NEW EUMETSAT POLAR SYSTEM ATTITUDE MONITORING SOFTWARE

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Abstract: *The original attitude monitoring software of the EUMETSAT Polar System Flight Dynamics Facility allowed the computation of mean attitude misalignment and AOCS sensor biases in a given processing time, assuming a nominal attitude law with constant misalignments. This system has been enhanced, computing the actual attitude evolution from the angular velocity reconstructed from the gyro data and estimating all the attitude and sensor parameters with a batch least square method based on the observations provided by the on-board optical sensors. The resultant attitude evolution can be propagated after a spectral analysis of the misalignment evolution through a Fast Fourier Transform. This software has been integrated in the Metop Flight Dynamics System and it is currently being used in routine operations to estimate the satellite short term attitude evolution (only one orbit of gyro data are daily available) providing a very useful information on the pointing stability of the platform.*

Keywords: *Attitude, software, Metop, EUMETSAT.*

1. Introduction

Metop-A is the first operational European meteorological satellite collocated in a Low Earth Orbit (LEO). Based on a three axis stabilised platform developed by Astrium for observation spacecrafts, it is the first of the three satellites of the EUMETSAT Polar System (EPS).

Metop's nominal attitude law is a Local Normal Pointing, corrected by a Yaw Steering Mode that compensates the apparent drift of the sub-satellite point due to the Earth's rotation. The on board Attitude Control System is designed in closed loop and, therefore, autonomous from the Ground Segment. Equipped with Digital Earth Sensors, Digital Sun Sensors and gyroscopes, the S/C downlinks the sensor telemetry for monitoring purposes on ground.

The Flight Dynamics Facility (FDF) was developed by GMV based on the ESA's Navigation Package for Earth Observation Satellites (NAPEOS). It processes the sensor telemetry to ensure that the nominal attitude law is kept within certain thresholds. Even if no prediction is actually required to the system, the attitude evolution has to be used for the computation of the instrument visibilities of different Earth regions, points or satellites.

The original attitude monitoring software assumed that the actual evolution of the satellite attitude only differed from the nominal law in some constant misalignments. With this assumption, an iterative batch estimation is performed to obtain the sensor biases, gyro drifts and attitude misalignments that minimise the observation residuals processed from the telemetry. Although this method provides good results for the mean sensor biases and misalignments of the platform, it does not allow to analyse the platform stability or to obtain the actual pointing of the satellite at a given time.

2. New Attitude Monitoring Software.

The limitations of the original attitude monitoring system have to be solved by the new one, and several technical solutions were analysed.

The first requirement of the software was to be able to compute the spacecraft attitude evolution from an initial state. Being the attitude controlled on board, a dynamical propagation of the attitude was quite complex to implement, as it would require a precise modelling of the on board control system, the duplication on ground of the inertia wheels laws and a precise knowledge of the platform inertia matrix.

Having the gyroscope telemetry available from the satellite, it was possible to compute the actual angular rates of the S/C, including any dynamic perturbation and allowing the direct integration of the kinematic equations [1]. This method, known as *gyro modelling*, significantly simplifies the attitude computation with respect to the dynamic problem.

However, this method is limited by the gyro data. The attitude can only be integrated while this telemetry is available, which would impose a heavy constrain in the satellite operations and could not be used to propagate the attitude into the future.

This problem has been solved performing a Fast Fourier Transform (FFT) on the difference between the integrated attitude and the nominal law. Assuming an orbital periodicity of the attitude evolution and truncating the FFT results, it is possible to propagate the satellite attitude out of the integration intervals.

The integrated attitude with the gyro modelling method depends on several parameters, the initial attitude state and the gyro drifts used to correct the input telemetry. In order to optimise this parameter, a determination process has to be performed, based on the observations provided by the optical sensors.

Even if most of the attitude determination systems are based on a Kalman filter, the new software uses a batch least square estimation. The Metop's Digital Sun Sensor provides one single observation per orbit and, in routine operations, the telemetry of two single gyros is received on ground. As a combination of these two factors, the observability of the yaw pointing of the spacecraft is very limited, so a Kalman filter could have a lot of convergence problems.

The parameters that can be estimated have also improved from the original attitude monitoring software. While the old system was only able to compute the mean values of gyro drifts and optical sensor biases, the new SW allows modelling any of these parameters as a linear evolution between two states at any given time.

2.1. Algorithms and Processing

The gyro telemetry can be pre-processed to extract the angular velocity of the spacecraft in the two gyro axes perpendicular to their own rotation axis. Two gyros are required at every time to obtain the spacecraft angular rates, but the four measurements have to be reduced to get the angular velocity in three axes. This conversion is performed through an ordinary least squares method, applying the Gauss pseudo-inverse.

$$\boldsymbol{\omega} = [(\boldsymbol{H^T} \cdot \boldsymbol{H})^{-\mathbf{1}} \cdot \boldsymbol{H^T}][\boldsymbol{M} - \boldsymbol{G}] \tag{1}$$

Where $\boldsymbol{\omega}$ is the satellite angular velocity, $\boldsymbol{H}$ represents the conversion matrix from gyro axes to S/C frame (4x3 matrix), $\boldsymbol{M}$ is the vector containing the gyro measurements of the angular velocity and $\boldsymbol{G}$ the gyro drift.

The integration of this angular velocity to obtain the attitude is performed through the quaternion ($\boldsymbol{q}$) representation of the satellite pointing, with the following equations [1], [2]:

$$\frac{d\boldsymbol{q}}{dt} = \dot{\boldsymbol{q}} = \frac{1}{2}\boldsymbol{\Omega} \cdot \boldsymbol{q} \tag{2}$$

With $\boldsymbol{\Omega}$ being the extended angular velocity matrix computed as follows:

$$\boldsymbol{\Omega} = \begin{bmatrix} 0 & \omega_3 & -\omega_2 & \omega_1 \\ -\omega_3 & 0 & \omega_1 & \omega_2 \\ \omega_2 & -\omega_1 & 0 & \omega_3 \\ -\omega_1 & -\omega_2 & -\omega_3 & 0 \end{bmatrix} \tag{3}$$

The numerical integration of the angular velocity has to be performed through a method adapted to the data characteristics: Only a discrete number of values are known from the incoming telemetry, and there is no information of any function derivative (through it could be numerically computed). Furthermore, small gaps in the data have to be managed to prevent errors during the integration.

The Samphine-Gordon method has been chosen for this integration [3]. This method is a variable-step size PECE (predictor-evaluate-corrector-evaluate) method based on the integration of the interpolating polynomial fitted with the input data. The order of the corrector polynomial is one degree higher than the predictor.

Equation 1 shows how the gyro drift affect the angular velocity, and therefore, the whole attitude evolution. The optical sensors can provide the information on the optimal values of these drifts, together with the initial attitude state and their own sensor biases.

The batch estimation process is, therefore, prepared to estimate a set of n parameters $\boldsymbol{X}$¸ consisting in the initial attitude state, $\boldsymbol{\alpha_0}$; the gyro drifts, $\boldsymbol{G}$, and the optical sensor bias parameters. The iterative method can be summarised in this equation [1]:

$$[\boldsymbol{P_0}^{-1} + \boldsymbol{F}_k^T \cdot \boldsymbol{Q_0}^{-1} \cdot \boldsymbol{F}_k] \cdot (\boldsymbol{X}_{k+1} - \boldsymbol{X}_k) = \boldsymbol{F}_k^T \cdot \boldsymbol{Q_0}^{-1} \cdot \Delta\boldsymbol{y}_k + \boldsymbol{P_0}^{-1} \cdot (\boldsymbol{X}_0 - \boldsymbol{X}_k) \tag{4}$$

Being $\boldsymbol{Q_0}$ the covariance matrix of the observations, $\boldsymbol{P_0}$ the covariance matrix of the parameters, $\boldsymbol{\Delta y}$ the observation residuals and the matrix $\boldsymbol{F}$ at the iteration number k containing the partials of the observation functions with respect to the estimated parameters:

$$\boldsymbol{F}_k = \begin{bmatrix} \frac{\partial f_1}{\partial x_1} & \frac{\partial f_1}{\partial x_2} & \cdots & \frac{\partial f_1}{\partial x_n} \\ \frac{\partial f_2}{\partial x_1} & \frac{\partial f_2}{\partial x_2} & \cdots & \frac{\partial f_2}{\partial x_n} \\ \vdots & \vdots & \ddots & \vdots \\ \frac{\partial f_m}{\partial x_1} & \frac{\partial f_m}{\partial x_2} & \cdots & \frac{\partial f_m}{\partial x_n} \end{bmatrix} \tag{5}$$

The nominal observations can be computed as a purely geometrical problem, since the input sensor telemetry is being pre-processed to extract the significant physical values from the incoming data, in order to simplify the determination process with simpler models of the observation functions. The partials being computed depend on the parameter x_i being estimated. The previous system used constant attitude biases, so the new monitoring software has included all the dependencies on the time and the variable attitude, implementing these new algorithms.

The derivatives of the observations functions with respect to the initial attitude misalignment are computed with the following equation:

$$\frac{\partial f}{\partial \boldsymbol{\alpha_0}} = \frac{\partial f}{\partial \boldsymbol{\alpha}} \cdot \frac{d\boldsymbol{\alpha}}{d\boldsymbol{\alpha_0}} \tag{6}$$

With the derivative of the attitude with respect to the initial state computed with:

$$\frac{d\boldsymbol{\alpha}}{d\boldsymbol{\alpha_0}} = \frac{d\boldsymbol{\alpha}}{d\boldsymbol{q}_k} \cdot \left(\prod_{i=1}^{k} \frac{d\boldsymbol{q}_i}{d\boldsymbol{q}_{i-1}}\right) \cdot \frac{d\boldsymbol{q_o}}{d\boldsymbol{\alpha_0}} \tag{7}$$

The Jacobean matrix of the transformation between quaternion and attitude misalignments is numerically computed at any required time. The partial of the observations with respect to the attitude at a given instant ($\delta f/\delta\alpha$ in Eq. 6) can be computed geometrically, since the observations have been previously preprocessed to reduce the input data to geometrical angles directly dependant on the position and attitude. Finally, the derivative of a quaternion with respect to the previous one is integrated during the same process of attitude integration, and it is computed by making a linear approximation of the Eq. 2, resulting:

$$\frac{d\boldsymbol{q}_{i+1}}{d\boldsymbol{q}_i} = \boldsymbol{I_4} + \frac{1}{2} \cdot \frac{d(\boldsymbol{\Omega}\cdot\boldsymbol{q}_i)}{d\boldsymbol{q}_i} \cdot \Delta t \tag{8}$$

Being $\boldsymbol{I_4}$ a unitary matrix of 4th order.
The estimation of the gyro drift requires the computation of the derivative of the observation functions with respect to the related parameters at a given time.

$$\frac{df}{d\boldsymbol{G}} = \left.\frac{\partial f}{\partial \boldsymbol{G}}\right|_{\alpha=K} + \frac{\partial f}{\partial \boldsymbol{\alpha}} \cdot \frac{d\boldsymbol{\alpha}}{d\boldsymbol{G}} \tag{9}$$

The derivative of the attitude with respect to the gyro drift ($\delta\alpha/\delta G$ in Eq. 9) is computed numerically by integrating the attitude in parallel with a small offset in the required parameter. This integration is performed simultaneously with the integration of the other partials and the main angular velocity with the Samphine-Gordon method. The derivatives at constant attitude (the first addend in Eq. 9) were already computed by the previous version of the system. The partial of the observations with respect to the optical sensor bias is a linear one to one problem already solved in the original system assuming constant attitude.

Since the optical sensor bias and gyro drifts can be defined through a segmented linear function, the parameters being computed are the values of these function at a given time. The lever rule has to be applied on the partials to take into account the derivative of the function over the time.

2.2. Software design and data flow

One of the drivers of the EPS Flight Dynamics Facility design is the modularisation of the software. The different parts of the system can be executed separately, even if the high level modules make use of the others. Another important characteristic of the system is the high degree of configurability allowed to the user. Figure 1 shows the data flow and the high level design of the new attitude monitoring system.

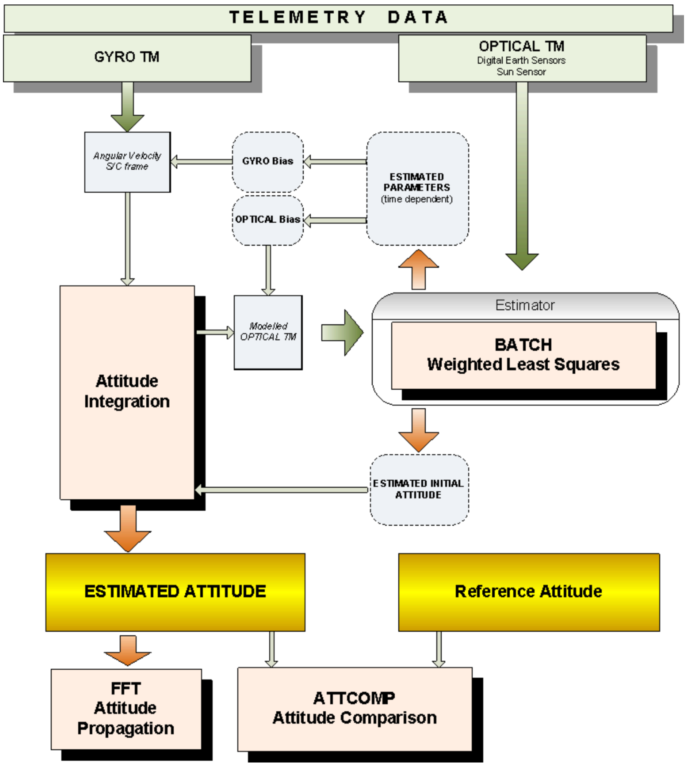

Figure 1. System design

The input telemetry data is pre-processed applying the calibration to every sensor and extracting the relevant physical information from each sensor measurements, i.e. angular velocity in each gyro axis, off pointing of the Earth centre in the Earth sensor and angle between the sun and the Digital Sun Sensor field of view axis. This per-processing has been also enhanced during the implementation of the attitude determination system, applying the infra-red correction to the observations received from the Digital Earth Sensor.

The computation of these observations from the S/C attitude and orbit is merely a geometrical problem that can be easily solved and will provide the modelled observations for the determination problem.

The gyro angular velocity is corrected with the estimated gyro drifts to compute the spacecraft angular velocity, which is integrated from an initial attitude state to generate an attitude arc. The module in charge of this integration process has been called ATTINT.

The implementation allows the user to define the degree of the interpolating polynomial and the maximum number of corrector steps to be performed by the Shampine-Gordon method. These options allows the operators to select the integration method that best fits the input data and the processing-time requirements, from a simple Euler integration (constant angular velocity without corrector steps) to complex 8th degree polynomials with several corrections every integration step.

The data from the optical sensor are used as observations to feed a batch least squares estimator. The parameters being computed at this point are the gyro drifts and initial attitude angles, which fully define the attitude integration, and the biases of the sensors, to correct systematic errors in the input observations. In case of more than two gyros providing data simultaneously, the angular velocities not being used for the integration of the spacecraft attitude can be used as additional observations to feed the estimator, together with the optical sensor data. The complete loop of the batch estimation is included in the ATTDET module.

At this level, the user is allowed to modify numerous parameters. The number of parameters to be estimated, or just considered in the computation, is highly configurable, as their evolution on the time as constant or linear evolution in different intervals. The data has to be characterised by the user, configuring the expected deviation from the nominal values, so it can be properly used in the batch process. Furthermore, the convergence criteria can be modified in ATTDET.

The differences of the integrated attitude with respect to the nominal attitude law are analyzed in the FTATTPROP module, generating a Fast Fourier Transform of these differences. Assuming an orbital periodicity of the attitude evolution it is possible to propagate the integrated attitude into the future. Actually, the period being analyzed (and repeated) can be configured by the user, but the maximum amount of data being provided by the satellite at a time is limited to a single orbit.

In the previous software (ATTMON), the attitude was modelled as a nominal law with constant misalignments so the differences between two attitude arcs for the same orbit were always introduced by different biases or a time shift. Since the attitude variation is now given by the evolution of the gyro angular velocity, it has been necessary to implement a new module (ATTCOMP) that permit the comparison between two attitude arcs.

3. Operational approach

3.1 Analysis of Attitude Estimation Process

The main limitation of this method is currently imposed by the amount of data provided by the satellite. A maximum of one orbit of continuous data is provided by the system; during the other 13 orbits of the day the instrument telemetry is downloaded instead of the sensor data, since more critical for the mission. Within that orbit no more than one Digital Sun Sensor observation is present and sometimes ever none, due to the very low sampling of this TM point (1/16Hz) and being the duration of the sun visibility normally very short (around 12 seconds); the absence of sun measurements limits significantly the observability of the yaw angle during the process.

With this limitation, the yaw bias has to be estimated solely through the gyro data (see Eq. 7 and Eq. 8), due to the projection of the orbital angular velocity (in the out-of plane direction) in the Y satellite axis (along the orbital velocity for zero yaw) caused by a rotation in yaw (Fig . 2). Having at least two observations of the yaw angle would improve significantly the accuracy of the system, since they would be used to pivot the whole solution around the yaw values obtained from them, by increasing the weight of this two observations in the batch determination process.

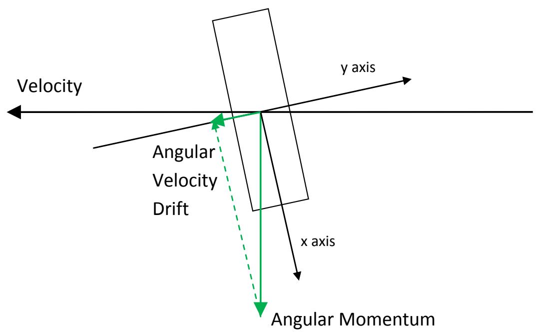

Figure 2. Impact of yaw angle on angular velocity drift

The direct integration of the gyro data with initial attitude bias and gyro drifts set to zero (Fig. 3) returns values close to the expected nominal evolution of the attitude, which proves the validity of the method for the attitude computation of the attitude evolution. Nevertheless, the differences with respect to the nominal attitude are higher than expected from the on board control system and present a monotonic divergence, which can only be caused by some initial misalignment of the platform or by non considered gyro drifts.

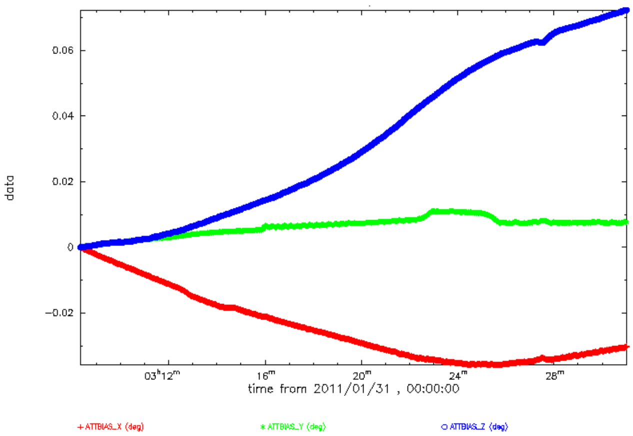

Figure 3. Difference between integrated attitude and nominal pointing

The non-linearity of the attitude problem increases significantly the number of iterations required for the convergence of the determination process. In order to improve the performances of the system, the original simplified monitoring system, which assumes constant attitude misalignments and sensor biases, is executed. This method is much faster than the new implementation, since there is no need to integrate the attitude and the partials of the parameters. The results of the simplified method are used as a-priori values for the determination based on the gyro angular velocities.

The attitude determination system can be used to estimate all the parameters simultaneously. By assigning the expected standard deviations to the parameters being estimated in order to define an a-priori covariance matrix it is possible to determine all the parameters in a single execution. The attitude variations introduced by the integration of the gyros allow the discrimination of the optical sensor bias (fixed in time) and the platform misalignments (changing in time), which was impossible with the previous system.

The output obtained from the determination process (Fig. 4) gives a maximum platform misalignment of around 0.05 degrees in the yaw direction (the one with worst control on-board, as not affecting the sub-satellite-point geolocation budget), around 0.03 in the pitch direction (a large pitch angular velocity is commanded to compensate for the orbital pulsation, increasing the controller error) and around 0.01 in the roll-direction (the best controlled axis). These results are consistent with the data obtained from the on board attitude control system and with the analysis of the geolocation from the mission instruments.

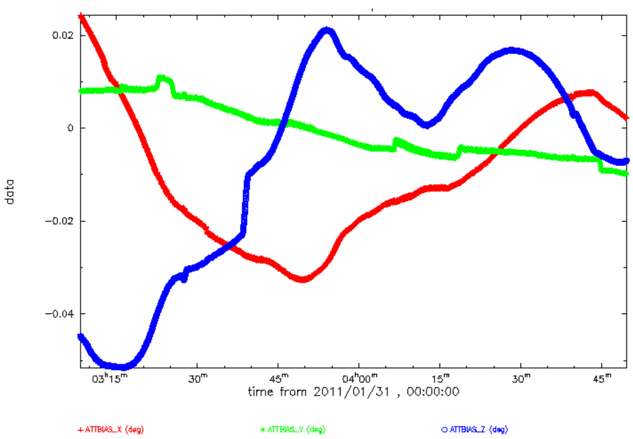

Figure 4. Difference between estimated attitude and nominal pointing

This value seems to be in the limit of the accuracy that can be obtained with the current sensors. The residuals obtained for the Earth sensor (Fig. 5) have the similar order of magnitude than the infra-red correction applied to this sensor (described in detail in next section, Fig. 8) and of the observed errors in pitch and roll. A residual signal is however visible, above all for the X component, which may be responsible to the somehow large evolution in pitch reported above.

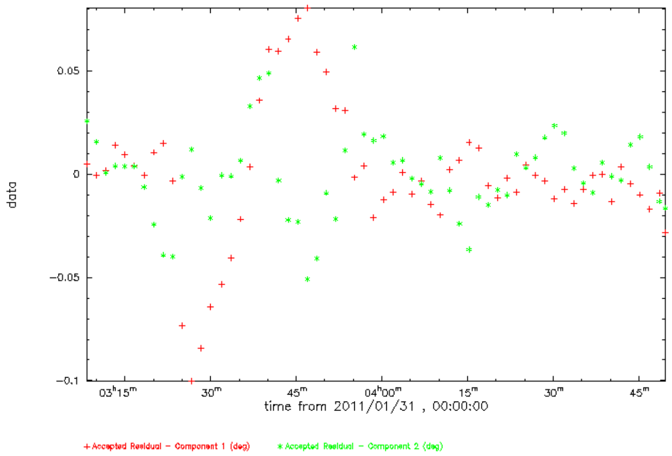

Figure 5. Digital Earth Sensor residuals

The differences of the angular velocity (shown in Fig. 6) with respect to the expected values following the nominal attitude law are close to the gyroscope maximum accuracy that can be obtained due to the data definition (truncation error), so part of the variations in the attitude would be introduced by the errors in the angular velocity inherited from the gyro data.

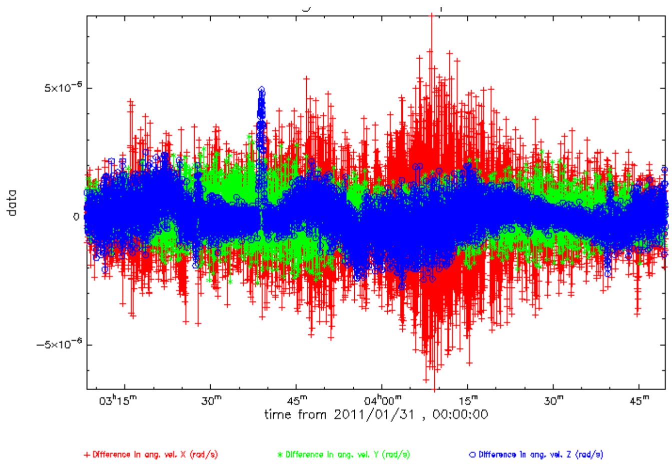

Figure 6. Difference between measured and nominal angular velocity

In order to mitigate this effect, the system includes an option in the preprocessing modules to filter the gyro data, extracting a normal point per second, and therefore, reducing the frequency of the input data from the nominal 8 Hz. Applying this option (as done when computing the attitude solution shown in Fig. 4) softens the evolution of the satellite attitude, but the frequencies above 1 Hz are filtered in the process.

It is interesting to observe in Fig. 6 the peak of angular velocity in Z occurring around 40 minutes after the start of the data; that peak correspond to the response of the platform to a detection of a misalignment in yaw observed by the Digital Sun Sensor. If Fig. 4 it can be observed how the attitude converges back to the nominal attitude in yaw afterwards; that shows that the estimated yaw evolution is in line with the expected behaviour of the attitude control loop.

3.2 Automatic Attitude Estimation

In the last months EUMETSAT is performing automatically attitude determination based on the full orbit of sensor data received daily from the satellite; these data are collected on the first full orbit having ascending node after 2:00 AM. For the remaining orbits, only 10 days of data are collected. Being the Flight Dynamic Scheduler linked not to ascending node time but to the Svalbard station visibility times and being the data downloaded in X-band with up to one orbit delay with respect to their collection on board (download happening only on Svalbard), it is necessary to properly configure both the Scheduler and the ATTDET (attitude determination) module to set the time of start of the integration before the start of the full orbit of sensor data. Automatic adjustment of the start of the integration to the first TM point found after the configured time permits to ensure proper synchronisation.

To initialise correctly the value of the mean attitude bias and of the gyro drift the ATTMON (attitude monitoring) module is executed on nominal attitude beforehand on the full set of data received in the day (being the assumed attitude the nominal one, that module is not sensitive to data gap). The estimated parameters are automatically set as a-priori parameters for the ATTDET module.

In both ATTMON and ATTDET modules the Digital Earth sensor data are compensated for infrared correction. The effect of the infrared can be observed in Fig. 7; on 24/11/2010 the infrared correction was activated, causing a clear stabilisation in the earth sensor biases estimated by ATTMON and a large reduction of the mean bias.

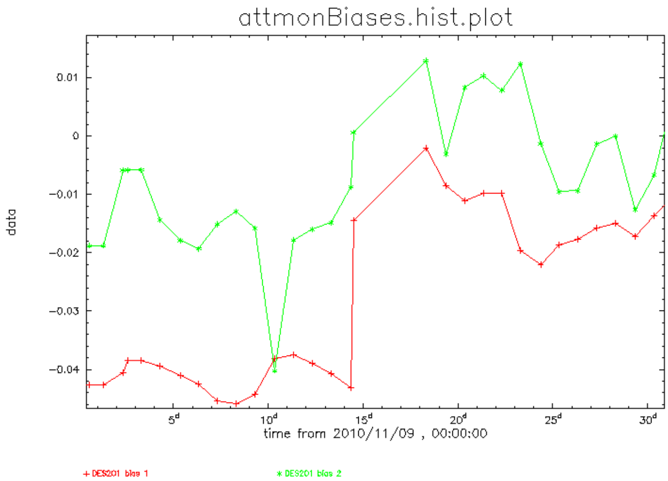

Figure 7. Impact of IR correction on estimated Earth Sensor biases

The infrared correction causes also a clear reduction of the Earth Sensor residual, as shown in Fig. 8, where the residuals computed by ATTMON for the same data are shown in the two cases, with (left) and without (right) compensation. The compensation cancels out the main orbital signal, but a residual signal remains, being probably the tables, derived from Envisat ones, not fine-tuned for the Metop orbit (as less pointing accuracy is required for the EPS mission).

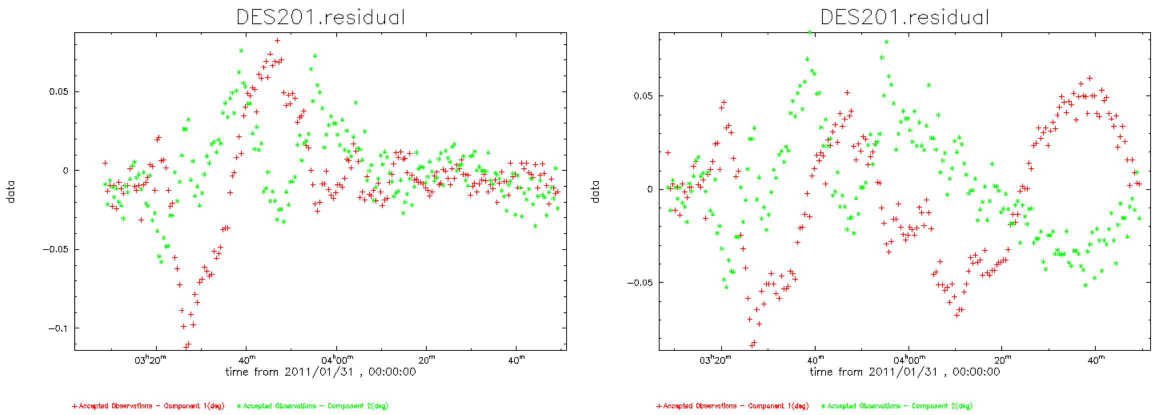

Figure 8. Impact of IR correction on Digital Earth Sensor residuals

A comparison with Fig. 5 shows that the two modules, ATTMON and ATTDET, are very consistent in this estimation. The same can be observed when comparing the gyro drift estimated with the two approaches; the obtained values are shown in Fig 9 for ATTMON (left) and ATTDET (right); the outlier observed at the end of the analysis arc is linked to an anomaly at instrument level, that obliged to reduce the duration of the platform TM data in favour of instrument TM data, affecting thus the estimation process.

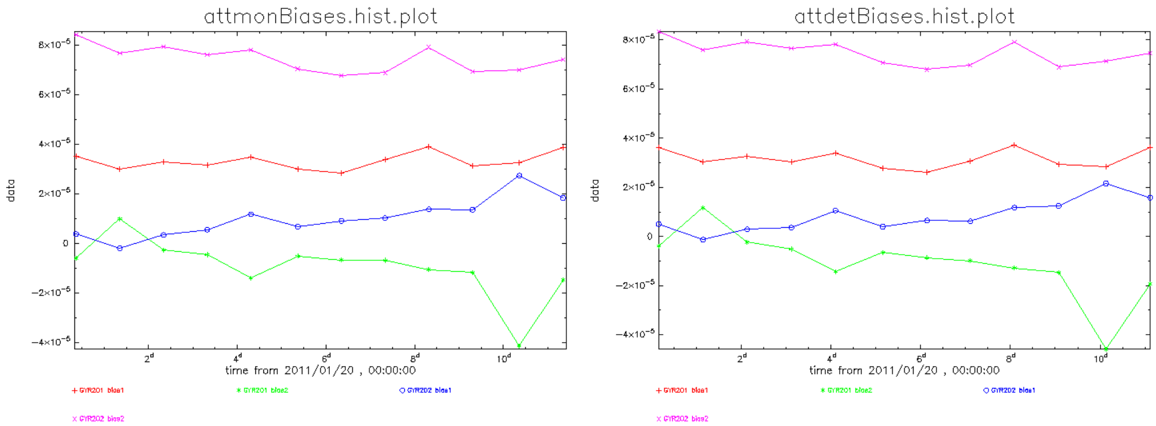

Figure 9. Estimated gyro drifts using different approaches

When analysing the estimated attitude for consecutive days it can be observed that a somehow repetitive pattern on the attitude could be identified as shown in Fig. 10 (above all on day 2 and 3).

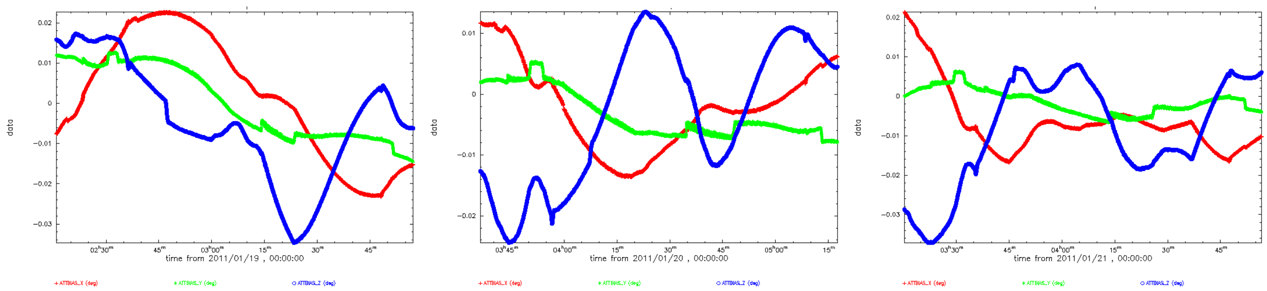

Figure 10. Estimated attitude for 3 consecutive days

That lead to consider that the attitude observed deviation could be correlated with the orbital position of the satellite, at least for short time periods

With that in mind further upgrades of the system can be foreseen to even further improve the performances of the system. It can be observed that the attitude deviation at the beginning and the end of the arc tends to be different; if the assumption of a certain dependency of the attitude deviation with the orbital position is accepted it could be possible to force that difference to be bounded to a limited value, even if still unclear how that could be implemented. Then, it may be interesting to limit the Fourier analysis on the deviation to only few frequencies (orbital pulsations and fractions of the orbital pulsation) instead of to the full frequency domain, to try to identify possible physical causes of the perturbation. Finally, as the observed evolutions are well centred around the reference attitude, proving the good performances of the AOCS system, the option of using pseudo observables derived from the nominal attitude (similar to what done in orbit determination when state vectors coming from the navigation solution are added to the process) could be considered.

For future missions the usage of land-marks or star-trackers is being considered to improve the accuracy of the observations provided by the spacecraft. Increased amount of data on several consecutive orbits would also be beneficial to improve the accuracy of the algorithm and ensure better observability on the yaw direction thanks to the sun visibility.

4. Conclusions

The integration of the angular velocity from the spacecraft gyroscopes telemetry can be used to compute the attitude, providing very useful information on the pointing stability of the platform. This method simplifies the integration of the dynamical equations, replacing them by a much simpler kinematical formulation without significant loss of information.

An initial analysis of the problem based on a computation of the mean attitude with simplified evolution, can provide the system with valid a-priori mean values to speed up the convergence of the estimation process. The initial values provided by this simplified method allow the estimation of every parameter defining the attitude evolution from an a-priori covariance matrix.

More accurate sensor data (like land-marks or star-trackers) and a higher amount of observations to increase the observability of the platform yaw angle would improve the accuracy of the algorithm and are foreseen for future upgrades of the system.

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