FAULT DIAGNOSE IN THE AUTONOMY MICRO-SATELLITE ATTITUDE DETERMINATION USING GPS AND GYROS

Arcélio Costa Louro
INPE – São Jose dos Campos - Brazil
e-mail: arcelio@dss.inpe.br

Roberto Vieira da Fonseca Lopes
INPE – São Jose dos Campos - Brazil
e-mail: roberto@dss.inpe.br

Hélio Koiti Kuga
INPE – São Jose dos Campos - Brazil
e-mail: roberto@dss.inpe.br

Abstract. *This work presents an algorithm for three axis autonomous attitude determination, developed to be applied in micro-satellites. This approach is based in the fusion of MEMS inertial sensor technologies observations with GPS receiver observations. The algorithm uses a Kalman filter that permits to make diagnosis of simple failures considering the residues statistics. Is defined the failure modes of each sensor and is also defined control parameters to detect failures. The influence of control parameters above the percentage of failures not detected and false alarms are analyzed. Results of the proposed algorithm are also presented.*

***Keywords**: GPS, Inertial sensors, Attitude determination and Kalman filter.*

1. INTRODUCTION

This work presents an algorithm for three axis autonomous attitude determination, developed to be applied in micro-satellites with low cost and moderate requirements of pointing accuracies. The algorithm presents robustness characteristics to eliminate temporary GPS signal loss and simply temporary faults, and is based in fusion of MEMS (Micro Electro-Mechanical Systems) technologies inertial sensor observations with GPS (Global Positioning System) receiver observations. Specifically, silicon ring Gyro, with vibratory structure, based on Coriolis Effect, with the characteristics of low cost, weight, volume and power consumption are employed as primary sensor for attitude propagation between auxiliary sensor sample instants.

The auxiliary sensor is one or more GPS receivers using 4 antennas and with the following outputs: each antenna position, each GPS satellite position and each antenna carrier phase (L1). Based on the interferometry principle (Misra, 2001), double difference between two antennas - defining one baseline - and between two GPS satellites are used to update the attitude in the sample instants and correct the Gyro bias effect.

The robustness, essential for autonomous space applications, is processed with a Kalman filter and a propagator to detect and isolate simple fault, based on the residues statistics. This algorithm is an adaptation of McMillan, Bird and Arden (1993) and provides the attitude in a sufficiently autonomous, continuous and robustness mode to make feasible the application in real time on-board a micro-satellite with precision better than the attitude using GPS receiver carrier phase only. Fault detection nomenclature follows the definition of Nyberg (1999) and can be seen in Louro and Lopes (2004).

Gyro and GPS fault Modes are discussed in Louro and Lopes (2004). This work also presents characteristics of each type of sensor, establishing MEMS Gyro as a good choice for primary attitude propagation, considering the possible applications, low cost and moderate requisites of attitude.

Louro and Lopes (2004) also presents simulation results of GPS part of the algorithm. Louro and Lopes (2005) shows simulation results of the GPS and Gyro part considering only one delay in the algorithm (part of Fig. 2). This work presents an extension of the previous effort with an update of the algorithm, experimental results of the GPS part and simulation of more than one delay in the Gyro part of the algorithm. This work shows parts of doctoral studies at INPE Space Engineering and Technologies Course.

2. GYRO AND GPS INTEGRATION

When integrated with Gyro, a GPS receiver provide means to calibrate the drift-rate Bias, and consequently allow precise attitude propagation with a sample rate better than the supplied only by GPS receivers. Moreover, the attitude propagated by Gyros provides important information to detect fault on the GPS, not implemented in this work. Because of that, this arrangement is advantageous for real time applications, and in particular for closed loop attitude control.

As can be seen in Kaplan (1996), the integration between GPS receiver and Gyro can be made by structures with different degrees of coupling as described below.

The structure with minimum coupling is the indirect feedback, where a linearized Kalman filter is supplied by the attitude residues locally estimated by the GPS related to the attitude propagated by the Gyro, and produces as output, attitude and drift-rate bias corrections. Although the structural simplicity, it has a serious disadvantage related with residues that accumulate propagation errors due to drift-rate Bias. In such case, the linearization error presents secular growth, which makes impossible the filtering. This obstacle is superseded by the direct feedback structure, where an extended Kalman filter process attitude residues locally estimated by the GPS related to the attitude propagated by the Gyro, is the same as the prior case, with the difference that the corrections estimated by the filter are incorporated during the propagation. In the way the linearization is made near the better attitude prediction available. This is basically the schema that will be used and is illustrated in Fig. 1.

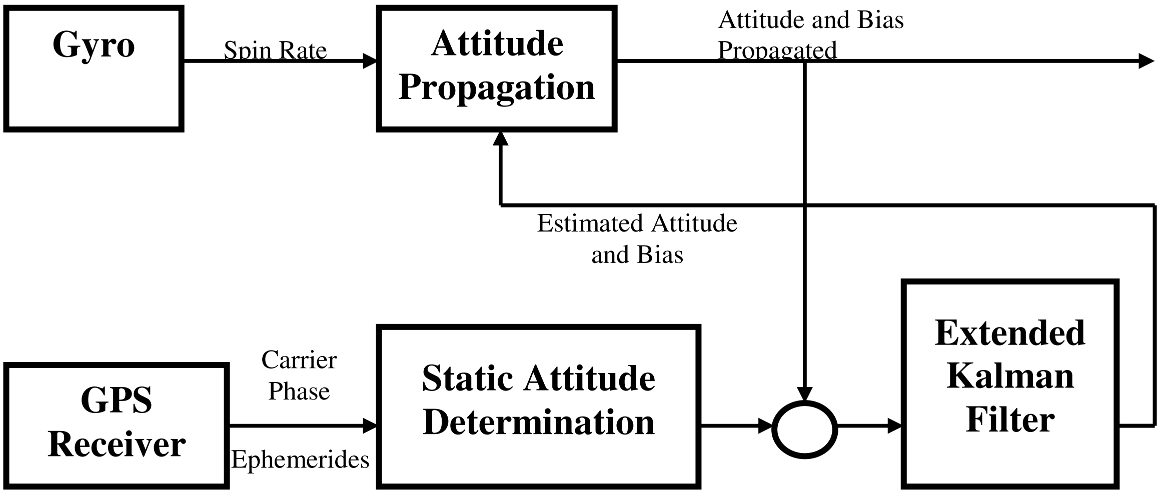

Figure 1. Gyro and GPS observations fusion Algorithm using extended Kalman filter: direct feedback coupling.

Structures with direct and indirect feedback are said to be loose coupling, because the auxiliary sensor in this case do not receive any feedback of the process. When this feedback exist, the structure is said to be tight or deep coupling, depending if the feedback is made in the receiver software or in the synchronizing loop of the satellites, considering the GPS case. This type of solution requires development of specific GPS receivers, while the loose coupling structures allow the use of “off the shelf” equipments, resulting in modularity, versatility and lower cost to the project.

2.1 STATIC ATTITUDE DETERMINATION

The GPS attitude determination used here is based on Lopes e Kuga (2002) that defines an algorithm to determinate 3-axis attitude using three or more GPS antennas.

The algorithm uses the carrier phase double difference, as defined in Eq. 1:

$$\varphi_{i,0}^{p,0} = \frac{1}{\lambda}(s^{p,0})' b_{i,0} + N_{i,0}^{p,0} + d_{i,0}^{p,0} + \varepsilon_{i,0}^{p,0} \tag{1}$$

where φ is the double difference between antenna i to the reference antenna 0 and between GPS satellites p to the reference satellite 0. λ is the L1 wavelength, s is the GPS satellite sightline, N is the integer ambiguity, d is delay due to multipath and other factors and ε is the random noise.

Initially the GPS data shall be pre-processed to match the data from the antennas, to resolve the integer ambiguity, calibration of the multipath and antenna phase center variation. After that, the static attitude determination is made minimizing the following quadratic cost function, Eq. 2, which takes in to account the intrinsic coupling due to the presence of the reference antenna and to the reference GPS satellite in all the double differences:

$$\hat{J} = tr\left\{\left[Y - \frac{1}{\lambda}B'AS\Delta_m{}'\right]\Lambda_m\left[Y - \frac{1}{\lambda}B'AS\Delta_m{}'\right]'\Lambda_n\right\} \tag{2}$$

where Y is the double difference after the integer ambiguity resolution, $tr\{\}$ is the matrix trace, B is the user position matrix, A is the attitude matrix, S is the GPS satellite sightline matrix, Δ_m is the matrix responsible to implement the double differences and Λ_m, Λ_n are the accessory matrices defined in Lopes e Kuga (2002).

The cost function, J, excluding any fault, has the Chi-square distribution with the following expected value, Eq. 3:

$$\left\langle \hat{J} \right\rangle = (n-1)(m-1)-3 \tag{3}$$

where $\langle \; \rangle$ is the expectation operator, *m* is the number of satellites and *n* is the number of GPS antennas.

2.2 ATTITUDE FILTERING

For the attitude filtering and propagation the algorithm is based on Lefferts, Markley and Shuster (1982), that defines an algorithm suitable for spacecraft equipped with three-axis Gyros as well as with attitude sensor, in our case, the Gyro is a simple MEMS device and the attitude sensor is a GPS receiver.

The rate of change of the attitude matrix with time defines the angular velocity vector ω in the satellite system, which can be represented in quaternion algebra by the Eq. 4:

$$\dot{\bar{q}}(t) = \tfrac{1}{2}\Omega(\omega(t))\bar{q}(t) \tag{4}$$

where the quaternion representation is defined by Eq. 5,

$$\bar{q} = \begin{bmatrix} q \\ q_4 \end{bmatrix} \tag{5}$$

where q and q_4 are, respectively, quaternion vector and scalar parts. And the matrix $\Omega(\omega(t))$ is defined in Eq. 6.

$$\Omega(\omega) = \begin{bmatrix} 0 & \omega_3 & -\omega_2 & \omega_1 \\ -\omega_3 & 0 & \omega_1 & \omega_2 \\ \omega_2 & -\omega_1 & 0 & \omega_3 \\ -\omega_1 & -\omega_2 & -\omega_3 & 0 \end{bmatrix} \tag{6}$$

Considering Eq. 4 and defining the attitude propagation model as simple cinematic model added by a first order Gauss-Markov Process which represents the Gyro drift-rate bias and the solution error supplied by the GPS, the following state equations can be established:

$$u = \omega + \beta_\omega + e_\omega(C_{\omega\omega}) \tag{7}$$

$$\dot{\beta}_\omega = -\frac{\beta_\omega}{\tau_\omega} + w_\omega(q_\omega) \tag{8}$$

$$Z = \theta + \beta_\theta + e_\theta(C_{\theta\theta}) \tag{9}$$

$$\dot{\beta}_\theta = -\frac{\beta_\theta}{\tau_\theta} + w_\theta(q_\theta) \tag{10}$$

where *u* and *Z* are the vector observations from the Gyro and GPS, β_ω is the gyro drift-rate bias, β_θ is the attitude correlated bias based only in the GPS data. Parameters q_θ, q_ω and τ_θ, τ_ω represents the intensity of the correlated bias β_θ and β_ω, and respective time constants. These values shall be adjusted empirically (see Lopes, Silva and Prado, 2002). e_ω is the drift-rate noise, $C_{\omega\omega}$ is the respective drift-rate noise variance, e_θ is the GPS attitude noise and $C_{\theta\theta}$ is the respective GPS attitude noise variance.

The covariance matrix associated with the state vector defined by Eqs. 4, 8 and 10 is singular. To manage this covariance matrix Lefferts, Markley and Shuster (1982) propose to reduce the state vector that will be effectively estimated, the reduced state vector is defined by Eq. 11:

$$\underline{x}(t) = \begin{bmatrix} \Delta \underline{q}(t) \\ \beta_\omega \\ \beta_\theta \end{bmatrix} \tag{11}$$

where $\Delta \underline{q}$ is the reduced quaternion difference between the quaternion estimated using only GPS (from Eq. 9) and the quaternion propagated by the Gyro (from Eq. 7), as Fig. 2 shows.

3. FAULT DIAGNOSIS

According with Nyberg (1999) a fault is a deviation not permitted of at least one characteristic property or variable of the system from acceptable or usual or standard behavior, while a failure is a permanent interruption of a system ability to perform a required function under specified operating conditions. Fault detection is 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. Multiple faults are not considered.

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 detected Fault in the Gyro signal, the level of uncertainty growth, but only until the level obtained by the solution using GPS alone. The occurrence of detected fault in the GPS signal can induce partial or completely degradation in the uncertainty level, depending on the duration and the extension of the interruption.

The number of GPS constellation satellites in view, considering an angle mask of 10°, in most of time is about 6 to 8 satellites for missions with earth pointing and in Low Earth Orbit. In this scenario, there is internal sufficient 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. Fig. 2b shows the proposed configuration.

The second part of the proposed algorithm uses a Kalman filter to implement the state estimation of Eq. 10. This equation estimates the quaternion corrections, Gyro drift-rate bias corrections and GPS error. After quaternion correction, the attitude is propagated using the Gyro corrected output, generating a sample rate better than the estimated using only GPS. Gyro output is correct using the drift-rate bias estimated by the filter.

In case of GPS or Gyro fault, attitude remains available by the GPS or by propagation based on the Gyro, that permits a completely fault diagnosis.

GPS and Gyros phases fault diagnosis control parameters shall be adjusted empirically and the efficiency of the algorithm depends direct from this adjustment. High values has the inclination to growth the percentage of fault not detected and increase the minimum fault intensity that can be detected, while low values of control parameters make those faults more perceptive, but accentuate the rate of false alarms.

3.1 GPS FAULT DIAGNOSIS

Figure 2b shows parallel processors results in the GPS fault diagnosis Box which makes the comparison of all the cost functions and sent to the Kalman filter the correct value of attitude matrix and variance.

If $J_p < J_o \ \forall p$, where p is the number of the processor, J_o is the control parameter, the algorithm of GPS fault diagnoses considers that no fault is detected and results $\theta^* = \theta$. In another case, if $J_p > J_o \ \forall p \neq q$, but $J_q < J_o$, then a fault is detected in satellite q, and results $\theta^* = \theta_q$. Where the number of processors *p* varies from 1 to (*m*+1).

In another cases, all the measured data are rejected following to the new measurement instant propagating the actual attitude. This condition is not simulated in this article.

3.2 GYRO FAULT DIAGNOSIS

In Fig. 2a can be seen that the Kalman filter is supplied with information by the difference between the Gyro propagated quaternion and the estimated attitude using only GPS data. The Filter is described in Section 2.2 and the propagation considered is described below.

If the rotation vector defined by Eq. 12,

$$\Delta\theta = \int_t^{t+\Delta t} \omega(t')dt' \tag{12}$$

is small, than the solution of Eq. 4 is

$$\overline{q}(t+\Delta t) = M(\Delta\theta)\overline{q}(t) \tag{13}$$

where

$$M(\Delta\theta) = \cos(|\Delta\theta|/2) I_{4x4} + \frac{sen(|\Delta\theta|/2)}{|\Delta\theta|}\Omega(\Delta\theta) \tag{14}$$

Gyro Fault is considered if the quadratic mean residues growth above the variance of the residues. This can be seen in next section, simulation results.

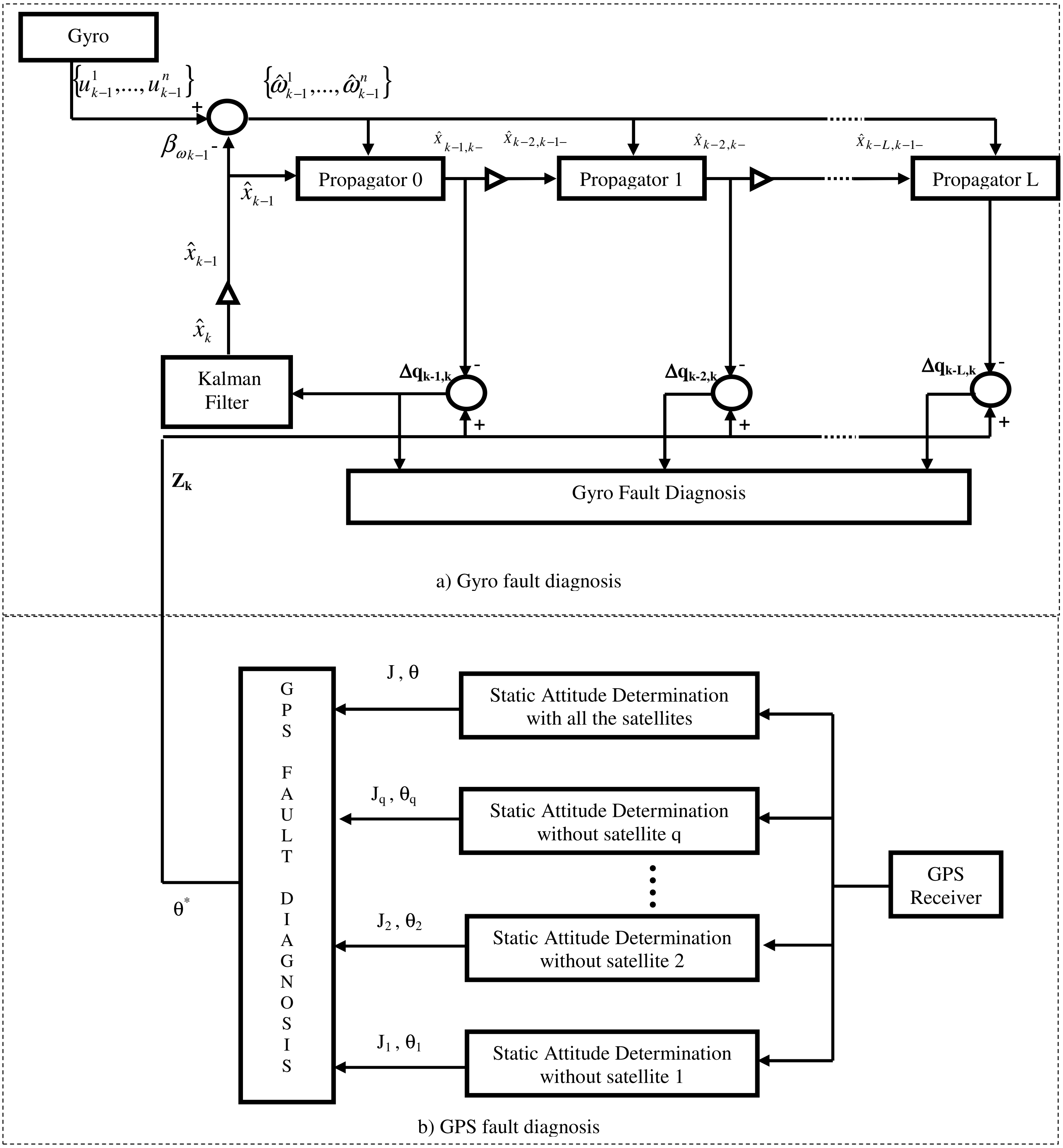

Figure 2. Fault diagnosis algorithm schema.

4. SIMULATION AND EXPERIMENTAL RESULTS

Louro and Lopes (2004) and shows simulation results of the GPS part of the algorithm (Fig. 2b). Louro and Lopes (2005) and shows simulation results of the GPS and Gyro part considering only one delay in the algorithm (part of Fig. 2) This work presents an extension of the previous effort with an update of the algorithm and experimental results of the GPS part and simulation of Gyro part of the algorithm.

Faults on the GPS signals may occur for instance due to receiver electronics instability and environment interferences, including multipath. In order to test the procedure, single faults were simulated on the GPS signals. These faults do not intend to represent any specific physical event, but they exemplify the sensitivity of the fault diagnosis procedure to the intensity of an arbitrary fault.

The simulation considers a square structure with lateral of 1 meter where three GPS antennas are positioned and the Gyro are attached to this structure to sense the same movement. Antenna number 1 are positioned at the reference of the tetrahedron, antenna number 2 positioned in x direction and antenna number 3 positioned in y direction, forming two baselines: baseline 12 and baseline 13.

4.1 GPS PART EXPERIMENTAL RESULTS

The Experimental data used as input for the algorithm were acquired using GPS CMC Allstar.

After generation of 200s of data with a sample rate of 1Hz, data was corrupted adding 5 cycles between instants 30s and 40s in GPS satellite 27 as an error, establishing with this a fault to be identified by the algorithm.

Figure 3 presents results for the cost function where can be seen that during the introduction of the failure the cost function for all other combination increase sensible, except for p = 27. Figure 4 presents the 3-axis attitude estimated using only GPS, after Fault detection. Those values are used to calculate the quaternion difference between GPS and Gyro to be used by the Kalman filter as input measurement (see Fig. 2). More results of this part of the algorithm can be found in Louro and Lopes (2004).

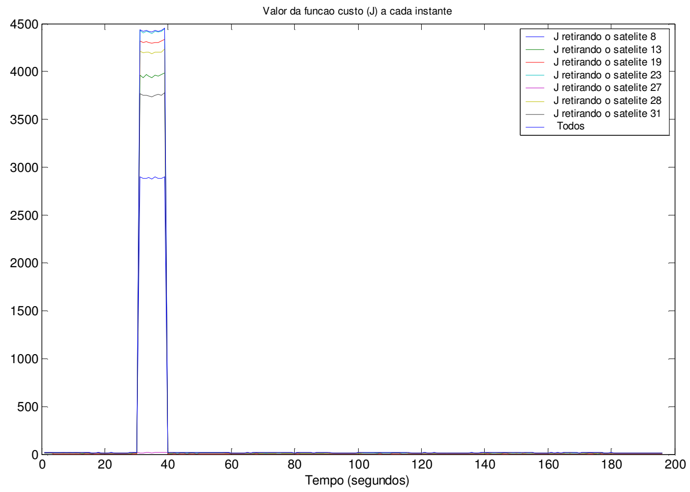

Figure 3. Cost function.

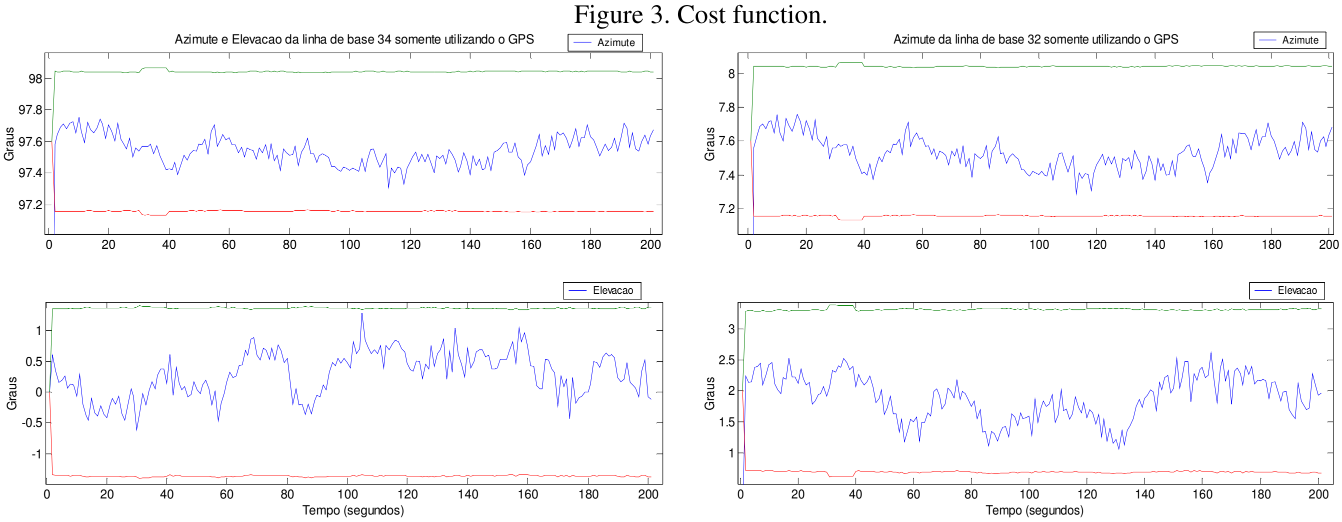

Figure 4. 3-axis attitude estimated using only GPS, 1 sigma. (Attitude matrix transformed for azimuth and elevation).

4.2 GYRO PART SIMULATION RESULTS

Gyro is simulated using commercial software Spacecraft Control Matlab Toolbox from Princeton Satellite Systems and using the characteristics of the Crista MEMS Gyro (Crista, 2005), used in Brown and Lu (2004).

Two cases were simulated: the first introduce an error of 0.03 radians/second in the drift-rate Bias from instants 70s to 80s, establishing with this a fault to be identified by the algorithm and the second case simulates a complete fault in the GPS sensor between instants 70 to 100 seconds.

First case - This part of the algorithm was simulated using two delays, Figure 5 shows the filtered and propagated 3-axis attitude. The filtered 3-axis attitude is showed in each exact second and between the exact seconds with a sample rate of 10Hz is plotted the propagated attitude.

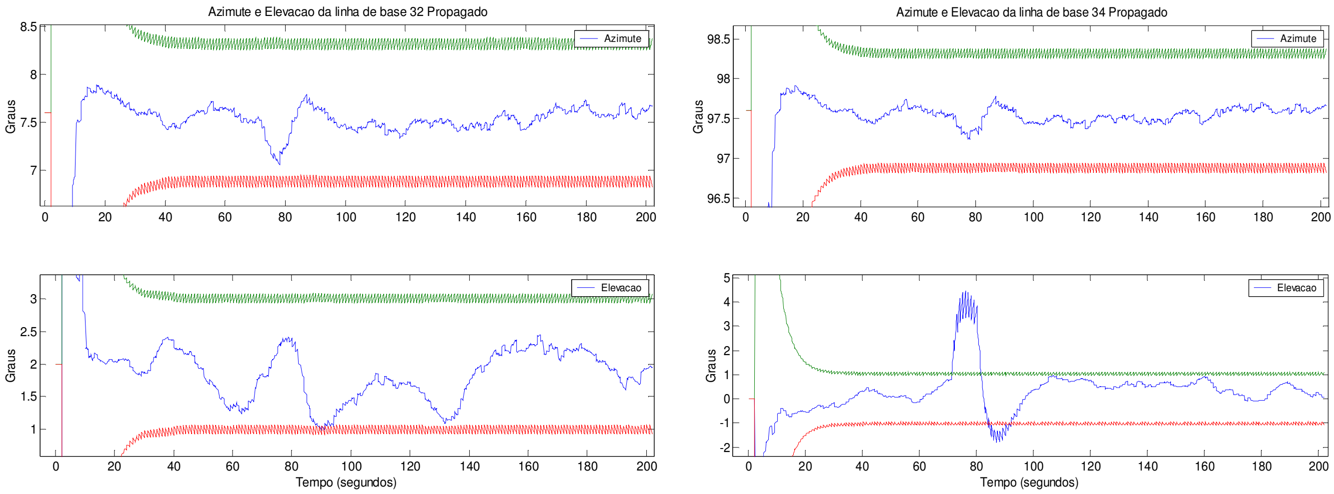

Figure 5. 3-axis filtered and propagated attitude, 1 sigma. (Attitude matrix transformed for azimuth and elevation).

Figure 6 shows the estimated state, where can be seen the instant that the algorithm detect a fault between instants 70s to 80s.

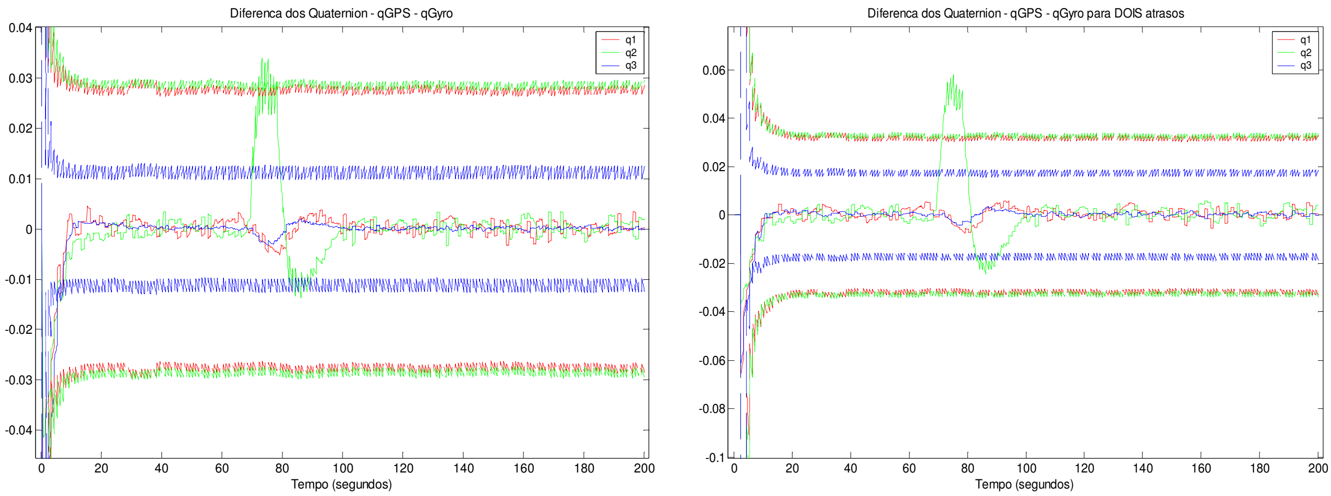

Figure 6. State Estimated, 1 sigma.

Second case - Figure 7 shows the quaternion difference between GPS quaternion and propagated quaternion, where can be seen the instant that the algorithm propagates the attitude without the GPS data update between instants 70s to 100s. During those instants can be seen in this figure that the propagation increases the error and the corresponding uncertainty also.

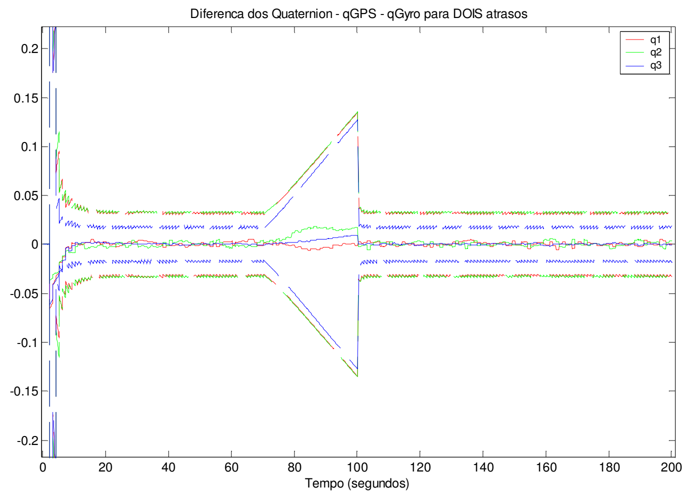

Figure 7. Difference between GPS estimated quaternion and Gyro propagated quaternion, 2 sigma.

5. CONCLUSIONS

An updated algorithm is presented for fault diagnosis based on the combination of two attitude sensors: low cost Gyro (MEMS) aided by a GPS receiver with multiple antennas. As showed above, GPS experimental results demonstrate that the algorithm has the capability to identity Gyro faults in real time and also propagates the attitude without the GPS reference during a certain period of time.

Experimental results were presented to demonstrate the capability of the algorithm to detect faults in the GPS or in the Gyro. The algorithm detects and isolates cycle error in the GPS, Figs. 3 and 4, also detects, and tries to minimize, the drift-rate bias error in the Gyro, Figs. 5 and 6, and propagates the attitude during a certain period of time without the GPS attitude data, Fig. 7.

This algorithm explores benefits of both sensors with different characteristics and is attractive to low cost micro-satellites.

6. REFERENCES

Brown, A.K. and Lu, Y, 2004, "Performance Test Results of an Integrated GPS/MEMS Inertial Navigation Package". ION GNSS 2004, Long Beach, CA, Sept.

Crista, 2005, "Crista IMU Specification", http://www.cloudcaptech.com/crista_imu.htm.

Kaplan, E. D., 1996, "Understanding GPS: principles and applications", London, UK.

Lefferts, E.J., Markley, F.L. e Shuster, M.D., 1982, "Kalman Filtering for Spacecraft Attitude Estimation". Journal of Guidance, Control, and Dynamics, Vol. 5, pp. 417-429.

Lopes, R. V. F.; Silva, A. R.; and Prado, A. F. B. A., 2002, "Navigation and Attitude Estimation from GPS Pseudorange, Carrier Phase and Doppler Observables." 53rd International Astronautical Congress IAC, Houston, Texas, USA, 10-19 Oct.

Lopes, R.V.F.; Kuga, H.K. 2002. Determinação de Atitude em 3 Eixos por Interferometria GPS. In: Durão, O.S.C.; Schad, V.R.; Kuga, H.K.; Lopes, R.V.F.; Carvalho, H.C.; Esper, M. Plataforma Integrada Sensores Inerciais/GPS, Anexo 1. São José dos Campos. [INPE-9293—PRP/234].

Louro, A.C., Lopes, R.V.F., "Diagnose de Falhas na Determinação Autônoma de Atitude de Micro-Satélites por meio de GPS e Giros". IV Simpósio Brasileiro de Engenharia Inercial – SBEIN, São José dos Campos – SP – Brasil Nov.

McMillan, J. C.; Bird, J. S. and Arden, D. A. G., 1993, "Techniques for Soft-Failure Detection in a Multisensor Integrated System." Journal of The Institute of Navigation, Vol. 40, No. 3, Fall, pp. 359-380.

Misra, P., Enge, P, 2001, "Global Positioning System: signals, measurements, and performance". Ganga-Jamuna Press.

Nyberg, M., 1999, "Model Based Fault Diagnosis Methods, Theory, and Automotive Engine Applications." Department of Electrical Engineering, Linköping University, Linköping, Sweden, [Dissertation No. 591].

7. RESPONSIBILITY NOTICE

The authors are the only responsible for the printed material included in this paper.