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Adaptive Sliding Mode Fault Tolerant Attitude Tracking Control for Flexible Spacecraft Under Actuator Saturation

Bing Xiao, Qinglei Hu, *Member, IEEE*, and Youmin Zhang, *Senior Member, IEEE*

***Abstract*—A novel fault tolerant attitude tracking control scheme is developed for flexible spacecraft with partial loss of actuator effectiveness fault. Neural networks are first introduced to account for system uncertainties, and an adaptive sliding mode controller is derived by using on-line updating law to estimate the bound of actuator fault such that any information of the fault is not required. To further address actuator saturation problem, a modified fault tolerant control law is then presented to ensure that the resulting control signal will never incur saturation. It is shown that the roll, pitch and yaw angle trajectories can globally asymptotically track the desired attitude in the face of faulty actuator, system uncertainties, external disturbances and even actuator saturation. A simulation example of a flexible spacecraft is given to illustrate the effectiveness of the proposed controller.**

***Index Terms*—Actuator saturation, adaptive sliding mode control, attitude tracking, fault tolerant control, flexible spacecraft, partial loss of actuator effectiveness.**

I. Introduction

ONE of the challenges in the design of attitude control is treating with uncertainties and external disturbances that spacecraft will encounter in operation, and this problem has been extensively studied in the literature based on several inspired control approaches, such as H-infinity control [1], [2], feedback linearization [3], [4] and adaptive control [5], [6]. However, most of the previous research deals only with uncertainties, assuming that there exists no actuator fault or failure during the entire attitude maneuvers. This assumption is rarely satisfied in practice because some catastrophic faults may occur due to the malfunction of actuators. As a result, if the designed attitude controller does not have any fault tolerance capability, then an abrupt occurrence of an actuator fault could ultimately fail the space mission. Therefore, fault tolerance capability is one of the main issues that need to be addressed in attitude control design. In order to enhance the system reliability and guarantee the control performance, a large number of researches have been carried out in the area of fault tolerant control (FTC) recently as one of very active research and development topics (see, e.g., [7]–[10]).

As for the application of FTC to spacecraft attitude control, to the best knowledge of the authors, there are minor investigations. In [11], a dynamics inversion and time-delay theory based fault-tolerant controller was developed to achieve attitude tracking control for a four momentum wheels actuated rigid satellite. A study on loss of thruster effectiveness faults for attitude tracking was performed in [12] by using adaptive control. Among the various design schemes of FTC, the robustness properties of sliding mode control to certain types of disturbances and uncertainties, especially to actuator faults, make it attractive in the field of spacecraft FTC. For instance, passive variable structure reliable control was proposed in [13] for the spacecraft attitude stabilization. Similarly, a sliding mode attitude tracking controller was presented for flexible spacecraft to compensate the additive and partial loss of reaction wheel effectiveness faults in [14]. Adaptive sliding mode control design technique has been thoroughly examined in [15] to perform attitude tracking maneuver for flexible spacecraft, wherein any given level of $\mathcal{L}_2$ gain disturbance attenuation from disturbance and system uncertainties to attitude output is successfully achieved.

Actuator saturation is another critical issue that needs to be tackled in attitude control design and also fault tolerant control systems design as pointed out in [7]. In practice, the output of spacecraft actuators is constrained or saturated. Therefore, if an unknown actuator fault occurs, then in spite of the fault, the attitude system would continue issuing its maneuver that may no longer be achievable. Consequently, the required control effort will quickly saturate the actuators while striving to maintain the attitude maneuvering performance, and the efforts to further increase the actuator output would not result in any variation in the output, and subsequently will destabilize the attitude. The most prominent method among all the solutions for actuator saturation is the anti-windup design, and Bang *et al.* [16] proposed an anti-windup control to achieve large angle spacecraft attitude control. In addition to anti-windup scheme, Xiao *et al.* [17], Bošković *et al.* [18], Panagiotis *et al.* [19] and the references therein have also developed a range of controllers to effectively handle the limited actuator output. Nevertheless, these controllers have not considered actuator faults, and thus these approaches cannot be directly applied to the FTC of spacecraft attitude.

This brief paper focuses on developing a novel fault tolerant control scheme that can achieve attitude tracking objective for a flexible spacecraft in the presence of partial loss of actuator effectiveness fault and actuator saturation under the existence of

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B. Xiao is with the Department of Control Science and Engineering, Harbin Institute of Technology, Harbin 150001, China (e-mail: bxiaobing@gmail.com).

Q. L. Hu is with the Department of Control Science and Engineering, Harbin Institute of Technology, Harbin 150001, China, and also with the Department of Mechanical and Industrial Engineering, Concordia University, Montreal, QC H3G 1M8, Canada (e-mail: huqinglei@hit.edu.cn).

Y. M. Zhang is with the Department of Mechanical and Industrial Engineering, Concordia University, Montreal, QC H3G 1M8, Canada (e-mail: ymzhang@encs.concordia.ca).

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modeling uncertainties and external disturbances using sliding mode control technique. The main contributions of this work can be summarized as three points. First, this brief integrates neural network technique to approximate the unknown system dynamics, and thus explicitly accounts for uncertainties in the plant. Second, by exploiting the online adaptive estimation of the bound of actuator fault, the proposed control scheme can achieve the goal of fault tolerant control without the need of any fault detection and isolation mechanism to determine the fault information. The third contribution is the ability of the proposed design methodology to protect the control law from actuator position saturation with globally asymptotically attitude tracking.

The remainder of this brief is organized as follows. Section II summarizes the faulty flexible spacecraft attitude control system and control problems formulation. Fault tolerant tracking controller design is performed in Section III. Simulation results of a spacecraft with the derived controller are given in Section IV, and conclusions and possibilities of future work comprises Section V.

TABLE I
MAIN PARAMETERS OF ONE FLEXIBLE SPACECRAFT

Mission	Imaging the earth
Mass (kg)	874.56
Inertia moments (kg·m^2):	
Principal moments of inertia	J_1=973.4, J_2=424.85, J_3=771.06
Products of inertia	Can be neglected
Orbit:	
Type	Circular
Altitude (km)	500
The inclination (deg)	97.4
Attitude control type	Three axis control by three reaction wheels
Reaction wheel:	
Speed range (rpm)	3500
Reaction torque (N·m)	0.15
Moment of inertia (kg·m^2)	0.0465
Coefficients of solar arrays	
Modal frequency (Hz)	0.7681, 1.1038, 1.8733
Damping ratio	0.003, 0.003, 0.003
Modal coupling coefficients	$\boldsymbol{\delta}=\begin{bmatrix} 6.4563 & 1.2781 & 2.1562 \\ -1.2581 & 0.9175 & -1.6726 \\ 1.1168 & 2.4890 & -0.8367 \end{bmatrix}$

II. Flexible Spacecraft Attitude Model and Problem Formulation

Consider a flexible spacecraft with a solar array moving in a circular orbit, and then the nonlinear equations of attitude motion are given by the attitude kinematics and the spacecraft dynamics [17].

A. Attitude Kinematics

$$\omega_1 = \dot{\phi} - \omega_0\psi, \quad \omega_2 = \dot{\theta} - \omega_0, \quad \omega_3 = \dot{\psi} + \omega_0\phi \tag{1}$$

where $\boldsymbol{\omega} = [\omega_1 \quad \omega_2 \quad \omega_3]^{\mathrm{T}}$ denotes the angular velocity of the spacecraft with respect to the inertial frame $\mathcal{I}, \omega_0$ denotes the orbital rate for the considered spacecraft, and $[\psi \quad \theta \quad \phi]^{\mathrm{T}}$ represents the attitude orientation of the spacecraft in the body frame $\mathcal{B}$ with respect to orbital frame $\mathcal{O}$ obtained by a yaw-roll-pitch sequence of rotations (ϕ, θ and ψ are, respectively, the roll, pitch, and yaw angles).

B. Spacecraft Dynamics Under Faulty Actuator

Now, let us consider the situation in which the actuator experiences partial loss of effectiveness fault, and then the faulty dynamics of the spacecraft with flexible solar array actuated by reaction wheels can be given by

$$\mathbf{J}\dot{\boldsymbol{\omega}} + \boldsymbol{\omega}^{\times}(\mathbf{J}\boldsymbol{\omega} + \mathbf{J}_s\boldsymbol{\Omega}_s) + \boldsymbol{\delta}\ddot{\boldsymbol{\eta}} = \mathbf{T}_d + \boldsymbol{\alpha}(t)\mathbf{u} \tag{2}$$

$$\mathbf{u} = -\mathbf{J}_s\dot{\boldsymbol{\Omega}}_s \tag{3}$$

$$\ddot{\boldsymbol{\eta}} + 2\boldsymbol{\xi}\Lambda\dot{\boldsymbol{\eta}} + \boldsymbol{\Lambda}^2\boldsymbol{\eta} + \boldsymbol{\delta}^{\mathrm{T}}\dot{\boldsymbol{\omega}} = \mathbf{0} \tag{4}$$

where $\mathbf{J} = \mathrm{diag}(J_1, J_2, J_3)$ denotes the spacecraft inertia parameters, $\mathbf{J}_s\boldsymbol{\Omega}_s = [J_{s1}\Omega_{s1} \quad J_{s2}\Omega_{s2} \quad J_{s3}\Omega_{s3}]^{\mathrm{T}}$ is the angular momentum of the reaction wheels, $\mathbf{u} = [u_1 \quad u_2 \quad u_3]^{\mathrm{T}}$ is the control torque and $\mathbf{T}_d = [T_{d1} \quad T_{d2} \quad T_{d3}]^{\mathrm{T}}$ the external disturbance; $2\boldsymbol{\xi}\boldsymbol{\Lambda} = \mathrm{diag}(2\xi_i\Lambda_i)$ and $\boldsymbol{\Lambda}^2 = \mathrm{diag}(\Lambda_i^2)$ are the damping matrices and stiffness matrices with $\boldsymbol{\xi}$ the modal damping and $\boldsymbol{\Lambda}$ the modal frequency ($i = 1, \ldots, N$, and N is the number of elastic modes considered); $\boldsymbol{\delta} \in \mathbb{R}^{3\times N}$ is the coupling matrix between the elastic structures and rigid body, and $\boldsymbol{\eta} \in \mathbb{R}^N$ the modal coordinate vector. Moreover, $\boldsymbol{\alpha}(t) = \mathrm{diag}(\alpha_1, \alpha_2, \alpha_3)$ is the actuator effectiveness matrix with $0 < \alpha_0 \le \alpha_i(t) \le 1$; the case of $\alpha_i(t) = 1$ means that the ith actuator works normally, while $0 < \alpha_i(t) < 1$ represents that the ith actuator loses its effectiveness partially.

C. External Disturbance Torques

For the considered flexible spacecraft as its physical parameters given in Table I, gravity-gradient torque, aerodynamic torque and earth magnetic torque are the primary external disturbances for $\mathbf{T}_d$ in (2). In addition, the orbital attitude is set to 500 km, and hence the solar radiation torque could be negligible.

1) Gravity-Gradient Torque: Due to the variation in the Earth's gravitational force over the object, the flexible spacecraft in orbit is subject to gravity-gradient torque, and this external torque expressed in the body frame $\mathcal{B}$ is given by [1]

$$\begin{cases} T_{g1} = 3\omega_0^2(J_3 - J_2)\cos^2\theta\cos\phi\sin\phi \\ T_{g2} = 3\omega_0^2(J_3 - J_1)\cos\theta\sin\theta\cos\phi \\ T_{g3} = 3\omega_0^2(J_1 - J_2)\cos\theta\sin\theta\sin\phi. \end{cases} \tag{5}$$

2) Magnetic Disturbance Torque: Magnetic disturbance torques are induced by the interaction between the spacecraft's residual magnetic field and the geomagnetic field, which is estimated as 4–6 Am^2 for the considered spacecraft in Table I. As such, the magnetic disturbance torque expressed in body frame $\mathcal{B}$ can be calculated as $\mathbf{T}_m = \mathbf{M}_m \times \mathbf{B}$,, where $\mathbf{M}_m$ is the sum of the individual magnetic moments caused by permanent and induced magnetism and the spacecraft generated loops, $\mathbf{B}$ is the geocentric magnetic flux density described in $\mathcal{B}$. According to the above analysis, $\mathbf{T}_m$ of the spacecraft is finally given as follows, where the residual magnetic moment of the spacecraft along the body axes is equal to 1.5 Am^2

$$\begin{cases} T_{m1} = -7.05 \times 10^{-5} \sin \omega_0 t \\ T_{m2} = -3.6 \times 10^{-5} \cos \omega_0 t + 7.05 \times 10^{-5} \sin \omega_0 t \\ T_{m3} = -3.6 \times 10^{-5} \cos \omega_0 t. \end{cases} \quad (6)$$

3) Aerodynamic Torque: Aerodynamic torque results from the spacecraft motion through the tenuous upper atmosphere, and the aerodynamic torque can be given by

$$\mathbf{T}_p = \mathbf{L}_P \times \mathbf{R}_{bi}(-0.5 C_D S_D \rho |\mathbf{V}| \mathbf{V}). \quad (7)$$

Based upon the values of the spacecraft parameters listed in Table I, the atmospheric density is $\rho = 2.35 \times 10^{-12}$ kg/m^3, the drag coefficient is $C_D = 2.2$, the velocity of the spacecraft $\mathbf{V} = 7613$ m/s, the area of the spacecraft cross-section is $S_D = 8.654$ m^2, the offset from the center of mass to the center of the pressure is taken to be $\mathbf{L}_P = [0.045 \quad 0.045 \quad 4.23]^{\mathrm{T}}$, and $\mathbf{R}_{bi}$ denotes the orthogonal rotation matrix between $\mathcal{I}$ and $\mathcal{B}$. Consequently, the worst-case aerodynamic disturbance torque is 1.45×10^{-5} Nm based on the given parameters.

D. Transformed Open-Loop Attitude System

Now, let us define the state vector

$$\begin{aligned} \mathbf{x} &= [\phi \quad \dot{\phi} \quad \theta \quad \dot{\theta} \quad \psi \quad \dot{\psi}]^{\mathrm{T}} \\ &= [x_1 \quad x_2 \quad x_3 \quad x_4 \quad x_5 \quad x_6]^{\mathrm{T}}. \end{aligned}$$

Then, the equations of motion for the flexible spacecraft in (2)–(4) can be decoupled into the following form:

$$\text{Roll subsystem:} \quad \begin{cases} \dot{x}_1 = x_2 \\ \dot{x}_2 = f_1(\mathbf{x}) + \frac{1}{J_1}\alpha_1 u_1 + d_1 \end{cases} \quad (8)$$

with $f_1(\mathbf{x}) = (J_2 - J_3)/(J_1)(x_4 x_6 + \omega_0 x_1 x_4 - \omega_0 x_6 - \omega_0^2 x_1) + \omega_0 x_6$ and $d_1 = (1)/(J_1)[-\sum_{i=1}^N \delta_{1i}\ddot{\eta}_i + T_{d1} + J_{s2}\Omega_{s2}(x_6 + \omega_0 x_1) - J_{s3}\Omega_{s3}(x_4 - \omega_0)]$.

$$\text{Pitch subsystem:} \quad \begin{cases} \dot{x}_3 = x_4 \\ \dot{x}_4 = f_2(\mathbf{x}) + \frac{1}{J_2}\alpha_2 u_2 + d_2 \end{cases} \quad (9)$$

with $f_2(\mathbf{x}) = (J_3 - J_1)/(J_2)(x_2 x_6 + \omega_0 x_1 x_2 - \omega_0 x_5 x_6 - \omega_0^2 x_1 x_5)$ and $d_2 = (1)/(J_2)[-\sum_{i=1}^N \delta_{2i}\ddot{\eta}_i + T_{d2} + J_{s1}\Omega_{s1}(x_6 + \omega_0 x_1) + J_{s3}\Omega_{s3}(x_2 - \omega_0 x_5)]$.

$$\text{Yaw subsystem:} \quad \begin{cases} \dot{x}_5 = x_6 \\ \dot{x}_6 = f_3(\mathbf{x}) + \frac{1}{J_3}\alpha_3 u_3 + d_3 \end{cases} \quad (10)$$

with $f_3(\mathbf{x}) = (J_1 - J_2)/(J_3)(x_2 x_4 + \omega_0^2 x_5 - \omega_0 x_2 - \omega_0 x_4 x_5) - \omega_0 x_2$ and $d_3 = (1)/(J_3)[-\sum_{i=1}^N \delta_{3i}\ddot{\eta}_i + T_{d3} + J_{s1}\Omega_{s1}(x_4 - \omega_0) - J_{s2}\Omega_{s2}(x_2 - \omega_0 x_5)]$.

Based on the process derived above, each subsystem can be summarized into the following single input nonlinear systems:

$$\begin{cases} \dot{x}_{2i-1} = x_{2i} \\ \dot{x}_{2i} = f_i(\mathbf{x}) + \frac{1}{J_i}\alpha_i u_i + d_i \end{cases} \quad (11)$$

with $d_i,\ i = 1, 2, 3$, viewed as the lumped disturbance for each subsystem.

E. Problem Statement

Let a smooth and bounded reference attitude angle trajectory $y_i^d \in \mathbb{R},\ i = 1, 2, 3$, be determined for each axis, given any initial attitude and angular velocity, the control objective is to develop a suitable control law for u_i such that the spacecraft attitude can globally asymptotically track the reference attitude y_i^d in spite of the bounded disturbances, the uncertainties in the system and unexpected actuator fault $\boldsymbol{\alpha}(t)$, i.e., $x_{2i-1}(t) \to y_i^d$ as $t \to \infty,\ i = 1, 2, 3$.

III. Adaptive Sliding Mode Fault Tolerant Control

In this section, we first consider the case in which each reaction wheel of the flexible spacecraft experiences fading actuation but is still active, and the result is then extended to the case in which actuator saturation is considered.

A. Fault Tolerant Attitude Tracking Control With Partial Loss of Actuators Effectiveness

Before giving the details of the fault tolerant control design by using sliding mode control technique, for $\forall i = 1, 2, 3$ the following tracking error is presented:

$$z_{i1} = x_{2i-1}(t) - y_i^d(t) \quad (12)$$

$$z_{i2} = x_{2i} - \dot{y}_i^d - v_i \quad (13)$$

where $v_i = -c_{i1} z_{i1}$ is the virtual control with $c_{i1} > 0$.

Now, we first design a sliding surface as

$$s_i(t) = k_i z_{i1}(t) + z_{i2}(t), \quad k_i > 0. \quad (14)$$

According to the result in [17], it can be known that if an appropriate controller $u_i(t)$ can be constructed to satisfy the sliding mode condition $s_i(t)\dot{s}_i(t) < 0$, and then z_{i2} and z_{i2} could asymptotically converge to zero.

Accordingly, the subsystem (11) can be rewritten as

$$\begin{cases} \dot{x}_{2i-1} = x_{2i} \\ \dot{x}_{2i} = f_i(\mathbf{x}) + \left(\frac{1}{J_{i0}} + \Delta J_i\right) \alpha_i u_i + d_i \end{cases} \quad (15)$$

where J_{i0} is the nominal inertia parameter of spacecraft, $\Delta J_i = (1)/(J_i) - (1)/(J_{i0})$ and $\delta J_i = J_i - J_{i0}$ are the uncertain terms.

Now, the following equation can be easily obtained:

$$\dot{x}_{2i} = f_i(\mathbf{x}) + \frac{1}{J_{i0}} u_i + g_i u_i + d_i \quad (16)$$

with $g_i = (\alpha_i - 1)/(J_{i0}) + \Delta J_i \alpha_i$. During operation the mass properties of the spacecraft may be uncertain or may change due to onboard payload motion, rotation of solar arrays, or fuel consumptions, making J_i time varying and also uncertain. However, inequalities $-J_{i0} \ll \delta J_i \ll J_{i0}$ and $-J_i \ll \delta J_i \ll J_i$ always hold, and then it follows that $0.5 < (J_{i0})/(J_i) < 2$. Imposing the bound $\alpha_0 < \alpha_i \leq 1$, we find that

$$|J_{i0} g_i| = \left| \alpha_i - 1 + J_{i0}\left(\frac{1}{J_i} - \frac{1}{J_{i0}}\right)\alpha_i \right| = \left| 1 - \frac{J_{i0}}{J_i}\alpha_i \right| < 1. \quad (17)$$

From (17), it is therefore reasonable to make Assumption 1.

Assumption 1: There always exists a positive but unknown constant π_i such that

$$|J_{i0} g_i| = \pi_i < 1. \tag{18}$$

On the other hand, the unknown inertia parameter and external disturbance would lead to the uncertainty of f_i and d_i. As a result, single layer neural network approximation technique will be utilized to represent f_i. Then, f_i can be viewed as the output of the neural network, which is given by

$$f_i(x) = f_i^* + \varepsilon_i = \mathbf{W}_{f_i}^{\mathrm{T}} \mathbf{X}_{f_i} + \varepsilon_i \tag{19}$$

where $\mathbf{W}_{f_i} \in \mathbb{R}^n$ is the optimal approximation weight, $\mathbf{X}_{f_i} \in \mathbb{R}^n$ is the bias function, ε_i denotes the approximation error and is supposed to be bounded by $|\varepsilon_i| \le \varepsilon_i^*$, in which ε_i^* is a positive constant.

Remark 1: During the entire attitude maneuver process, the angles ϕ, θ and ψ are bounded as well as angular velocity $\boldsymbol{\omega}$, small displacements for solar array are also considered with bounded elastic oscillation and its rate; that is to say, $\|\boldsymbol{\eta}\|$ and $\|\dot{\boldsymbol{\eta}}\|$ are bounded. It is therefore reasonable to assume that there exists a constant (unknown) $\chi_i > 0$ such that $|d_i + \varepsilon_i| \le \chi_i$.

Since the knowledge on π_i in (18) cannot be obtained exactly due to unexpected fault and uncertain inertia parameters, the adaptive technique will be employed to tackle this challenge. In the following theorem, we summarize our control solution to the underlying attitude tracking problem by incorporating adaptive sliding mode control action.

Theorem 1: Consider the faulty attitude control system given by (1)–(4) under the Assumption 1. Suppose that the following adaptive sliding mode fault tolerant control (ASMFTC) law is implemented by

$$\begin{aligned} u_i = J_{i0}[-k_i(z_{i2} - c_{i1} z_{i1}) + \ddot{y}_i^d + \dot{v}_i - \hat{\mathbf{W}}_{f_i}^T \mathbf{X}_{f_i} \\ -\hat{\chi}_i \mathrm{sgn}(s_i) - \Gamma_i \mathrm{sgn}(s_i)] \end{aligned} \tag{20}$$

and updated by

$$\dot{\hat{\mathbf{W}}}_{f_i} = \gamma_{i1} s_i \mathbf{X}_{f_i} \tag{21}$$
$$\dot{\hat{\chi}}_i = \gamma_{i2} |s_i| \tag{22}$$
$$\dot{\hat{\beta}}_i = \gamma_{i3} \lambda_i |s_i| \tag{23}$$

where $\lambda_i = |-k_i(z_{i2} - c_{i1} z_{i1}) + \ddot{y}_i^d + \dot{v}_i - \hat{\mathbf{W}}_{f_i}^T \mathbf{X}_{f_i}| + \hat{\chi}_i + \kappa_i$, $\Gamma_i = -\lambda_i + \hat{\beta}_i \lambda_i$; while $\gamma_{ij}(j = 1, 2, 3)$ and κ_i are positive control gains. Then, the closed-loop attitude tracking system is globally asymptotically stable. That is, the attitude tracking error $z_{i1} \to 0$ and $z_{i2} \to 0$ as $t \to \infty$, and thus $\lim_{t\to\infty} x_{2i-1} = y_i^d$.

Before we proceed with the proof for Theorem 1, the following remarks are in order.

Remark 2: Note that the control scheme (20) does not directly involve the inertia matrix $\mathbf{J}$ (particularly time varying and uncertain) and the coupling matrix $\boldsymbol{\delta}$. Thus, from the viewpoint of external disturbances and uncertainties rejection, the derived control law has great robust performance.

Remark 3: In order to avoid the chattering phenomenon due to the imperfect implementation of the sign function in controller (20), saturation function $\mathrm{sat}(x) = \mathrm{sgn}(x)\min\{|x|, \varepsilon\}$ is a simple choice to replace the discontinuous function, where ε is a small positive constant.

Proof of Theorem 1: From (16), the dynamics of states z_{i1} and z_{i2} are derived as follows:

$$\dot{z}_{i1} = z_{i2} + v_i = z_{i2} - c_{i1} z_{i1} \tag{24}$$
$$\dot{z}_{i2} = \dot{x}_{2i} - \ddot{y}_i^d - \dot{v}_i = f_i + \frac{1}{J_{i0}} u_i + g_i u_i + d_i - \ddot{y}_i^d - \dot{v}_i. \tag{25}$$

Hence, it follows that $\dot{s}_i$ satisfies

$$\dot{s}_i = k_i(z_{i2} - c_{i1} z_{i1}) + f_i + \frac{1}{J_{i0}} u_i + g_i u_i + d_i - \ddot{y}_i^d - \dot{v}_i. \tag{26}$$

Choose the Lyapunov function V_{i1} candidate as

$$V_{i1} = \frac{1}{2} s_i^2 + \frac{\tilde{\mathbf{W}}_{f_i}^{\mathrm{T}} \tilde{\mathbf{W}}_{f_i}}{2\gamma_{i1}} + \frac{\tilde{\chi}_i^2}{2\gamma_{i2}} + \frac{1 - \pi_i}{2\gamma_{i3}} \tilde{\beta}_i^2 \tag{27}$$

where $\tilde{\mathbf{W}}_{f_i} = \mathbf{W}_{f_i} - \hat{\mathbf{W}}_{f_i}, \tilde{\chi}_i = \chi_i - \hat{\chi}_i$ and $\tilde{\beta}_i = \beta_i - \hat{\beta}_i$; $\hat{\mathbf{W}}_{f_i}, \hat{\chi}_i$ and $\hat{\beta}_i$ are the estimates of $\mathbf{W}_{f_i}, \chi_i$ and $\beta_i = (1)/(1 - \pi_i)$, respectively.

From Remark 1, differentiating (27) and inserting (20)–(23) result in

$$\begin{aligned} \dot{V}_{i1} &= s_i \dot{s}_i - \frac{\tilde{\mathbf{W}}_{f_i}^{\mathrm{T}} \dot{\hat{\mathbf{W}}}_{f_i}}{\gamma_{i1}} - \frac{\tilde{\chi}_i \dot{\hat{\chi}}_i}{\gamma_{i2}} - \frac{1 - \pi_i}{\gamma_{i3}} \tilde{\beta}_i \dot{\hat{\beta}}_i \\ &\le -\Gamma_i |s_i| + s_i g_i u_i + |s_i|(|d_i + \varepsilon_i| - \hat{\chi}_i) \\ &\quad - \frac{1 - \pi_i}{\gamma_{i3}} \tilde{\beta}_i \dot{\hat{\beta}}_i - \frac{\tilde{\chi}_i \dot{\hat{\chi}}_i}{\gamma_{i2}} \\ &\le -\Gamma_i |s_i| + s_i g_i u_i \\ &\quad - \frac{1 - \pi_i}{\gamma_{i3}} \tilde{\beta}_i \dot{\hat{\beta}}_i \end{aligned} \tag{28}$$

in which the fact $|d_i + \varepsilon_i| \le \chi_i$ is used. In particular, from (18) and the controller (20), one has

$$\begin{aligned} s_i g_i u_i &\le \pi_i |s_i| \Big[\Big| -k_i(z_{i2} - c_{i1} z_{i1}) + \ddot{y}_i^d + \dot{v}_i \\ &\quad - \hat{\mathbf{W}}_{f_i}^T \mathbf{X}_{f_i} \Big| + \hat{\chi}_i + \Gamma_i \Big] \\ &= \pi_i |s_i| (\lambda_i - \kappa_i + \Gamma_i) \\ &= \pi_i (\hat{\beta}_i \lambda_i - \kappa_i) |s_i|. \end{aligned} \tag{29}$$

Imposing the bound (29) and $\beta_i(1 - \pi_i) = 1$, (28) can be further evaluated as

$$\begin{aligned} \dot{V}_{i1} &\le (1 - \hat{\beta}_i) \lambda_i |s_i| + \pi_i (\hat{\beta}_i \lambda_i - \kappa_i) |s_i| - \frac{1 - \pi_i}{\gamma_{i3}} \tilde{\beta}_i \dot{\hat{\beta}}_i \\ &= -\kappa_i \pi_i |s_i| + [\pi_i \hat{\beta}_i + (1 - \hat{\beta}_i)] |s_i| - \frac{1 - \pi_i}{\gamma_{i3}} \tilde{\beta}_i \dot{\hat{\beta}}_i \\ &= -\kappa_i \pi_i |s_i| < 0, \quad \forall s_i \ne 0 \end{aligned} \tag{30}$$

which implies that $\lim_{t\to\infty} V_{i1}(t) = V_{i1}(\infty)$ exists. By integrating $\dot{V}_{i1}$ from 0 to ∞, one obtains

$$\lim_{t\to\infty} \int_0^t |s_i(\tau)| d\tau \le \frac{1}{\kappa_i \pi_i} [V_{i1}(0) - V_{i1}(\infty)]. \tag{31}$$

Because the term on the right-hand side is bounded, and using the Barbalat's lemma, it follows that $\lim_{t\to\infty} s_i = 0$. Thus, globally asymptotical stability of the closed-loop system can be guaranteed, that is, $\lim_{t\to\infty} z_{i1} = 0$ and $\lim_{t\to\infty} z_{i2} = 0$; and with (12) the attitude angle x_{2i-1} can globally asymptotically track the desired attitude y_i^d. Thereby, the proof is completed. □

B. Fault Tolerant Attitude Tracking Control in the Presence of Both Actuator Fault and Saturation Limit

From a practical perspective, the signal u_i generated by the control law (20) might not be implemented due to actuator saturation limit. Let $u_{\max} > 0$ denote the maximum output torque of the actuator, now we consider the flexible spacecraft dynamic under actuator saturation, i.e.,

$$\begin{cases} \dot{x}_{2i-1} = x_{2i} \\ \dot{x}_{2i} = f_i(\mathbf{x}) + \frac{1}{J_i}\alpha_i \mathrm{sat}(u_i) + d_i \end{cases} \tag{32}$$

where $u_i = \mathrm{sat}(u_i) = \mathrm{sgn}(u_i)\min\{|u_i|, u_{\max}\}$.

For the purpose of designing control law to the actuator saturated system (32), an estimate of the domain of attraction of the origin is first determined, and the following level set of the Lyapunov function V_{i1} as an estimate is employed [20]:

$$\mathcal{D}(b_i) = \{z_{i1} \in \mathbb{R}, z_{i2} \in \mathbb{R} \mid V_{i2} \le b_i\} \tag{33}$$

in which b_i is such that $\max_{z_{i1}, z_{i2} \in \mathcal{D}(b_i)} |u_{i0}| \le u_{\max}$.

Theorem 2: Consider the faulty and actuator saturated attitude control system (32). With the application of the modified adaptive sliding mode fault tolerant control (MASMFTC)

$$u_i = u_{i0} - \tau_i s_i \tag{34}$$

where $\tau_i \ge 0$ is a positive constant or function, and u_{i0} is designed as the same as (20). Then, the control objectives as stated in Section II are met.

Proof: With the modified control law (34), we obtain

$$\begin{aligned} \frac{1}{J_i}\alpha_i \mathrm{sat}(u_i) &= \frac{1}{J_{i0}}\mathrm{sat}(u_{i0} - \tau_i s_i) + g_i \mathrm{sat}(u_{i0} - \tau_i s_i) \\ &= \frac{1}{J_{i0}} u_{i0} + g_i u_{i0} + \frac{1}{J_{i0}}[\mathrm{sat}(\tau_i s_i - u_{i0}) + u_{i0}] \\ &\quad + g_i[\mathrm{sat}(\tau_i s_i - u_{i0}) + u_{i0}]. \end{aligned} \tag{35}$$

Consider the same candidate Lyapunov function (27), with (35), we have

$$\dot{V}_{i1} \le -\kappa_i \pi_i |s_i| - s_i \left(\frac{1}{J_{i0}} + g_i\right)[\mathrm{sat}(\tau_i s_i - u_{i0}) + u_{i0}]. \tag{36}$$

Note that

$$\begin{aligned} &\mathrm{sat}(\tau_i s_i - u_{i0}) + u_{i0} \\ &\quad = \begin{cases} \tau_i s_i, \ \text{if } |\tau_i s_i - u_{i0}| \le u_{\max} \\ u_{\max} + u_{i0}, \ \text{if } (\tau_i s_i - u_{i0}) > u_{\max} \\ -u_{\max} + u_{i0}, \ \text{if } (\tau_i s_i - u_{i0}) < -u_{\max} \end{cases}. \end{aligned} \tag{37}$$

We know that $|u_{i0}| \le u_{\max}$ for $\forall z_{i1}, z_{i2} \in \mathcal{D}(b_i)$. This implies that $u_{\max} + u_{i0} \ge 0$ and $-u_{\max} + u_{i0} \le 0$. Further, it follows that $\tau_i s_i - u_{i0} > u_{\max} \Rightarrow \tau_i s_i > 0$ and $\tau_i s_i - u_{i0} < -u_{\max} \Rightarrow \tau_i s_i < 0$ when u_{i0} is outside the set $\mathcal{B}_1 \triangleq \{u_{i0} || u_{i0}| \le u_{\max}\}$.

To that end, we use the definition of g_i to obtain

$$g_i + \frac{1}{J_{i0}} = \frac{\alpha_i}{J_{i0}} + \left(\frac{1}{J_i} - \frac{1}{J_{i0}}\right)\alpha_i = \frac{\alpha_i}{J_i} > 0. \tag{38}$$

Using (37)–(38), (36) can be rewritten as (39), shown at the bottom of the page, which implies that $\dot{V}_{i1} < 0$ for $\forall z_{i1}, z_{i2} \in \mathcal{D}(b_i)$. Thus, it is able to show that for $\tau_i \ge 0$, s_i is asymptotically stable at the origin with $\mathcal{D}(b_i)$ included in the domain of attraction, thereby completing the proof of achieving the stated attitude stabilization objective with the same analysis as in the proof of Theorem 1. This completes the proof. □

Remark 4: It is worth mentioning that actuator saturation does not occur within the defined level set (33), and hence $\mathcal{D}(b_i)$ is a contractively invariant set and is within the domain of attraction. Moreover, as we can see in (20), the parameters k_i and κ_i can be tuned in such a way that $\mathcal{D}(b_i)$ is a contractively invariant set of a satisfactory large size.

Remark 5: As shown in (34), the modified fault tolerant controller increases the utilization of the actuator capacity by tuning the value of τ_i large.

Remark 6: Since any information of the actuator fault $\boldsymbol{\alpha}$ is not required in the control law (34), there is no need to include a health monitoring unit to identify which actuator is unhealthy. Knowledge of the fault time, patterns and values is not even required. The actuator fault compensation is done automatically and adaptively by the proposed control scheme. These imply that the controller (34) is able to perform attitude tracking maneuver regardless of the incipient or abrupt partial loss of reaction wheel fault or nominal condition as long as no actuator is completely failed.

Remark 7: The spacecraft considered is actuated by three reaction wheels, and as pointed out in [21], reaction wheels are sensitive devices that are vulnerable to the following different sources of faults: 1) failure to respond to control signals; 2) decreased reaction torque; 3) increased bias torque; and 4) continuous generation of reaction torque. Note that this brief only focuses on the fault of decreased reaction torque, and assumes

$$\dot{V}_{i1} \le -\kappa_i \pi_i |s_i| - \begin{cases} -\left(\frac{1}{J_{i0}} + g_i\right)\tau_i s_i^2, \ \text{if } |\tau_i s_i - u_{i0}| \le u_{\max} \\ -\left(\frac{1}{J_{i0}} + g_i\right) s_i(u_{\max} + u_{i0}), \ \text{if} (\tau_i s_i - u_{i0}) > u_{\max} \\ -\left(\frac{1}{J_{i0}} + g_i\right) s_i(u_{i0} - u_{\max}), \ \text{if } (\tau_i s_i - u_{i0}) < -u_{\max} \end{cases} \le -\kappa_i \pi_i |s_i| \tag{39}$$

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that three actuators are still active even if there exist faults in some actuators. If one or more actuators are totally failed, then the system will become under-actuated [19], and the designed controller (34) would not guarantee attitude tracking control due to the lack of necessary hardware redundancy available in the spacecraft for achieving fault-tolerant control. The under-actuated system is not further considered in this paper. Although reaction wheel faults 1), 3), and 4) are not considered in this brief, extension of the proposed control scheme to handle these three types of fault, together with spacecraft actuation hardware system design with redundancy will be carried out in our future works.

IV. Simulation and Comparison Results

To verify the effectiveness of the proposed fault tolerant controller (34), detailed responses are numerically simulated for the flexible spacecraft listed in Table I, and its initial attitude orientation is set to be $\phi(0) = \theta(0) = \psi(0) = 0$ deg with $\boldsymbol{\omega}(0) = [0 \quad 0 \quad 0.15]^{\mathrm{T}}$ deg/s, initial modal displacements $\eta_i(0) = 0.001$ and its time rate $\dot{\eta}_i(0) = 0.0005$. Moreover, the proportional-integral-derivative (PID) control is also carried out in the following simulation for the comparison purpose.

Actuator Fault Scenario: 1) The reaction wheels in the roll and pitch axis decrease 85% and 90% of their normal values after 10 s, respectively; and 2) the reaction wheel in the yaw axis loses 90% power in 2000 s.

Desired Attitude: In order to image some area on the earth within a mission's interest, the desirable attitude angle in each axis is designed as: $y_1^d = 0.2\sin(\bar{\omega}t)$, $y_2^d = 0.1\sin(\bar{\omega}t)$ and $y_3^d = 0.15\cos(\bar{\omega}t)$ deg with $\bar{\omega} = 4\pi/T$ and orbital period T.

A. Response With MASMFTC for Fault Free and Fault Cases

In order to implement the MASMFTC, the control parameters in (34) are chosen as $k_i = 0.15, \gamma_{i1} = 20, \gamma_{i2} = 5, \gamma_{i3} = 20, c_{i1} = 1.5, \kappa_i = 5$ and $\tau_i = 125$; moreover, radial basis function of neural network is applied to approximate $f_i(\mathbf{x})$ due to its good properties of linear-in-parameter form and the spatially localized structure, and the vectors $\mathbf{X}_{f_i} \in \mathbb{R}^n$ used in the simulation are chosen as Gaussian type functions:

$$\mathbf{X}_{f_{ij}}(\mathbf{x}) = \exp\left(-\frac{\|\mathbf{x}-\mathbf{p}_j\|^2}{\sigma_j^2}\right), \quad j = 1, 2, \ldots, n \tag{40}$$

where $\sigma_j \in \mathbb{R}$ and $\mathbf{p}_j \in \mathbb{R}^6$. Moreover, $n = 10$ and $\sigma_i^2 = 7.5$ are selected in the simulation, and the elements of $\mathbf{p}_j$ are chosen between -1 and 1 randomly.

As stated in Remark 6, the control law (34) can achieve the attitude tracking maneuver in case of no actuator fault. Therefore, when the MASMFTC is applied to the nominal attitude system in simulation, the responses with the controller (34) is shown in Figs. 1–2 (solid line). It is clear to see that this controller achieves excellent response despite uncertain inertia matrix and external disturbances on the spacecraft dynamics. Actually, the attitude tracking error is going to be stable in 150 s as shown in Fig. 1 (solid line). Moreover, due the effect of the high gain component in the MASMFTC, the control input sent to actuator is always within the maximum magnitude of the reaction wheel, as shown in Fig. 2 (solid line).

When three reaction wheels endowed in the spacecraft experience severe partial loss of effectiveness fault, the application

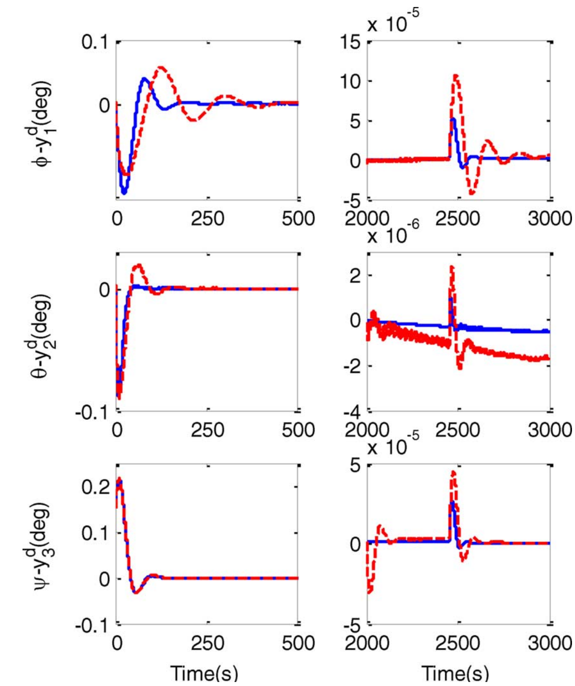

Fig. 1. Time response of attitude tracking error under MASMFTC: without fault (solid line) and with actuator fault (dashed line).

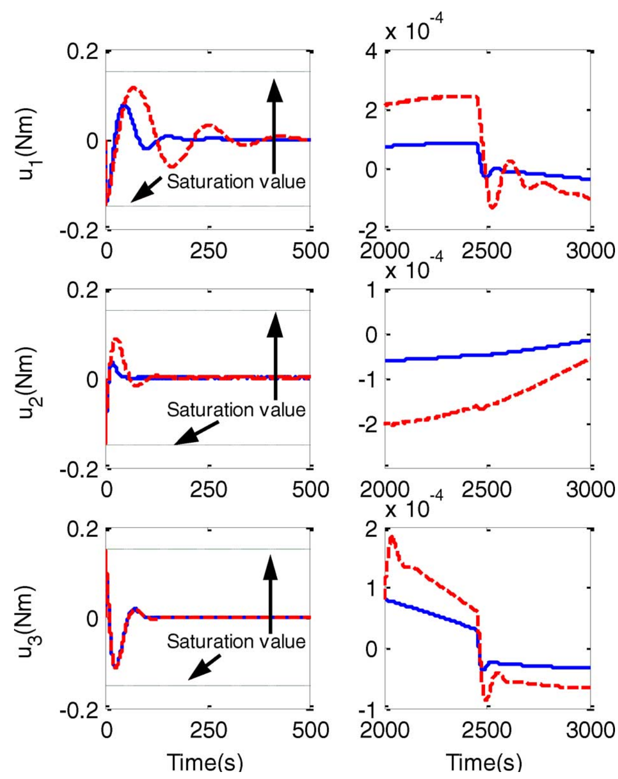

Fig. 2. Time response of control input under MASMFTC: without fault (solid line) and with actuator fault (dashed line).

of MASMFTC still leads to the attitude tracking with high accuracy in that the tracking errors asymptotically converge to the origin after 400 s as shown in Fig. 1 (dashed line). The corresponding fault tolerant control signals are shown in Fig. 2 (dashed line), where we can see that MASMFTC needs more control torque than the fault free case to compensate for actuator fault. However, the proposed control methodology successfully protects the control input of each reaction wheel from its position saturation. From the simulation results using MASMFTC for fault free and actuator fault cases, it can be known that the proposed algorithm has great fault tolerance capability together with great robustness to external disturbances and system uncertainties.

B. *Response With PID for Fault Free and Fault Cases*

For the purpose of comparison, the widely-used PID controller is also applied to the attitude tracking control of this flexible spacecraft. For handling the problem of reaction wheels saturation, the magnetic torquers are additionally equipped in the flexible spacecraft to unloading the saturated control power. The unloading strategy in details is discussed in [22] and is omitted here.

With application of PID control to the spacecraft, the attitude tracking errors are shown in Fig. 3, it can be clearly seen that in the case of no occurrence of actuator faults, the attitude tracking can be achieved with control accuracy of 10^{-3} deg, and the stability of the resulting closed-loop system can be guaranteed, as can be seen in Fig. 3 (solid line). However, when the above mentioned reaction wheel fault occurred, since the PID controller does not have the ability to accommodate the actuator faults, the closed-loop system is going to be unstable, as shown in Fig. 3 (dashed line). This is due to the fact that the magnitudes of control inputs, as shown in Fig. 4 (dashed line), are not big enough to compensate the faults, although they do not exceed the saturation limit.

For further comparison, the simulation is also conducted with the implicit fault tolerant control (IFTC) proposed by Bonivento *et al.* [23] under the above-mentioned two conditions: fault-free and the considered fault scenario. In order to gain more insight in a quantitative manner, the control performance is summarized in Table II (refer to Appendix) to make the comparison more apparent. From the control results as listed in Table II, it is noted that the proposed control scheme provides better control performance than the PID control and IFTC in both theory and simulations whenever actuator fault occurs or not. In addition, extensive simulations were also done using different control parameters and under different actuator faults. The results show that our proposed controller can be practically appealing to accomplish the attitude tracking maneuver with high control performance.

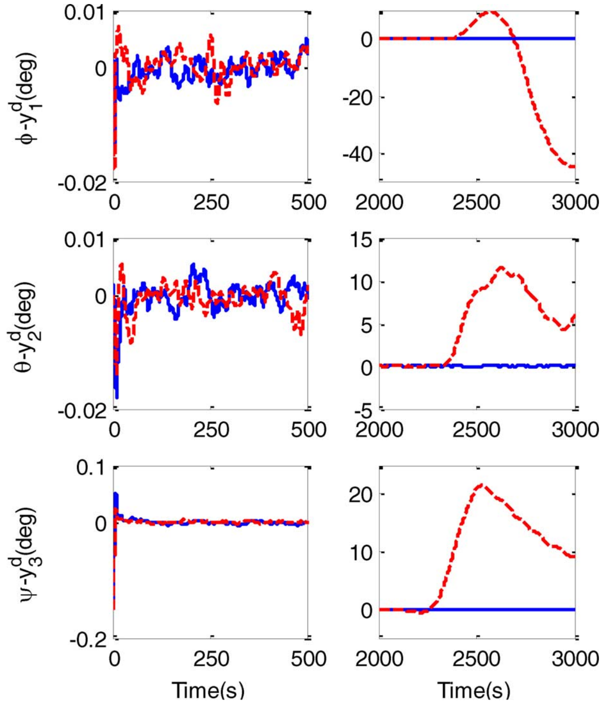

Fig. 3. Time response of attitude tracking error under PID: without fault (solid line) and with actuator fault (dashed line).

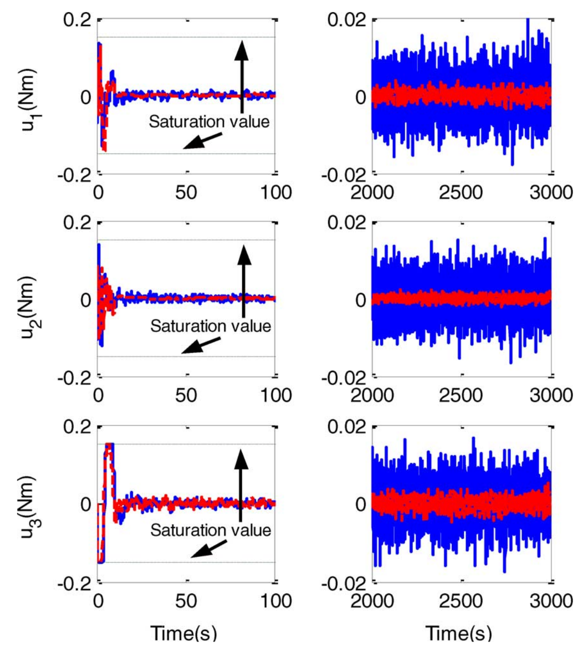

Fig. 4. Time response of control input under PID: without fault (solid line) and with actuator fault (dashed line).

V. Conclusion

A sliding mode-based adaptive fault tolerant attitude tracking control scheme was proposed in this paper for flexible spacecraft with unknown inertia parameters, external disturbances, partial loss of actuator effectiveness faults and even actuator saturation. This control scheme used the estimated parameters on-line to eliminate its dependence on the bound of actuator fault, and employed neural network to learn the unknown dynamics. It can be a reliable method in the sense that globally asymptotical convergence in tracking error was guaranteed. Numerical simulations were also carried out to verify the effectiveness of the developed control. We would like to point out that, in this brief paper, we studied the case of partial loss of reaction wheel effectiveness; however we did not consider the case that some actuators completely lose the control power or experience stuck faults based on consideration of limitation of

TABLE II
PERFORMANCE SUMMARY UNDER DIFFERENT CONTROL SCHEME

Control Performance		Actuator Healthy Status	Control Schemes		
			MASMFTC	IFTC [23]	PID
Attitude Tracking Accuracy (deg)	Roll	Normal	±5.0e-5	±3.0e-4	±0.003
		Fault	±1.0e-4	±1.0e-3	±40
	Pitch	Normal	±1.0e-6	±4.0e-5	±0.005
		Fault	±2.0e-6	±6.0e-5	±150
	Yaw	Normal	±2.5e-5	±2.8e-4	±0.002
		Fault	±4.0e-5	±6.2e-4	±180
Slew Rate Accuracy (deg/s)	Roll	Normal	±5.0e-6	±1.5e-5	±1.0e-3
		Fault	±4.0e-5	±1.5e-3	Infinity
	Pitch	Normal	±1.0e-6	±3.5e-5	±0.06
		Fault	±2.0e-6	±6.0e-4	±0.16
	Yaw	Normal	±2.5e-6	±4.0e-5	±1.0e-3
		Fault	±4.0e-5	±8.0e-4	±0.15
Attitude Tracking Maneuver Time (s) (attitude tracking error ⩽0.005deg)		Normal	150	600	1000
		Fault	400	1000	Infinity

available actuation hardware redundancy of the spacecraft. This last case should be investigated by fault tolerant control design together with spacecraft actuator hardware redundancy design and management, as one of our future works. We can also consider, as some of future works, improvement of the robustness to system uncertainties and different external disturbances, e.g., due to gravitational, aerodynamic, solar and magnetic effects in the face of different orbit altitudes; this may be done using feedback linearization control [3], [4] and disturbance rejection methodology [24].

APPENDIX

See Table II.

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