Attitude Control of a Satellite Using sliding mode

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***Abstract*— This paper is dedicated to the problem of a satellite attitude control. The control method is based upon a robust sliding mode control. A sliding mode controller has been designed in such a way that the state variables of the system converge to the desired values. Where the controller parameters can drive the state to hit the sliding surface. The steps of the control design are presented.**

Keywords— spacecraft attitude control; quaternion parameter; output tracking control; sliding mode control

INTRODUCTION

Most conventional techniques for controlling non-linear systems are based on the precise knowledge of the mathematical model. In case, we are confronted with inaccuracies due to uncertainties related to the studied process (non known or difficultly identifiable parameters), or dynamic neglected (simplified model), these techniques cannot be used.

The sliding mode control, due to its robustness to uncertainties and external disturbances can be applied to uncertain and disturbed [1], [2] nonlinear systems. However, the presence of the signum function in the sliding mode control, causing a phenomenon of chattering which consists of abrupt and rapid changes in the control signal, which can causes the high-frequency process and damages.

The sliding mode technique permits usage of lower order system model for generating control commands, which includes unmodeled dynamics or uncertainties, and stabilizes the plant faster and robustly under bounded disturbance [3].

The chattering at high frequencies is not desired because it may cause vibration. Chattering may be eliminated by replacing saturation instead of signum function [4]. However, in that case non-zero tracking errors exist, which can be made small by taking a tiny region for saturation and also, saturation is limited with hardware capability and reduction of accuracy and robustness as introduced [5] and [6]. On the other hand, chattering may be eliminated by pulse modulation as done [7].

I. BASIS DYNAMICS AND CONTROL THEORY:

I.2. Attitude dynamics and kinematics:

For the sake of simplicity of the analysis, the satellite is assumed to be a perfect rigid body.

The three basis axes of the dynamics of the rigid satellite are described in the following form [8]:

$$J\dot{\omega} = -\omega \times J\omega + u + d \tag{1}$$

Where:

J: stands for the inertia matrix and the satellite inertia moments;

ω**:** is the satellite angular velocity, $\omega \in \mathcal{R}^{3\times 1}$

u**:** is the control vector, $u \in \mathcal{R}^{3\times 1}$;

d: the bounded disturbances acting on the satellite;

For simplicity, let

$$f = -I^{-1}[\omega \times I\omega],\ b = I^{-1}$$

Therefore,

$$\dot{\omega} = f + bu + d \tag{2}$$

Representation through quaternion parameter has the property of non-singularity and it is free from the trigonometric component. Therefore, this representation is widely used to study the attitude behavior of spacecraft [9].

$$q = \begin{bmatrix} q_{13} \\ q_4 \end{bmatrix} \tag{3}$$

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Where each element satisfies the following relation [10]:

$$q_{13} = \begin{bmatrix} q_1 \\ q_2 \\ q_3 \end{bmatrix} = \mathrm{I}\sin\frac{\phi}{2}, q_4 = \cos\frac{\phi}{2} \quad (4)$$

I: is the principal Euler vector axis and $\emptyset$ is the corresponding main angle.

The kinematics of the satellite model is the part which expresses the relation between the attitude and angular velocities of the body [9]. The kinematics equations through unit quaternion representation are given as the temporal derivative of the quaternion as [11]:

$$\dot{q} = \frac{1}{2}\Omega(\omega)q = \frac{1}{2}\Xi(q)\omega \quad (5)$$

For which:

$$\Omega(\omega) = \begin{bmatrix} -[\tilde{\omega}] & \omega \\ -\omega^T & 0 \end{bmatrix}, \Xi(q) = \begin{bmatrix} q_4 \mathrm{I}_{3\times 3} + [\tilde{q}_{13}] \\ -q_{13} \end{bmatrix} \quad (6)$$

The notation $[\tilde{\omega}]$ represents the matrix

$$[\tilde{\omega}] = \begin{bmatrix} 0 & -\omega_3 & \omega_2 \\ \omega 3 & 0 & -\omega_1 \\ -\omega_2 & \omega_1 & 0 \end{bmatrix} \quad (7)$$

And the same definition is applied to $[\tilde{q}_{13}]$ [11].

The quaternion must also satisfy the constraint in (4) as:

$$q^T q = q_{13}^T q_{13} + q_4^2 = 1 \quad (8)$$

2.2 The controller

The controller can be extended to a sliding mode controller for a system of relative degrees $\rho_1, \rho_2, ..., \rho_m$ [6], [12], [13].

The sliding surface is defined by:

$$s_i(t) = e_i^{(\rho i-1)} + c_{i(\rho i-1)} e_i^{(\rho i-2)} + \ldots + c_{i1} e_i - c_{i0} \int e_i dt = 0 \quad (9)$$

Where: i=1, 2...m.

The stable sliding conditions are given by:

$$\frac{1}{2}\frac{ds_i^2}{dt^2} \le -\eta_i |s_i|, \ \eta_i > 0 \ \text{(i=1, 2… m)} \quad (10)$$

After algebraic calculations the final form of sliding mode controller that satisfies the sliding condition in (10) is represented in the following form, [14]:

$$u = G^{-1}(-\dot{s} + Y^{(\rho)} - F - K\,\mathrm{sgn}(s)) \quad (11)$$

Where S= $[S_1,\ldots,S_m]^T$: is the sliding surface vector

K= $[K_1\ldots K_m]^T$: is a gain matrix to be determined

$Y^{(\,)}$ et G^{-1} and F: are matrix available in the feedback control design.

And Sgn(s) = $[\mathrm{sgn}(s_1),\ldots,\mathrm{sgn}(s_m)]^T$ represents the signum function vector.

For the the control the signal of the error is defined by [9]:

q_e= [q_{13e} q_{4e}] is the relative attitude error from a desired reference frame to the body-fixed reference frame of the spacecraft, which equal to:

$$q_e = q \times q_d^{-1} \quad (12)$$

With: q_d the desired attitude quaternion

The relative attitude error can be obtained by:

$$\begin{bmatrix} \dot{q}_{13e} \\ \dot{q}_{4e} \end{bmatrix} = \frac{1}{2}\begin{bmatrix} q_{4e} I + [q_{13e} \times] \\ -q_{13e}^T \end{bmatrix} \omega_e(t) \quad (13)$$

$$\omega_e(t) = \omega(t) - \omega_d(t)$$

Where: $\omega_d(t)$ is the desired angular velocity, and it must equal to zero. That's leads to:

$\omega_e(t) = \omega(t)$; So the error state vector (q_e, ω) is fed to the controller module to compute the control signal command.

2.3. Sliding Mode Design:

The sliding mode control can be constructed from the vector equation of the sliding surface of the form:

$$s = \dot{e} + c_1 e + c_0 \int e dt = 0 \quad (14)$$

Note that $e = [e_1, e_2, e_3]^T = q_e$ represents the error vector.

And $s = [s_1, s_2, s_3]^T$, the sliding mode controller is constructed according to the condition $\dot{s} = 0$ which by consequence leads to :

$$u = \beta^{-1}[\ddot{r} - \alpha - c_1(\dot{y} - \dot{r}) - c_0(y - r) - K\,\mathrm{sgn}(s)] \quad (15)$$

Where: $\dot{r}(t) = (q_{13}^f - q_{13}^0)(1/\tau)e^{-t/\tau}$

and $\ddot{r}(t) = (q_{13}^f - q_{13}^0)(1/\tau^2)e^{-t/\tau}$

K: is a third order diagonal square matrix, and « sgn » represents the signum function, and τ is the time constant

The signum function is replaced by a saturation function to minimize the phenomenon of "chattering" potential. In other words:

$$\mathrm{sgn}(f) = \begin{cases} \mathrm{sgn}(f) & pour |f| > \varepsilon \\ f/\varepsilon & pour |f| < \varepsilon \end{cases} \quad (16)\ (22)$$

II. Analysis and Simulation:

1. Simulation by the proposed control approach has been conducted. The mass moment of inertia of the satellite model is supposed to be:

J=diag [300, 320,250] (kg-m^2).

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The elements of the proposed initial attitude quaternion for the simulation are taken as follows:
q_{13}^{0}= [0.2, 0.4, 0.5]^{T}, q_{4}^{0} =0.742.
While the components of the target quaternion are given by:
q_{13}^{f}= [0.6,-0.2,-0.4]^{T}, q_{4}^{f} =0.663.
The controller gain matrices C_1, C_0 and K are set to be diagonal; they contain identical numbers of 0.2, 0.1and 0.05, respectively. The time constant (τ) is selected for the reference output to 10 seconds.

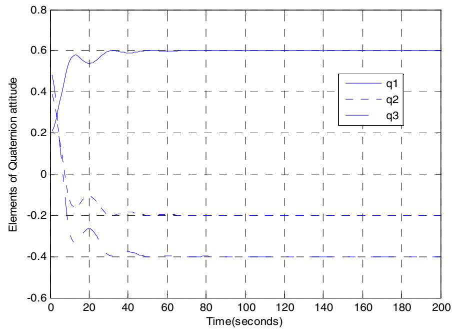

Fig.1: Parameter variation in attitude quaternion.

The simulation results of quaternion responses are plotted in Fig. 1. As it can be shown the quaternion parameter follows the final target

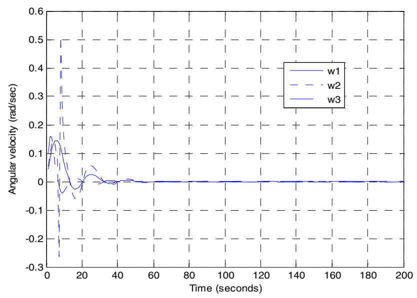

Fig.2: Angular velocity variation.

The responses of the angular velocity are obtained from the calculation of equations describing the behavior of the quaternion, and from (5). The convergence to zeros for the angular velocity is evident.

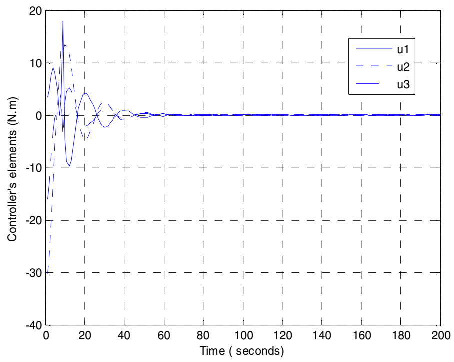

Fig.3: Components of Control.

The fig.3 shows that controller's elements convergence also to zeros.

The responses of the angular velocity of the satellite, the attitude quaternion and the components of control are presented. Overall, the performance of asymptotic tracking is achieved.

2. And with using other components of the attitude quaternion: q_{13}^{f}= [0.5, - 0.5, 0.1] ^{T}, q_{4}^{f} =0.7 we obtain these simulation results presented in fig.5, fig. 6 and fig.7. Identical initial condition and control parameters adopted. It is not easy to see the difference between the results in Fig.2 and Fig.5.

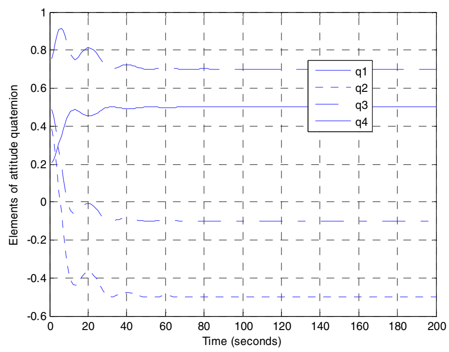

Fig.4: Variations in the parameters of the quaternion attitude.

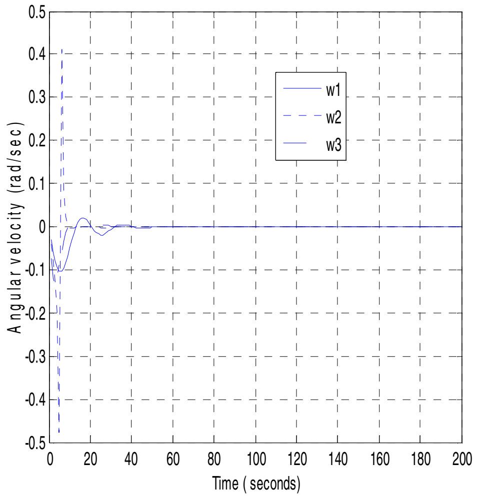

Fig.5: Variation in the angular velocity.

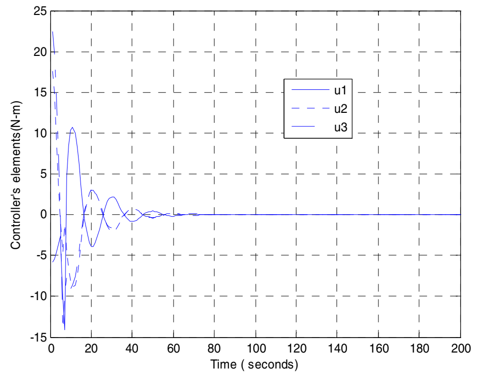

Fig.6: Control components

III. CONCLUSION:

In this paper, we presented a design of satellite attitude control. The sliding mode controller is used to designing a control law tracking. The quaternion vector defines the attitude parameters for global representation without singularities or trigonometric functions calculation.

In addition the computation of controller parameters is simple and its tuning is straightforward.

Therefore the use of attitude only as output function, leads us to a closed loop system stable.

The overall system performance has improved with respect to the classical sliding mode controller.

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