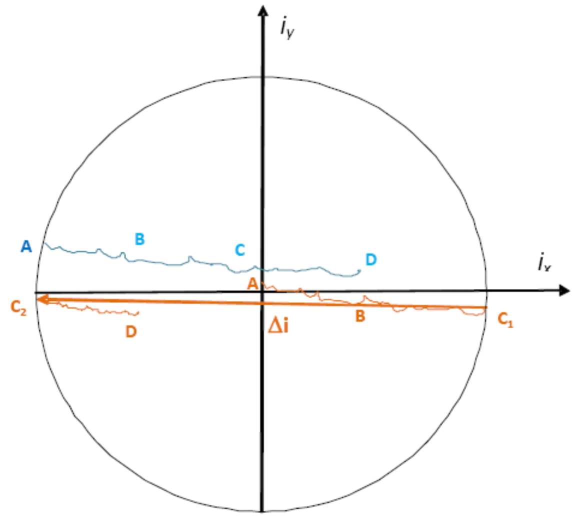


**Figure 2. Inclination Evolution of Collocated Satellite**

Three antenna control strategies have been considered for analysis:

1) Antenna kept in the centre of the nominal control box (longitude/latitude 0/0 degrees)
2) Antenna following continuously the mean angular location of the two satellites
3) Antenna step control during high inclination seasons

### 3.1. Antenna kept in the centre of the nominal control box

When this strategy is implemented, a satellite can be tracked only if the angular separation of the satellite from the nominal fixed pointing is lower than half of the beam-width.

The daily evolution of this angular separation (in degrees) is shown in Fig. 3 for an inclination of 0.3 degrees (left) and 0.5 degrees (right); longitude is set to 0.1 degrees in both cases.

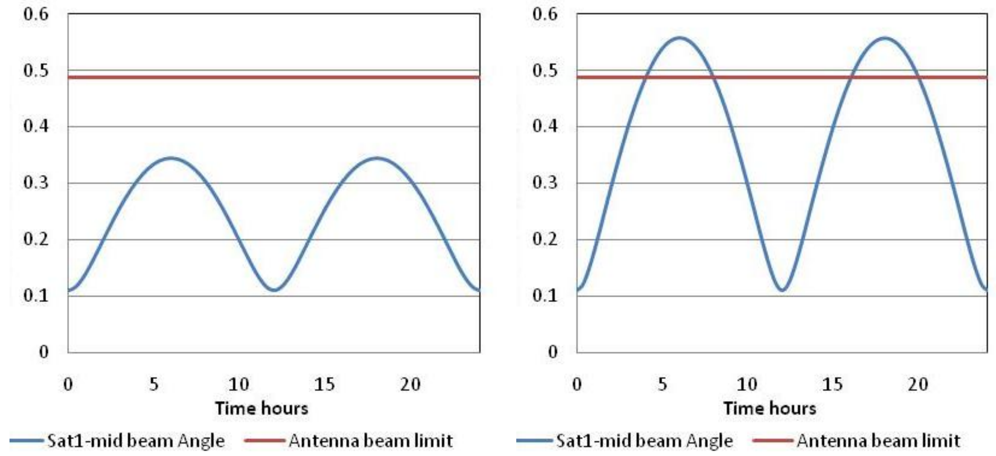


**Figure 3. Angular Separation [deg] from Nominal Pointing**

It can be observed that this strategy, extremely simple, provide satisfactory tracking only if the inclination is bounded under a bit more than 0.4 degrees, and thus not optimal for MTG.

### 3.2. Antenna following continuously the mean angular location of the two satellites

To implement this strategy it is necessary to continuously command the antenna pointing in order to maintain the same angular separation from both satellites. When this strategy is implemented, both satellites can be tracked only if the angular separation between them is lower than the beam-width.

The daily evolution of the angular separation (in degrees) between the two satellites is shown in Fig. 4 for an inclination of 0.3degrees for both satellite (points B and D in Fig. 2, left) and 0.5 degrees for one satellite and 0.1 for the other (points A and C in Fig. 2, right); in both cases the longitude is set respectively to +0.1 and -0.1 degrees.

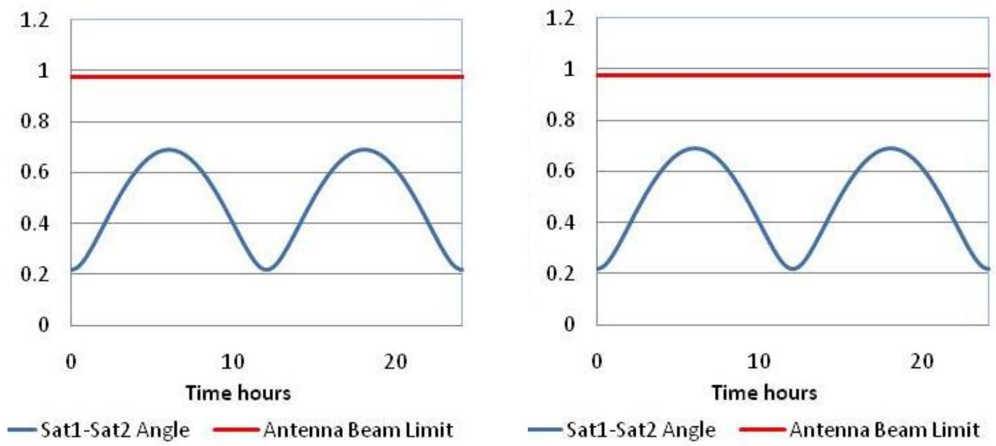


**Figure 4. Relative Angular Separation [deg]**

It can be observed that this strategy, extremely complex to implement, provide more than satisfactory tracking for all ranges of inclination considered, and thus fully applicable for MTG.
It is also interesting to note that the large relative angular separation does not permit to implement a strategy where one satellite is tracked in auto-track, as the other satellite would exit the antenna beam width; antenna pointing angle cannot be used for orbit determination.

### 3.3. Antenna step control during high inclination seasons

In 3.1 was observed that the strategy of keeping the pointing fixed in the centre of the nominal window is applicable only if both satellites have an inclination below 0.42 degrees; for the period of time when one of the satellites presents an inclination larger than 0.42 degrees, and taking into account the fact that the satellite with higher inclination has a libration period of one sidereal day, it is possible to conceive a simple strategy permitting to follow it while maintaining also the satellite with higher inclination properly tracked.

Let's consider just two antenna pointing targets defined as:

1) longitude 0 degrees, latitude 0.08 degrees
2) longitude 0 degrees, latitude -0.08 degrees

The daily evolution of the angular separation (in degrees) of the first satellite, assumed to have an inclination of 0.5 degrees (left) and the second satellite (assumed to have an inclination of 0.1 degrees) from the first target is shown in Fig. 5.

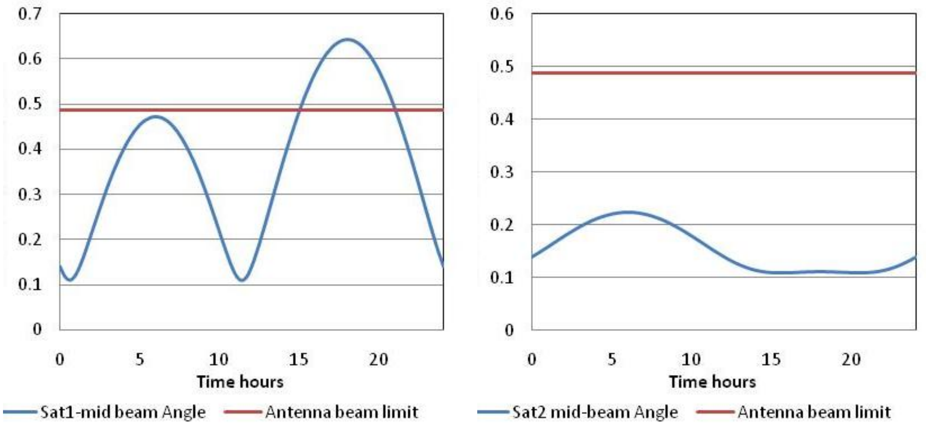


**Figure 5. Angular Separation [deg] from Target: long 0deg, lat 0.08deg**

The same angular separation from the second target is shown in Fig. 6.

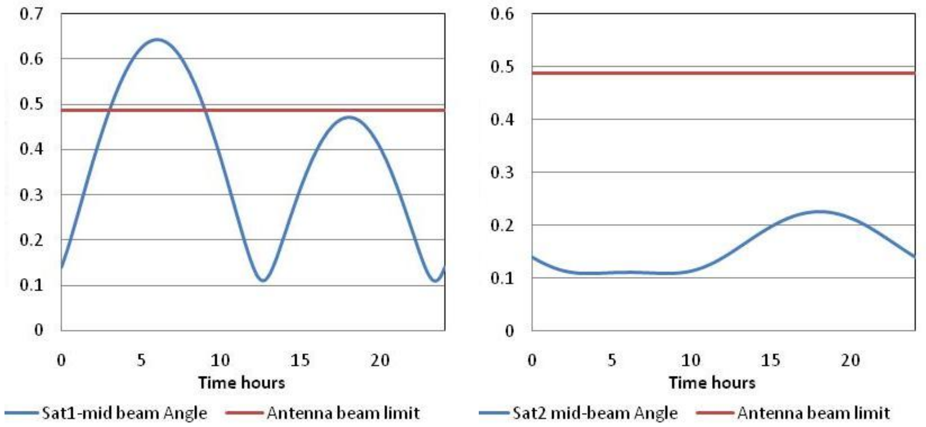


**Figure 6. Angular Separation [deg] from Target: long 0deg, lat -0.08deg**

It can be observed that satisfactory tracking of both satellites is provided during the first 12 hours of the day by target 1, while during the second 12 hours it is provided by target 2.

This strategy, relatively simple to implement (only two position swap per day is sufficient during seasons when one of the two satellites presents very large inclination), provides satisfactory tracking for all ranges of inclination considered, and thus fully applicable for MTG.

### 3.4. Orbit determination performances using single antenna for more satellites

In order to verify if it is still possible to achieve a satisfactory orbit determination even if the same S-band station is used to track more satellites, an extra simulation run has been executed using the following ranging data (see Section 2):

- M5: Ranging from prime station (Usingen) at 5min frequency with 1m noise and data collected only 5 hours every 20 hours.

This data represent the case where up to 4 collocated satellites are operated with a single station; the 1m noise is deriving from the fact that a measurement rate of 30 seconds is assumed (noise reduction by a factor √10 for normal point generation at 5 minutes rate); the selection of a duty cycle of 20 hours ensures observability of the entire orbit arc after 4 days.

The accuracy obtained for the determined orbit is equivalent to the one of scenario S3, proving the feasibility of the strategy, provided that sufficiently high ranging data rate is available. It is to be noted that, being the antenna no more operated in autotrack, antenna pointing data cannot be used for orbit determination anymore.

## 4. Impact of Maneuver Execution Errors on Collocation Strategy

The goal of this analysis (described in detail in [3]) is to evaluate if it is possible to implement a safe collocation strategy of two satellites within a longitude window of ±0.1 degrees and an inclination window of ±0.5 degrees when very large inclination maneuvers are used. For the collocation a standard strategy based in separation in eccentricity and inclination is assumed (see [4]).

In order to consider a safe collocation strategy we have to ensure that:

1) after an inclination maneuver the satellite does not exit the allocated longitude window before a longitude control touch-up maneuver is performed (normally 24 hours later);
2) after an inclination maneuver the satellite eccentricity vector remains sufficiently separated from the one of the collocated satellite (collision risk);
3) after the inclination maneuver the satellite inclination vector remains sufficiently separated from the one of the collocated satellite (collision risk).

Assuming that one inclination maneuver per year is executed, which is compatible with the ±0.5 degrees of inclination window available (see Fig. 2), then the size of the resulting maneuver is over 50m/s. Moreover, separation in inclination is always ensured if inclination maneuvers on the two satellites are de-correlated in time (condition 3 above).

In case of perfect execution of an inclination maneuver, neither longitude drift nor eccentricity degradation are observed after the maneuver itself; however no maneuver is executed perfectly and parasitic thrusts are to be expected both in the along-track and radial direction, whose effects are:

- the along-track parasitic thrust causes a shift in mean longitude $D$ (in degrees) proportional to the size of the thrust itself ($\Delta v_{at}$, in m/s) to the time elapsed from the maneuver execution ($\tau$, in days), to the geostationary orbital velocity ($v_{geo}$ = 3074.66m/s) and to the orbital arc flown by the satellite in a solar day ($\alpha_{day}$ = 360.985deg), as in Eq. 2;

$$D = 3 \cdot \tau \cdot \Delta v_{at} \cdot \frac{\alpha_{day}}{v_{geo}} \tag{2}$$

- the along-track and the radial parasitic thrusts cause an eccentricity degradation $\Delta e$ proportional to the size of the thrusts themselves ($\Delta v_{at}$ and $\Delta v_r$) and to the orbital velocity ($v_{geo}$) as in Eq. 3;

$$\Delta e = \frac{1}{v_{geo}} \cdot \sqrt{4 \cdot \Delta v_{at}^{\ 2} + \Delta v_r^{\ 2}} \tag{3}$$

- the satellite eccentricity $e$ (sum of the eccentricity before the maneuver $e_0$ and of the eccentricity degradation caused by the maneuver $\Delta e$) causes an oscillation in longitude $E$ (in degrees) around the mean longitude with amplitude proportional to the eccentricity itself as in Eq. 4.

$$E = 2e\frac{180}{\pi} \tag{4}$$

The longitude band $L$ required to allocate the effect of the parasitic is then the sum of the terms $D$ and $E$ and of the initial deviation from the centre of the window $I$. If $L$ is lower than the longitude window ($W = 0.1$ degrees), then condition 1 above is satisfied.

To minimize the effect of the two last terms ($E$ and $I$) it is therefore necessary to perform the inclination maneuver when the eccentricity is as close as possible to zero ($e_0$ as small as possible) and when the longitude is as close as possible to the centre of the window.

To ensure that the longitude is close to the centre of the window at a certain time, is quite trivial. However, to make sure that the eccentricity is close to zero when the inclination maneuver is performed is more complex. Assuming a six month time de-correlation between the execution of the inclination maneuvers on the two satellites (as done in Section 3), then it is possible to conceive an eccentricity strategy that brings every six months one of the satellite very close to the zero eccentricity, as described in Fig. 7 (A, B, C, D being separated by around 3 months). Each satellite is controlled in eccentricity on a circle having half of the size of the allocated eccentricity window (assumed to be 0.0003 as for MSG) and tangent to the origin of the eccentricity plan. So each satellite goes through the origin every year (satellite on the right at time C, satellite on the left at time A); to ensure that the transit occurs at the desired point in time, the orientation of the centre of the eccentricity control centre has to be properly adjusted.

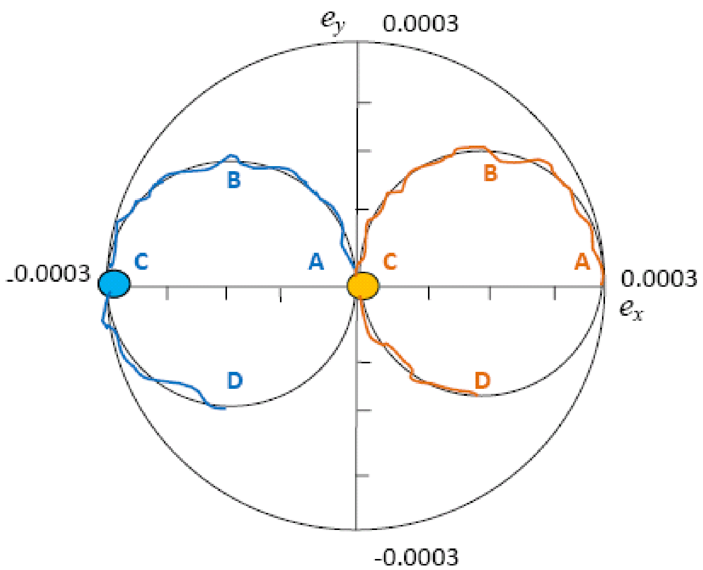


**Figure 7. Eccentricity Evolution of Collocated Satellites**

To minimize the first term ($D$) it is necessary to implement a longitude control maneuver, stopping the drift, as soon as possible; normally 24 hours of time are needed to properly calibrate the inclination maneuver and to prepare the drift stop maneuver. It may be possible to compress the operations and be able to achieve a faster calibration in 12 hours; however, whereas a drift stop maneuver after 1 day counteract the along-track parasitic also in eccentricity, to have it performed after 12 hours would cause a further increase of the eccentricity (and thus of the term $E$), cancelling out the benefit on the term $D$; therefore the value of 1 day will be assumed for the $\tau$ term.

The maximum acceptable eccentricity degradation is linked to the nominal eccentricity separation between the two satellites ($\delta e$ = 0.0003, for the case in Fig. 7); if $\Delta e$ is significantly lower than this separation then condition 2 above is satisfied.

Parasitic thrusts are composed by two components:

- predictable, normally linked to thruster mounting, attitude control actuation in rigid body dynamics, solar panel position and platform biases; this component is normally proportional to the size of the maneuver, can be calibrated by orbit determination and, if the satellite is designed for, can be cancelled by satellite commanding (roll and yaw bias);
- unpredictable, depending very much on the behavior of the propulsion system maneuver execution time, on the flexible modes excitation and on the angular momentum adjustment; this component is normally random and independent from the size of the maneuver (so relatively big for small maneuvers and small for big maneuvers), cannot be calibrated by orbit determination and cannot be compensated at satellite level.

The following 4 cases are considered (all percentages referring to the maneuver size for both along-track and radial direction):

1) large predictable parasitic (1%) and large unpredictable parasitic (0.4%);
2) large predictable parasitic (1%) and small unpredictable parasitic (0.2%);
3) small predictable parasitic (0.1%) and large unpredictable parasitic (0.4%);
4) small predictable parasitic (0.1%) and small unpredictable parasitic (0.2%).

Large predictable parasitic belongs to not calibrated thrusters, small to calibrated thrusters and compensation at satellite level (being the maneuvers executed yearly, a very good calibration can be expected, due to the repetitive conditions of execution).
The size of the unpredictable parasitic is linked to the quality of the satellite systems (mainly propulsion and attitude control).

The obtained results are presented in Tab.7 ($I$ and $e_0$ assumed to be zero).

**Table 7. Longitude and Eccentricity Deviations**

| Case | D [deg] | Δe | E [deg] | L [deg] | L/W | Δe/δe |
|---|---|---|---|---|---|---|
| 1 | 0.247 | 0.00051 | 0.058 | 0.305 | 3.05 | 1.70 |
| 2 | 0.211 | 0.00044 | 0.050 | 0.261 | 2.61 | 1.45 |
| 3 | 0.088 | 0.00018 | 0.021 | 0.109 | 1.09 | 0.61 |
| 4 | 0.053 | 0.00011 | 0.013 | 0.065 | 0.65 | 0.36 |

The ratio $L/W$ represents the percentage of the longitude window occupied by the post maneuver evolution; if larger than 1 the satellite exceeds the allocated window. In order to allocate for initial deviation in longitude and eccentricity ($I$ and $e_0$ different from zero) and to provide satisfactory margins, no more than 80% of the window should be used.
The ratio $\Delta e/\delta\varepsilon$ represents the percentage of the eccentricity separation consumed by the post maneuver evolution; if larger than 1 then the eccentricity of the commanded satellite may fall onto

the eccentricity of the collocated satellite, with consequent collision risk. In order to allocate for initial deviation in eccentricity ($e_0$ different from zero) and to provide satisfactory margins, no more than 80% of the window should be used.

The following can be observed:

- in order to implement a collocation strategy using one yearly inclination maneuver, very little values of the predictable parasitic thrusts are needed (cases 1 and 2 are very far away from the required performances); that can be achieved if the platform is able to compensate for the expected parasitic thrusts (estimated via calibration in the orbit determination) by biasing the platform in yaw and roll; due to the yearly repetitively of the strategy, very accurate calibration could be achieved;
- the value of the unpredictable parasitic thrusts shall also be maintained limited; case 3 presents acceptable performances in eccentricity but marginal violation in longitude; the performances in longitude are fully acceptable in case 4, when the unpredictable component is made smaller; that can only be achieved at satellite design level.

## 5. Conclusions

The proposed strategy, to track two collocated satellites with only one ground station, seems really feasible; the same for the proposed collocation strategy with single yearly inclination maneuver.
The benefits of the proposed strategy are quite evident:

- to require only one ground station for orbit determination of up to 4 satellites would reduce significantly the cost of the ground segment, in terms of set-up, maintenance and operations; no secondary ground station with large geographical separation would be needed anymore to perform dual ranging, as done for MSG; in case of need, the secondary ground station may be made available through a service agreement instead;
- to track two collocated satellite with only one ground station permits to reduce the number of antenna required in the primary site, with consequent economical benefit; the need of a second antenna in the primary site for redundancy seems however unavoidable, noting that this antenna may be dedicated also to other activities (as tracking of other satellites on different longitude);
- to reduce the number of inclination maneuvers presents a clear benefit in terms of operational load; inclination maneuvers are handled as special operations and then to have tenths of them per year instead of only few would imply a large human cost escalation; benefits in terms of mission return are also evident, as fewer interruptions are experienced and as full recovery of mission performances after an inclination control maneuver requires always a long period of the order of one day, regardless from the size of the maneuver.

However the following limitations shall be taken into account:

- even if tracking from one station is sufficient in routine cases for orbit determination, there is a clear benefit to add auxiliary data to the process, as for instance image or telescope data; the need of a second station for calibration of large maneuvers is in any case clear;
- the standard satellite tracking strategy, based on autotrack, has to be modified to allow tracking of two satellite at a time; as a consequence angular data cannot be used for orbit determination (currently not used for EUMETSAT GEO satellites);
- the impact of having a large inclination in the quality of the image data needs to be carefully assessed; even if degradations may be still acceptable it may be recommended to maintain the inclination better controlled to ensure optimal quality of the images;
- the satellite shall allow execution of very large inclination maneuvers (up to 50m/s) with very high accuracy; the option of compensating for repetitive errors in along-track and radial direction by biasing the platform in yaw and roll may be considered; moreover the satellite

shall ensure very small unpredictable parasitic thrusts; these performances are currently outside the envelope of standard GEO platforms and thus extremely challenging; the option of executing a sequence of smaller maneuvers in a short time span could be however considered;

- the need of dumping the angular momentum accumulated by the wheel may require the regular execution of large maneuvers, as the size of the foreseen longitude control maneuvers does not seems sufficient for that; in this case the option of performing one big inclination maneuvers per year may not be anymore the optimal one to minimize the mission outages;
- it is still not confirmed whether two satellites can be tracked by the same antenna, as RF interferences may prevent it.

Due to the presence of these limitations the strategy described in this paper, even if feasible from a Flight Dynamics point of view, cannot be considered valid at system level; therefore this strategy is not to be taken as baseline for the implementation of the MTG operations.

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