External Disturbance Detection and Its Application on the Identification of Fault in CMG System

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Abstract: CMGs (control moment gyros) as satellite actuators have intrinsic structures generating disturbance, like bearings, motors, high-speed rotating wheels, steering gimbal, etc. Disturbances induced by faults in some of these parts shall be detected immediately and identified in real time. A continuous second order sliding mode observer can be applied for the detection of disturbances. In this paper, a nonlinear sliding mode observing algorithm based on the gyro sensor is suggested for the detection of external disturbances. The algorithm is then applied to detect a fault in a CMG, here the wheel fluctuation fault. By distinguishing the direction of disturbance torque by a diagnosis algorithm, the fault CMG can be then identified and isolated from other normal CMGs. The performance of detecting algorithm is verified on the hardware satellite simulator, in which four CMGs are installed.

Key words: Satellite actuator, fault detection, identification, sliding mode observer.

1. Introduction

CMGs (control moment gyros) as momentum exchanging actuators, have been used as primary attitude control actuators for a variety of spacecraft. Slew or reorientation maneuvers of the spacecraft are executed by steering the gimbals of CMGs [1]. For the precision control of spacecraft, it is thus necessary that the gimbals are steered precisely while each wheel holding the nominal speed. Severe speed fluctuation either in wheel rotation or gimbal steering may induce critical disturbances [2-4]. While monitoring the spacecraft health, this type of severe faults in actuators thus need to be detected and isolated as quickly in real time in orbit as possible. Since the magnitude and frequency characteristics of such fault disturbances differ from those in normal conditions, the algorithm for the detection and identification of external disturbances can be used as a detector of actuator faults.

Since a spacecraft model is intrinsically nonlinear in large angle maneuvers, a proper nonlinear fault

detection and isolation strategy should be adopted [5-7]. Second order SMOs (sliding mode observers) have been suggested in several studies [8-12]. In this paper, a second order nonlinear SMO is applied to detect CMG fault-induced disturbances. The chattering in the conventional sliding observer is avoided by using the continuous type observer law as in Ref. [11]. The magnitude and frequency of the oscillating disturbance are estimated properly. The gimbal orientation of CMGs needs to be considered when isolating faults. First, the working concept of the suggested continuous type sliding mode observer is explained theoretically. Next, the observing algorithm is applied to detect the disturbances induced by the CMG fault—the high speed wheel motor fault. A vector transpose isolation scheme in the single fault case is suggested, and the performances are proven through several simulations with hardware simulator equipped with four CMGs.

2. Attitude Control with CMGs

A spacecraft with four single gimbal CMGs installed in a pyramid configuration is considered as shown in Fig. 1.

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Fig. 1 CMGs installed in pyramid configuration.

The motion of the spacecraft is then governed by the following equations:

$$\dot{q} = \frac{1}{2} E(q) \omega \quad (1)$$

$$I \dot{\omega} + \omega^\times I \omega + \omega^\times A h = u_c + u_f \quad (2)$$

$$D \dot{\sigma} = -u_c \quad (3)$$

where q is the spacecraft attitude quaternion, ω the angular velocity, and $h = [h_1 \ h_2 \ h_3 \ h_4]^T$ the wheel momentum vector. The matrix $E(q)$ is the quaternion matrix and I is the spacecraft inertia matrix. Matrix D , composed of the output torque vectors d_i s as $D = [d_1 \ d_2 \ d_3 \ d_4]$, changes with the CMG gimbal angles $\sigma = [\sigma_1 \ \sigma_2 \ \sigma_3 \ \sigma_4]^T$. Vector u_c is the control torque generated by steering σ . Matrix A is the gimbal configuration matrix, which is composed of the momentum wheel axis vectors a_i s as $A = [a_1 \ a_2 \ a_3 \ a_4]$. To go to the target attitude q_T , a conventional quaternion feedback control law Eq. (4) and a singular robust steering law Eq. (5) can be used as follows [13-15]:

$$u_c = \omega^\times (I \omega + A h) - K_D \omega + K_P E^T(q) q_T + K_I \int E^T(q) \omega dt \quad (4)$$

$$\dot{\sigma} = -D^T (D D^T + \alpha)^{-1} u_c \quad (5)$$

The disturbance torque vector u_f consists of several kinds of disturbances [11, 12]. The disturbances in this study are confined to one due to u_s , the wheel speed fluctuation disturbance. The

wheel momentum h is constant in normal conditions as $u_s = 0$. However, in the wheel speed fault due to the motor control fault, the wheel momentum is disturbed by a speed fluctuation, in which case u_s would have a nonzero value in a component corresponding to the fault wheel.

3. Disturbance Observer Design

When a disturbance u_f is exerted on the spacecraft body, the spacecraft attitude is then affected immediately. A state observer can be used as a tool to monitor the disturbance. Since the attitude equations are nonlinear, a nonlinear sliding observer is considered here as [9, 11]

$$\hat{I} \dot{\hat{\omega}} + \hat{\omega}^\times \hat{I} \hat{\omega} + \hat{\omega}^\times A h = u_c - \hat{I} v \quad (6)$$

where $\hat{I}$ is the nominal estimate value of I , and $\hat{\omega}$ of ω . If the state estimate error is defined as $e = \omega - \hat{\omega}$, then the error dynamics becomes

$$\dot{e} = f + I^{-1} u_f + v \quad (7)$$

where f is the system error due to the error in the inertia and angular velocity as follows:

$$f \equiv I^{-1} (-\omega^\times I \omega - \hat{\omega}^\times A h) - \hat{I}^{-1} (-\hat{\omega}^\times \hat{I} \hat{\omega} - \hat{\omega}^\times A h) \quad (8)$$

The observer switching term v is then designed by the second order sliding mode [8-11]. Let S be a sliding surface satisfying

$$\ddot{S} + z_o \dot{S} = \dot{e} + c e \quad (9)$$

where z_o and c are stable gains to make $S \rightarrow 0$. Consider a Lyapunov function candidate V with a positive l .

$$V \equiv \frac{1}{2} (\dot{S}^T \dot{S} + S^T l S) \quad (10)$$

By choosing v as $v \equiv z_o \dot{S} - c e - l S - d \operatorname{sgn}(\dot{S})$, then $\dot{V} \leq 0$ and $S, e \rightarrow 0$ with the selection of a proper large positive gain d . However, since chattering exists around the $S=0$ surface in this type of algorithm, the function $\operatorname{sgn}(\dot{S})$ has been occasionally substituted with $\operatorname{sat}(\dot{S})$ [9, 10]. In this

paper, however, we use a different continuous term (written as $d\dot{S}$) instead of $sat(\dot{S})$, as suggested in Ref. [11]. With a sufficiently large gain in d , the switching law $v \equiv z_o \dot{S} - ce - lS - d\dot{S}$ stabilizes the observer around $e = 0$ [11]. Finally, the estimate of disturbance $\hat{u}_f \equiv -\hat{I}(\hat{f} + v)$ measures the disturbance u_f within some error bound of the inertia.

4. External Disturbance Detection

4.1 Secular Disturbance

On orbit, the satellite occasionally experience secular disturbance torques, like gravity gradient torque, and solar pressure, etc., which induce the wheel momentum increase and finally saturation. The detection of secular disturbance in real time is thus helpful to provide proper curing actions. The observer designed above can be used as a detector of this type of disturbance. Consider an example case with a mid-class spacecraft, as MOI $I = \text{diag}\{200, 80, 130\}$ (kg/m²). The observer and controller gains are set as $[K_D \ K_P \ K_I] = [53 \ 7 \ 0]$ and $[z_o \ c \ l \ d] = [0.6 \ 100 \ 0.6 \ 1.65]$.

Assume a secular disturbance $u = [0.1 \ 0 \ 0]$ acts on x -axis of a satellite during time $t = (20, 60)$ sec as in Fig. 2c.

The angular velocity estimate $\hat{\omega}$ in Fig. 2b obtained from the observer follows well the real angular velocity ω in Fig. 2a. The disturbance by using the suggested switching term is well detected immediately at time $t = 20$ sec and the magnitude is also identified exactly in Fig. 2d. The Lyapunov function $\dot{V} \rightarrow 0$ in Fig. 2f and the sliding surface state $\dot{S} \rightarrow 0$ in Fig. 2e as expected. Conclusively, the designed observer works well as a secular disturbance detector.

4.2 Oscillating Disturbance

As satellite rotates around the earth, it may be affected by oscillating disturbances like aerodynamic force and earth oblateness, etc. For the precision attitude control, the oscillating disturbance needs to

detect. The performance of detecting the oscillating disturbance depends on the observer processing frequency. For the detection of high frequency disturbance, the observer frequency shall be greater 5 ~ 10 times than the disturbance frequency for more precise prediction. Assume the oscillating disturbance $u = [0 \ 0.1 \sin 2\pi f_d t \ 0]$ acts on the satellite during time $t = (20, 60)$ sec on y -axis. Here, the external disturbance frequency $f_d = 5$ Hz, and the observer is then set to process every 0.02 sec, i.e., 10 times faster than the disturbance frequency. The time constant of error e shall be then reduced by using bigger gain c . As shown in Fig. 3, the oscillating disturbance in Fig. 3a is well detected and the magnitude is estimated exactly as in Fig. 3b.

5. Actuator Fault Detection and Identification

We consider now the application of the above continuous observer on the detection and identification of a fault, e.g. momentum wheel motor speed fluctuation in a CMG.

5.1 Disturbances Due to Momentum Wheel Speed Fault

The required torque for a spacecraft maneuver is generated by steering the gimbal of each CMG. To generate the exact amount of required torque, CMG wheel momentum should be maintained constant in a nominal value. When a malfunction occurs in the wheel speed controller or bearing friction, etc., then the wheel speed might fluctuate and the disturbances are generated. The speed fluctuation can be detected by using the same disturbance detector suggested above.

Assume that the speed control of wheel No. 1 in Fig. 1 malfunctions. The wheel speed fluctuates as $\dot{h} = [0.01 \times 2\pi f_d \cos 2\pi f_d t \ 0 \ 0 \ 0]^T$ where $f_d = 1$ Hz during the maneuver time $t = (20, 40)$ sec interval. Even though the wheel speed fluctuates and thus induces the oscillating disturbance torque on the spacecraft body as $u_f = -A\dot{h}$, the spacecraft attitude

Fig. 2 Responses on a secular disturbance.

Fig. 3 Responses on oscillating disturbance.

Fig. 4 Observer responses to a momentum wheel speed fault.

is finally stabilized as shown in Fig. 4a. The angular velocity of the spacecraft is stabilized to the origin as shown in Fig. 4b. The angular velocity estimate $\hat{\omega}$ starts from an arbitrarily initial guess but aptly follows the real angular velocity ω , as before. The disturbance inserted from $t = 20$ sec in Fig. 4c is detected immediately with the magnitude identified exactly as in Fig. 4d. The designed observer works well as a oscillating disturbance detector.

5.2 Fault Identification Algorithm of Malfunction Wheel

The wheel induced disturbance torque is propagated to the all three body axes. The disturbances are thus detected in all three axes by the disturbance observer, as shown in Fig. 4d. This is due to the change of CMG

gimbal angles during maneuvers. The spacecraft operator needs to identify the abnormal wheel among the normal wheels. The disturbance effects on the spacecraft body depend on the instantaneous gimbal orientation, written as $u_f = -A\dot{h}$. The disturbance estimate $\hat{u}_f$ is obtained by the disturbance detector as before. To identify the malfunction wheel, $\hat{u}_f$ is projected on the column space of D and A as

$$\hat{h} = A^T \hat{u}_f \quad (11)$$

$$\hat{h}_N = D^T \hat{u}_f \quad (12)$$

We can obtain the candidate responses for $\hat{h}$, as shown in Fig. 5a. However, ambiguity still exists as to which wheel—No. 1 or No. 3 in this simulation—is malfunctioning. From the fact that $\hat{u}_f$ is maximally projected on the wheel axis as in Eq. (11) and

Fig. 5 Fault wheel identification.

minimally projected (i.e., orthogonal) to the output torque axis as in Eq. (12) of the malfunctioning wheel, by choosing the wheel axis with the direction nearest to $\hat{u}_f$, we can finally isolate No. 1 as the fault wheel, as shown in Fig. 5b.

6. Hardware Simulator Verification

We now verify the performance of the fault detector with the hardware ground simulator, i.e. an air-bearing test platform recently designed and manufactured at Spacecraft Control Laboratory in Korea Aerospace University as Fig. 6. The simulator has a table-top configuration capable of floating 100 kg-class platform, the scale model for a mid-class satellite. System MOI is designed as $I = \text{diag}\{20, 15, 15\}$, 1/10 scale model of mid-class spacecraft. Attitude is measured by reference sensors: a sun-sensor and an accelerometer, which simulate the sun vector and earth vector, respectively. The angular velocity is measured by gyros. All sensors and actuators are not for the space flight class, but for the ground vehicle class with lower grade. CMGs are developed in SCL KAU as Fig. 7. Each CMG has a momentum wheel of 2/3 Nms angular momentum and a gimbal with maximum acceleration of 3 rad/sec.

6.1 Fault Identification of Malfunction Momentum Wheel

The fault is so injected on the momentum wheel

speed control of CMG1 that the wheel speed fluctuates as $\dot{h} = [h_o \times 2\pi f_d \cos 2\pi f_d t \ 0 \ 0 \ 0]^T$ where $f_d = 0.1$ Hz at time $t = 10$ sec as in Fig. 8. The disturbance detection algorithm detects the disturbance as in Fig. 9. We can see that all body axes are fluctuated by disturbance torque. Thereafter, the fault identification algorithms of Eqs. (11) and (12) are applied to identify the fault wheel among four wheels. By projection of the

Fig. 6 Hardware simulator on an air-bearing.

Fig. 7 SCL KAU CMG.

Fig. 8 MWA speed.

Fig. 9 Estimated disturbances.

Fig. 10 Fault wheel identification.

estimated disturbance on each CMG frame, the comparison of four results in Fig. 10 leads to the final conclusion that there exists the speed fault in momentum wheel of CMG1.

7. Conclusion

Application of a continuous nonlinear second order sliding mode observer is suggested to detect disturbances in spacecraft with CMGs. Both slowly and highly oscillating external disturbances are aptly

detected, and the magnitude is identified successively. Internal wheel faults that result from wheel speed error can be detected using the suggested observer and the fault CMG can be identified by applying the simple vector transpose identifying logic. The algorithm performance is verified in real hardware simulator.

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