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Rigid Spacecraft Attitude Regulation with Delayed Feedback and Actuator Faults

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ABSTRACT: The work presented in this paper discusses time delay compensation of a rigid spacecraft with faulty actuators. The proposed method consists of a nominal controller and an extended state observer. Based on the backstepping method, the nominal control is designed to stabilize the spacecraft in the presence of delayed inputs. Then, the discrepancy between the nominal plant and real system which is influenced by faulty actuators, model uncertainties, and external disturbances is estimated by the extended state observer and actively compensated. The proposed controller does not require exact knowledge of delay, actuator faults and disturbances. By adjusting controller parameters, using the Lyapunov-krasovski method and properties of modified Rodrigues parameters; it is proved that the investigated control scheme can stabilize the system with respect to a small neighborhood of the origin. Numerical simulation results demonstrate that the acceptable performance of the controlled system is guaranteed in the presence of retreated inputs, the considered faults are tolerated and disturbances are rejected.

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1. INTRODUCTION

Due to the fact that in real engineering systems reactions to actions never take place instantaneously, time delays are ubiquitous in control systems [1]. Having actuators in systems can introduce delays. Actuators take a certain time to respond to input signals which can be modeled as a delay [2]. Time delay can be arisen from limitations of operational sensors used to determine spacecraft attitude. For example, star tracker sensors require a certain time for star identification or data latency of global positioning system (GPS) sensors can cause a delay [3]. Furthermore, the onboard computational capability of most spacecraft (particularly, small and micro ones) is limited. It results in a computational delay [4], i.e., an amount of time is required to create control decisions, and to execute these decisions. For mentioned reasons, delays almost always exist in practical engineering systems. Thus, to prevent undesirable features caused by delay such as instability, it is needed to derive a control system which guarantees overall system stability even in the presence of delay.

Arguably, existing researches which are devoted for stabilizing a rigid spacecraft with delayed inputs have been quite limited [5]-[7]. Although these works provide a stable closed-loop system in the presence of delay, there is a critical limitation which is common throughout them; that is, all of the results are local. This leads to the problem that we can stabilize spacecraft systems only for a certain finite set of initial conditions, but not be for any arbitrary large attitude maneuver. Therefore, the need for developing an attitude control system to regulate a spacecraft system globally in the presence of retreated inputs is clear. Also, from practical viewpoint, it is highly desired for delay length to be considered as an unknown value. In recent papers [8] and [9], we tackled this problem by employing the backstepping approach. Although *global* attitude regulation was archived in the presence of unknown long actuator delay by employing the methods proposed in [8] and [9]. But owning to the increasing demand on maintaining high-pointing accuracy, requiring system reliability and ensuring the acceptable spacecraft performance in wide operating conditions, it is appreciated to design attitude control systems in a way that the system under control retains the nominal performance when external disturbances, modeling uncertainties and component malfunctions influence the overall system.

Due to the facts that (i) spacecraft commonly act in the presence of various disturbances (e.g., aerodynamic, gravitational and radiation torques), (ii) the inertia matrix of a spacecraft is usually not known exactly, and (iii) the movement of payload like camera and telescope causes the change of moments of inertia, disturbance attenuating control strategies are highly desired. On the other hand, reaction wheels (RWs) are typical actuating devices which are utilized to provide high pointing accuracy for three-axis stabilized spacecraft [7]. Despite their great advantages, these wheels are vulnerable to faults. Doubtlessly, the performance of an attitude control system is extremely impressed by the

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performance of its actuators. So any fault in these devices can seriously jeopardize the mission of a spacecraft. The JAXA Hayabusa spacecraft, the NASA Kepler space telescope, the far ultraviolet spectroscopic explorer satellite, and GPS BII-07 are examples which their missions are degraded because of faulty RWs.

In this light, to ensure increased autonomy of current spacecraft, their control systems must be able to accommodate control effector faults without substantially affecting the performance and stability of the overall system. So, the issues of high precise pointing, reliability, and cost efficiency necessitate designing attitude control systems which ensure robustness of the overall system against parametric uncertainties, un-parametric uncertainties and faults. Here, we pursue a approach to design an additional control component to robustify the proposed method in [8] and [9] to a class of large uncertainties and faults. This is the main contribution of the paper. Namely, we develop a scheme which makes the effects of disturbance and RW faults to be actively rejected in the presence of unknown delay in actuators. To meet this request, we use the extended state observer (ESO) in the control structure. However the ESO has obtained successful achievements in many engineering problems (see [10] and references therein) and some methods are suggested in [11]-[13] to design an ESO for systems with delay; but all aforementioned researches are applicable to systems with known time delay. Note that, as previously mentioned, the assumption of knowing the delay is not necessarily satisfied for practical systems. Thus, this problem becomes one of the most crucial factor restricting the applicability of the ESO. The novel aspect of this paper is a modification on the original ESO which makes it applicable for systems with unknown delays. The modified ESO is then integrated with the nominal controller. It will be proved that closed-loop states reach to a small neighborhood of the origin even considering the effects of time delay, modeling uncertainties, external disturbances and faulty RWs, simultaneously.

The rest of the paper is organized as follows: spacecraft attitude model and problem formulation are outlined in Sec. 2; the controller development is presented in Sec. 3; simulation results are illustrated in Sec. 4 and the conclusions of Sec. 5 wrap up the paper.

2. PROBLEM FORMULATION

Among several existing methods for describing attitude representation of the spacecraft body with respect to the inertia frame [14], we use modified Rodrigues parameters (MRPs) to represent spacecraft attitude. MRPs offer the advantage of (i) being well-suited to onboard real-time implementation; since MRPs are characterized by a set of three parameters (i.e., a minimal set) the processing burden on the guidance and control spacecraft system reduces, and (ii) being valid for eigenaxis rotations up to 360^o.

The kinematic equations of the system under studying in terms of MRPs can be described by [6]

$$\dot{\sigma}(t) = B(\sigma)\omega(t) \tag{1}$$

With

$$B(\sigma) = \frac{1}{4}\left(\left(1 - \|\sigma\|^2\right)\right)I + 2S_\sigma + 2\sigma\sigma^T \tag{2}$$

where $\sigma = [\sigma_1, \sigma_2, \sigma_3]^T \in \Re^3$ is the MRP vector, $\omega = [\omega_1, \omega_2, \omega_3]^T \in \Re^3$ represents the angular velocity vector in the body-fixed frame, I denotes the 3×3 identity matrix, and $S_x \in \Re^{3\times3}$ is the cross-product operator for a vector $[x_1, x_2, x_3]^T$ defined by $S_x = [0, -x_3, x_2; x_3, 0, -x_1; -x_2, x_1, 0]$.

By tacking $J_r = J + \Delta J$ as real inertia matrix of the spacecraft in which $J = \mathrm{diag}[J_1, J_2, J_3], J_i \in \Re_+, i = 1,2,3$ is the nominal inertia matrix of the spacecraft and $\Delta J \in \Re^{3\times3}$ denotes uncertainties in the inertia matrix and $T_a = [T_{a_1}, T_{a_2}, \ldots, T_{a_m}]^T \in \Re^m$ as applied (actuator) torque with m is the number of actuators, the dynamics of the rigid spacecraft is given by [6]

$$\dot{\omega}(t) = -J_r^{-1}S_\omega J_r\omega(t) + J_r^{-1}DT_a(t-\tau) \tag{3}$$

where $\tau \in \Re_+$ is a time delay satisfying $0 \le \tau \le \overline{\tau}$ with $\overline{\tau}$ being known, and $D \in \Re^{3\times m}$ is the actuator configuration matrix.

Remark 1: To achieve safety in attitude control and to obtain optimality with respect to control effort, the actuators with redundancy ($m > 3$) are usually considered for attitude control system design. In this case, even if any faults/failures occur in actuators, it is still capable to stabilize the spacecraft and attain performance as close as nominal condition. Also, to avoid the singularity present in MRPs, we introduce a different set of MRPs, known as the shadow MRPs which abide by the same differential kinematics equation. The shadow MRPs, say σ^S, are related to the MRPs by $\sigma^S = -\sigma/(\sigma^T\sigma)$. By switching between the original and alternate sets of MRPs, we achieve a globally nonsingular attitude parameterization.

Three types of the most common catastrophic actuator failure including loss of effectiveness (fading actuation), lock failure (stock), and complete failure (outage) are considered in this study. Without going into the details of the possible nature of actuator faults, such faults can be modeled as

$$T_a = T + \Gamma_a(\overline{T} - T) \tag{4}$$

where $T = [T_1, T_2, \ldots, T_m]^T \in \Re^m$ represents the actuators' desired control commanded by the controller, $\overline{T} = [\overline{T}_1, \overline{T}_2, \ldots, \overline{T}_m]^T \in \Re^m$ is a (not necessarily constant) vector that cannot be manipulated, and $\Gamma_a = \mathrm{diag}[\Gamma_{a_1}, \Gamma_{a_2}, \ldots, \Gamma_{a_m}] \in \Re^{m\times m}$ is the actuator fault indicator in which Γ_{a_i} s are scalers satisfying $0 \le \Gamma_{a_i} \le 1$.

In this way, if $\Gamma_{a_i} = 0$, the i th actuator is working normally. If $\Gamma_{a_i} > 0$, a fault is present in the i th actuator, and if $\Gamma_{a_i} = 1$, the i th actuator has completely failed.

Substituting the actuator fault model (4) in the spacecraft dynamics (3) and incorporating with external disturbance $T_d \in \Re^3$ result the following model:

$$\dot{\omega}(t) = -J^{-1}S_\omega J\omega(t) + J^{-1}DT(t-\tau) + T_{td} \tag{5}$$

where $T_{td} \in \Re^3$ is the total disturbance which is defined by

$$T_{td}(t,\tau) = -\tilde{J}S_\omega J_r\omega(t) - J^{-1}S_\omega \Delta J\omega(t) - \tilde{J}DT(t-\tau) + J_r^{-1}D\left(\Gamma_a(\bar{T}(t-\tau) - T(t-\tau))\right) + J_r^{-1}T_d(t) \quad (6)$$

and $J_r^{-1} = (J+\Delta J)^{-1} = J^{-1} + \tilde{J}$ and $\tilde{J} = -J^{-1}\left(I + \Delta J J^{-1}\right)^{-1}\Delta J J^{-1}$.

Problem Statement: Given the rigid spacecraft model in (1) and (5), the control problem objective is to find the control law for T such that the closed-loop signals remain bounded and σ and ω reach a small region around the origin.

The above control objective should be met in the presence of (i) input time delay without knowing its exact value, (ii) uncertain spacecraft mass moment of inertia, (iii) unknown bounded external disturbances, and (iv) unknown actuator faults.

3. CONTROLLER DESIGN

The first object is to design a simple controller which ensures the system states tend to the origin for all initial values in the presence of delayed inputs when $T_{td} = 0$. The proposed control is given by [8], [9]

$$\omega_d(t) = -P\sigma \quad (7)$$

$$T(t) = D^\dagger J e^{\rho\bar{\tau}}\left(\rho\alpha + \dot{\omega}_d + J^{-1}S_\omega J\omega\right) \quad (8)$$

where $\alpha = \omega + P\sigma$, $D^\dagger = D^T(DD^T)^{-1}$ is the pseudo-inverse of D, $P = \mathrm{diag}[P_1, P_2, P_3]$ and ρ denotes are control gains, satisfying

$$\rho > 2.5\bar{P} + \|\alpha(0)\|_\infty - \frac{1}{2.1\bar{\tau}} \quad (9)$$

$$\rho > 2.5\bar{P} + \|\alpha(0)\|_\infty - \frac{1}{2.1\bar{\tau}} \quad (10)$$

$$\frac{\bar{P}^2}{\underline{P}} < \frac{1}{1.5\sqrt{5.01\bar{\tau}}} \quad (11)$$

in which $\bar{P}$ and $\underline{P}$ are the largest and smallest eigenvalues of the positive definite matrix P, respectively.

Although the controller given in (7) and (8) provides a fundamental contribution to the input-delay problem for nonlinear systems, as mentioned before, we require a robust and safe attitude control system which guarantees system performance while the plant is exposed to various disturbances and faults. To address this requirement, a ESO-based strategy, also known as active disturbance rejection control (ADRC), is employed. ADRC is a new paradigm that significantly reduces the tuning effort to achieve desired closed-loop performance while the system is operating in the presence of uncertainties and failures.

The idea of ESO-based control is simple; that is, the model of the dynamics subsystem (5) is extended with a new state variable which includes all uncertainties, disturbances, and Rws faults that are left unnoticed in the normal plant description; this new state is estimated by the ESO; then, the total disturbance is attenuated by adding a feedforward compensation law based on the estimated signal. But, as previously indicated, the original ESO needs a modification to make it suitable for systems with unknown delay time. Toward this end, let us rewrite the dynamics model (5) as

$$\dot{\omega}(t) = -J^{-1}S_\omega J\omega(t) + J^{-1}DT(t-\bar{\tau}) + T_{gd}(t) \quad (12)$$

$$\dot{T}_{gd}(t) = g(t)$$

where the unknown functional $T_{gd} \in \Re^3$ is a generalized concept in which effects of uncertainties, disturbances, and RWs faults, i.e., T_{td}, are combined with discrepancy between $J^{-1}DT(t-\tau)$ and $J^{-1}DT(t-\bar{\tau})$. We call this term as *Total Generalized Disturbance* (TGD). The unknown functional g is the derivation of TGD.

Next, to provide an effective and relatively easy to implement ESO to the prevailing practical applications with unknown delayed inputs, we propose the following modified ESO (MESO)

$$\begin{aligned} E_o &= \varpi_{o_1} - \omega \\ \dot{\varpi}_{o_1} &= -J^{-1}S_\omega J\omega + J^{-1}DT(t-\bar{\tau}) + \varpi_{o_2} - \beta_{o_1}E_o \\ \dot{\varpi}_{o_2} &= -\beta_{o_2}\,\mathrm{fal}(E_o, \wp_o, \delta_o) \end{aligned} \quad (13)$$

where $E_o \in \Re^3$ is an observer estimation error, $\varpi_{o_1} \in \Re^3$ and $\varpi_{o_2} \in \Re^3$ are observer states, $\beta_{o_1} = \mathrm{diag}[\beta_{o_{11}}, \beta_{o_{12}}, \beta_{o_{13}}]$ and $\beta_{o_2} = \mathrm{diag}[\beta_{o_{21}}, \beta_{o_{22}}, \beta_{o_{23}}]$ are observer gain matrices and the function $\mathrm{fal}(\cdot)$ is defined as [12], [15]

$$\mathrm{fal}(E_o, \wp_o, \delta_o) = \begin{bmatrix} \mathrm{fal}_1(E_o, \wp_o, \delta_o) \\ \mathrm{fal}_2(E_o, \wp_o, \delta_o) \\ \mathrm{fal}_3(E_o, \wp_o, \delta_o) \end{bmatrix} \quad (14)$$

With

$$\mathrm{fal}_i(E_o, \wp_o, \delta_o) = \begin{cases} |E_{o_i}|^{\wp_o}\,\mathrm{sgn}(E_{o_i}), & |E_{o_i}| > \delta_o \\ E_{o_i}/\delta^{1-\wp_o}, & |E_{o_i}| \le \delta_o \end{cases}$$

where $0 < \wp_o \le 1$ and $\delta_o > 0$ are predetermined observer parameters which denote the slop and range of the linearity of the fal function, respectively.

In the MESO, the maximum delay block $\bar{\tau}$, as shown in Fig. 1, is added to the control signal T before it goes into the ESO and estimation of TGD is provided by ϖ_{o_2}.

Finally, from integration of the nominal control with the MESO, the control law given in (8) is modified to

$$T(t) = D^\dagger J\left(e^{\rho\bar{\tau}}\left(\rho\alpha + \dot{\omega}_d + J^{-1}S_\omega J\omega\right) - \varpi_{o_2}\right). \quad (15)$$

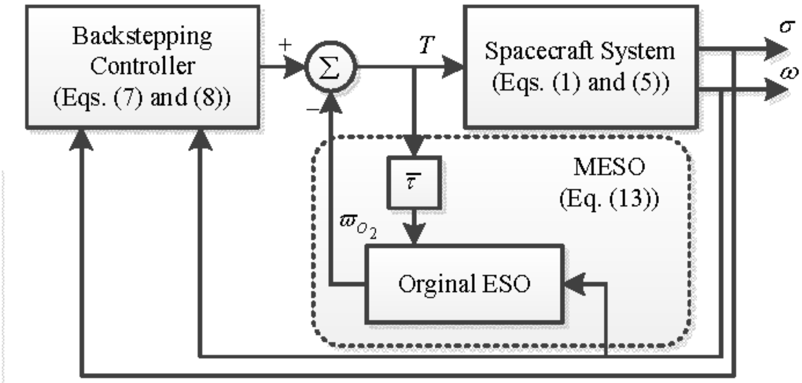


Fig. 1. Block diagram of the MESO-based control strategy

It should be emphasized that the defined control system needs to verify the following assumptions.

Assumption 1: Both attitude (non-inertial) and angular velocity (inertial) measurements are assumed to be available for feedback law. Furthermore, the angular velocity which is provided by gyros is incorrupt by various errors, such as misalignment, scale factor, bias errors, and noise.

Assumption 2: TGD and its first time derivation are unknown but bounded i.e., $T_{gd}, \dot{T}_{gd} \in L_\infty$.

The main results appeared in this paper are summarized in the following theorem:

Theorem: Consider a nonlinear spacecraft system described by (1) and (5) with the control law given by (7) and (15) and let Assumptions 1 and 2 satisfy. Furthermore, P and ρ exist such the conditions (9)-(11) hold and

$$\beta_{o_{1_i}}^2 > 4\beta_{o_{2_i}}\delta^{\wp-1}, \quad i=1,2,3 \tag{16}$$

Then, for all given initial conditions we have the following results:

i. The estimation errors i.e., E_o and $\varpi_{o_2} - T_{gd}$, converge to a small neighborhood around zero in finite time.

ii. $\sigma(t)$ and $\omega(t)$ are bounded for all $t \geq 0$ and these signals converge to a small set containing the origin.

Proof: The first statement can be proved by similar procedures which were presented in [15]. Hence, it is omitted here. To prove the second statement, we just need to show that the closed-loop system is input-to-state stable (ISS) with respect to the estimation error $\varpi_{o_2} - T_{gd}$.

Let the ISS Lyapunov-Krasovskii functional be

$$v(\sigma,\alpha) = \sum_{0}^{5} v_i \tag{17}$$

Where

$$v_0(\sigma) = 2\ln\left(1 + \| \sigma \|^2\right) \tag{18}$$

$$v_1(\sigma) = \frac{1}{2\overline{\tau}} \int_{t-\overline{\tau}}^{t} \int_{\lambda}^{t} \sigma^T(\eta) P \sigma(\eta) \mathrm{d}\eta \mathrm{d}\lambda \tag{19}$$

$$v_2(\alpha) = \gamma_1 \int_{t-\overline{\tau}}^{t} \int_{\lambda}^{t} \| \alpha(\eta) \|^2 \mathrm{d}\eta \mathrm{d}\lambda \tag{20}$$

$$v_3(\sigma) = \gamma_2 \int_{t-\overline{\tau}}^{t} \sigma(\eta)^T P \sigma(\eta) \mathrm{d}\eta \tag{21}$$

$$v_4(\alpha) = \gamma_2 \int_{t-\overline{\tau}}^{t} \| \alpha(\eta) \|^2 \mathrm{d}\eta \tag{22}$$

$$v_5(\sigma,\alpha) = \frac{1+802\overline{\tau}\gamma_1}{2\gamma_3} \| O \|^2 \tag{24}$$

in which $\gamma_1 = \left(22.545\overline{\tau}^3 \overline{P}^4 / \underline{P}\right)^{-1}$, $4\gamma_2 = 0.5 + \overline{\tau}\gamma_1 - \sqrt{0.25(1+\underline{P}) + \tau^2\gamma_1^2 - \tau\gamma_1}$, and $O = \alpha(t) + \int_{t-\tau}^{t} e^{\rho(t-\tau-\eta)} J^{-1} D T(\eta) \mathrm{d}\eta$, satisfying $\dot{O}^T O \leq -\gamma_3 \| O \|^2$ for some $\gamma_3 \in \Re_+$.

Taking the time derivative of (17) with some algebraic manipulations[1], we conclude that there exist a function $\kappa_1 \in K_\infty$ and a positive constant γ_4 such that

$$\dot{v} \leq -\kappa_1 + \gamma_4 \| \varpi_{o_2} - T_{gd} \|^2 \tag{23}$$

On the other hand, the definition of v allows us to state that

$$\kappa_2 \leq v(\sigma,\alpha) \leq \kappa_3 \tag{24}$$

where $\kappa_i \in \mathcal{K}_\infty, i = 2,3$.

Finally, using (23), (24) and according to Theorem 3.1 in [16], we can conclude that the functional v is a Lyapunov-Krasovskii functional which completes the proof of the theorem.

Remark 2: In the proposed method, the knowledge of the degree of failure for each RW is not needed. Indeed, the RW fault accommodation is done automatically without requiring any fault detection and isolation schemes. This feature and less computing power are advantages which make the proposed controller in Theorem 1 favorable to build affordable and effective fault-tolerant attitude control systems.

1 Computational details are omitted due to space limitations.

4. SIMULATION AND COMPARISON RESULTS

In this section, simulation results of the control strategies mentioned in this paper are presented. The complete set of physical parameters used in the numerical simulations is adopted from [17]. That is, the nominal inertia matrix of the spacecraft is $J = \text{diag}[60, 57, 65]\text{Kgm}^2$. The variation range of the inertia moments is not more than 8%. The initial attitude orientation is set to be $\sigma(0) = [0.6932, 0.1304, -0.2043]^T$ with a zero initial angular velocity.

The maximum delay time is supposed to be $\bar{\tau} = 750\text{ms}$ while the actual delay is considered to be 500ms. The configuration matrix D of four reaction wheels is given by

$$D = \begin{bmatrix} \sqrt{1/3} & \sqrt{1/3} & -\sqrt{1/3} & -\sqrt{1/3} \\ \sqrt{2/3} & -\sqrt{2/3} & 0 & 0 \\ 0 & 0 & -\sqrt{2/3} & \sqrt{2/3} \end{bmatrix}.$$

According to Theorem 1, the controller gains are select as $\rho = -0.06236$, $P = 0.114I$, $\beta_{o_1} = 25I$, $\beta_{o_2} = 30I$, $\wp_o = 0.2$ and $\delta_o = 0.25$.

For simplicity of the representation and discussion of the results, the controller described by (7) and (8) will be referred as BS and the modified control law governed by (7) and (15) is mentioned as MBS.

4.1. Attitude Stabilizing in Nominal Condition

In this case, an ideal situation is simulated in which not only no actuator fault occurs but also there are not any internal or external disturbances. Fig. 2(a) shows the time histories of MRPs. Time response of angular velocity is depicted in Fig. 2(b). Fig. 3 demonstrates applied control torque. As shown in Fig. 3 (dashed line), we observe the meager oscillatory behavior in the control torque which is produced by MBS. It is caused by the transient estimation error of MESO to estimate the fictitious disturbance $J^{-1}D(T(t-\tau) - T(t-\bar{\tau}))$ which is added by MESO; actually, in this scenario there are not any disturbances (external or/and enteral), and also the system is fault-free; the fictitious disturbance is injected to the closed-loop system because of introduction MESO in the control structure.

Attitude stability with high attitude pointing accuracy in both schemes are guaranteed in the presence of unknown input delay.

4.2 Attitude Stabilizing in Worst Case Condition

In order to demonstrate the performance and robustness of the proposed control laws in the presence of actuator faults and external disturbances, a set of numerical simulations is performed. The true inertia matrix $J_r = \text{diag}[57, 61.56, 59.8]\text{Kgm}^2$ is used in place of the nominal one in the spacecraft dynamics

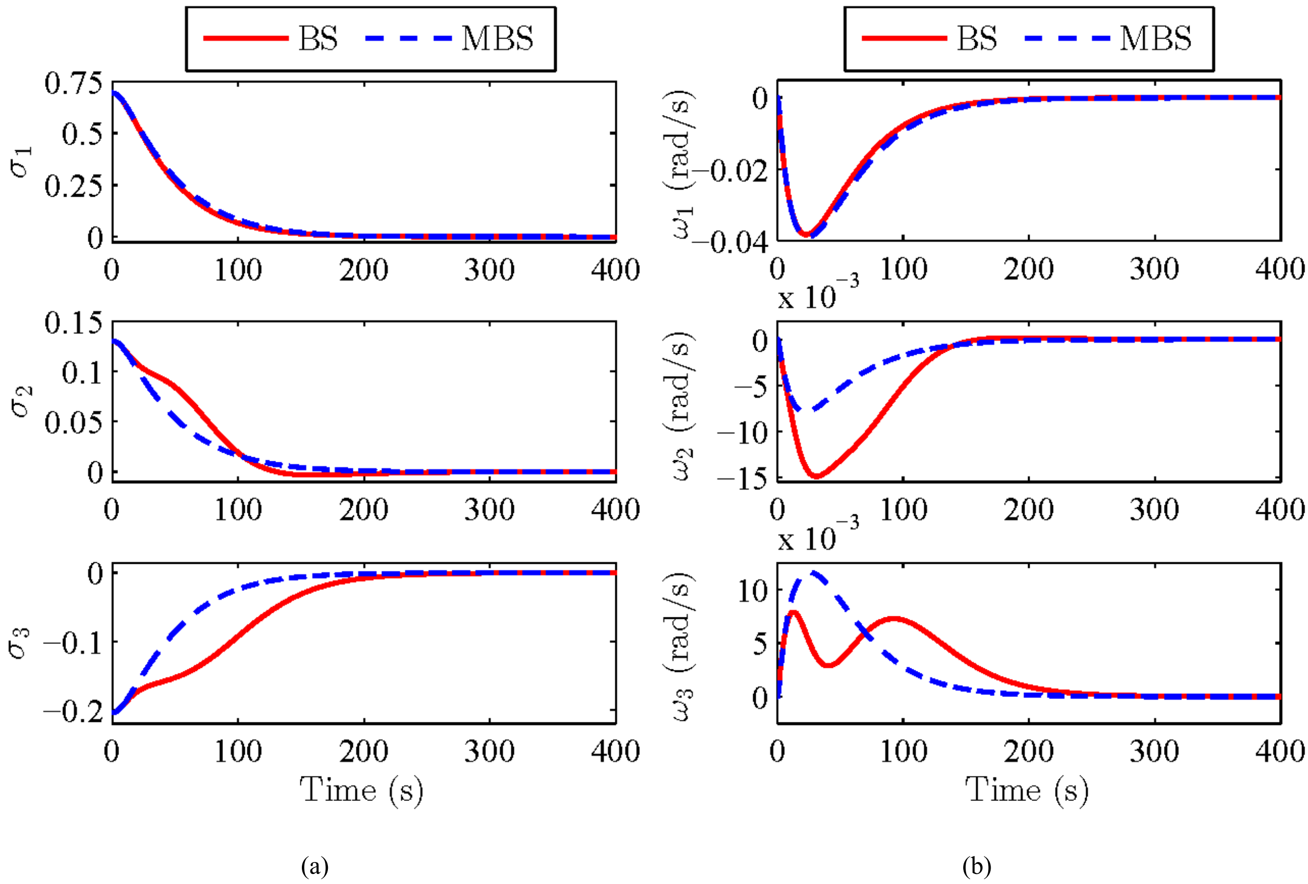


Fig. 2. Simulation results in the nominal condition for BS (solid line) and MBS (dashed line) for $\tau = 500\text{ms}$ and $\bar{\tau} = 750\text{ms}$ (a) MRP vector (b) Angular velocity.

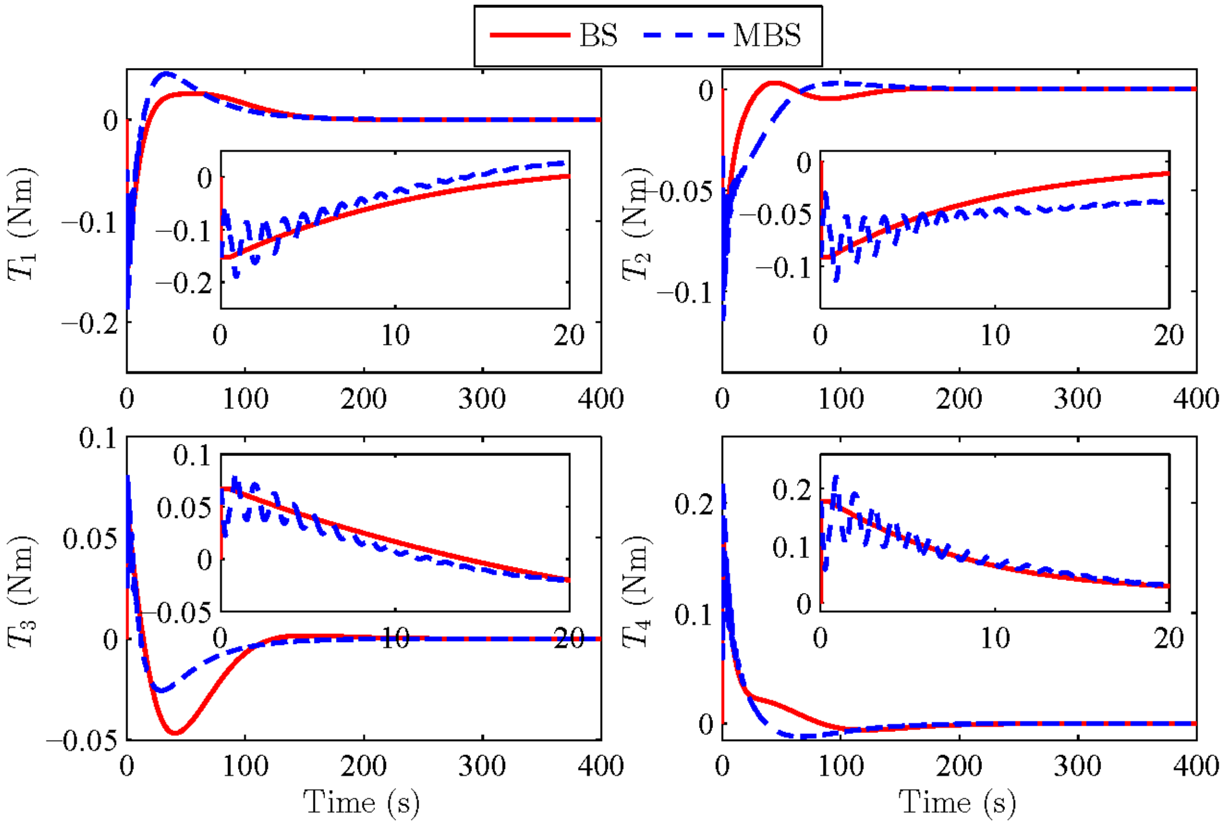


Fig. 3. Applied torque by actuators versus time in the nominal

while the inertia matrix used by the controllers remains the nominal one. The external disturbance is assumed to be $T_d(t)=\left(\|\omega\|^2+0.005\right)\left[\sin(0.8t)+1,\cos(0.5t)-2,\cos(0.3t)-4\right]^T$ Nm. Furthermore, actuator faults is considered as

$$\Gamma_{a_1}=\begin{cases}1, & t\le 35\text{s}\\ 0.6, & \text{othewise}\end{cases}\qquad \Gamma_{a_2}=\begin{cases}1, & t\le 25\text{s}\\ 0, & \text{othewise}\end{cases}$$

$$\Gamma_{a_3}=0\quad \Gamma_{a_4}=\begin{cases}1, & t\le 50\text{s}\\ 0.2, & \text{othewise}\end{cases}$$

$$\bar{T}_1=0\quad \bar{T}_2=\begin{cases}0.05, & t\le 25\text{s}\\ 0, & \text{othewise}\end{cases}$$

$$\bar{T}_3=0\quad \bar{T}_4=0$$

The fault scenarios occur under these situations: 1) the first RW starts to act after 35s with 40% of its power (during the period of $[0,35]\text{s}$, no toque is executed by this RW); 2) the second RW starts to act with lock failure. At time instant $t=50\text{s}$, this RW recovers its health and works normally; 3) in the third RW no any fault occurs; 4) the fourth wheel start to operate at $t=50\text{s}$ with 20% loss of its power (during the period of $[0,50]\text{s}$, no toque is executed by this RW).

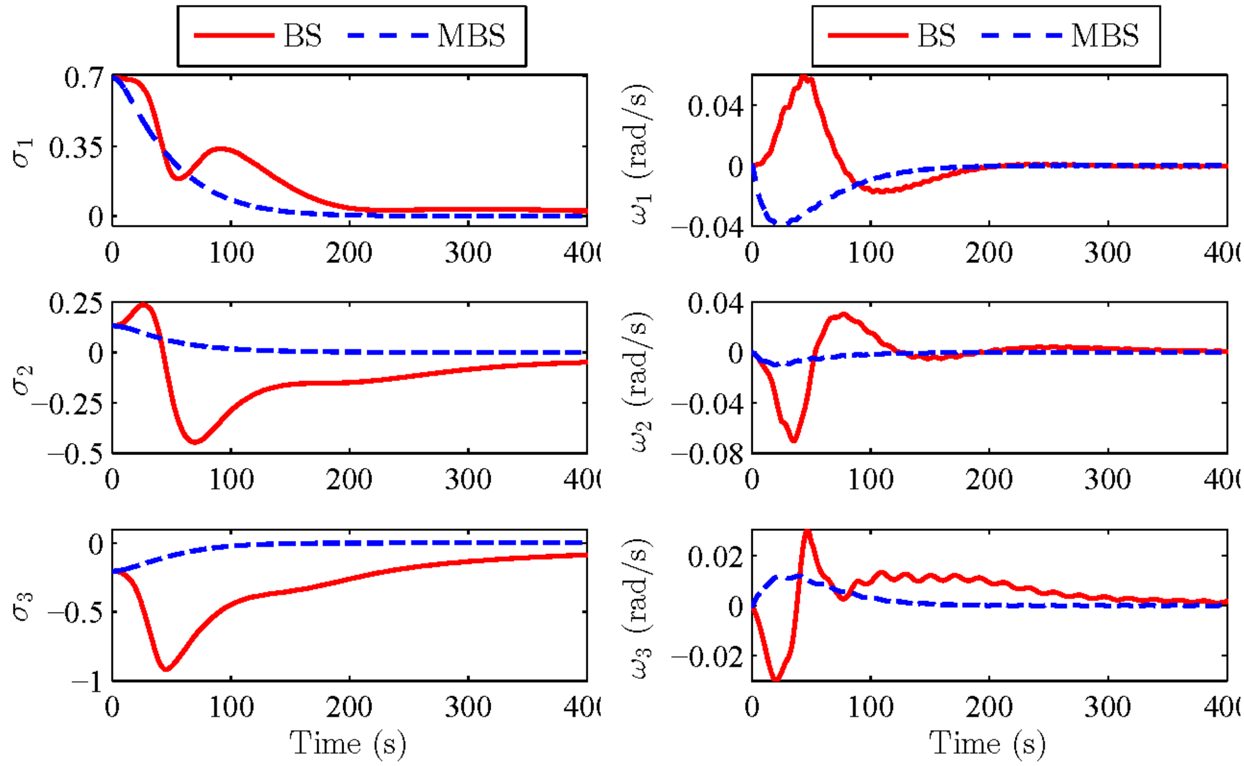
Figs. 4 and 5 show the closed-loop system states and driving torque, respectively. Observer states in the nominal and worst case conditions are depicted in Fig. 6(a) and Fig. 6(b), respectively.

Note that BS is still capable to drive system trajectories to a neighborhood of the origin despite of different faults and disturbances. This is duo to the fact that the closed-loop system with BS is ISS [9]; i.e., while disturbances are bounded, closed-loop states remain bounded.

In order to make a comparison between two schemes (BS vs. MBS) more clear, the steady attitude stability and pointing accuracy, the absolute workload and attitude recovery time are summarized in Table 1. Note that the absolute workload for interval $t\in[0,t_f]$ is defined as $W=\int_0^{t_f}\sum_{i=1}^{3}|\omega_i(t)T_{e_i}(t)|dt$ with $T_{e_i}(t)\in\Re^3$ as i th column of the matrix DT and the attitude recovery time, say T_S, is the time after which $\sigma_i<10^{-4}$, $\omega_i<10^{-4}$, $i=1,2,3$ always hold.

Based on simulation results, in the nominal condition, both of the proposed methodologies can achieve almost the same attitude pointing accuracy and attitude stability. When actuator fault and disturbance effects are considered, MBS can successfully perform the attitude stabilization maneuver and results clearly demonstrate that MBS is much more robust than BS in the presence of disturbance and achieves superior performance (the faster recovery time and the less absolute workload).

To show the advantage of the proposed MESO over the regular ESO, the simulations using the following two control schemes are conducted: (i) the proposed backstepping



(a) (b)

Fig. 4. Time response of (a) MRP (b) angular velocity in the presence of actuator faults, external disturbance and modeling uncertainties for $\tau = 500\text{ms}$ **and** $\overline{\tau} = 750\text{ms}$.

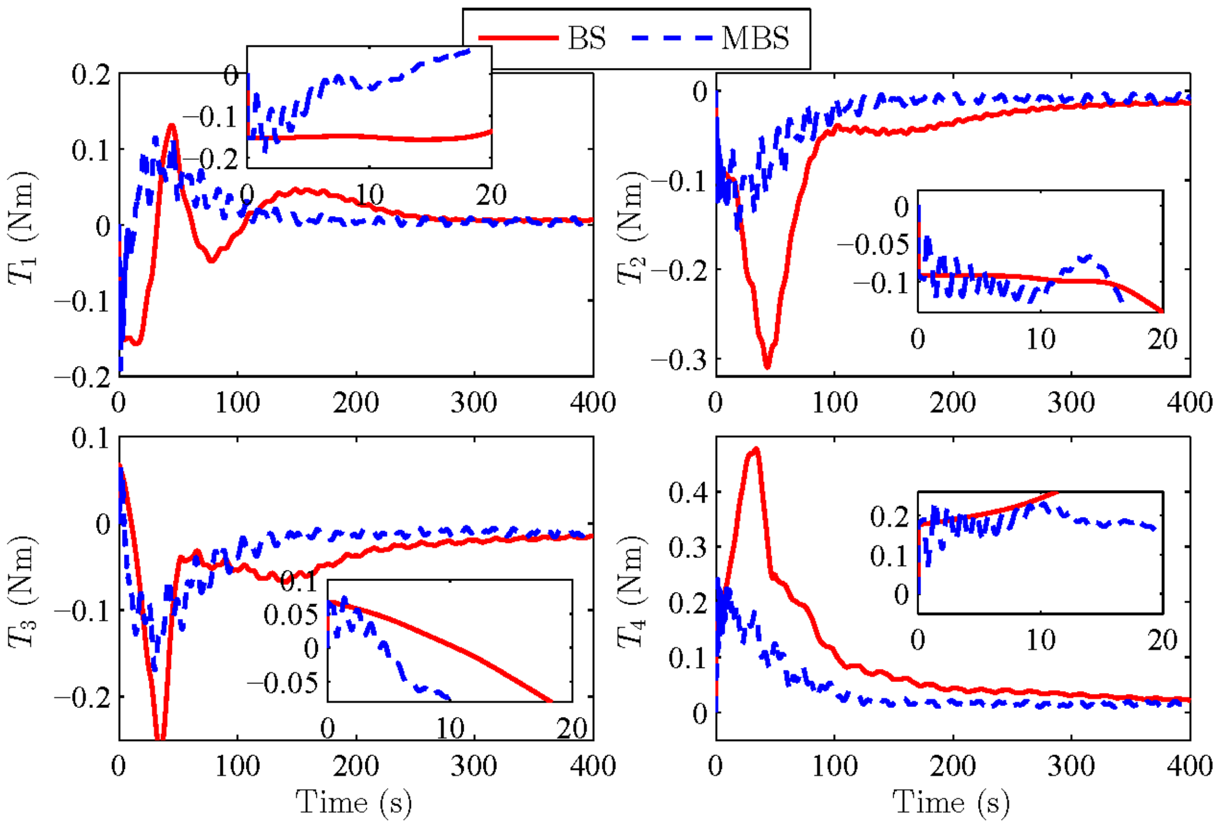
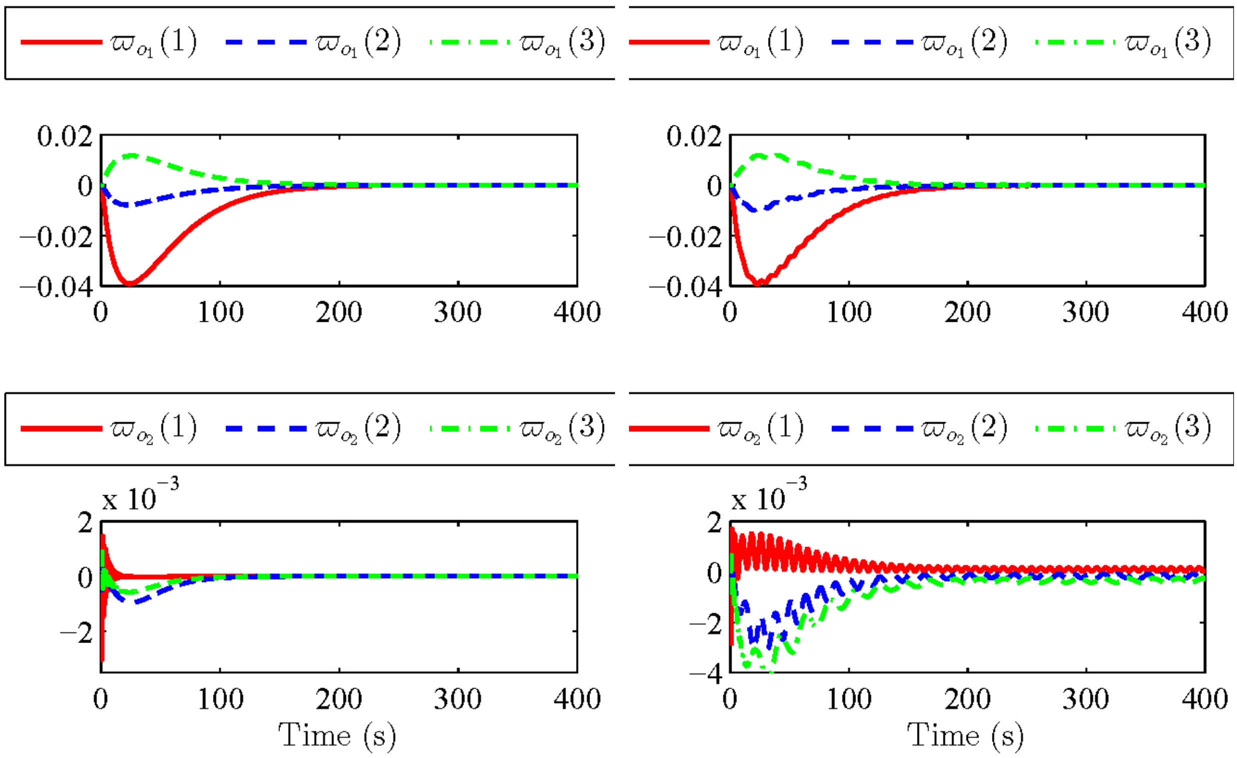


Fig 5. Desired control commanded by the controller in the worst case scenario.



(a) (b)

Fig. 6. MESO states in (a) the nominal condition (b) the worst case scenario.

Table 1. Performance comparison between BS and MBS

Control Performance		Simulation Scenario	Control Schemas	
			BS	MBS
Pointing Accuracy	σ_1	Nominal	2.12e-8	3.21e-8
		Worst	0.02	1.97e-5
	σ_2	Nominal	1.96e-8	6.91e-9
		Worst	0.03	1.14e-6
	σ_3	Nominal	4.76e-8	8.81e-9
		Worst	0.06	5.04e-6
Slew Rate Accuracy (rad/s)	ω_1	Nominal	2.49e-9	3.47e-9
		Worst	2.17e-5	4.51e-6
	ω_2	Nominal	1.99e-9	1.05e-9
		Worst	2.29e-5	8.76e-6
	ω_3	Nominal	5.58e-8	1.38e-9
		Worst	4.07e-4	5.08e-6
Absolute Workload (mJ)	W	Nominal	126.90	123.42
		Worst	1583.52	305.75
Attitude Recovery Time (s)	T_s	Nominal	349.3	349.3
		Worst	-	335.6

controller incorporating the original ESO, (ii) the proposed controller in Theorem 1, i.e., the backstepping controller incorporating the proposed disturbance observer in (13). The performance index $\mathrm{ISE}=\int_0^{400}(\varpi_{o_2}(t)-T_{gd}(t))\mathrm{d}t$ is evaluated for different ESOs. This index shows how efficient the observer is.

From Table 2, it is founded

that MESO provides an overall superior performance for all delays. But the performance of ESO becomes worst as delay increases and the closed-loop system with the regular ESO is not even stable for $\tau > 0.3\mathrm{s}$.

Table 2. Performance comparison between ESOs

Performance Index	Schemas	Actual Time delay (s)					
		0	0.15	0.30	0.45	0.60	0.75
ISE	ESO	0.032	0.032	0.032	_	_	_
	MESO	0.033	0.033	0.034	0.033	0.033	0.033

5. CONCLUSION

In this paper, we developed a novel MESO based backstepping scheme to achieve attitude stabilization maneuver in the presence of various actuator faults, uncertainties in the inertia parameters, external disturbances, and retreated inputs. The control approach guaranteed the closed-loop attitude system states to reach to a small neighborhood of the origin. With the modification of ESO and the proposed nominal backstepping control law, the exact value of delay time is not needed in designing the controller. Also any system identification process to identify the faults or any method of fault detection and isolation were not required. Simulation results demonstrate that the modified backstepping improve system performance and the considered disturbances are effectively rejected.

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