APPLICATION OF SELF-TUNING CONTROLLERS TO A REMOTE SENSING SATELLITE ATTITUDE CONTROL SYSTEM

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Abstract. An attitude control system for a remote sensing satellite is designed by using self-tuning techniques. The satellite attitude control system is modelled by a linear vector differential equation which is corrupted by random disturbances. By employing a cost function with penalized control terms, it is possible to find an implicit self-tuning controller that will adapt itself to all mission phases and will be simple enough to be implemented with microcomputers. Parameter estimation is performed by Kalman filtering techniques. The controller performance is verified by numerical simulation using a model of a real satellite and the results show good compliance with the satellite point requirements.

INTRODUCTION

The purpose of this work is to design an attitude controller for the Brazilian Remote Sensing Satellite that will keep its picture taking camera aligned with the local vertical. This system should have a good performance in all mission phases and should be simple enough in order to be implemented in an onboard microcomputer.

The satellite in question has the following characteristics: the principal moments of inertia are 30.0 Kgm^2, 28.0 Kgm^2 and 17.0 Kgm^2, respectively in the roll, pitch and yaw axes; polar synchronous orbit with a semimajor axis of 7,028,120 m, eccentricity of 7.7×10^{-6}, inclination of 176.01^0, perigee argument of 5.19^0 and mean anomaly of 229.7^0. The employed sensors are non inertial (horizon sensor and sun sensor) and inertial (gyroscopes). The actuators are gas jets and/or reaction wheels.

The approach employed in the design of the attitude control systems considers that the attitude dynamics is modelled by a linear vector differential equation subject to random disturbances. Then, following the same approach proposed by Borison (1979) and Koivo (1980), a cost function is defined taking into account the pointing accuracy and the control effort. By minimizing this function a controller is obtained. Since the parameters of the attitude dynamics model change with the mission phases, a parameter estimation procedure is performed using Kalman filter techniques, leading to a implicit self-tuning controller that adapts itself to the changes. In order to implement the algorithm so derived a preprocessing of the non inertial sensor measurements is performed by using the QUEST algorithm, developed by Shuster and Oh (1981). The verification of the controller behaviour is accomplished by numerical simulation of the system controller plus satellite dynamics in a digital computer.

DERIVATION OF THE CONTROLLER STRUCTURE

The controller to be used is the one proposed by Koivo (1980). So the system to be controlled is supposed to be represented by a linear vector difference equation:

$$A(q^{-1})\underline{y}(k) = B(q^{-1})\underline{u}(k-d) + C(q^{-1})\underline{e}(k) + \underline{D}$$

(1)

where: $\underline{y}(k) \in R^m$ is the output variable vector of the system to be controlled; $\underline{u}(k-d) \in R^m$ is the control variable vector and d is the implicit system delay; $\underline{e}(k) \in R^m$ is a disturbance vector; $\underline{D} \in R^m$ is a contant vector corresponding to the system response to a null input; q^{-1} is the backward shift operator and $A(q^{-1}), B(q^{-1})$ and $C(q^{-1})$ are polynomial mxm matrices given by:

$$A(q^{-1}) = I + A_1 q^{-1} + \ldots + A_{na} q^{-na}$$ (2)

$$B(q^{-1}) = B_0 + B_1 q^{-1} + \ldots + B_{nb} q^{-nb}$$ (3)

$$C(q^{-1}) = I + C_1 q^{-1} + \ldots + C_{nc} q^{-nc}$$ (4)

The cost function to be considered will be given by:

$$J(\underline{y},\underline{u},\underline{W},k) = E\{||P(q^{-1})\underline{y}(k+d) - R(q^{-1})\underline{W}(k)||^2 + ||Q(q^{-1})\underline{u}(k)||^2\}$$ (5)

where: $\underline{W}(k) \in R^m$ is a known reference vector; $P(q^{-1})$, $R(q^{-1})$ and $Q(q^{-1})$ are polynomial mxm matrices given by:

$$P(q^{-1}) = P_0 + P_1 q^{-1} + \ldots + P_{np} q^{-np}$$ (6)

$$R(q^{-1}) = R_0 + R_1 q^{-1} + \ldots + R_{nr} q^{-nr}$$ (7)

$$Q(q^{-1}) = Q_0 + Q_1 q^{-1} + \ldots + Q_{nq} q^{-nq}$$ (8)

$E\{.\}$ is the expected value operator and $||\underline{v}||$ is the vector $\underline{v}$ norm.

The cost function is minimized over all admissible strategies. A control strategy is admissible if

the value of the control signal at time t_k, $\underline{u}(k)$, is a function of all observed outputs up to time t_k and all previously applied control signals. By using an estimator for the future term resulting from the $P(q^{-1})\underline{y}(k+d)$ product that appears in the cost function and minimizing in relation to $\underline{u}(k)$, one obtains the following expression for the optimal strategy:

$$F(q^{-1})\underline{y}(k) + G(q^{-1})\underline{u}(k) + Z(q^{-1})\underline{w}(k) + \underline{D} = 0 \qquad (9)$$

where $F(q^{-1})$, $G(q^{-1})$ and $Z(q^{-1})$ are polynomial mxm matrices given by:

$$F(q^{-1}) = F_0 + F_1 q^{-1} + \ldots + F_{nf} q^{-nf} \qquad (10)$$

$$G(q^{-1}) = G_0 + G_1 q^{-1} + \ldots + G_{ng} q^{-ng} \qquad (11)$$

$$Z(q^{-1}) = Z_0 + Z_1 q^{-1} + \ldots + Z_{nz} q^{-nz} \qquad (12)$$

The expression (9) can be used to obtain an implicit version for a self-tuning controller that allows the direct estimation of the controller parameters when there is no complete knowledge of the system model. So, in this way, the elements of matrices F_i, G_i, and Z_i, and vector $\underline{D}$ are estimated at each sample interval and, by the substitution of their estimated values in equation (9), it is possible to obtain directly the value of the control variable $\underline{u}(k)$ to be applied to the system.

In order to update the controller parameter it was decided to use a Kalman filtering procedure instead of the usual least square procedure. The reason was to verify the efficiency of this method in the case of rapid changes of parameters in the satellite dynamics.

IMPLEMENTATION OF THE SELF TUNING CONTROLLER TO THE ATTITUDE CONTROL OF A REMOTE SENSING SATELLITE

Usually for a remote sensing satellite the attitude control system has to keep the satellite three orthogonal axes (roll, pitch and yaw axes) aligned with a fixed reference system. In the present situation the objective is to maintain the reference system fixed in the satellite that coincides with the satellite principal axes of inertia lined up with the orbital reference system (see Fig. 1) in such a way that the angles and angular rates in roll, pitch and yaw are very near zero values. Therefore, for a good attitude control it is necessary to have information about the satellite orientation and angular rates in the orbital reference system. The information about orientation that are going to be used is given by the relative quaternions. To obtain them a pre-processing of the measurements made by the non-inertial sensors(earth sensor and sun sensors) are performed using the algorithm called QUEST, QUaternion ESTimator (Shuster and Oh, 1981). This is a static algorithm employed usually in attitude determination, generating an optimum least-square estimator for the quaternion vector and its error covariance matrix. So the set of measurements to be available for the controller will be the QUEST algorithm output plus the outputs of three gyroscopes, each one aligned with one of the satellite axes which gives a total of seven variables to be handled. The number of actuators either in the case of using gas jets or in the case of using reaction wheels is always three (one actuator per axis). Since the proposed controller, described above, has to have the number of inputs equal to the number of outputs, it is necessary to reduce the number of inputs by four. This is achieved by defining an artificial three dimension

measurement vector $\underline{y}(k)$, considered as the output of the system at time t_k, and defined as follows:

$$\underline{y}(k) = K_1 \underline{\tilde{q}}(k) + K_2(\underline{w}(k) - \underline{w}_r(k)) \qquad (13)$$

where $\underline{\tilde{q}}(k)$ is a three dimension vector whose components correspond to the vectorial part of the quaternion representing the satellite attitude at time t_k, in relation to the orbital reference frame; $\underline{w}(k)$ is the satellite angular velocity at time t_k measured by the gyroscopes; $\underline{w}_r(k)$ is the reference angular velocity, i.e., the velocity that the satellite should have in order to be aligned with the local vertical; K_1 and K_2 are 3x3 weighting matrices. It can be proved that the condition $\underline{y}(k) = \underline{0}$ is necessary and sufficient for keeping the satellite body axes aligned with the orbital reference frame. Taking into account these facts, the cost function is defined as:

$$J(\underline{y},\underline{u},k) = E\{||P(q^{-1})\underline{y}(k+d)||^2 + \\ + ||Q(q^{-1})\underline{u}(k)||^2\} \qquad (14)$$

with $P(q^{-1}) = P_0$; $Q(q^{-1}) = Q_0 + Q_1 q^{-1}$; $\underline{u}(k)$ is a three dimensional control variable corresponding to torques applied to the satellite either by gas jets or by reaction wheels; P_0, Q_0, Q_1 are 3x3 diagonal matrices. After some experiments it was found out that for a good compromise between control performance and computational requirements the attitude dynamics should be modelled by:

$$[I + A_1 q^{-1} + A_2 q^{-2}]\underline{y}(k) = [B_0 + B_1 q^{-1} + \\ B_2 q^{-2}]\underline{u}(k-2) + \underline{e}(k) + \underline{D} \qquad (15)$$

For this system the implicit self-tuning controller is of the form (Koivo and Tanttu, 1985):

$$\underline{u}(k) = -G_0^{-1}\{F_0\underline{y}(k) + F_1\underline{y}(k-1) + G_1\underline{u}(k-1) + \\ G_2\underline{u}(k-2) + \underline{H}\} \qquad (16)$$

where G_0, F_0, F_1, G_1, G_2 are 3x3 matrices and $\underline{H}$ is three dimensional vector whose values are estimate using the model:

$$\Phi(k) = F_0\underline{y}(k-2) + F_1\underline{y}(k-3) + G_0\underline{u}(k-2) + \\ G_1\underline{u}(k-3) + G_2\underline{u}(k-4) + \underline{H} \qquad (17)$$

$\Phi(k)$ being given by:

$$\Phi(k) = P_0\underline{y}(k) + Q_0\underline{u}(k-2) + Q_1\underline{u}(k-3) \qquad (18)$$

The estimation of G_0, F_0, F_1, G_1, G_2 and $\underline{H}$ are performed considering that at each interval of time their values are random variables with a prescribed variance. In this way it is possible to keep tracking of sudden changes in the attitude dynamics and also to use Kalman filter techniques in the estimation process.
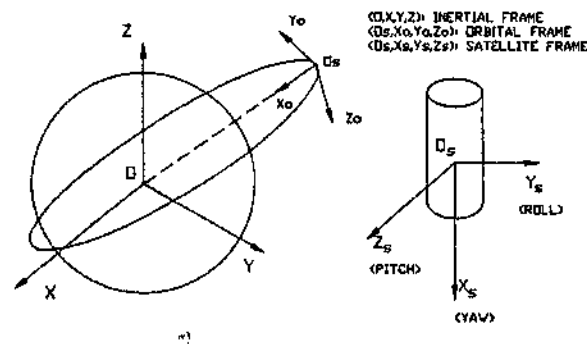

(O,X,Y,Z): INERTIAL FRAME
(Os,Xo,Yo,Zo): ORBITAL FRAME
(Os,Xs,Ys,Zs): SATELLITE FRAME

Fig. 1. Reference frames.

TESTS PERFORMED USING THE PROPOSED CONTROLLER

In order to verify the performance obtained by the self-tuning controllers being proposed , two kinds of tests were done. First the initial satellite attitude was considered to be outside the range for the nominal mode at the initial time. Then , by using the self-tuning controller, it was verified the attitude dynamical behaviour. The second test considered that the satellite had its solar arrays retracted at initial time, with an attitude outside nominal mode, then the attitude controller is turned on and the attitude after some time reachs the nominal mode region. After this moment, the solar arrays are deployed and the self-tuning controller should adapt itself to the new situation, keeping the satellite aligned.

For both tests the following parameters were used:

$$K_1 = \begin{bmatrix} 1.5 & 0.0 & 0.0 \\ 0.0 & 1.5 & 0.0 \\ 0.0 & 0.0 & 1.5 \end{bmatrix} \tag{19}$$

$$K_2 = \begin{bmatrix} 30 & 0 & 0 \\ 0 & 25 & 0 \\ 0 & 0 & 25 \end{bmatrix} \tag{20}$$

$$P_0 = Q_0 = Q_1 = I \text{ (identity matrix)} \tag{21}$$

A sample of the results obtained in the first test is shown in Fig. 2 to Fig. 13. The initial position for the attitude is specified by a 25^0 in roll, 25^0 in yaw and -25^0 in pitch. Figure 2 to Fig. 7 describe the attitude dynamical behaviour, whereas Fig. 8 to Fig. 13 show the angular rates in the three body axes. By these results it can be seen that nominal mode attitude is reached in about 3 minutes in the worst case. The deviations in angular displacement and angular rate are 0.5^0 and $0.15^0/s$ in the worst case for all axes.

A sample of the results obtained in the second type of tests is presented in Fig. 14 to Fig. 25. The attitude initial value is the same as in the first test. The initial values for the satellite moments of inertia are 24.0 Kg.m^2 in roll, 12.0 Kg.m^2 in pitch and 24.0 Kg.m^2 in yaw. After 1000 seconds the moment values are changed to the nominal ones (28.0 Kg.m^2 in roll, 17.0 Kg.m^2 in pitch and 30.0 Kg.m^2 in yaw).Examining the results presented in the figures, it can be seen that the proposed controller keeps the pointing precisions around the values of 0.5^0 and $0.15^0/s$ in angular displacement and angular rate respectively.

CONCLUSIONS

Comparing the results from the simulations with actual remote sensing satellite data it can be checked that the controller performance is satisfactory. For instance, the pointing precisions for the LANDSAT satellite are 0.7^0 in angular deviation and 0.04^0 /s in angular rate deviation. Using the self-tuning controller, the deviations achieved were in the range of 0.5^0 and $0.06^0/s$ respectively. The result in angular rate deviation can be explained by the fact that the Brazilian Remote Sensing Satellite has very small moments of inertia when compared with the LANDSAT,so this satellite has a better performance.
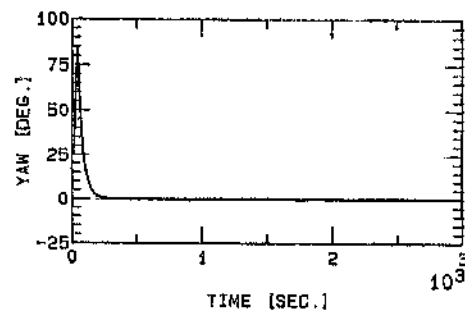
Fig. 2. Yaw angle.
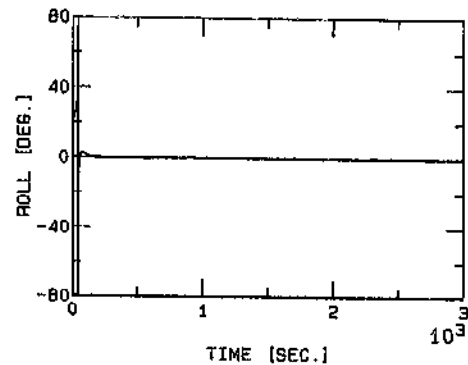
Fig. 3. Roll angle.
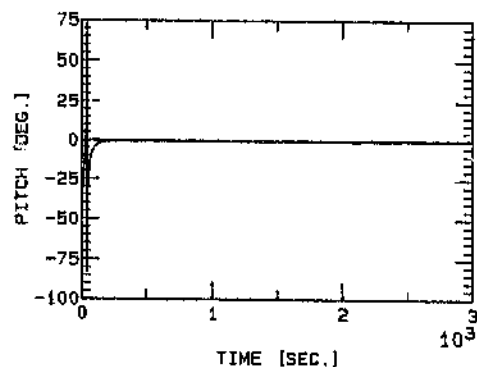
Fig. 4. Pitch angle.
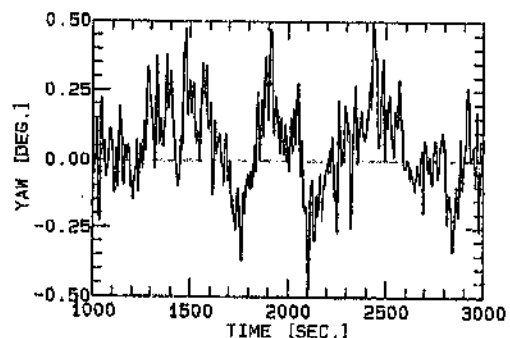
Fig. 5. Yaw angle after 1,000 seconds.
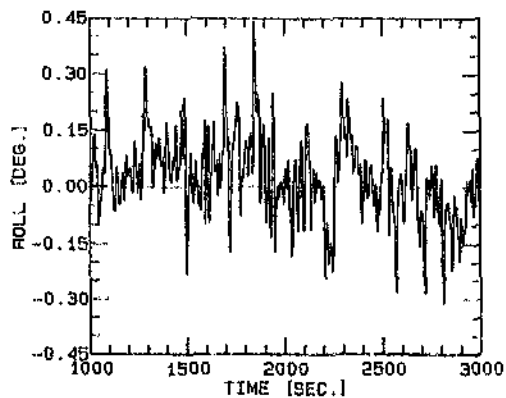
Fig. 6. Roll angle after 1,000 seconds.
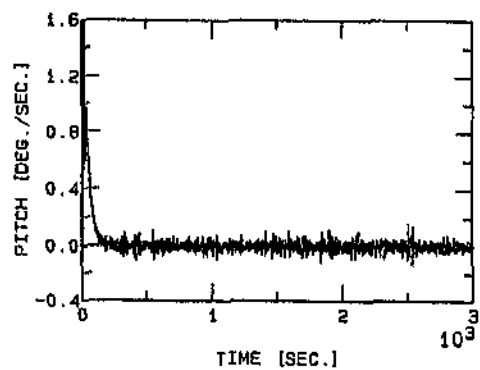
Fig. 10. Pitch rate.
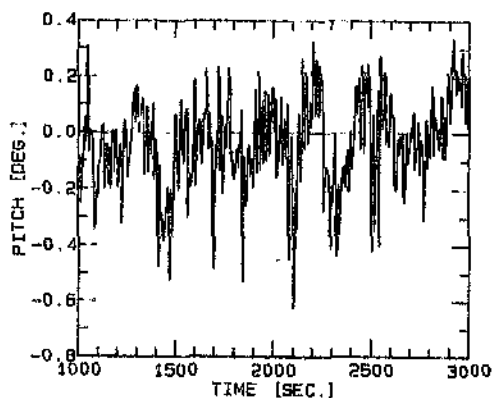
Fig. 7. Pitch angle after 1,000 seconds.
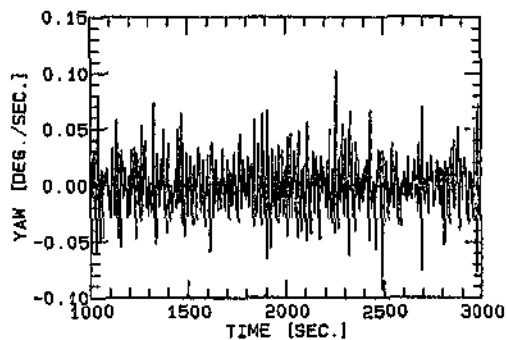
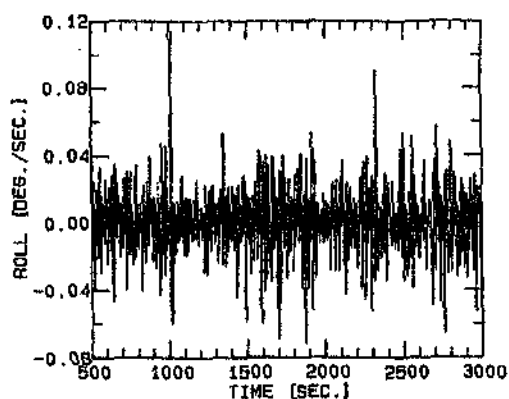Fig. 11. Yaw rate after 1,000 seconds.
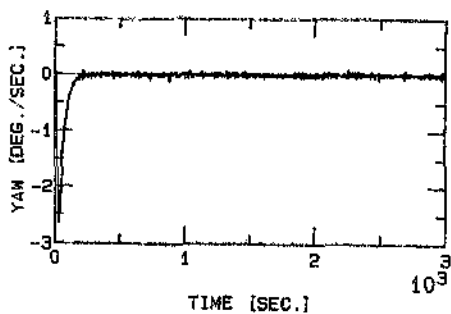
Fig. 8. Yaw rate.
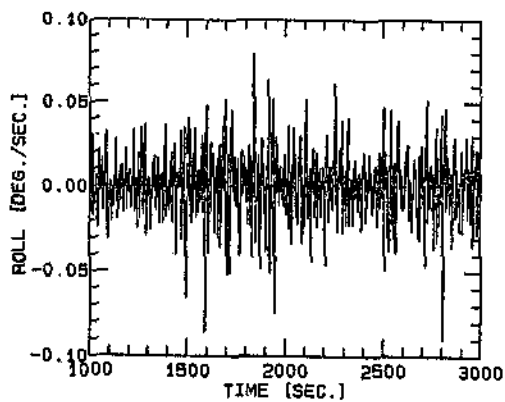
Fig. 12. Roll rate after 1,000 seconds.
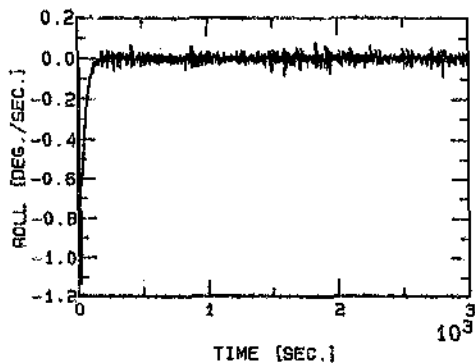
Fig. 9. Roll rate.
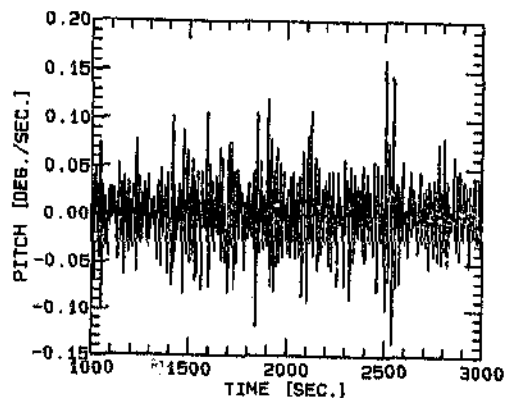
Fig. 13. Pitch rate after 1,000 seconds.
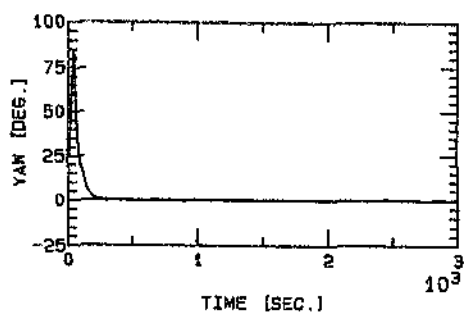
Fig. 14. Yaw angle.

Fig. 18. Roll angle after 500 seconds.

Fig. 15. Roll angle.
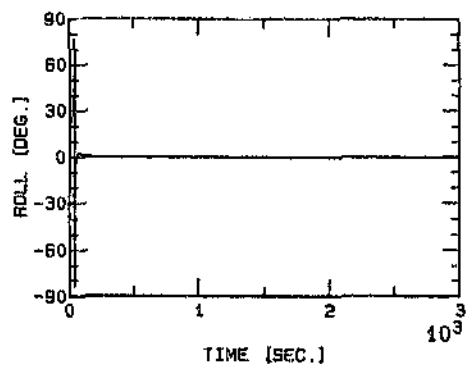
Fig. 19. Pitch angle after 500 seconds.
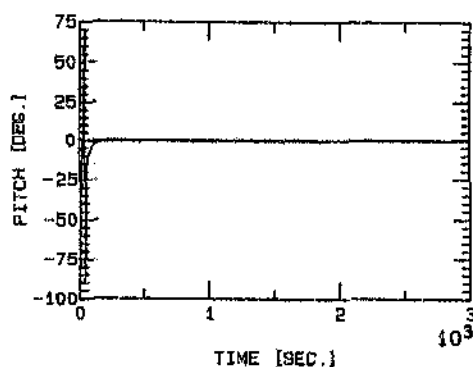
Fig. 16. Pitch angle.
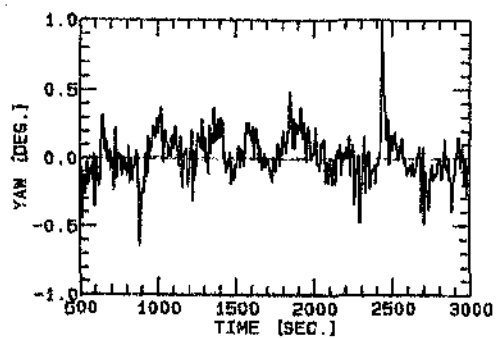
Fig. 20. Yaw rate.

Fig. 17. Yaw angle after 500 seconds.
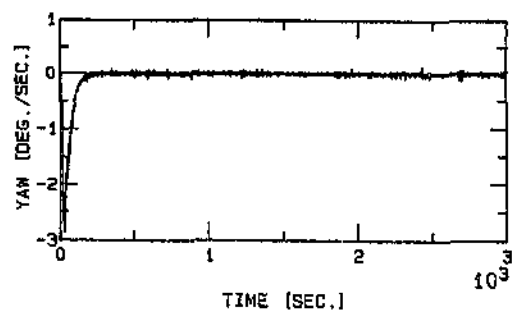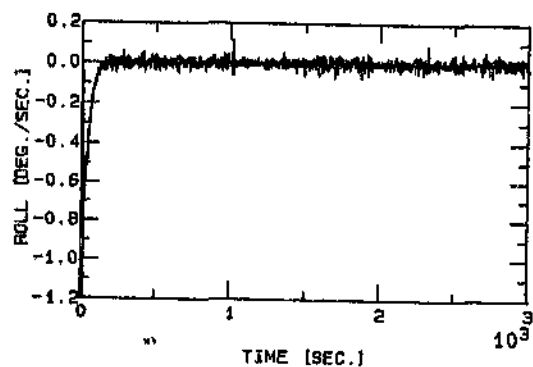
Fig. 21. Roll rate.
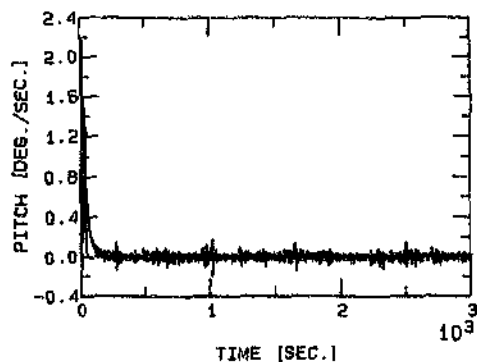
Fig. 22. Pitch rate.

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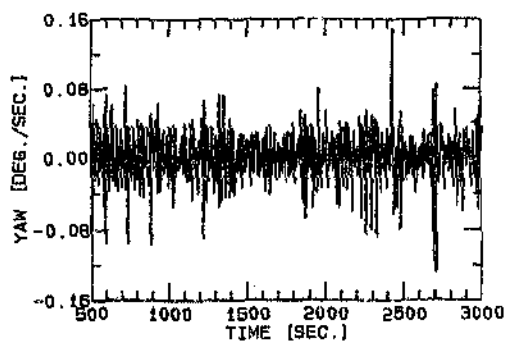
Fig. 23. Yaw rate after 500 seconds.

Fig. 24. Roll rate after 500 seconds.
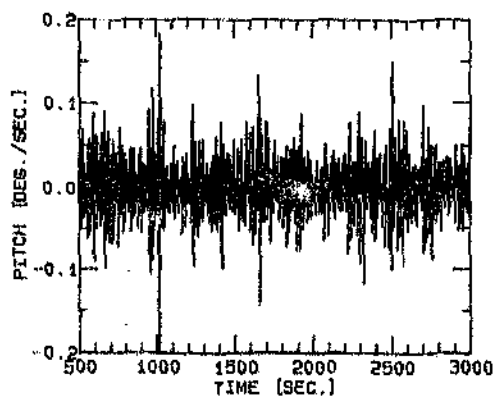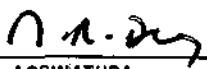
Fig. 25. Pitch rate after 500 seconds.

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TÍTULO

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Attitude Control System

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