AUTONOMOUS FAULT DETECTION ON A LOW COST GPS-AIDED 3-AXIS ATTITUDE DETERMINATION SYSTEM

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BIOGRAPHY

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ABSTRACT

This paper presents a three-axis attitude determination procedure based on the Global Positioning System (GPS) with special emphasis to the autonomous integrity monitoring issue. One envisages a class of low cost navigation applications requiring low accuracy (around 0.5 degrees) but continuous attitude knowledge. A snapshot algorithm first estimates the three-axis attitude from interferometry on double differences of GPS carrier phase L1. Then, the attitude estimate is improved by fusing it with angular rate measurements from low cost, MEMS gyros. The procedure is especially designed to detect single faults on either the GPS or the gyros autonomously from residuals monitoring. Both GPS and gyro measurements are assumed as corrupted by colored Gaussian noises whose effects are mitigated by a stochastic dynamic compensation model. The state vector, which includes the attitude quaternion and parameters of the error model, is estimated from a bank of extended Kalman filters with different time delays. Attitude propagation model between GPS sampling times is based on the gyro output after drift calibration. One defines the fault modes of each sensor and some control parameters to the fault detection procedure. The algorithm is tested with numerical simulation and real data. The simulation scenarios include LEO micro-satellites with different orbit inclinations and three different failure modes: complete loss of track from the whole GPS constellation; temporary abnormal interference on a single GPS satellite; and gyro drift higher than its specified level. The results show that the algorithm is suitable to cope with different fault types and intensity levels.

INTRODUCTION

This work presents an algorithm for three-axis autonomous attitude determination, envisaging future applications mainly in micro-satellites. The algorithm is based on inertial sensors using MEMS (Micro Electro-Mechanical Systems) technology aided by GPS (Global Positioning System) and is able to keep working under temporary single faults either in the gyros or in the GPS. Specifically, we use a set of three-axis silicon Vibrating Ring Gyros based on the Coriolis Effect that presents a promising long term stability [1] compatible with a class of space missions with moderate pointing requirements and stringent budgets in terms of cost, weight, volume and power consumption. However, the algorithm is not constrained to space applications of any special type.

The gyro drift is compensated by interferometry on double differences of GPS carrier phase L1 as observed by a set of at least three GPS receivers, each one with single antenna inputs. The robustness required for autonomous space applications, is ensured by fault detection and diagnosis techniques basically relying on the GPS redundant information.
As discussed in Reference [2], we concentrate in three fault modes of Gyro and GPS: GPS signal contingency outages; GPS signal under unexpectedly strong interference; and gyros with long term drifts higher than the specified limits. Fault detection nomenclature follows Reference [3] (see also [2]). The algorithm is adapted from the work of McMillan, Bird and Arden [4]. Under a temporary interruption of GPS signal lock, attitude is propagated from gyros alone. High interference on a signal from one GPS receiver is detected and solved by checking the residuals of the remaining healthy ones. High gyro noise level is detected and isolated by a bank of attitude propagators with different time delays in a Kalman filter scheme.

This work was developed in the frame of the doctoral thesis of the first author [5] and is part of a series of developments in the field of GPS-based attitude determination that have been carried out at INPE by the second author and his collaborators in the past ten years. It started with experimental studies on spin-axis attitude determination from double difference of GPS carrier phases on a rotating baseline, first at low spin rates only [6,7] and afterwards extended to arbitrary high spin rates [8]. The practical subject of multipath mitigation was then addressed as a calibration problem with different solutions for both ground applications [9] and space applications [10,11]. In the first case the attitude is a priori known, while in the second case the attitude is considered unknown and needs to be estimated together with the calibration coefficients. Additionally, dynamic based algorithms for integer ambiguity resolution were developed for three-axis attitude determination applications [12] and for two-axis (azimuth and elevation) attitude determination applications [13]. Also, a time correlation experimentally detected on GPS carrier phase observables was successfully modeled by a dynamic compensation technique [14].

Envisaging future space applications, the contribution of the present work to that sequence is concerning the robustness issue. Partial results with focus on the GPS solution were already presented in Reference [2]. Preliminary results of the GPS and gyro data fusing were also presented in Reference [15] showing simulation results with a single attitude propagation delay to detect Gyro faults. In this work we present the algorithm on its plain capacity, with emphasis on the robustness aspects. Results with real GPS experimental data and simulation of multiple delays in the Gyro fault detection part of the algorithm are presented in details and compliment the authors’ parallel paper [16].

GYRO AND GPS INTEGRATION

When integrated with Gyros, a GPS receiver provides means to calibrate the long term gyro drift, and consequently allows precise attitude propagation with a sample rate higher than the GPS receiver usually operates. Moreover, the attitude propagated by Gyros provides important information during GPS loss of signal. In addition, in the absence of faults, the attitude is estimated with better accuracy than the GPS-based only solution. This makes the approach particularly suitable for real time spacecraft application, including closed loop attitude control. That is why it is so important to detect, isolate and as much as possible to correct any fault in the system autonomously.

The integration between GPS receiver and Gyro can be made by structures with different degrees of coupling [17]. In this work we follow the loosely coupled solution, with direct feedback that is suitable for use of independent, off-the shelf equipment.

The algorithm is divided in two steps. First, attitude is determined by a snapshot algorithm from the GPS data. Then, the gyro data is added and the attitude is estimated together with other system parameters by the Kalman filter, as described in the sequence.

Static Attitude Determination

The GPS attitude determination used here follows directly from Reference [14] for 3-axis attitude determination using three or more GPS antennas.

The algorithm uses double differences of carrier phase:

\[
\phi_{i,0}^{p,0} = \frac{1}{\lambda} \left( s_{p,0}^{p,0} - \lambda N_{i,0}^{p,0} + d_{i,0}^{p,0} + \epsilon_{i,0}^{p,0} \right)
\]

(1)

where \( \phi_{i,0}^{p,0} \) is the double difference between antenna \( i \) to the master antenna 0 and between GPS satellites \( p \) to a reference satellite 0; \( \lambda \) is the L1 wavelength; \( s \) is the GPS satellite line of sight; \( N \) is the integer ambiguity; \( d \) is the delay due to multipath among other factors and \( \epsilon \) is a random noise.

First the GPS data is pre-processed to deal with integer ambiguity and antenna calibration. After that, a static (snapshot) attitude determination is carried out, that minimizes the following quadratic cost function, which takes into account the intrinsic coupling due to the presence of master antenna and reference GPS satellite in every double difference:

\[
J = tr \left( Y - \frac{1}{\lambda} B' A S A' A_m Y - \frac{1}{\lambda} B' A S A' A_m A_n \right)
\]

(2)

where \( Y \) is the matrix of double differences of carrier phase after integer ambiguity resolution, \( tr(\cdot) \) is the matrix
trace operator, $B$ is the antenna baseline matrix in the body frame, $A$ is the attitude matrix, $S$ is the matrix of line of sights of locked GPS satellites in the reference frame, $A_n$ is the matrix operator responsible to implement between satellite differences and $A_n$ and $A_m$ are the decoupling matrices defined in Reference [14].

Excluding faults, the minimum value of the cost function $J$ should present a Chi-square distribution with $\mu = (n-1)(m-1)-3$ degrees of freedom, where $m$ is the number of satellites and $n$ is the number of GPS antennas. This property will be useful to detect faults on GPS signals in next session.

**Attitude Filtering**

For the attitude filtering and propagation the algorithm is based on Reference [18] that applies the Kalman filter to three-axis attitude estimation based on the kinematic model. In our case, the 3-axis Gyros are a simple MEMS device and the attitude sensor is the GPS receiver baselines, whose attitude snapshot solution works as a quaternion sensor.

The rate of change of the attitude matrix with time is driven by the angular velocity vector $\omega$ in the satellite system, and may be represented in quaternion algebra by:

$$\dot{q}(t) = \frac{1}{2}\Omega(\omega(t))q(t)$$

where the unit quaternion $\overline{q}$ is composed of a vector $q$ and a scalar $q_4$:

$$\overline{q} = \begin{bmatrix} q \\ q_4 \end{bmatrix}$$

(4)

Skew symmetric matrix $\Omega(\omega(t))$ is defined below:

$$\Omega(\omega) = \begin{bmatrix} 0 & -\omega_3 & \omega_2 \\ \omega_3 & 0 & -\omega_1 \\ -\omega_2 & \omega_1 & 0 \end{bmatrix}$$

(5)

The attitude propagation model is given by a simple kinematic model given by Equation 3, and a gyro model disturbed by a first order Gauss-Markov stochastic process which represents the Gyro long term drift-rate bias, and a white noise. Similarly, the GPS-based attitude is modeled as an attitude observation corrupted by a white noise stochastic sequence and a first order Gauss-Markov stochastic process that takes into account the time correlation observed in the GPS phase signal. So, the dynamic model is given by the following equations:

$$u = \omega + \beta_\omega + e_\omega(C_{\omega\omega})$$

$$\dot{\beta}_\omega = -\frac{\beta_\omega}{\tau_\omega} + w_\omega(q_\omega)$$

$$Z = \theta + \beta_\theta + e_\theta(C_{\theta\theta})$$

$$\dot{\beta}_\theta = -\frac{\beta_\theta}{\tau_\theta} + w_\theta(q_\theta)$$

where $Z$ is the input vector observation to the Kalman filter, coming from the GPS, $\beta_\omega$ is the long term gyro drift, $\beta_\theta$ is a time correlated noise in the GPS-based attitude estimate. The model parameters $q_\theta \cdot q_\omega$ and $\tau_\theta \cdot \tau_\omega$ represents respectively the noise intensity and time constants of the correlated bias $\beta_\theta$ and $\beta_\omega$; $e_\omega$ is the gyro noise, $C_{\omega\omega}$ is its covariance matrix, $C_{\theta\theta}$ represents the GPS-based attitude noise and $C_{\theta\theta}$ its covariance matrix.

All those model parameters were fitted empirically from experimental data (see [14]).

The well-known singularity on quaternion covariance matrix is handled following the reduced covariance approach presented by [18]. Then, the reduced state vector that will be actually estimated is:

$$\tilde{x}(t) = \begin{bmatrix} \Delta \overline{q}(t) \\ \Delta \beta_\omega \\ \Delta \beta_\theta \end{bmatrix}$$

(10)

where $\Delta \overline{q}$ is the reduced quaternion increment from the GPS-based attitude to the attitude propagated by the Gyro (from Eq. 9) and the quaternion propagated by the Gyro.

**FAULT DIAGNOSIS**

A fault is a deviation not permitted of at least one characteristic property or variable of the system from acceptable, usual or standard behavior [3], in contrast to a failure, which is considered a permanent interruption of the system ability to perform any required function under specified operating conditions. Fault detection is here understood as the determination that the system presents a fault; fault isolation determines the kind and location of a fault; and fault identification determines the fault intensity. Fault diagnosis deals with fault detection, isolation and identification. The proposed procedure envisages single fault diagnosis only, either on the GPS or on the gyro. Permanent failures or multiple faults are not considered.

A fault occurrence has degradation consequences in the level of uncertainty of the estimated attitude. The level of
degradation depends on the fault and from the algorithm success to detect faults, established by the percentage of false alarms and by the percentage of faults not detected. In case of temporary fault detected in the Gyro signal, the level of uncertainty grows, but only up to the level obtained by the solution using GPS alone. The occurrence of a fault in the GPS signal can induce partial or complete degradation in the uncertainty level, depending on the fault duration and intensity.

Considering a mask angle of 10°, the number of GPS constellation satellites in view from an Earth pointing satellite in Low Earth Orbit ranges from 6 to 8 most of the time. This number may even increase significantly with the advent of other GNSS constellations. In this scenario, there is sufficient internal redundancy to detect fault on the GPS during the phase of static attitude determination. This detection is made by a group of parallel processors where in each processor one of the GPS satellites is suppressed, as indicated in part b of diagram shown in Fig. 1.

The second part of the proposed algorithm uses a Kalman filter to update the quaternion, the Gyro drift-rate bias and the GPS error corrections. After the state update step, the attitude is propagated using the drift-compensated Gyro output, with a sample frequency higher than the attitude obtained from GPS alone.

In case of a single fault either on GPS or Gyro, the attitude solution continuity is assured by the remaining healthy sensor, which allows a complete fault diagnosis.

The fault diagnosis algorithm has some control parameters that need to be tuned empirically.

The efficiency of the algorithm follows directly from this adjustment. More specifically, they determine the false alarm rate and the missing fault detection rate.

**GPS Fault Diagnosis**

Part b of Fig. 1 shows the parallel processors that implement the GPS fault diagnosis box which makes the comparison of all the cost functions and send to the Kalman filter the fixed estimate of attitude matrix and its error covariance matrix.

If \( J_p < J_\bullet \), where \( p \) is the number of the processor that excludes data coming from \( p \)-th GPS satellite, and \( J_\bullet \) is the fault detection threshold, the algorithm considers that there is no detectable fault and takes \( \theta^* = \theta \). Otherwise, if \( J_p > J_\bullet \), but \( J_q < J_\bullet \), then a fault is detected in satellite \( q \), and one takes \( \theta^* = \theta_q \). There are \( m \) processors: one to the full constellation and \( m-1 \) to isolate possible faults. If \( J_p > J_\bullet \), it is necessary to repeat the test with a different reference satellite because it may be the faulty one.

Since in absence of faults \( J \) has chi-square distribution with \( \mu \) degrees of freedom, it is convenient to express the threshold \( J_\bullet \) as \( \mu_j + \kappa \sigma_j \), where \( \sigma_j = \sqrt{2\mu_j} \) is the standard deviation of \( J \) and \( \kappa \) is the control parameter that should be somewhere in the practical range from 0 to 5 and was set to 3 from simulation analysis.

**Gyro Fault Diagnosis**

Part a of Fig. 1 shows the Kalman filter fed by information from the difference between the gyro propagated quaternion and the estimated attitude coming from the GPS. The Filter equations are standard and follow directly from Reference [18] with a minor variation due to the additional state variable \( \beta_\theta \) which has a very simple dynamics and is not coupled with the other state variables. Therefore we may concentrate in the quaternion propagation equation. If the rotation vector defined as:

\[
\Delta \theta = \int_{t'}^{t} \omega(\theta') dt'
\]

is small, then the solution of Eq. 3 is given by:

\[
\bar{\theta}(t + \Delta t) = M(\Delta \theta) \bar{\theta}(t)
\]

where

\[
M(\Delta \theta) = \cos(\Delta \theta/2) I_{4x4} + \frac{\sin(\Delta \theta/2)}{\Delta \theta} \Omega(\Delta \theta)
\]

(14)

Gyro is considered faulty if the residues of Equation 8 grow above a tolerance level proportional to its theoretical standard deviation for any propagator. This is illustrated in next section, with some numerical results.

**SIMULATION AND EXPERIMENTAL RESULTS**

Reference [2] shows simulation results of the GPS part of the algorithm (part b of Fig. 1). Reference [6] shows simulation results of the GPS and Gyro parts considering only one delay in the gyro part of the algorithm (part a of Fig. 1). This work completes those previous works with an arbitrary number of time delays and present experimental results of the GPS part taken on ground, in connection with simulated Gyro data. Also, we present more extensive simulation results with two different orbit scenarios.
Figure 1. Fault diagnosis algorithm schema.

a) Gyro fault diagnosis

b) GPS fault diagnosis

Figure 1. Fault diagnosis algorithm schema.
Faults on the GPS signals may occur for instance due to receiver electronics instability and environment interference including multipath. In order to test the procedure, single faults were simulated on the GPS signals. These faults are not associated with any specific physical event, but they intend to demonstrate the sensitivity of the fault diagnosis procedure to the intensity of an arbitrary fault.

A hybrid simulation with hardware in the loop was carried out with a square structure with lateral size of 1 meter where three GPS antennas were positioned (see Fig. 2). The Gyros were simulated perfectly aligned with this structure to sense its movement in three axes. Antenna number 0 was positioned at the corner of the square and was taken as the master antenna. Antenna number 2 was positioned in the x direction with respect to the master antenna and antenna number 1 in the y direction, forming two orthogonal baselines, namely baseline 0-2 and baseline 0-1. A fourth antenna at the last square corner played as a cold redundant antenna and was not used in this experiment.

![Figure 2. The antennas set frame.](image)

GPS Experimental Data

The Experimental data taken was carried out using three commercial Ashtech Z-XII GPS receivers with a sample rate of 1Hz. An intentional fault injection was performed by adding 5 cycles between instants 30s and 40s in GPS satellite 27 that simulates undetected cycle slips or an error in the integer ambiguity resolution algorithm in order to illustrate the fault detection performance.

As shown in Figure 3, during the introduction of the failure the cost function for all data combination increased dramatically but for the faulty satellite \( p = 27 \). Figure 4 presents the 3-axis attitude estimated from GPS only, after fault detection. Those values feed the Kalman filter as input measurements and do not present any visible degradation due to the fault. More details are found in [5].

![Figure 4. Azimuth and elevation estimates for each antenna baseline from GPS only, and respective 1 sigma uncertainty boundaries.](image)

GYRO DATA SIMULATION

A MEMS Gyro with the characteristics described in Reference [19] (See also [20]) was simulated using commercial software Spacecraft Control Matlab Toolbox from Princeton Satellite Systems. Two different
Simulations are presented here to explore different aspects, as described in the sequence.

**Gyro - First Case: effect of number of time delays in the bank of attitude propagators**

In this case, intentional fault injection was performed by introducing an error of 0.03 rad/s in the drift-rate bias during the interval from 70s to 80s. The gyro fault detection was performed using two delayed propagators. Figure 5 shows both filtered and propagated 3-axis attitude by the propagator with two time delays, while Figure 6 shows the estimates of quaternion increment ($\Delta q$) by both filter-propagator processes. In Figures 5 and 6, the filtered 3-axis attitude is shown with the GPS sampling rate of 1 Hz, while gyro propagated attitude is shown with a higher sampling rate of 10 Hz.

The algorithm was able to detect the gyro fault between instants 70s to 80s since $q_2$ clearly exceeded its 1-sigma uncertainty level in both propagators. However, the effect was much more pronounced in the propagator with the longest delay.

As a marginal general observation that may be useful in some applications, one can see from Figures 4 to 6 that azimuth estimates are more accurate than elevation estimates. This fact is confirmed by simulations and may be related with the spatial distribution of GPS constellation with respect to the antennas frame.

**Gyro - Second Case: effect of GPS signal outage**

Figure 7 shows the quaternion increment ($\Delta q$) considering complete loss of the GPS signal in the interval from 70s to 100s. In this case, the gyro propagates the attitude not considering the GPS. As a consequence, the error level clearly increases during this interval but quickly recover its normal accuracy level after the instrument recovers itself from the faulty condition.
Flight Conditions Simulation

In addition to the experiments described in the previous sessions, simulations were performed to test the algorithm in orbit conditions. Two orbit scenarios are selected with this purpose: one circular polar orbit and one circular, low inclined orbit. In both cases the orbit altitude is set to 650 km and the satellite is in Earth-oriented three-axis stabilization with residual oscillations caused by attitude control errors (pointing accuracy better than 1° and drift rate below 0.1°/s). The simulations run for one orbit period to check the influence of different relative geometry of line of sight of the GPS satellites. Other simulation parameters are given in Table 1. Besides addition of noise, the L1 phase is affected by multipath delays with medium level corresponding to 50% of the strong multipath scenario caused by a pair of metal plates in the close neighborhood of the antennas, as described in [12]. Gyro fault injection happens in a ten minutes extra time after end of the one orbit simulation, which is time enough to the algorithm to detect it.

Intentionally, the integer ambiguity is solved every sample time in order to check the performance of this part of the algorithm under different orbit and attitude conditions.

Several tests were run in order to check the effect of control parameters of the fault diagnosis algorithm as well as to explore the effect of error levels in both attitude sensors. Typical results from the Equatorial orbit scenario are summarized in Table 2. No special difference was found in the polar orbit scenario. Figures 7 and 8 show the attitude errors and their statistics. The advantage of data fusion in terms of accuracy is clear in all axes. Furthermore, the problem of worse accuracy given by the GPS alone around pitch and roll-axes is minimized. Also, as the GPS is more accurate around yaw axis, it clearly improves the accuracy of the Kalman filter estimate when compared with the gyro propagated solution in a profitable cooperation between these sensors.

Table 1 – Attitude Sensor Parameters in the Simulation

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
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</thead>
<tbody>
<tr>
<td>Long term Gyro stability</td>
<td>48μ-rad/s (10°/h)</td>
</tr>
<tr>
<td>Gyro Random walk</td>
<td>20 μ-rad/√s (0.066°/√h)</td>
</tr>
<tr>
<td>Gyro sampling rate</td>
<td>10 Hz</td>
</tr>
<tr>
<td>GPS sampling rate</td>
<td>0.2 Hz</td>
</tr>
<tr>
<td>Antenna baseline</td>
<td>0.6 m</td>
</tr>
<tr>
<td>L1 Phase Uncertainty (before double difference)</td>
<td>1% of L1 wavelength</td>
</tr>
<tr>
<td>GPS mask angle</td>
<td>15°</td>
</tr>
<tr>
<td>Mean time between fault injection on GPS</td>
<td>60 s</td>
</tr>
<tr>
<td>Intensity of GPS fault injection</td>
<td>35% of L1 wavelength</td>
</tr>
<tr>
<td>Intensity of gyro fault injection</td>
<td>10 times the gyro drift</td>
</tr>
</tbody>
</table>

The results give evidence of the advantage of having more than one time delay in the Gyro fault detection algorithm. After 5 time delays, the gain in the fault detection was marginal.

Table 2 – Summary of Simulation Results

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Failure rate on Integer Ambiguity Solution:</td>
<td>0 %</td>
</tr>
<tr>
<td>Rate of GPS Missed Faults:</td>
<td>2 %</td>
</tr>
<tr>
<td>Rate of GPS False Alarms:</td>
<td>40%</td>
</tr>
<tr>
<td>Time delay to detect gyro fault by 1st propagator:</td>
<td>450s</td>
</tr>
<tr>
<td>2nd propagator:</td>
<td>220s</td>
</tr>
<tr>
<td>3rd propagator:</td>
<td>190s</td>
</tr>
<tr>
<td>4th propagator:</td>
<td>190s</td>
</tr>
<tr>
<td>5th propagator:</td>
<td>125s</td>
</tr>
<tr>
<td>Rate of Gyro False Alarms:</td>
<td>0%</td>
</tr>
</tbody>
</table>

CONCLUSIONS

An attitude determination algorithm is presented based on MEMS gyros aided by GPS. Tested under different flight simulated conditions as well as with real data taken on ground, the algorithm presented robustness to the following fault modes of the attitude sensors: GPS temporary loss of signal; high interference on GPS carrier phase signal; and unpredicted high level of gyro drift.

The proposed algorithm explores the benefits of both sensors with different characteristics that may be attractive to low cost micro-satellites. The algorithm is potentially promising specially considering that the number of GNSS systems is foreseen to increase in the near term and the stability of MEMS gyros are improving systematically with time and may become a key instrument to cheap attitude determination systems. In this context, the paper intends to give a contribution to make it a reality.
Figure 8 – Attitude errors in Roll, Pitch and Yaw axes from GPS, gyro and Kalman filter, and histogram of normalized residuals of GPS carrier phase.

Figure 9 – Observed Probability Distribution Function of attitude errors in Roll, Pitch and Yaw axes and global attitude error from GPS, gyro and Kalman filter.
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REFERENCES


