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ORIGINAL PAPER

An anti-disturbance PD control scheme for attitude control and stabilization of flexible spacecrafts

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Abstract This paper studies the attitude control problem of spacecrafts with flexible appendages. It is well known that the unwanted vibration modes, model uncertainty and space environmental disturbances may cause degradation of the performance of attitude control systems for a flexible spacecraft. In this paper, the vibration from flexible appendages is modeled as a derivative-bounded disturbance to the attitude control system of the rigid hub. A disturbance-observer-based control (DOBC) is formulated for feedforward compensation of the elastic vibration. The model uncertainty and space environmental disturbances as well as other noises are merged into an "equivalent" disturbance. We design a composite controller with a hierarchical architecture by combining DOBC and PD control, where DOBC is used to reject the vibration effect from the flexible appendages. Numerical simulations are performed to demonstrate that by using the composite hierarchical control law, disturbances can be effectively attenuated and the robust dynamic performances be enhanced.

Keywords Flexible spacecraft · PD control · Disturbance observer (DO) · Multiple disturbances · Vibration control · Composite hierarchical anti-disturbance control

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1 Introduction

High precision attitude control for flexible spacecrafts has been a difficult and important problem especially in communication, navigation, remote sensing, and other space-related missions. It is because modern spacecrafts often employ large, deployed and light damping structures (such as solar paddles and antenna reflectors) to provide sufficient power supply and reduce launch costs. Consequently, the complex space structure may lead to the decreased rigidity and low-frequency elastic modes. The dynamic model of a flexible spacecraft usually includes the interaction between the rigid and elastic modes [1, 2]. The unwanted excitation of the flexible modes during the control of the rigid body attitude, together with other external disturbances, measurement and actuator error, and unmodeled dynamics, may cause degradation of the performance of attitude control systems (ACSs). Thus, the control scheme must provide not only adequate stiffness and damping to the rigid body modes, but also actively damp or reject the flexible modes. The desired control scheme should be robust enough to overcome the model uncertainty and unmodeled nonlinearity, various disturbances from environment and structural vibrations of the flexible appendages.

Most present ACSs of spacecrafts are based on PID (proportional-integration-derivative) or PD (proportional-derivative) laws for its simplicity and reliability. When the model uncertainties and coupling vibration exist or external disturbance varies, it is difficult for PID/PD controller to get satisfactory performances for flexible spacecrafts. From the 1990s, instead of the conventional PID/PD control, optimal control of flexible spacecraft has once been studied for vibration suppression problems in [3, 4]. It is noted that robustness cannot be guaranteed for optimal control in the presence of model uncertainties and nonlinearity. From then on, control schemes with robustness have been designed for the attitude control and vibration suppression problem. Sliding mode control (SMC), known as an efficient and simple control strategy to systems with strong nonlinearity and model uncertainty is effectively applied to the ACS design [5–7]. SMC is combined with active vibration control by piezoelectric actuator, but the chattering phenomenon caused by SMC has limited its practical applications. In [8, 9], the influence of the flexibility on the rigid motion, the presence of disturbances acting on the structure, and parameter variations have been considered in robust controller design. In [10, 11], a series of variable structure control schemes have been provided systematically for vibration suppression problems. H_∞ control has been used in ACS design in [12, 13] where external disturbance and model uncertainty are considered. An H_∞ multi-objective controller based on the Linear Matrix Inequality (LMI) framework, is designed for flexible spacecraft in [14]. It is shown that H_∞ controller may lead to large conservativeness for the active vibration control problem, especially in case of the fast dynamic disturbance and high-frequency perturbations.

As an efficient anti-disturbance control strategy, DOBC has attracted considerable attention. DOBC has also been considered as a robust control scheme where the modeling error or the exogenous disturbance can be estimated and compensated through feedforward (see e.g. [15, 16]). Many different effective schemes have been provided for robots, hard disks and missiles (see e.g. [17, 18]). In [15], the DOBC approach in state space framework has been presented for a class of nonlinear uncertain systems, where the disturbance was generated by a linear exogenous system. It has been shown that DOBC may have less conservativeness for many types of disturbances and is easy to integrate with other conventional feedback controllers such as PD, H_∞, and variable structure controllers [15, 19].

In this paper, we will design a composite controller based on DOBC and PD control schemes for flexible spacecrafts, where DOBC can compensate the effect of vibration from flexible appendages, and PD controller can control the attitude of the spacecraft. Different from [15, 19], the disturbance is not confined to be a constant, a harmonic signal or a norm-bounded variable. With the proposed method, some new composite anti-disturbance composite methods (by combining DOBC with other feedback controllers) can be provided for ACS of spacecrafts. Simulations for a flexible spacecraft shows that the performance of ACS can be guaranteed by the proposed method.

Notation Throughout this paper, for a vector $s(t)$, its Euclidean norm is defined by $\|s(t)\|^2 = s^T(t)s(t)$. A real symmetric matrix $P > (\geq 0)$ denotes P being a positive definite (positive semi-definite) matrix. The identity and zero matrix are denoted by I and 0, respectively. Matrices, if not explicitly stated, are supposed to have compatible dimensions. The symmetric terms in a symmetric matrix are denoted by symbol $*$. For a square matrix M, we denote $\mathrm{sym}(M) := M + M^T$.

2 Problem formulation

To simplify the problem, only single-axis rotation is considered. We can obtain the single-axis model derived from the nonlinear attitude dynamics of the flexible spacecraft (see also [9, 10]). It is assumed that this model includes one rigid body and one flexible appendage (see Fig. 1) and the relative elastic spacecraft model is described as follows

$$J\ddot{\theta} + F\ddot{\eta} = u + w, \tag{1}$$

$$\ddot{\eta} + 2\xi\omega\dot{\eta} + \omega^2\eta + F^T\ddot{\theta} = 0, \tag{2}$$

where θ is the attitude angle, J is the moment of inertia of the spacecraft, F is the rigid-elastic coupling matrix, u is the control torque, w represents the merged disturbance torque including the space environmental torques, unmodeled uncertainties and noises from

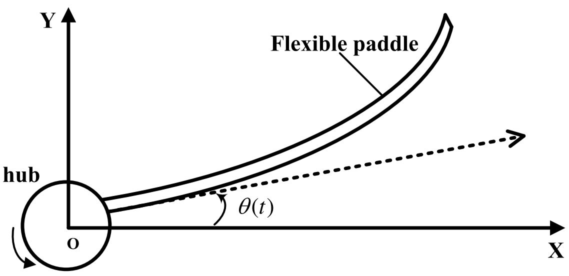

Fig. 1 Spacecraft with flexible appendages

sensors and actuators, η is the flexible modal coordinate, ξ is the damping ratio, and ω is the modal frequency. Since vibration energy is concentrated in low-frequency modes in a flexible structure, its reduced-order model can be obtained by modal truncation. In this paper, only the first two bending modes are taken into account.

Combining (1) with (2), we can get

$$(J - FF^T)\ddot{\theta} = F(2\xi\omega\dot{\eta} + \omega^2\eta) + u + w. \tag{3}$$

To (3), we consider $F(2\xi\omega\dot{\eta} + \omega^2\eta)$ as the disturbance due to elastic vibration of the flexible appendages. Denote $x(t) = [\theta(t), \dot{\theta}(t)]^T$, then (3) can be transformed into the following form

$$\dot{x}(t) = Ax(t) + B_u u(t) + B_f d_0(t) + B_d d_1(t), \tag{4}$$

where the coefficient matrices are denoted by

$$A = \begin{bmatrix} 0 & 1 \\ 0 & 0 \end{bmatrix},$$

$$B_u = B_f = B_d = \begin{bmatrix} 0 \\ (J - FF^T)^{-1} \end{bmatrix},$$

$d_0(t) = F(2\xi\omega\dot{\eta} + \omega^2\eta)$ is the disturbance from the flexible appendages, $d_1(t)$ is the "equivalent disturbance" including the space environmental disturbances, unmodeled uncertainties and noises from sensors and actuators, and $u(t)$ is the control input.

Obviously, the disturbance considered in this paper generalizes the disturbance types studied in previous works. In fact, we consider a new model with two different disturbance types $d_0(t)$ and $d_1(t)$. In this paper, we suppose $\|\dot{d}_0(t)\| \le W_0$ and $\|d_1(t)\| \le W_1$, where W_0 and W_1 are known positive constants.

The assumption that $\|\dot{d}_0(t)\|$ (upper bound of the derivative of the disturbance) is bounded by a constant is need to guarantee the stability. In the context of anti-disturbance control, this assumption is not very strong. In fact, $d_0(t)$ includes the neutral stable disturbances or harmonic disturbances with known frequency as studied in [20, 21]. Different from [15, 20, 21], in this paper the disturbance model is not required so that the proposed controller can be used directly and widely.

As the PD controller has been widely employed in many practical systems, we consider the classical PD controller described by $u_c(t) = K_P\theta(t) + K_D\dot{\theta}(t)$. Denote $K = [K_P, K_D]$, where K is the control gain to be determined.

3 Composite controller design

3.1 Disturbance observer design

According to system (4), we formulate the disturbance observer as

$$\begin{cases} \dot{\tau}(t) = -NB_f(\tau + Nx(t)) \\ \qquad\quad - N(Ax(t) + B_u u(t)), \\ \hat{d}_0 = \tau + Nx(t), \end{cases} \tag{5}$$

where N is the gain of the observer to be designed.

The estimation error of disturbance observer is defined as $e(t) = d_0(t) - \hat{d}_0(t)$. Then we have

$$\dot{e}(t) = \dot{d}_0 - NB_f e(t) - NB_d d_1(t). \tag{6}$$

The first step of DOBC framework is to estimate the disturbance via disturbance observer. According to the practical situation of the flexible spacecraft, we should design an appropriate N such that $e(t) \to 0$. In the DOBC scheme, the controller is constructed as

$$u(t) = -\hat{d}_0(t) + u_c(t) = -\hat{d}_0(t) + Kx(t).$$

The DOBC scheme can be described by Fig. 2 (where $\hat{d}_0(t)$ is the estimation of $d_0(t)$). From Fig. 2, it can be seen that the composite hierarchical controller consists of two parts, the inner loop is the disturbance observer and feedforward compensation, and the outside loop is the PD attitude controller. Thus, the composite controller can effectively control the spacecraft attitude and attenuate disturbances. The vibration caused by the flexible appendages is observed and compensated, and the attitude of the spacecraft is controlled by PD controller.

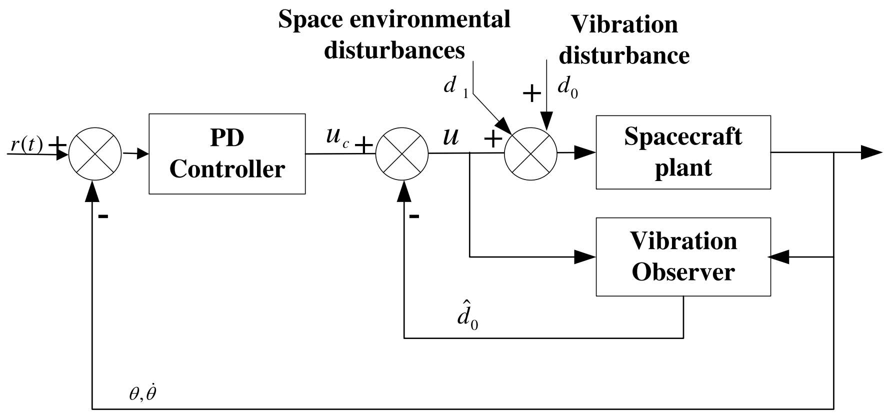

Fig. 2 Block diagram of composite attitude controller

Substituting $u(t)$ to (4) and (6), we can get the augmented closed loop system as follows

$$\begin{bmatrix} \dot{x}(t) \\ \dot{e}(t) \end{bmatrix} = \begin{bmatrix} A + B_u K & B_f \\ 0 & -N B_f \end{bmatrix} \begin{bmatrix} x(t) \\ e(t) \end{bmatrix} + \begin{bmatrix} 0 & B_d \\ 1 & -N B_d \end{bmatrix} \begin{bmatrix} \dot{d}_0(t) \\ d_1(t) \end{bmatrix}. \tag{7}$$

3.2 Stability of the composite systems

In this section, we will consider the uniformly ultimately boundedness (UUB) of the augmented system (7) (see e.g. [22]). We will give an LMI based design method to compute the controller gain and the observer gain simultaneously.

Theorem 1 *To augmented system (7), for $\alpha_1 > 0$, $\beta_1 > 0$ and $\beta_2 > 0$, if there exists matrices $Q_1 > 0$, $P_2 > 0$, R_1, and Q_2 satisfying*

$$\Theta = \begin{bmatrix} \Theta_{11} & B_f & 0 & 0 \\ * & \Theta_{22} & Q_2 & P_2 \\ * & * & -\beta_1^{-2} I & 0 \\ * & * & * & -\beta_2^{-2} I \end{bmatrix} < 0, \tag{8}$$

where

$$\Theta_{11} = \mathrm{sym}(A Q_1 + B_u R_1) + \alpha_1^2 I,$$

$$\Theta_{22} = -\mathrm{sym}(Q_2 B_f),$$

then composite system (7) is uniformly ultimately bounded, the observer gain and the controller gain can be computed via $N = P_2^{-1} Q_2$ and $K = R_1 Q_1^{-1}$.

Proof Denote

$$\begin{aligned} V(x, e, t) &= \begin{bmatrix} x^T & e^T \end{bmatrix} \begin{bmatrix} P_1 & 0 \\ 0 & P_2 \end{bmatrix} \begin{bmatrix} x \\ e \end{bmatrix} \\ &= x^T P_1 x + e^T P_2 e \\ &= V_1(x, t) + V_2(e, t). \end{aligned}$$

Computing the derivative of $V_1(x, t)$ along the trajectories of (7), we can obtain

$$\begin{aligned} \dot{V}_1(x, t) &= x^T \big(\mathrm{sym}(P_1 A + P_1 B_u K)\big) x \\ &\quad + 2x^T P_1 B_f e + 2x^T P_1 B_d d_1 \\ &\le x^T \big(\mathrm{sym}(P_1 A + P_1 B_u K) \\ &\quad + \alpha_1^2 P_1 P_1^T\big) x + \frac{1}{\alpha_1^2} \|B_d\|^2 \|d_1\|^2 \\ &\quad + 2x^T P_1 B_f e. \end{aligned}$$

Similarly, we can compute the derivative of $V_2(e, t)$ in the following

$$\begin{aligned} \dot{V}_2(e, t) &= e^T P_2 \dot{e} + \dot{e}^T P_2 e \\ &= \mathrm{sym}\big(e^T P_2 \dot{d}_0 - e^T P_2 N B_f e \\ &\quad - e^T P_2 N B_d d_1\big) \\ &= -e^T \mathrm{sym}(P_2 N B_f) e + 2e^T P_2 \dot{d}_0 \\ &\quad - 2e^T P_2 N B_d d_1 \\ &\le e^T \big(-\mathrm{sym}(P_2 N B_f) + \beta_1^2 P_2 N (P_2 N)^T \\ &\quad + \beta_2^2 P_2 P_2^T\big) e \\ &\quad + \frac{1}{\beta_1^2} d_1^T B_d^T B_d d_1 + \frac{1}{\beta_2^2} \dot{d}_0^T \dot{d}_0. \end{aligned}$$

Then, it can be verified that

$$\begin{aligned}\dot{V}(x,e,t) &= \dot{V}_1(e,t)+\dot{V}_2(x,t)\\ &\le x^T\left(\mathrm{sym}(P_1A+P_1B_uK)+\alpha_1^2P_1P_1^T\right)x\\ &\quad+e^T\left(-\mathrm{sym}(P_2NB_f)+\beta_1^2P_2N(P_2N)^T\right.\\ &\quad\left.+\beta_2^2P_2P_2^T\right)e\\ &\quad+\frac{1}{\alpha_1^2}\|B_d\|^2W_1^2+2x^TP_1B_fe\\ &\quad+\frac{1}{\beta_1^2}\|B_d\|^2W_1^2+\frac{1}{\beta_2^2}W_0^2\\ &=\left[x^T\ e^T\right]\Gamma\begin{bmatrix}x\\e\end{bmatrix}+\frac{1}{\beta_2^2}{W_0}^2\\ &\quad+\left(\frac{1}{\alpha_1^2}+\frac{1}{\beta_1^2}\right)\|B_d\|^2W_1^2\\ &=\left[x^T\ e^T\right]\begin{bmatrix}P_1&0\\0&I\end{bmatrix}\Upsilon\begin{bmatrix}P_1&0\\0&I\end{bmatrix}\begin{bmatrix}x\\e\end{bmatrix}\\ &\quad+C_b,\end{aligned}$$

where

$$\Gamma=\begin{bmatrix}\Gamma_{11}&P_1B_f\\B_f^TP_1&\Gamma_{22}\end{bmatrix},$$

$$\Gamma_{11}=\mathrm{sym}(P_1A+P_1B_uK)+\alpha_1^2P_1P_1^T,$$

$$\Gamma_{22}=-\mathrm{sym}(P_2NB_f)+\beta_1^2P_2N(P_2N)^T+\beta_2^2P_2P_2^T,$$

$$C_b=\frac{1}{\beta_2^2}{W_0}^2+\left(\frac{1}{\alpha_1^2}+\frac{1}{\beta_1^2}\right)\|B_d\|^2W_1^2, \tag{9}$$

$$\Upsilon=\begin{bmatrix}P_1^{-1}&0\\0&I\end{bmatrix}\Gamma\begin{bmatrix}P_1^{-1}&0\\0&I\end{bmatrix}=\begin{bmatrix}\Upsilon_{11}&B_f\\B_f^T&\Upsilon_{22}\end{bmatrix},$$

$$\Upsilon_{11}=\mathrm{sym}\left(AP_1^{-1}+B_uKP_1^{-1}\right)+\alpha_1^2I,$$

$$\Upsilon_{22}=-\mathrm{sym}(Q_2B_f)+\beta_1^2Q_2(Q_2)^T+\beta_2^2P_2P_2^T.$$

Denote $Q_1=P_1^{-1}$ and $R_1=KP_1^{-1}$, then by using the well-known Schur complement formula, it is shown that $\Theta<0\Leftrightarrow\Upsilon<0$. Denote

$$\Xi=\begin{bmatrix}P_1&0\\0&I\end{bmatrix}\Upsilon\begin{bmatrix}P_1&0\\0&I\end{bmatrix},$$

then $\Theta<0\Leftrightarrow\Xi<0$ holds, which leads to the following conclusion that there exists a positive scalar $\sigma_1>0$ such that $\Xi<-\sigma_1I$. So we can get

$$\dot{V}(x,e,t)<-\sigma_1\left\|\begin{bmatrix}x\\e\end{bmatrix}\right\|^2+C_b,$$

where C_b is defined by (9). The conclusion can be obtained. □

In Theorem 1, $\alpha_1>0$, $\beta_1>0$ and $\beta_2>0$ are necessary parameters to get the controller gain and observer gain. We can adjust the appropriate values to get the desired attitude control performances.

4 Simulations

In this section, the effectiveness of the present algorithm is demonstrated by numerical simulations. It will be shown that the effect of the elastic vibration can be estimated and compensated by DOBC, then enhanced anti-disturbance attitude control performance can be obtained. The composite controller will be applied for the attitude control of a spacecraft with solar paddles.

In this paper we only consider the attitude in the pitch channel. Four bending modes are considered for the practical spacecraft model at $\omega_1=3.17$, $\omega_2=7.38$, $\omega_3=16.954$, and $\omega_4=57.938$ rad/s with damping $\xi_1=0.0001$, $\xi_2=0.00015$, $\xi_3=0.000173$, and $\xi_4=0.0001576$, respectively. Because low-frequency modes are generally dominant in a flexible system, only the first two bending modes are used to represent the displacement of the flexible appendage in the simulation of the system. We suppose $F=[F_1\ F_2]$, where the coupling coefficients of the first two bending modes are $F_1=1.2781$ and $F_2=0.9176$, respectively. $J=35.72$ kg m^2 is the nominal principal moment of inertia of pitch axis. In addition, in order to enhance the robustness, ±20% perturbation of the nominal moment of inertial is also considered.

The flexible spacecraft is supposed to move in a circular orbit with the altitude of 500 km, then the orbit rate $n=0.0011$ rad/s. The space environmental disturbance torques acted on the satellite are supposed as follows

$$\begin{cases}T_{dx}=4.5\times10^{-5}(3\cos nt+1),\\ T_{dy}=4.5\times10^{-5}(3\cos nt+1.5\sin nt),\\ T_{dz}=4.5\times10^{-5}(3\sin nt+1).\end{cases}$$

The initial pitch attitude of the spacecraft are

$$\theta=0.08\ \text{rad},\qquad\dot{\theta}=0.001\ \text{rad/s}.$$

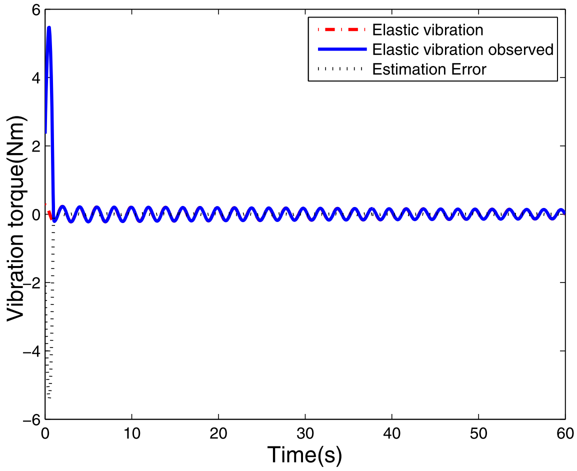

Fig. 3 Time responses of vibration and vibration observed

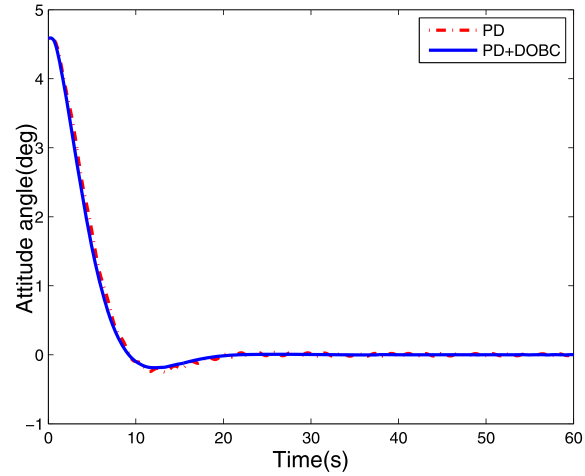

Fig. 5 Time responses of attitude angle

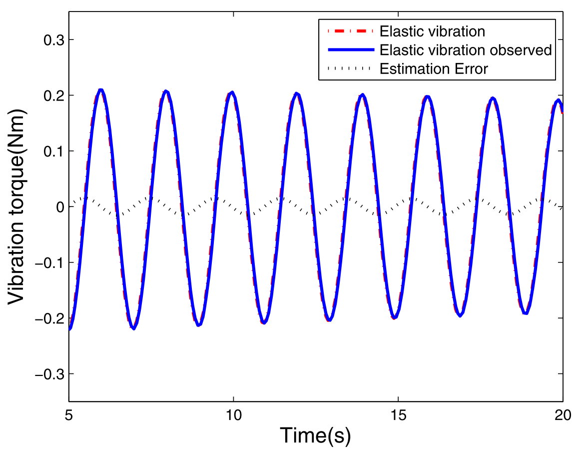

Fig. 4 Vibration estimation error in disturbance observer

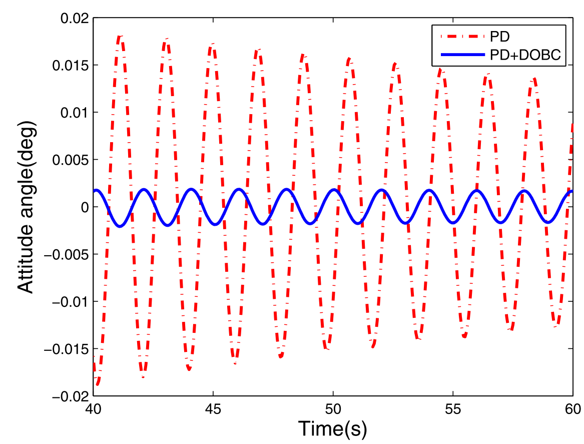

Fig. 6 Pointing precision under different controller

The intermediate design parameters in Theorem 1 are $\alpha_1 = 1600$, $\beta_1 = 1$ and $\beta_2 = 1000$, then we can get the parameters of the PD controller

$$K_P = 4.3515, \qquad K_D = 17.2488,$$

and the observer gain

$$N = [0\ 1481.3].$$

In this paper, $F(2\xi\omega\dot{\eta} + \omega^2\eta)$ is considered as the vibration torque caused by the flexible appendages. It can be easily verified that $d_1(t)$ and $\dot{d}_0$ are bounded. Figure 3 shows the time responses of elastic vibration, vibration observed and estimation error. Figure 4 is obtained by partial amplification of Fig. 3, and it shows that the vibration from the flexible appendages can be effectively estimated by disturbance observer, where the estimation error is less than 5% of practical elastic vibration. Thus, the effect of the elastic vibration to the rigid hub is reduced to the lowest by feedforward compensation. Figure 5 shows that the attitude angle of the spacecraft has fine dynamic response performance under the composite controller. To reveal the detail, the time response in steady state in Fig. 5 is zoomed in as shown in Fig. 6, where it shows that the attitude control accuracy is improved by composite controller compared with pure PD controller. At 50 s, the accuracy can be raised from about 0.0150 to 0.0020. Figure 7 shows that the attitude stabilization is improved obviously under composite controller compared with

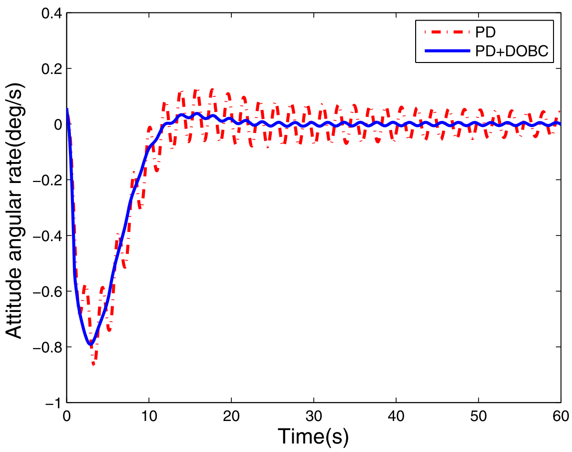

Fig. 7 Time responses of attitude angular velocity

pure PD controller in the presence of flexible vibration.

5 Conclusion

In this paper, a new composite control scheme is designed for attitude control of flexible spacecrafts in the presence of model uncertainty, elastic vibration and external disturbances. The composite controller combined DOBC and PD control law, where DOBC can reject the effect of the elastic vibration from the flexible appendages, and PD controller can effectively perform attitude control for the rigid hub in the presence of multiple disturbances. Simulation results showed that the composite controller can improve the pointing accuracy and stabilization capability of the flexible spacecraft. This method provides a useful and promising way for the flexible spacecraft attitude control, and more detailed work would be done in our future work. Further research is required to consider more general disturbance models and provide new composite control laws with enhanced anti-disturbance performance.

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