Enhanced MetOp Attitude Monitoring

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A new attitude monitoring software has been developed by GMV for the EUMETSAT Polar System program. Based on the integration of the angular velocity provided by the satellite gyroscopes telemetry and a weighted least squares batch estimator using Earth and Sun sensor data, it is able to compute accurately the satellite attitude and the AOCS sensor biases used during the determination process. The propagation into the future of the integrated attitude can be performed through a Fourier Transform of the differences with respect to the nominal attitude, assuming an orbital periodicity of the misalignments evolution.

\section*{Nomenclature}

\begin{align*}
q &= \text{attitude quaternion} \\
\alpha &= \text{attitude misalignments (Euler angles)} \\
\omega &= \text{spacecraft angular velocity} \\
\Omega &= \text{extended angular velocity (4x4 matrix)} \\
\Delta t &= \text{time step} \\
X_k &= \text{parameters at iteration } k \\
G &= \text{gyro drift parameters} \\
I_n &= \text{identity matrix (dimension } n) \\
\Delta y_k &= \text{observations residuals at iteration } k \\
n &= \text{number of parameters being estimated} \\
m &= \text{number of observations} \\
P_0 &= \text{covariance matrix of the parameters (nxn)} \\
Q_0 &= \text{covariance matrix of the observations (mxm)} \\
F &= \text{matrix of observation equations coefficients (mxn)}
\end{align*}

I. Introduction

\textbf{M}ETOP A is the first European operational meteorological satellite in a Low Earth Orbit (LEO) and it is the first of the three satellites of the EUMETSAT Polar System (EPS). It is based upon a three axis stabilized platform developed by Astrium for observation satellites and used in several missions like Envisat, Helios or ERS. The nominal attitude law is a Local Normal Pointing mode augmented by a Yaw Steering Mode to compensate the apparent drift of the sub-satellite point due to the Earth rotation. The Attitude Control System is design in close-loop, and therefore, completely managed on board.

The EPS Flight Dynamics Facility (FDF),\textsuperscript{1} developed by GMV is based on the ESA’s NAvigation Package for Earth Observation Satellites (NAPEOS). Even if no attitude prediction is required on ground for the spacecraft

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control, the sensor telemetry is processed on the FDF to check that the reference attitude is kept within certain defined thresholds. In routine operations, the telemetry provided by the satellite contains data from two gyroscopes and two optical sensors—a digital Infra-red Earth Sensor plus a digital Sun sensor.

The original design of the attitude monitoring system within the FDF assumed a nominal attitude or constant attitude biases during the entire arc of analysis. Gyro and optical sensor data were handled as observations in a weighted least squares batch process that allows the estimation of the attitude misalignments, gyro drifts and optical sensor biases. Even if this implementation was enough to cover the monitoring requirements, it only provided information of the average behavior of the satellite, not allowing a proper modeling of the attitude evolution in time.

A new system has been developed as a technological demonstrator for the concept of attitude estimation through the integration along the time of the satellite angular velocity provided by the gyro data, optimizing the integration parameters with a batch least squares estimator based on the optical sensor data.

II. System design and algorithms

The architecture of the Flight Dynamics facility focuses on the modularization of the system so the most basic parts of the system can be accessed separately from the Graphic User Interface (GUI) even if they are mainly intended to be used within the highest level modules.

The input telemetry data is preprocessed 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. 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 modeled observations for the determination problem.

As it is shown in the Fig. 1, 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 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 gyro 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.

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.

Figure 1. Attitude monitoring system design. Data flow and design of the attitude determination module.
The previous attitude was assumed 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.

A. Attitude Integration Module (ATTINT)

The gyro modeling method for attitude computation replaces the dynamical model of the attitude so the kinematic equations can be integrated directly,\textsuperscript{2} pp. 558-566. As the gyro telemetry is directly taken from the S/C, the retrieved values of the angular rates include any dynamic perturbation and this method notably simplifies the attitude computation with respect to the dynamic problem, which would require a precise modeling of the on board control system.

The angular velocity provided by the gyro telemetry (Fig. 2) can be parameterized with the drifts of the two gyroscopes. Whereas in the previous attitude monitoring system this drift was assumed constant during the whole processing interval, the new software allows the definition of the gyro drift as a linear segmented function in time.

The integration of the attitude is performed using the quaternion representation, that reduce the kinematic equations to the following formula,\textsuperscript{2,3}:

\[
\frac{dq}{dt} = \dot{q} = \frac{1}{2} \Omega q
\]  

With:

\[
\Omega = \begin{bmatrix}
0 & \omega_3 & -\omega_2 & \omega_1 \\
-\omega_3 & 0 & \omega_1 & -\omega_2 \\
\omega_2 & -\omega_1 & 0 & \omega_3 \\
-\omega_1 & \omega_2 & -\omega_3 & 0
\end{bmatrix}
\]  

The numerical integration of the angular velocity has to be performed through a method that copes with the data characteristics. Only a discrete number of values are known, 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 the attitude integration.\textsuperscript{4} 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. The implementation allows the user to define the degree of the interpolating polynomial and the maximum number of corrector steps to be performed. 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 8\textsuperscript{th} degree polynomials with several corrections every integration step.
B. Attitude Determination Module (ATTDET)

The attitude determination module implements a batch least squares method to minimize the observations residuals through the variation of some parameters selected by the user. The parameters that can be optimized by this method are the initial attitude state, the gyro drifts within some given intervals and the optical sensor biases also within certain time intervals.

The batch estimator has been preferred to a Kalman filter to prevent divergences on the estimation. Having a maximum of one orbit of continuous gyro and Earth sensor telemetry and one single Sun sensor measurement to fix the yaw angle, the confidence on the Kalman filter was not high enough for its implementation on the ground segment attitude monitoring system.

The iterative batch estimation process can be summarized in the following equation, \(^\text{pg. 451}\):

\[
[P_0^{-1} + F_k^T \cdot Q_0^{-1} \cdot F_k] \cdot (X_{k+1} - X_k) = F_k^T \cdot Q_0^{-1} \cdot \Delta y_k + P_0^{-1} \cdot (X_0 - X_k)
\]

(3)

With the matrix \(F\) at the iteration number \(k\):

\[
F_k = \begin{bmatrix}
\frac{\partial f_1}{\partial x_1} & \frac{\partial f_1}{\partial x_2} & \cdots & \frac{\partial f_1}{\partial x_n} \\
\frac{\partial f_2}{\partial x_1} & \frac{\partial f_2}{\partial x_2} & \cdots & \frac{\partial f_2}{\partial x_n} \\
\vdots & \vdots & \ddots & \vdots \\
\frac{\partial f_m}{\partial x_1} & \frac{\partial f_m}{\partial x_2} & \cdots & \frac{\partial f_m}{\partial x_n}
\end{bmatrix}
\]

(4)

The partials being computed depend on the parameter \(x_i\) being estimated. Since the dependencies on the time and the variable attitude have been included in the new version of the software, most part of the source code has been developed for the new system.

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

\[
\frac{\partial f}{\partial \alpha_0} = \frac{\partial f}{\partial \alpha} \cdot \frac{da}{da_0}
\]

(5)

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

\[
\frac{da}{da_0} = \frac{da}{dq_k} \cdot \left(\Pi_{l=1}^{k} \frac{dq_l}{dq_{l-1}}\right) \cdot \frac{dq_0}{dq_0}
\]

(6)

The Jacobian 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 (\(df/d\alpha_0\) in Eq. 5) 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. 1, resulting:

\[
\frac{dq_{i+1}}{dq_i} = I_4 + \frac{1}{2} \cdot \frac{d(\tilde{q}_i)}{dq_i} \cdot \Delta t
\]

(7)

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.

\[
\frac{df}{dg} = \left. \frac{\partial f}{\partial \alpha} \right|_{\alpha=K} + \frac{\partial f}{\partial \alpha} \cdot \frac{da}{dg}
\]

(8)

The derivative of the attitude with respect to the gyro drift (\(d\alpha/dG\) in Eq. 8) 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 in ATTINT. The derivatives at constant attitude (the first addend in Eq. 8) were already computed by the previous version of the system.\(^1\)
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.

C. FFT Analysis Module (FTATTPROP)

The implemented method allows a precise computation of the attitude while the gyro telemetry is available. In order to allow the propagation of the spacecraft attitude into the future, a Fast Fourier Transform is performed on the differences of the attitude with respect to the nominal guiding law. The corresponding sinusoidal functions are extrapolated in the future to represent the periodicity of this evolution. Nominally, this analysis will be performed on a complete orbit, so the attitude misalignments will have the same period than the orbit itself. Higher period perturbations cannot be analyzed due to the amount of data available from the satellite.

The change of the guiding law has to be taken into account for this analysis, as the expected misalignments in Yaw Steering mode are not the same that can be assumed for other attitude modes, furthermore when alternative guiding laws are usually considered only during maneuvers in normal operations.

D. Attitude Comparison Module (ATTCOMP)

Although the attitude is stored in form of Euler angles and misalignments with respect to the nominal mode, the comparison of two arcs using angles presents several problems, due to the high variation of the roll, pitch and yaw values to represent two similar attitudes close to the limits in the range of these angles.

The comparison of two attitudes is directly performed through the rotation matrix between the two satellite frames. Since the expected difference between two arcs should be relatively small, this matrix is transformed back into Euler angles to be analyzed by the users.

III. Operational approach

The main limitation of this method is currently 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 its more critical for the mission), including only one Sun sensor observation (sometimes ever none, due to the very low sampling of this TM point - 1/16Hz, being the duration of the sun visibility sometimes shorter), which significantly limits 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 Eqs. (6) and (7)), due to the projection of the orbital pulsation angular velocity in satellite axis with a rotation in yaw. 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 3. Integrated attitude in J2000 without initial bias and no gyro drift.](image-url)
The direct integration of the gyro data without any initial attitude bias or gyro drifts (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, which is probably caused by some misalignment of the platform or a non considered gyro drift.

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 is executed, assuming constant attitude misalignments and sensor biases. 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 lack of observability in the yaw parameter is also mitigated by the usage of the gyro data during the integration process.

The output obtained from the determination process (Fig. 4) gives a maximum platform misalignment of 0.02 degrees. 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.

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 same order of magnitude than the infra-red correction applied to this sensor. The differences of the angular velocity (shown in Fig. 2b) 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, so part of the variations seen in the attitude have been introduced by the errors in the angular velocity inherited from the gyro data.

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 softens the evolution of the satellite attitude, but the frequencies above 1 Hz are filtered in the process.

Further upgrades of the system are foreseen 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 desirable in order to improve the accuracy of the algorithm.

Figure 4. Differences of the determined attitude with the nominal. Initial attitude bias, constant gyro drifts and optical sensor bias estimation.

Figure 5. Earth sensor residuals after the attitude determination.

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IV. Conclusion

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.

References

1Eumetsat Polar System Flight Dynamics Facility, Software Package, Ver. 4.0, GMV S.A.U., Tres Cantos, Madrid 2009.