METEOSAT THIRD GENERATION NAVIGATION APPROACH

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Abstract: *The EUMETSAT Meteosat Third Generation (MTG) is the next series of the European operational meteorological geostationary satellite system following on MSG. Separate platforms will be carrying the imaging and sounding payload missions, the first of them planned to be launched in 2017. The navigation approach foreseen for the MTG satellites to fulfill stringent mission and operational requirements is presented and discussed, together with the relation between the Image Navigation (INR) and Flight Dynamics (FD) subsystems.*

Keywords: *image navigation, orbit determination, attitude determination.*

1. INTRODUCTION

This paper presents an overview of the image navigation approach foreseen for EUMETSAT MTG (Meteosat Third Generation, three axis stabilized) satellites to fulfill the orbit and pointing accuracy requirements imposed by the spacecraft observation missions and by operational constraints. MTG is the next series of the European operational meteorological geostationary satellite system following on MSG (Meteosat Second Generation). The launch of the first MTG satellite is planned end 2017.

To address the size, development schedules and operating characteristics of the main instruments, separate satellites will be carrying the imaging (MTG-I) and the sounding (MTG-S) missions. Three in-orbit satellites are needed to support the complete and total set of missions. To span the operational life time of the programme over 20 years, there will be in total 4 satellites dedicated to support the imagery missions, and 2 satellites to support the sounding missions.

The problematic of two different satellites, with two different types of observations and geolocation requirements and the challenge to define one image navigation and flight dynamics system approach for both satellites and all observation missions has to be deeply analyzed. For reference, the EUMETSAT MSG (Meteosat Second Generation, spin stabilized) mission requirements and the orbit and attitude performances obtained with the current EUMETSAT operational image navigation and flight dynamics systems will be compared with the MTG mission requirements.

The MTG missions demand finer spatial resolutions and require better geo-location and pointing accuracies. This creates the necessity of enhanced on-ground processing for orbit and attitude determination to bring those processes to performance levels compatible with the requirements.

A spacecraft prime contractor has been selected, but the MTG Programme is still in Phase B, therefore the preliminary design is not fully finalized. However, various simulation studies have been performed based on realistic assumptions. The S/C design is expected to drive the overall Flight Dynamics and INR architecture, including the ground processing. In this paper a summary of these studies is provided, including a proposed complementary approach of Flight Dynamics and Image Navigation to meet the relevant MTG platform and instrument navigation requirements as well as operational needs.

The basic orbit and attitude estimation performances that can be obtained with standard tracking systems and autonomous on-board attitude control (i.e. as for communication satellites) are taken as reference. Then the performance of this basic approach is enhanced by inclusion of additional INR information derived from instrument observations processing to assess the derived performance estimates against MTG requirements.

2. METEOSAT THIRD GENERATION PROGRAM (MTG)

The Meteosat Second Generation (MSG) system has become the primary European source of geostationary observations over Europe and Africa with the start of nominal operations in January 2004. It is one of the key EUMETSAT contributions to the Global Observing System (GOS) of the World Meteorological Organization (WMO). The series of four MSG satellites will deliver observations and services at least until at least 2017 with the high level of availability expected from an operational system. However, considering the typical development cycle for a new complex space system, plans for the Meteosat Third Generation (MTG) system are in place since 2001. MTG needs to be available end 2017, before the end of the nominal lifetime of MSG.

2.1. MTG Observation Missions

The basically reflect the observing capabilities expected from the MTG system. The requirements of the observation missions are generally expressed in terms of the expected level 1b or level lc data outputs (navigated and calibrated radiance) and their characteristics, and will be directly used by real time assimilation applications in the 2017 timeframe.

Five observation missions have been identified for MTG:

- The Full Disc High Spectral resolution Imagery (FDHSI) mission in support of the Full Disc Scanning Service (FDSS) and the High spatial Resolution Fast Imagery (HRFI) mission in support of the Rapid Scanning Service (RSS) will be implemented by a single instrument, i.e. the Flexible Combined Imager (FCI). The FCI has heritage from the SEVERI instrument on MSG. Like SEVIRI it is a scanning imaging radiometer, but with an increased number of channels, better spatial resolution, and enhanced temporal coverage. The Flexible Combined Imager (FCI) mission allows to scan either the full disc in 16 channels every 10 minutes with a resolution in the range 1-2km (FDHSI mode) or a quarter of the earth in 4 channels every 2.5 minutes with a resolution twice better (HRFI mode).
- The Infrared Sounder (IRS) has not been flown in a geo-stationary orbit before, but it has similarities to the IASI instrument on METOP. Both instruments are Fourier Transform Spectrometers which acquire interferograms, which are subsequently processed to spectral radiances. The IRS mission covers the full disc, providing hyperspectral sounding information in two bands, a Long Wave InfraRed (LWIR: 700 - 1210 cm-1) and Mid Wave InfraRed (MWIR: 1600 -2175 cm-1) band with a resolution around 4km.
- The Lighting Imager (LI) has no heritage in EUMETSAT. A similar instrument is the GLM (Geostationary Lightening Mapper) which will be flown on NOAA's GOES-R series of environmental satellites. The LI mission detects continuously the lightning discharges taking place in clouds or between cloud and ground with a resolution around 10km over almost the full disc. It is a non-scanning instrument which continuously transmits detected raw lightening event and background images to ground for refined processing.
- The Ultraviolet-Visible & Near Infrared (UVN) Sounder is implemented with the GMES Sentinel-4 instrument accommodated on MTG-S satellites. The UVN sounding mission covers Europe every hour, taking measurements in three spectral bands (UV: 290 - 400 nm; VIS: 400 - 500 nm, NIR: 755 - 775 nm) with a resolution around 10km.

The three imagery missions (FDHSI, HRFI, and LI) are dedicated to operational meteorology and climate, with emphasis on nowcasting (NWC) and very short range forecasting (VSRF). The two sounding missions (IRS and UVN) are focused on operational meteorology and climate, and to atmospheric chemistry. Studies conducted by ESA during the previous phases have already lead to preliminary instrument concepts for the implementation of the above observation missions.

Most of our analysis and evaluation will focus on the FDHSI and HRFI imagery missions, hosted on the MTG-I platform, which have the most challenging navigation requirements.

3. DRIVING OBSERVATION MISSION REQUIREMENTS

For MTG, the most stringent requirement is the absolute geolocation knowledge error needed for the HRFI imaging mission: the Earth location of acquired samples needs to be determined with accuracy (knowledge) better than 250m at sub-satellite point (SSP) (i.e. 7μrad for a sample distance of 0.5km) with a 68.26 % confidence level over the HRFI coverage.

For the IRS sounder, the requirement is more relaxed, and amounts to 800m at SSP (i.e. 22 μrad) and at 68.26% confidence over a Local Area Coverage (equivalent to a quarter of the Earth). However the sounder has a demanding pointing stability requirement of 300 m at SSP and at a 68.26% confidence level, over the dwell time of about 9 seconds.

Most geometric requirements express image navigation needs: the main task is to estimate a posteriori the position and attitude of the spacecraft from available measurements in order to reconstruct the best possible knowledge of the localization of each sample for subsequent image re-sampling needs (however there are exceptions, most notably the pointing stability requirement).

Therefore, it is not mandatory to obtain the accurate orbit and attitude determination with on-board sensing and on-board processing alone. The best architecture needs to make the most efficient use of on-board processing (i.e. AOCS attitude determination), ground measurements (i.e. ranging) and ground processing in order to obtain the best localization knowledge for the downloaded samples.

Further, the timeliness requirements are also very demanding. For the HRFI imagery level 1b data need to be delivered within 150 seconds of their acquisition. This sets constraints on the number of observations that can be accumulated and used in the image navigation processing. Therefore, efficient algorithms for the processing of image and on-board data are requested.

For comparison: absolute navigation knowledge error performance for the SEVIRI instrument of the MSG platform is about 1130 m at SSP and at a 68.26% confidence level. Hence the performance for the absolute navigation has to be increased by a factor of about four.

3.1. Operational Constraints

A frequency allocation constraint on the geostationary ring and the cost benefits of reducing the number of ground stations demands operation of the MTG satellites within a dedicated longitude and latitude window (respectively of +/-0.1 degrees and +/-0.5 degrees). The reduced angular separation between collocated satellites allows tracking of two satellites using a single S-Band station. On the other side, the collocation requirement creates additional operational constraints on the orbit accuracy (always maintain a minimum safety separation between the collocated satellites) together with more operational complexity because of increasing number of station keeping maneuvers.

Regarding the MTG missions, the satellite availability shall be at least 96% calculated on an annual basis for the duration of the satellite nominal operational life. From this outage, 1% is allocated to unscheduled outages (e.g. safe mode) and 3% to scheduled outages (e.g. station keeping maneuver and operations like satellite decontamination). Therefore disruptions due to orbit control maneuvers must be minimized, using appropriate station keeping strategies and satellite performances.

As seen in [2], station keeping maneuvers may induce larges transients on the orbit recovery of several hours, depending on the strategy considered (i.e. available measurements for processing after maneuver); the transients can be less or more disrupting.
Concerning the imaging outages, initial orbit accuracy is required for the INR to be able initialize their filter processing and converge to good image quality. Any greater deviation from the predicted orbit after maneuver will cause INR problem (and thus FD then needs to provide orbit information to resume INR).

3.2. Use of parallax effect

As described in [2], in case measurements are based on inertial data (star tracker, star imaging or sun sensor), any error in the estimated orbital position translates directly into error localization on Earth (1 to 1 ratio) as shown in Figure 1. In case the attitude measurements are based on Earth data (Earth sensor, landmark matching), the error in the orbital position is only observed through parallax effect (see Figure 2). Samples near the horizon will appear fixed, while samples near the sub-satellite point will appear to move by 15% (i.e. ratio of the Earth radius over the orbit semi-major axis). The apparent localization error of samples in the image is only 15% of the error with respect to the estimated orbital position 1 to 6.56 ratio. Therefore, depending on the concept for attitude reference (Earth or star based), the allocation position knowledge error is respectively 50m or 330m. All performances quoted are three-sigma values.

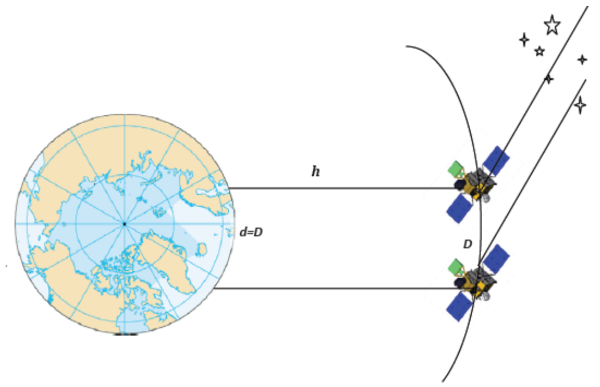

Figure 1: Attitude estimated in inertial frame: direct effect of position D error on geolocation

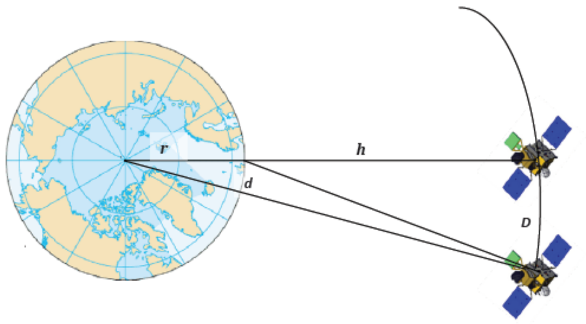

Figure 2: Attitude estimated in Earth-fixed frame: parallax effect of position error D on geolocation

When several landmarks (LM) are available in the same image, it is possible to extract orbital information from the relative displacement of these landmarks. This is true, if the same observation time (so same orbital position for all LMs) and the same attitude are assumed (meaning that attitude errors have already been previously removed via gyro and star measurements).

As shown in picture Figure 3, two landmarks are observed at an angle α in nominal position and the same landmarks are observed at angle β in position affected by position error D. As it can be seen from the picture, there is a correlation between difference of observation angles α and β and position errors. The differential view angle can therefore be used as additional observable in the orbit determination process. The differential view angle provides E/W observability for LM separated in E/W and N/S observability for LMs separated in N/S. E/W accuracy normally improves more than N/S due to smaller time de-correlation (several scanning swaths in between N/S separated LMs).

In order to perform orbit determination using landmarks differential information, the flight dynamics requires regular information from INR. The LM view angle is satellite reference frame as shown in Figure 4 (corrected from instrument mounting misalignments and scan law errors).

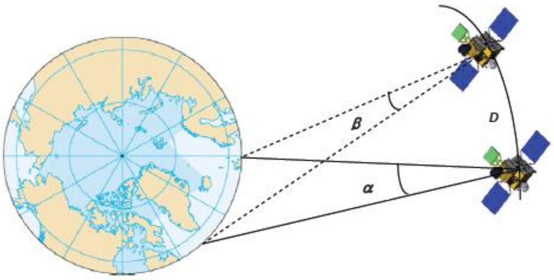

Figure 3: Parallax effect on differential LM view angle

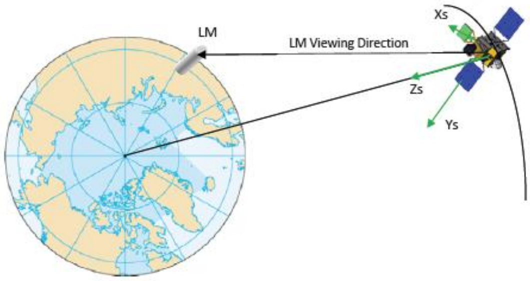

Figure 4: LM view direction in satellite reference frame

In addition, landmarks need to be corrected for any variations in the platform attitude: Attitude errors can be removed from common LM displacement (assuming an average attitude error over the full image): if gyro data are available on-ground, integration of the gyro data and fitting through the available LMs enables to reconstruct accurately the attitude evolution over the image time [6], and to remove its effect from the LMs displacement more accurately. Differential displacement can then be used for orbit determination as explained above.

Similarly, using LMs and star for orbit determination could be considered as well. The displacement between star and earth center is a direct measurement of the orbital error (see Figure 1).

4. STUDY RESULTS OVERVIEW

In order to assess the performance and operational relevance of different orbit and attitude determination concepts several studies (internal and external) at Flight Dynamics and INR level have been performed during MTG Phase A and B. Compliance with mission requirements and operational constraints together with mission costs were taken into account for the assessment.

4.1. Flight Dynamics Studies

In a phase A feasibility study with Astrium, Toulouse [1] it has been demonstrated that a classical tracking system based on S-band ranging can have its performance considerably improved if complemented by regular executions of an image-based orbit determination filter: the combined attitude and orbit determination approach can extract an accurate observation of the longitude error from the image data, and this provides the required observability on the ranging system's bias errors.

Similarly, the achievable performances of the selected attitude determination solutions were computed using a Kalman covariance analysis filter as prototyped by ASTRIUM Toulouse (KalmanSandBox). Causal formulation (only information in the past is used in the estimation process) and fixed lag formulation (also information in a time window in the future is used in the estimation process) were analyzed.

4.2. Orbit determination accuracy results

The main contribution to the position error, for a standard 2-station tracking scenario, is the longitude error. Combining ranging measurements with image data allows improving observability on the longitude error. This in turns results in a much better convergence of the orbit determination process. For comparison, the following table shows the orbit performance and the impact on the geolocation error for a scenario with 2 S-band ranging stations without and with image based data; the availability of LM permits moreover a much finer estimation of the station bias.

Table 1. Orbit tracking system performances

System	Orbit performance [m 1σ]	Impact on geolocation SSP [m 1σ]	Assumptions and considerations
2 stations ranging only	450	70	Large baseline: 3200 km. Measurements every 15 min. Poor estimation of station bias: ~10 m
2 stations ranging and LM	60	9	Large improvement in orbit accuracy. Good estimation of station bias: ~ 2m
1 station ranging and LM	180	30	Good orbit accuracy. Good estimation of station bias: ~ 2m

4.3. Attitude determination accuracy results

A Kalman covariance analysis tool was also used to compute the achievable performances of the selected attitude determination solutions; the evolution of the attitude estimation error in time for different configuration is shown in Figure 5.

As reference solution the one obtained processing only landmarks is presented (blue line); the performances appears to be not satisfactory for all latitudes; the maximum error is reached just after the North Pole, due to landmark unavailability around the Antarctic and Arctic region (scanning from North to South and retrace from South to North); the performance degradation is caused by the random walk of the gyros used to propagate cinematically the attitude and filter the landmark noise.

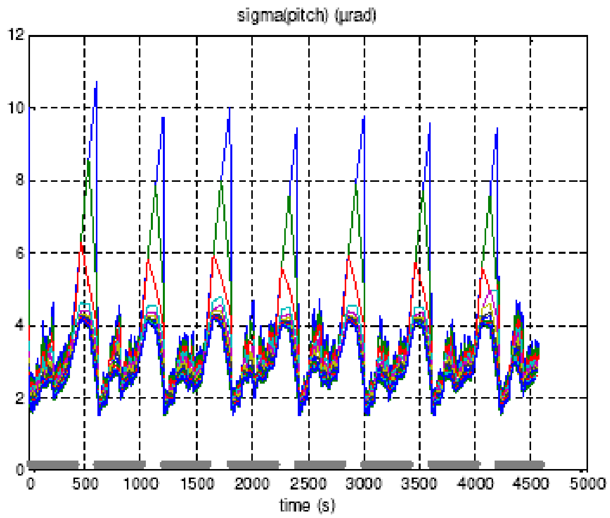

Figure 5: Attitude determination performance enlarging the processing window

In order to improve the system performances additional features are needed:

- An high stability gyro, with limited random walk, to ensure limited divergence in case of landmark gaps;
- Star data, either from star sensors of from star detection in the image, integrated in the processing, to mitigate the effect of the landmark gaps;
- Enhanced landmark detection, to limit the size of the gaps;

- Enlarged processing window (fixed lag), permitting to bridge the solution across two landmarks (the effect in the accuracy obtained is shown in Figure 5).

For comparison, the attitude determination performances and the effect on geolocation for different scenarios (average and max) are presented in the following table: it can be noticed that the addition of classical sensors does not cause any benefit due to their large measurement noise; the benefit of adding star data detected in the image is also limited due to their limited number.

Table 2: Attitude determination system performances

System	Attitude performance [µrad 1σ]	Impact on geolocation error at SSP [m 1σ]
LM alone with standard gyro	5-11	180-400
LM and star trackers	4.5-7	160-250
LM and improved gyro	3.2-3.5	120-130
LM and star imaging (mag. > 3)	5-11	180-400
LM and Earth and Sun sensors	5-11	180-400

To achieve the required attitude accuracy all alignment biases (from launch, aging or time varying thermal distortion) must be calibrated, which makes the use of landmarks and stars within the image data compulsory.

4.4. Orbit Transients after maneuvers

The results above show that accurate orbit determination is feasible if image data are available to enhance the classical S-Band ranging with landmarks. During MTG phase B with the evolution of the system and operation requirements the need of collocation together with the need of ground segment cost optimization created the necessity to investigate the feasibility of a single ranging S-band station for orbit determination.

Several analyses for the feasibility of orbit determination with only one ranging station, including the case of station keeping maneuvers and collocation were performed at EUMETSAT using a full dynamical orbit model of operational flight dynamics software. Detailed results with the main assumptions are presented in another paper presented at this conference [5]. It was confirmed that orbit determination using ranging data from one station in the routine case without maneuver is feasible even without landmarks, provided that sufficiently high ranging data rate (i.e. ranging every 15 minutes) is available, even if a quite large cross-track error remains. For the maneuver case, with regular (i.e. monthly) inclination and longitude station keeping of the order of 0.11m/s to 4m/s respectively, as each maneuver has an impact on the orbit determination performance, a sufficient arc of measurements needs to be performed in order to recover a satisfactory estimation. Even if the usage of auxiliary observables from images improves the orbit accuracy obtained with single station ranging only, 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.

These results have been complemented with a Kalman covariance analysis. A filter was used together with a covariance analysis tool in order to verify the transient time after a 4m/s inclination maneuver. The following can be observed in the pictures below (initial transient linked to the initial orbital error at simulation start):

- with two ranging stations the transient time to recover the orbit accuracy is about 2 hours (Figure 6);
- in case of dual ranging plus landmarks every 30 minutes the error decreases but not the convergence time (Figure 7);

- in case of having only one ranging station and landmarks, the recovery time after a maneuver increases to about 6 hours, as visibility is lost in the cross-track direction (Figure 8); no convergence is achieved at all without landmarks;
- even worse, in case ranging data are not available and only landmarks every 30 minutes are used the radial component increases up to 900m and more than 10 hours are needed to recover the orbit to the initial accuracy (Figure 9).

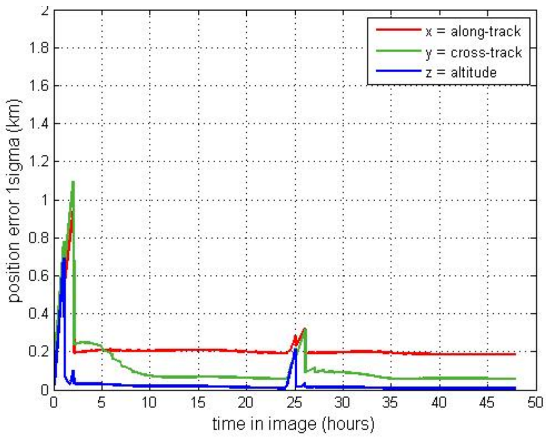

Figure 6: Transient time with dual ranging every hour

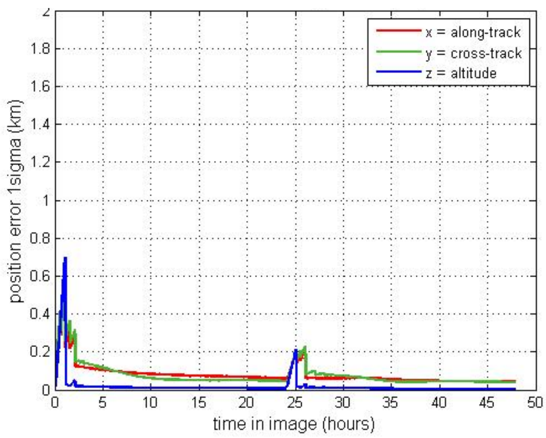

Figure 7: Transient time with dual ranging every hour and LM every 30 minutes

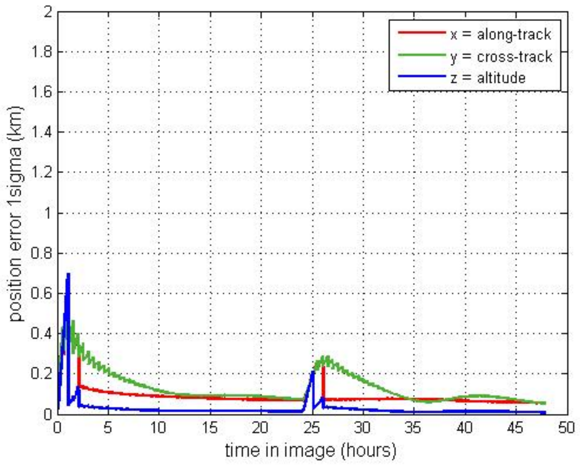

Figure 8: Transient time with 1 station ranging; LM every 30 min

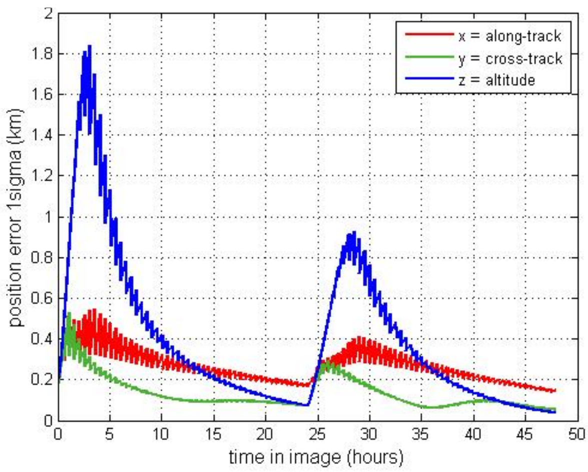

Figure 9: Transient time with no ranging; LM every 30 min

4.5. Flight Dynamics Conclusions on Attitude and Orbit determination processes

For attitude determination, few solutions appear capable of meeting the required performance. The selected candidates are essentially combinations of landmark observations, star trackers and star imaging and gyro:

- Imager: LM + gyro + star tracker (and/or star imaging if available)
- Sounder: star tracker + gyro (and image capabilities if available)

For orbit determination the preferred solution rely on one single S-Band station complemented by landmarks (and star imaging if available). In case of image data not being available, usage of a second ground station to ensure collocation safety and fast convergence after maneuvers is required. The following scheme can be then conceived:

- S-band ranging from 2 stations for initializing INR and when INR data are not available;
- S-band ranging from 1 station combined with image data from INR when INR available.

5. INR STUDIES

An INR study was launched in Phase A with the objective to prove feasibility of the MTG INR requirements [3]. A central part of the study was based on model-driven INR simulations. The combined FD-INR architecture that has been assumed for the simulations is presented in Figure 10 and described in more detail below.

5.1. INR Architecture

An INR system concept, generally applicable to all MTG instruments, but exemplified here for the FCI only, has been developed during the Phase A study. The concept has some affinity to the GOES-N INR system.

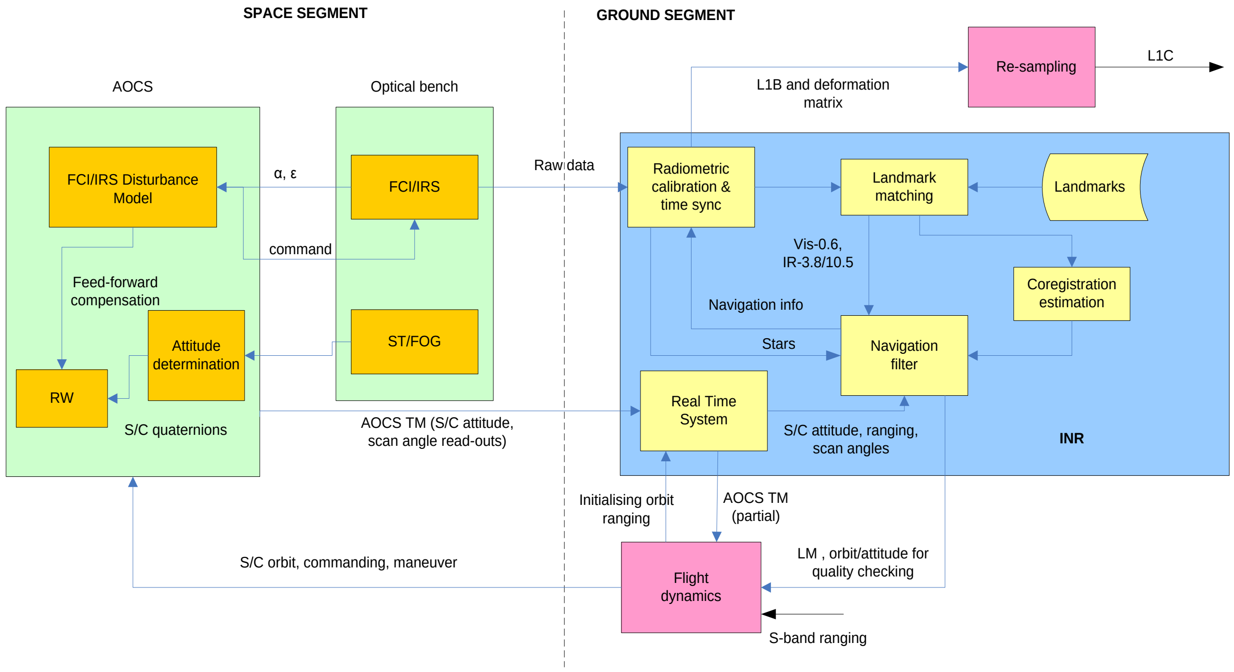

Figure 10: INR and FD system architecture design

The INR system shown in Figure 10 has been chosen for the simulation to prove the feasibility to MTG geolocation requirements. On the satellite, a combination of star tracker and gyroscope are the essential elements to determine the space bus attitude. The star tracker determines an absolute pointing reference while the gyroscope records relative changes in the accelerations. The star tracker uses an on-board star catalogue, which is uploaded to the spacecraft (about every six months). Star tracker and gyro data are inputs to the Attitude Determination unit which calculates the necessary updates of the attitude. A control signal is sent from here to the reaction wheels (RW) to point the spacecraft towards the centre of the earth and correct for bus disturbances. The bandwidth of the control signal is typically significantly smaller than the bandwidth of the signals coming from the ST/Gyro unit, which is providing typically gyroscope measurements of some tenth of Hertz. The reaction wheels control can also take the anticipated instrument perturbation from the scanning movement into account, which is indicated in Fig.12 by the FCI Disturbance Model and its feed-forward compensation to the reaction wheels. The scanning mirror of the instrument does not get any compensatory signals in this architecture, but the scan angles in azimuth and elevation are provided to the FCI Disturbance Model. Therefore any high-frequency perturbations, which may affect the LOS stability has to be minimized by design (e.g. prevent micro-vibration where possible, using dampening devices, suitable material choices). It is further assumed that the instrument and the star tracker/gyro unit are placed in such a way that thermal distortion between the two are minimized (e.g. by placing both instrument and ST/gyro on the same optical bench).

Raw imagery is down-linked to the Ground Segment. In addition an AOCS telemetry stream is provided (possibly embedded in the raw imagery). The latter must at least provide the scan angle readouts, but AOCS telemetry of the S/C attitude is also part of the proposed baseline. The spacecraft attitude could be provided as quaternion or in form of gyroscope and star-tracker measurements.

The propulsion system used for platform maneuvers, i.e. station keeping and momentum unloading, is based on chemical thrusters only.

Ranging measurements are important for orbit localization and therefore part of the set of observations used in the INR component. The INR component receives these measurements – possibly corrected for atmospheric delays – via the FD component. The ranging is coming from a single S-band ranging station as baseline in the reference architecture. This has also been the baseline in the INR MTG study simulation.

Landmark and star observations in images are extracted on ground and together with the corresponding scan angles and tracking/ranging data from the ground station fed into a Navigation Filter, which maintains a model of the LOS for each instrument. The observations are used to update the adjustable LOS model parameters, i.e. the so called state vector. The estimated state vector is then distributed to the re-sampling function of the instrument data processing. Landmark observations are also sent to the Flight Dynamics component to improve the FD orbit determination. The FD orbit is uploaded to the spacecraft by FD, where it is used to steer the spacecraft to the earth centre.

5.2. Determination of INR budgets

The main contributors to the pixel localization errors can be summarized as:

- Orbital position knowledge error (primarily along-track and cross-track)
- Platform attitude knowledge error (mainly pitch and roll)
- Instrument scan errors (scan repeatability)
- Thermal deformation (between instrument and attitude sensors)
- Micro-vibrations (from reaction wheels, solar panel, coolers)

If we assume an allocation of a fifth of the total error budget of 250m to each of the contributors this result in an orbit position knowledge error requirement of 50m and an attitude knowledge error requirement of 1.4 μrad.

Assumptions made in the simulations on sensor hardware and error budgets from the identified contributors are detailed below.

5.2.1. Orbit knowledge error budget

S-band ranging measurements are provided with a frequency of 4 per hour; antenna angle tracking measurements are not used. Ranges are simulated with a bias of 5m and white noise of 10m (3σ).

5.2.2. Attitude knowledge error budget

The sensor combination in the simulations included two possible configurations: the HYDRA SED36 star tracker (ST) with either the FOG 120 HR or 200 HR. Both sensor configurations are expected to have flight heritage in time for MTG. An analysis of the two configurations was conducted offline using the method described in [4]. The analysis took into account the ST and FOG noise parameters, ST orientation, and the ST update rate. The results showed that even when the FOG's angular random walk (ARW) is reduced by a factor of 10 (as is the case between the 120 and 200 HR), the pointing errors are only reduced by a factor of 2. The pitch axis is worse because the ST boresight must be inclined away from the equatorial plane to avoid intrusions. Table 3 shows only the conservative configuration (based on 120 HR) and this also applies to the reaction wheel jitter where it was assumed that current ball bearing technology would be used.

Table 3: Noise contributors for the AOCS hardware

Error Component	Error (3σ)	Description
Star Tracker Noise (Based on 3 head Hydra-SED36)	42.7 μrad	RSS of Low spatial frequency, high spatial frequency, and temporal noise within FOV; assuming 12 stars; along BS error>200 μrad)
Startracker boresight misalignment	20 μrad	Due to thermal deformation of ST (does not include mounting bracket)
FOG 120 HR ARW	0.0017 deg/rt(hr)	Angular random walk
FOG Bias stability	0.002 deg/(hr)	@1sec
FOG AWN	0.2 deg/(hr)	Angle white noise
Scale factor stability	360 ppm	-
Reaction wheel jitter (with ball bearings)	6.1 μrad	Pointing error due to RSS of wheel noise (static and dynamic imbalance, friction, bearing noise, μ-vibrations)
Solar Array Jitter	7.5 μrad	Pointing error induced by stepping SA
Scan mirror disturbance	1.0 μrad	@0.1HZ step rate; bus depointing for pitch axis; roll/yaw is 0.5 μrad
Stability of onboard clock	10 μs (bias) 1 ms (drift)	Bias and drift (after cal)

The ST noise includes star catalog error, low spatial frequency errors, and high spatial frequency errors. To a large extent the on-board filter will reduce these error contributors as was shown in the analysis. The ST misalignment error is assumed to be residual error after calibration. The scan mirror disturbance after compensation to the bus controller is assumed to be 1 μrad for the pitch axis and 0.5 μrad for the roll/yaw axes. In addition to scan mirror disturbances, reaction wheel, coolers and solar array jitter are the main contributors to micro-vibration and therefore included in the AOCS error budget.

5.2.3. Thermal error budget

Thermal deformation is the dominant but deterministic error that is caused by a deformation to the scan mirror and telescope optics. The scan mechanism, by contrast, introduces a smaller but far less deterministic error. In accounting for optical thermal deformation, a model has been developed from an analysis of GOES-N flight data. It is recognized that the actual FCI/IRS deformation will perhaps be an improvement over GOES-N due to advances in mirror construction and thermal management. Therefore, while preserving the diurnal trend, the amplitude was multiplied by a factor of 0.75. The thermal deformation profile is modeled in terms of five angles: three Euler angles for the scan mirror and two misalignments for the telescope. The profiles are strongly influenced by the location of the sun with respect to instrument boresight: around midnight Spacecraft Local Time (SLT), the pointing errors undergo significant changes. The profiles will experience daily and seasonal non-repeatable errors. To account for this in the simulations, we consider a daily non-repeatable error of 5 μrad. The off-axis components (yaw, roll misalignment, pitch misalignment) are scaled by approximately 1/6 in their impact to the LOS. In order to maintain INR specification, special ground processing will be required to mitigate the LOS errors due to thermal deformation. The peak-to-peak thermal deformation error budget (i.e. peak errors in

LOS depointing for roll/pitch) has been set to 450 μrad for nominal situations (i.e. outside of eclipse) and an additional 250 μrad for eclipse transients.

5.2.4. Instrument scanning errors

The scan mechanism errors are more difficult to account for. An optical encoder was assumed to be used for controlling and determining the scan angles. Since the GOES scan system relied upon an inductosyn, it will not be possible to utilize flight data in this case to develop an error model. Instead the simulation relied upon currently available sensors. In general, the encoder errors are due to instrument error, quadrature error, and interpolation error. Instrument and interpolation are the dominant errors. Instrument errors tend to be periodic and are the sum of disc pattern error, disc eccentricity, and bearing runout. Quadrature error depends of phasing and duty cycle. Interpolation error occurs when the resolution has been increased by electric interpolation. The summarized RSS budget for all encoder errors was assumed to be 10 μrad (3σ).

5.3. INR Processing

The orbit and instrument attitude state is determined from the INR measurements (star observations, landmarks, and ranges) by an estimation filter. In the performance simulations it is assumed that the update occurs once per day (i.e. batch filter), although more frequent updates are possible as needed to maintain pointing specification (e.g. Kalman filter).

In nominal situations visible landmarks and star observations are obtained from the Vis-0.6 channel. The IR landmarks are obtained from the IR-3.8. The Vis landmark acquisition rate is on the order of 15 landmarks per hour (a very conservative figure) and with a measurement accuracy of 0.5 SSD (a very conservative figure). The sun angle threshold for Vis landmarks illumination is 74deg.

The difference between the IR and Vis residuals are due to the measurement error in the landmark matching process (IR measurement error is a factor 2x larger than Vis due to SSD) and the co-registration error, which is reduced only slightly during the estimation process. The star measurement error is equivalent to the Vis landmark measurement error.

5.4. INR Performance

The Absolute Pixel Position Knowledge Error (APPKE) performance metrics is shown in Figure 10. The time series begins on day 2 of the simulation; the first day is removed because it takes at least one day for characterization and INR startup. The error for 99.73% of the samples must be below the specification value for an image to be considered meeting specifications. APPKE specification of 21 μrad (east-west and north-south) is just being met in the east-west and has 2 μrad of margin in the north-south. The Relative Pixel Position Knowledge Error (RPPKE, not shown) is also within specification of 29.4 μrad.

The Phase A study results showed that even with pessimistic assumptions, MTG geolocation requirement will be met given an INR architecture as outlined in Figure 10 and hardware specifications and error budgets as shown previously. In fact, the assumptions on the accuracies of observables are conservative and will probably be better.

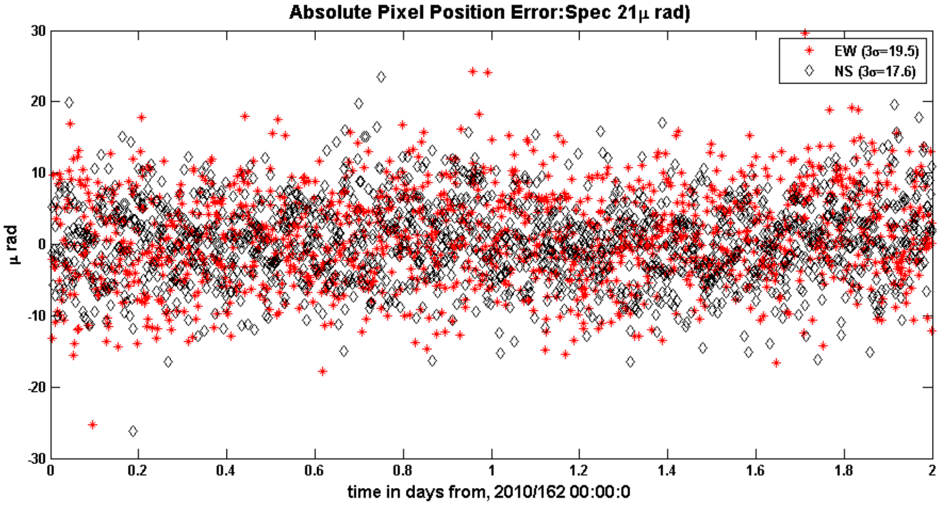

Figure 11: APPKE error in EW and NS

One important result from the INR study is the critical role of star observations in order to separate orbit and attitude errors unambiguously. When not considering star observations, orbit errors increase by a factor of 3 after the second navigation process has completed as shown in Figure 12, which shows along track (AT) and cross track errors (CT). AT error does not increase significantly because of the single-station range measurements provide excellent AT information. However, the large CT error increase is due to the fact that, without star observations, the navigation filter is not able to fully decouple the orbit and attitude from landmarks. The attitude errors increase by a factor of 2 when star observations are removed (not shown). ,

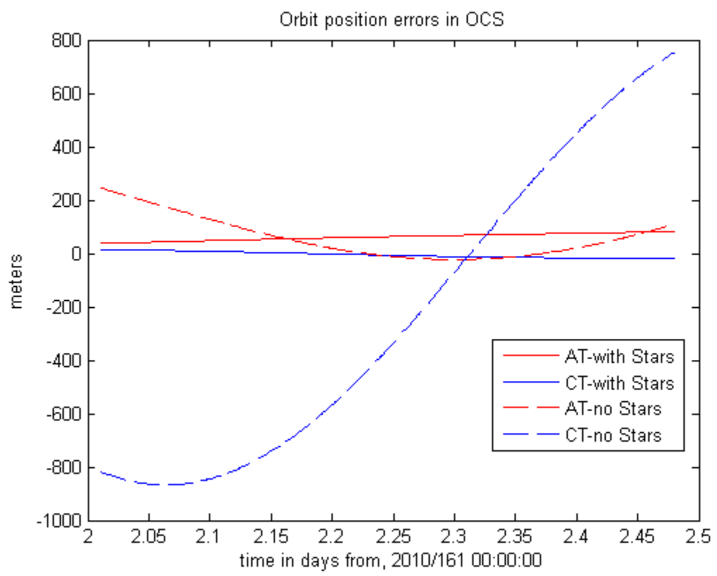

Figure 12: Cross-track and Along-track errors with and without stars

Further, improved frequency and accuracy of landmark determination (e.g. using better a landmark database) can greatly enhance INR capabilities. Therefore a revision of the landmark detection scheme as used in MSG will be considered for MTG.

6. IMPACT ON SYSTEM ARCHITECTURE DESIGN

The actual performance in respect to any budget allocation is strongly depending on the available observables (e.g. ranging, landmarks, star observations, etc.) and their measurement accuracy, which drive the overall INR architecture. An important driver comes also from operational requirements to keep the platforms under safe control even in the absence of instrument data (thus no INR but only FD is available).

The range-based orbit determination and on-board attitude control are designed to fulfill the platform requirements, but are not accurate enough to meet the more demanding pointing accuracy requirements of the instruments and geo-location accuracy requirements of the derived products.

For these, a refined knowledge of the instrument attitude and orbit is required and implemented on ground based on instrument observations and estimation techniques as part of the Image Navigation and Registration (INR) processing. Observations acquired by the instrument, i.e. landmarks and star observations in the image, together with inertial measurements are used to estimate both the attitude (of platform and instrument) and the orbit.

To ensure flight dynamics operations covering all mission scenarios including when no image data available, an orbit and attitude determination process independently of the image navigation process need to be implemented. The flight dynamics system needs to work complementary and independently of the image navigation in order to ensure satellite survival in case of payload switch off.

The ground segment system architectural design will be optimized in order to ensure these complementary functionalities are useable independently or jointly while minimizing duplications between the Flight Dynamics (FD) and the Image Navigation and Registration (INR) subsystems.

7. CONCLUSIONS AND RECOMMENDATIONS

The basic orbit and attitude estimation accuracy performance that can be obtained with standard tracking system and on-board attitude sensors is insufficient for meeting the image navigation requirements of MTG. The performance of this basic approach is therefore complemented by a simultaneous INR processing based on instrument observations. The derived performance estimates from simulations indicate that the combined approach is sufficient for fulfilling the MTG INR and operational requirements. These EUMETSAT assumptions are in-line with respect to the latest available reference approach provided by the Space Segment Manufacturer.

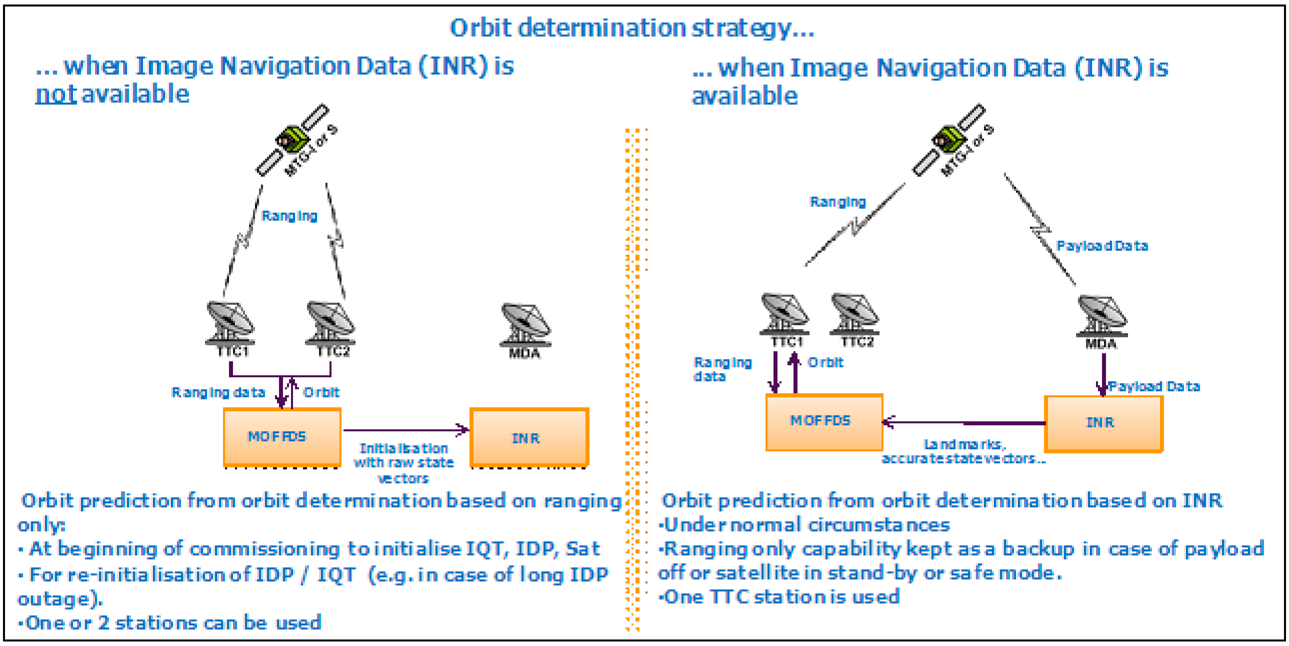

Figure 13: Orbit Determination with & without INR

The approach adopted splits the MTG navigation filter architecture in two parts as shown in Figure 13: Flight Dynamics (essentially based on ranging and image data if available) and INR (based on instrument observations and ranging provided from Flight Dynamics). There is exchange of information between the two elements, but each of them will be designed to work independently from the other.

The stringent accuracy requirements for instrument data navigation and rectification cannot be satisfied by traditional ranging measurement based orbit determination systems. This is therefore performed directly by the INR using instrument data. Attitude determination for image data navigation and rectification is performed within the Image Data Processing function by matching known landmarks to the instrument data. Planned disruptions to the satellites orbit, e.g. due to maneuvers, are notified to INR in advance.

8. References

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