Approach To Model Reference on Orbit Fault Diagnosis of Satellite Attitude Control System Based on Measurement of Environment Torques

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Abstract: According to principle of model reference adaptation, an approach to model reference on-orbit fault diagnosis for satellite attitude control system was proposed based on measurement and calculation of environmental torques acted on satellite. In view of satellite kinematics and attitude dynamics, a linear model for satellite attitude control system is established. On the basis of system parameter and structural characteristics, the reference model of satellite attitude control system is built and improved through measurement and calculation of environmental torques. In this way, the fault residual vector is generated through comparative analysis of the output of reference model with the one of real system and the disturbance is estimated by using extended state observer and augmented matrix method so as to accomplish the sensor fault detection and isolation. Simulation experiment results demonstrate its effective for on-orbit real-time fault diagnosis of satellite attitude control system.

Key Words: Satellite Attitude Control System, Fault Diagnosis, Environmental Torque, Reference Model, Extended State Observer.

1. Introduction

Satellite attitude control system mainly undertakes several tasks such as on-orbit satellite attitude measurement, attitude information calculation and attitude control command generation and implementation, so its performance is directly related to carrying out works of the correlated payload. As the complexity of satellite structure, the variety interfere of environmental factors in orbit and the increasing precise of the assembled components, it is difficult to ensure that satellite attitude system has not any problems during on-orbit. In case of failure of the satellite attitude control system, it not only prejudices the accomplishment of tasks but also may jeopardize the safety of the whole-spacecraft if without timely diagnosis and effective intervention[1-3]. Therefore, it is necessary to develop fault diagnosis technology of satellite attitude control system in orbit to provide effective protection for the safe and reliable operation of satellite and the works achievement efficiently and smoothly.

Analytical model based diagnosis method is to a powerful tool for troubleshooting of satellite attitude control system in orbit. Wang et.al[4] proposed a fault diagnosis method for satellite attitude sensor based on Kalman filter. The maximum likelihood estimation is employed to design Kalman filter, and then the fault detection and estimation is achieved according to the state vector obtained from filter. Gao et.al[5] built multiple failure modes based on the interacting multiple model (IMM) method. In which, the state augmentation method was combined with the non-destructive Kalman filter to estimate the fault parameters and realize fault diagnosis of the satellite attitude actuator. Lee et.al[6] estimated the system state and implemented fault detection and diagnosis of the satellite attitude sensors and actuators by designing cascaded filters composed of a decentralized Kalman filter and a set of IMM dispersion filter. Baldi et.al [7] estimated the system state by designing a set of nonlinear geometric algorithms filter and made fast Fourier transform over the residual vector, then fault detection and isolation of the satellite attitude control system actuator was achieved by power spectrum analysis. Aiming at the mutation and soft faults of the satellite attitude control system actuator, Wang et.al[8] presented a real-time detection and diagnosis algorithm for parameter deviation fault based on strong tracking filter theory. Cen et.al[9] employed the extended state observer (ESO) to achieve the estimation of failure quantity depending on the satellite attitude control system actuator output and controller output by using adaptive observer. Analytical model based diagnosis method usually build multiple models associated with potential failure modes to generate residual sequence which contains the fault characterization information, and then determine the location and type of fault failure by identifying the residual sequence to achieve fault detection and diagnosis[10].

In this paper, the reference model is modified based on measurement and calculation of environmental torques acted on satellite. The residual signals of sensors are generated based on the comparison of corresponding output vector between the reference model and the actual system. The ESO is used to estimate the system disturbance, and then the fault residual signals of actuator are generated according to the estimated disturbance value and the measurement value of environmental torque. Finally, the estimated fault residual vector amplitude is integrated with the fault detection thresholds to realize fault detection and location and then achieve the diagnostic decision.

2. Scheme design of fault diagnosis approach for on-orbit satellite

According to the structural characteristics of the satellite attitude control system, the reference model and ESO are built. By real-time calculation of the suffered environmental torques for on-orbit satellites, the faults residual vectors of sensor and actuator are acquired respectively, and then the fault detection and isolation of on-orbit satellite is realized.

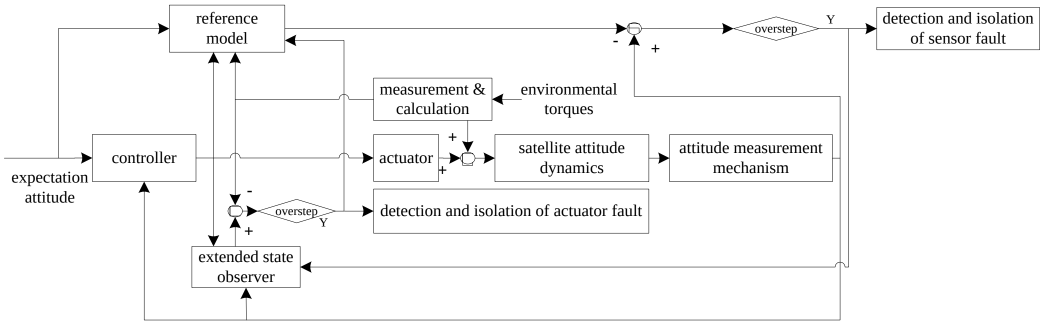

Fig. 1.1. Fault diagnosis of on-orbit satellite attitude control system based on measurement of environment torques

2.1. Approach to on-orbit fault diagnosis with model reference

Fig. 2.1 depicts the proposed model reference fault diagnosis approach for on-orbit satellite attitude control system based on measurement and calculation of environmental torques in this paper. According to on-orbit satellite operation mode, the structural characteristics and effect mechanism of the spatial environmental torques, the environmental torque suffered by satellite is calculated in real time. The extended state observer of attitude control system is designed. When no sensor failure is detected, the difference between the attitude measurement output and the observer output will be fed back as error, and the disturbance suffered by the system will be estimated by using augmented matrix method, in which the estimated disturbance and measurement and calculation of environmental torques will be used to obtained actuator failure impact vector to achieve detection and isolation of actuator. The reference model of the attitude control system is constructed. When no actuator fault is detected, the controller output will be regard as incentive for the reference model, and then the reference model will be corrected according to the environmental torque calculation, in which fault residual vector will be generated by the actual system output and the reference model output to detect and isolate the sensor faults. In the case of some actuator fault is detected, the estimated impact vector will be used correct reference model incentive. In the case of some sensor fault is detected, the generated residual vector will be employed to correct the error feedback of the ESO.

2.2. Reference model and its performance verification

According to the three-axis stabilized satellite kinematics and attitude dynamics[11)], the zero momentum flywheel is regard as the actuator of attitude control system, and then the satellite attitude kinematics and attitude dynamics are formulated as following where $\boldsymbol{J} = \mathrm{diag}(J_x, J_y, J_z)$ is the astral inertia matrix; the satellite attitude in orbital coordinate system is represented by rolling angle φ, pitching angle θ and yaw angle ψ; The rotate speed of orbital coordinate system is $(0, -\omega_0, 0)$; d_x, d_y and d_z are environmental torques suffered by satellite; u_x, u_y and u_z are control torques provided by the three-axis flywheel; ω_x, ω_y and ω_z are speed components of satellite rotary plate in the system of proprio-coordinate.

$$\begin{cases} J_x\ddot{\varphi} + \omega_0^2(J_y - J_z)\varphi + \omega_0(J_y - J_x - J_z)\dot{\psi} = d_x + u_x \\ J_y\ddot{\theta} = d_y + u_y \\ J_z\ddot{\psi} + \omega_0^2(J_y - J_x)\psi - \omega_0(J_y - J_x - J_z)\dot{\varphi} = d_z + u_z \end{cases} \tag{1.}$$

$$\boldsymbol{\omega} = \begin{bmatrix} \omega_x \\ \omega_y \\ \omega_z \end{bmatrix} = \begin{bmatrix} \dot{\varphi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} + \begin{bmatrix} 1 & \psi & -\theta \\ -\psi & 1 & \varphi \\ \theta & -\varphi & 1 \end{bmatrix}\begin{bmatrix} 0 \\ -\omega_0 \\ 0 \end{bmatrix} = \begin{bmatrix} \dot{\varphi} - \omega_0\psi \\ \dot{\theta} - \omega_0 \\ \dot{\psi} + \omega_0\varphi \end{bmatrix} \tag{2.}$$

According to Eq. (1) and (2), the state space model of satellite attitude control system can be obtained as following

$$\begin{cases} \dot{\boldsymbol{x}}(t) = \boldsymbol{A}\boldsymbol{x}(t) + \boldsymbol{B}\boldsymbol{u}(t) + \boldsymbol{G}\boldsymbol{d}(t) \\ \boldsymbol{y}(t) = \boldsymbol{C}\boldsymbol{x}(t) + \boldsymbol{w} \end{cases} \tag{3.}$$

where $\boldsymbol{x} = [\dot{\varphi} \quad \dot{\theta} \quad \dot{\psi} \quad \varphi \quad \theta \quad \psi]^{\mathrm{T}}$, $\boldsymbol{w} = [0 \quad -\omega_0 \quad 0 \quad 0 \quad 0 \quad 0]^{\mathrm{T}}$, $\boldsymbol{u} = [u_x \quad u_y \quad u_z]^{\mathrm{T}}$, $\boldsymbol{d} = [d_x \quad d_y \quad d_z]^{\mathrm{T}}$, $\boldsymbol{y} = [\omega_x \quad \omega_y \quad \omega_z \quad \varphi \quad \theta \quad \psi]^{\mathrm{T}}$, $a_1 = -(\omega_0(J_y - J_x - J_z))/J_x$, $a_2 = -(\omega_0^2(J_y - J_z))/J_x$, $a_3 = (\omega_0(J_y - J_x - J_z))/J_z$, $a_4 = -(\omega_0^2(J_y - J_x))/J_z$,

$$\boldsymbol{A} = \begin{bmatrix} 0 & 0 & a_1 & a_2 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 \\ a_3 & 0 & 0 & 0 & 0 & a_4 \\ 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 \end{bmatrix}, \quad \boldsymbol{B} = \boldsymbol{G} = \begin{bmatrix} \frac{1}{J_x} & 0 & 0 \\ 0 & \frac{1}{J_y} & 0 \\ 0 & 0 & \frac{1}{J_z} \\ 0 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \end{bmatrix}, \quad \boldsymbol{C} = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 & -\omega_0 \\ 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & \omega_0 \\ 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 \end{bmatrix}.$$

The reference model of satellite attitude control system is constructed as following

$$\begin{cases} \dot{\hat{\boldsymbol{x}}}(t) = \boldsymbol{A}\hat{\boldsymbol{x}}(t) + \boldsymbol{B}\boldsymbol{u}(t) + \boldsymbol{G}\hat{\boldsymbol{d}}(t) \\ \hat{\boldsymbol{y}}(t) = \boldsymbol{C}\hat{\boldsymbol{x}}(t) + \boldsymbol{w} \end{cases} \tag{4.}$$

where $\hat{\boldsymbol{x}} = [\dot{\hat{\varphi}} \quad \dot{\hat{\theta}} \quad \dot{\hat{\psi}} \quad \hat{\varphi} \quad \hat{\theta} \quad \hat{\psi}]^{\mathrm{T}}$ is the state of reference model. $\hat{\boldsymbol{d}} = [\hat{d}_x \quad \hat{d}_y \quad \hat{d}_z]^{\mathrm{T}}$ is the estimated value of environmental torque for correction of the reference model. $\hat{\boldsymbol{y}} = [\hat{\omega}_x \quad \hat{\omega}_y \quad \hat{\omega}_z \quad \hat{\varphi} \quad \hat{\theta} \quad \hat{\psi}]^{\mathrm{T}}$ is the output of the reference model.

When some actuator fault of satellite attitude control system occurs, the system model will be depicted as following[12,13)]

$$\begin{cases} \dot{\boldsymbol{x}}(t) = \boldsymbol{A}\boldsymbol{x}(t) + \boldsymbol{B}\boldsymbol{u}(t) + \boldsymbol{G}\boldsymbol{d}(t) + \boldsymbol{H}\boldsymbol{f}_s(t) \\ \boldsymbol{y}(t) = \boldsymbol{C}\boldsymbol{x}(t) + \boldsymbol{w} \end{cases} \tag{5.}$$

where $\boldsymbol{E}$ is an identity matrix; $\boldsymbol{f}_s = [f_{sx} \quad f_{sy} \quad f_{sz}]^{\mathrm{T}}$ is the impact vector of actuator fault; $\boldsymbol{H} = \begin{bmatrix} \boldsymbol{E}_{3\times 3} \\ \boldsymbol{0}_{3\times 3} \end{bmatrix}$ is the interaction matrix of actuator fault.

According to Eqs. (3) and (4), the state error of the reference model can be determined as $\boldsymbol{e}_1(t) = \boldsymbol{x}(t) - \hat{\boldsymbol{x}}(t)$, and we can obtain the following

$$\dot{\boldsymbol{e}}_1(t) = \dot{\boldsymbol{x}}(t) - \dot{\hat{\boldsymbol{x}}}(t) = \boldsymbol{A}\boldsymbol{x}(t) + \boldsymbol{B}\boldsymbol{u}(t) + \boldsymbol{G}\boldsymbol{d}(t) - (\boldsymbol{A}\hat{\boldsymbol{x}}(t) + \boldsymbol{B}\boldsymbol{u}(t) + \boldsymbol{G}\hat{\boldsymbol{d}}(t)) = \boldsymbol{A}\boldsymbol{x}(t) - \boldsymbol{A}\hat{\boldsymbol{x}}(t) + \boldsymbol{G}\boldsymbol{d}(t) - \boldsymbol{G}\hat{\boldsymbol{d}}(t) \tag{6.}$$

If $\hat{x}(0) = x(0)$ and $\hat{d}(t) \to d(t)$, then $\hat{x}(t) \to x(t)$, so as to $e_1(t) \to 0$. When the parameters is consistent between the reference model and the actual system, the system state of the reference model will approximately equal to the actual system state in the case of the estimated environmental torques close to the actual environmental torques. Subsequently, the estimation error of actuator fault is formulated as following

$$e_2(t) = f_s(t) - \hat{f}_s(t) = f(t) - J^{-1}d(t) - (\hat{f}(t) - J^{-1}\hat{d}(t)) \tag{7.}$$

If $\hat{f}(t) \to f(t)$ and $\hat{d}(t) \to d(t)$, then $e_2(t) \to 0$. When the system extended state observed by the ESO is approximate to the total disturbance of actual system, the estimated fault vector of actuator is consistent with the practical situation in the case of the estimated calculation of environmental torques are approximately equal to the actual environmental torques.

2.3. Environmental torques calculation and its effect on fault diagnosis

Analysis the effect of various torque generated by space environment imposing on on-orbit satellite is precondition for the satellite attitude control system designation and system components selection. During the running of satellite in orbit, it mainly suffers the impact of environmental torques such as solar radiation pressure torque, gravity gradient torque, geomagnetic torque and pneumatic torque. Although the environmental torques is generally small, the environment moment will cause bigger effect on the satellite attitude control system with the accumulation over time[11-16)]. According to the structural characteristics and operation mode of the satellite as well as the mechanism of space environmental torques, accurate calculation various environmental torques can be realized. The calculation of environmental torques is not only conducive to the high precision attitude control of on-orbit satellite but also help for fault diagnosis accurately and timely. In addition, the impact of space environment is one of the important factors that induce satellite on-orbit fault occurring, while the calculation of environmental torques can be used to implement fault determination and exclusion caused by the environment. In this paper, on the basis of calculation of environmental torques, the reference model and the extended state observer is employed for fault diagnosis of the satellite attitude control system, so the calculation results directly affect the performance of the diagnosis method. The higher consistency between the calculation results and the practical results means the better diagnosis accuracy.

3. Algorithm implementation of on-orbit fault diagnosis

On the basis of the real-time calculation of environmental torques suffered by on-orbit satellite in, an algorithm of model reference fault diagnosis of satellite attitude control system in orbit is proposed. According to fault residual vector between the reference model and the extended state observer, the faults of the attitude control system sensor and actuator will be detected and isolated.

3.1. Real-time calculation of environmental torques

According to the interaction mechanism of space environment and the characteristics of the satellite structure, the environmental torques suffered by on-orbit satellite can be calculated.

The gravity gradient torque can be calculated from the satellite inertia, attitude Angle and orbit angular velocity[11)], and formulated as following

$$d_g = \begin{bmatrix} d_{gx} \\ d_{gy} \\ d_{gz} \end{bmatrix} = \begin{bmatrix} 3\omega_0^2 (J_z - J_y)\cos^2\theta \sin\varphi \cos\varphi \\ 3\omega_0^2 (J_z - J_x)\cos\varphi \sin\theta \cos\theta \\ 3\omega_0^2 (J_x - J_y)\sin\theta \cos\theta \sin\varphi \end{bmatrix} \tag{8.}$$

where d_{gx}, d_{gy} and d_{gz} are components at three directions of the gravity gradient torque in the system of proprio-coordinate.

The geomagnetic torque can be calculated by satellite magnetic torque and the magnetic induction intensity of the geomagnetic field at the location of satellite[11)], and formulated as following

$$\boldsymbol{d}_m = \begin{bmatrix} d_{mx} \\ d_{my} \\ d_{mz} \end{bmatrix} = \begin{bmatrix} M_{my}B_{mz} - M_{mz}B_{my} \\ M_{mz}B_{mx} - M_{mx}B_{mz} \\ M_{mx}B_{my} - M_{my}B_{mx} \end{bmatrix} \tag{9.}$$

where d_{mx}, d_{my} and d_{mz} are components at three directions of the geomagnetic torque in the system of proprio-coordinate; M_{mx}, M_{my} and M_{mz} are equivalent magnetic moments at three directions of the satellite in the system of proprio-coordinate; B_{mx}, B_{my} and B_{mz} are components at three directions of the geomagnetic field in the system of proprio-coordinate.

The pneumatic torque can be calculated by some parameters of satellite such as the relative velocity, atmospheric density in the orbit and the incident flow surface area[11].

$$\boldsymbol{d}_a = \begin{bmatrix} d_{ax} \\ d_{ay} \\ d_{az} \end{bmatrix} = \begin{bmatrix} \frac{\rho v_R^2}{2} C_d S(C_{py}\boldsymbol{v}_z - C_{pz}\boldsymbol{v}_x) \\ \frac{\rho v_R^2}{2} C_d S(C_{pz}\boldsymbol{v}_x - C_{px}\boldsymbol{v}_z) \\ \frac{\rho v_R^2}{2} C_d S(C_{px}\boldsymbol{v}_y - C_{py}\boldsymbol{v}_x) \end{bmatrix} \tag{10.}$$

where d_{ax}, d_{ay} and d_{az} are components at three directions of the pneumatic torque in the system of proprio-coordinate; ρ is atmospheric density; v_R is atmospheric speed relative to the satellite; C_d is resistance coefficient; S is incident flow surface area; C_{px}, C_{py} and C_{pz} are components at three directions of the distance vector between the satellite centroid and the center of atmospheric pressure; $\boldsymbol{v}_x$, $\boldsymbol{v}_y$ and $\boldsymbol{v}_z$ are unit vectors at three directions of the flow direction of atmosphere in the system of proprio-coordinate.

The solar radiation pressure torque can be calculated by some parameters such as the sunlight pressure, incident angle, reflection and scattering coefficients[11]. If the action area is S_s, the sunlight pressure can be calculated as following

$$\boldsymbol{p}=\boldsymbol{p}_n+\boldsymbol{p}_\tau = p_0 \cos\theta_s \left\{ \frac{2}{3}\rho_s\mu_s + [1+\rho_s(1-\mu_s)]\cos\theta_s \right\} \boldsymbol{n} + p_0(1-\rho_s+\rho_s\mu)\cos\theta_s \sin\theta_s \boldsymbol{\tau} \tag{11.}$$

where p_0 is the light pressure of satellite surface; θ_s is the incident angle of sunlight; ρ_s and μ_s are reflection and scattering coefficients of irradiated surface respectively; $\boldsymbol{n}$ and $\boldsymbol{\tau}$ are normal unit vector and tangent unit vector of irradiated surface respectively. Consequently, the solar radiation pressure torque can be calculated by following

$$\boldsymbol{d}_s = \begin{bmatrix} d_{sx} \\ d_{sy} \\ d_{sz} \end{bmatrix} = \begin{bmatrix} r_{sy}F_{sz} - r_{sz}F_{sy} \\ r_{sz}F_{sx} - r_{sx}F_{sz} \\ r_{sx}F_{sy} - r_{sy}F_{sx} \end{bmatrix} = \begin{bmatrix} (r_{sy}S_{sz} - r_{sz}S_{sy})\,p \\ (r_{sz}S_{sx} - r_{sx}S_{sz})\,p \\ (r_{sx}S_{sy} - r_{sy}S_{sx})\,p \end{bmatrix} \tag{12.}$$

where r_{sx}, r_{sy} and r_{sz} are components at three directions of the distance vector between the satellite centroid and the center of solar radiation pressure; S_{sx}, S_{sy} and S_{sz} are projected area at three plane (YOZ, XOZ and XOY) of irradiated surface S_s respectively.

Totally, the environmental torques suffered by on-orbit satellite can be calculated by following

$$\hat{\boldsymbol{d}}(t) = \boldsymbol{d}_g(t)+\boldsymbol{d}_m(t)+\boldsymbol{d}_a(t)+\boldsymbol{d}_s(t) \tag{13.}$$

3.2. Model reference fault diagnosis based on measurement of environmental torques

According the approach to on-orbit model reference fault diagnosis based on measurement of environmental torques, an on-orbit fault diagnosis algorithm of satellite attitude control system is presented, as shown in algorithm 1.

Algorithm 1 On-orbit model reference fault diagnosis based on measurement of environmental torques

Input: $\boldsymbol{v}$, $\boldsymbol{y}$, $\boldsymbol{u}$
Output: fault residual vector $\boldsymbol{f}_{CF}$, $\boldsymbol{f}_{SF}$ and the corresponding number of fault component
Initialize $\boldsymbol{A}$, $\boldsymbol{B}$, $\boldsymbol{C}$, $\boldsymbol{L}$, $\boldsymbol{G}$, ω_0, $\boldsymbol{\alpha}$, $\boldsymbol{\beta}$
Calculate environmental torques $\hat{\boldsymbol{d}}(t)$ according to Eqs. (8), (9), (10), (12) and (13)
Estimate system disturbance $\hat{f}(t)$ according to the ESO shown in Eq. (6)
Obatin fault vector of actuator $\hat{\boldsymbol{f}}_s(t)$, $\boldsymbol{f}_{SF}(t) = \hat{\boldsymbol{f}}_s(t)$
If $|\boldsymbol{f}_{SFi}(t)| \geq \alpha_i$ then
    Determine an actuator fault correspond to $\boldsymbol{f}_{SFi}(t)$
    Calculate $\dot{\tilde{\boldsymbol{x}}}(t) = \boldsymbol{A}\tilde{\boldsymbol{x}}(t) + \boldsymbol{Bu}(t) + \boldsymbol{G}\hat{\boldsymbol{d}}(t) + \boldsymbol{Hf}_{SF}$ in Eq. (4)
    Obtain output of the reference model $\tilde{\boldsymbol{y}}$, $\boldsymbol{f}_{CF}(t) = \boldsymbol{y}(t) - \tilde{\boldsymbol{y}}(t)$ by Eq. (4)
End if
If $|\boldsymbol{f}_{CFj}(t)| \geq \beta_j$ then
    Determine an sensor fault correspond to $\boldsymbol{f}_{CFj}(t)$ and $\bar{\boldsymbol{y}}(t) = \boldsymbol{y}(t) - \boldsymbol{f}_{CF}(t)$
End if

In algorithm 1, the input parameters include expectation attitude $\boldsymbol{v}$, measurement matrix $\boldsymbol{y}$ and controlled variable $\boldsymbol{u}$. According to system structure characteristics and performance requirements, some parameters such as system matrix $\boldsymbol{A}$, control matrix $\boldsymbol{B}$, calculate matrix $\boldsymbol{C}$, incremental matrix $\boldsymbol{L}$, environmental interaction torque $\boldsymbol{G}$, orbit angular velocity ω_0, sensor fault detection threshold vector $\boldsymbol{\alpha}$ and actuator fault detection threshold vector $\boldsymbol{\beta}$ are initial setup. Subsequently, the ESO and the reference model are respectively constructed to obtain fault residual vector $\boldsymbol{f}_{CF}$, $\boldsymbol{f}_{SF}$ and the number of fault component. In addition, i and j are dimensions of system control vector and output vector respectively with $i = 1, 2, 3$ and $j = 1, 2, \ldots, 6$.

3.3. Implement conditions and algorithm performance analysis

The on-orbit fault diagnosis method for three-axis stabilized satellite attitude control system in which the zero momentum flywheel is executor is presented in this paper. On the basis of the accurate real-time calculation of main environmental torques suffered by satellite, the potential impact on satellite can be corrected according to space environment factors such as the earth's radiation torque, high energy particles, space debris in practical applications. The proposed method has strong dependence on model, and the model parameter should be has high consistency with the actual system parameters in order to rapidly and accurately reflect the actual operation system.
The proposed algorithm is basis on fault residual vector generated by combination of the calculation of environmental torques and the designed mode, and the process is synchronizing with the system operation to achieve real-time diagnosis. Based on the fault residual vector of model correspond to the sensors and actuators in the attitude control system, the fault diagnosis of various components are implemented to ensure the completeness of diagnosis. Under the condition of parameters consistency between the constructed model and the actual system, accurate fault diagnosis will be achieved.

4. Simulation experiment and results analysis

For the performance verification of the presented diagnosis algorithm, some digital simulation experiments are carried out. In the three-axis stabilized satellite attitude control system, three positive installed star sensors are employed for to measure the information of attitude angle; three positive installed gyroscope is employed to measure attitude angular velocity, and three positive installed reaction flywheel provides attitude control torques.
The satellite parameters given in previous reference[17)] are adopted to conduct simulation experiments.

The moment of inertia matrix of satellite is $\mathrm{diag}(80 \quad 90 \quad 70)\,\mathrm{kg}\cdot\mathrm{m}^2$; the maximum output torque of reaction flywheel is $\bar{\mathrm{u}}_{\mathrm{c}} = 2\,\mathrm{N}\cdot\mathrm{m}$; the orbit angular velocity of satellite is $\omega_0 = 0.2^{\circ}/\mathrm{s}$; The initial attitude angle is $[-0.1^{\circ} \quad -0.1^{\circ} \quad -0.1^{\circ}]^{\mathrm{T}}$; the initial attitude angular velocity is $[0.01^{\circ}/\mathrm{s} \quad 0.01^{\circ}/\mathrm{s} \quad 0.01^{\circ}/\mathrm{s}]^{\mathrm{T}}$; the expectation attitude angle is $[-0^{\circ} \quad -0^{\circ} \quad -0^{\circ}]^{\mathrm{T}}$; the initial attitude angular velocity is $[0^{\circ}/\mathrm{s} \quad -0.2^{\circ}/\mathrm{s} \quad 0^{\circ}/\mathrm{s}]^{\mathrm{T}}$; the space disturbance torques are set to $\boldsymbol{d}_x = \mathrm{d}_0(3\cos(\omega_0 t)+1)$, $\boldsymbol{d}_y = \mathrm{d}_0(1.5\sin(\omega_0 t)+3\cos(\omega_0 t))$ and $\boldsymbol{d}_z = \mathrm{d}_0(3\sin(\omega_0 t)+1)$, in which the torque amplitude space environment is set to $\mathrm{d}_0 = 1.5\times10^{-5}\ \mathrm{N}\cdot\mathrm{m}$.

4.1. Simulation experiment of typical fault diagnosis

If no fault occurs, the attitude angle, attitude angular velocity, control torque and fault residual amplitude of star sensor, fault residual amplitude of gyroscope and fault residual amplitude of reaction fly wheel are shown in Fig. 4.1.1.

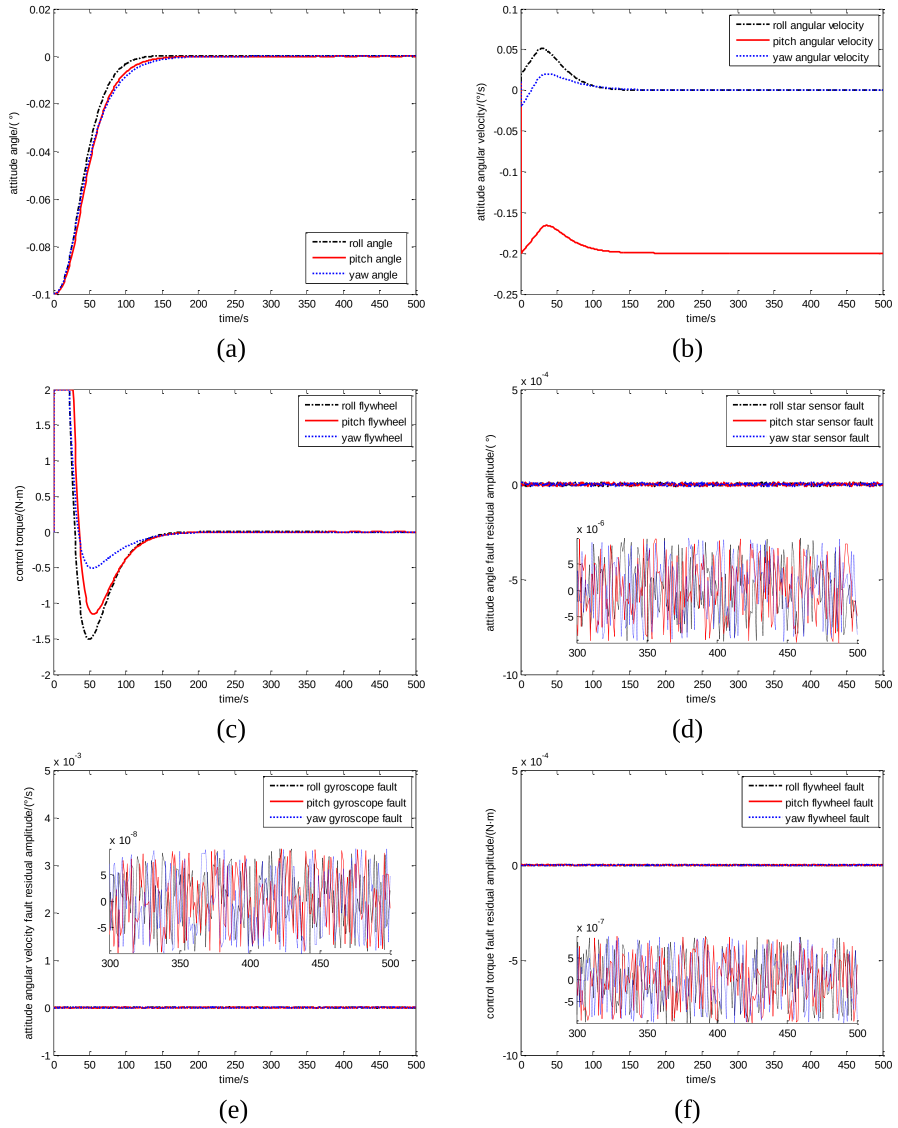

Fig. 4.1.1. Simulation experiment results in case of no fault. (a) satellite attitude angle, (b) satellite attitude angular velocity, (c) satellite control torque, (d) attitude angle fault residual amplitude of star sensor, (e) attitude angular velocity fault residual amplitude of gyroscope and (f) control torque fault residual amplitude of reaction flywheel

If the roll axis star sensor occurs too large random noise fault in 320 seconds (Gaussian white noise with intensity is 0 dBw and variance is 0.0002), the satellite attitude angle, attitude angular velocity, control torque, fault residual amplitude of star sensor, fault residual amplitude of gyroscope and fault residual amplitude of reaction flywheel are shown in Fig. 4.1.2.

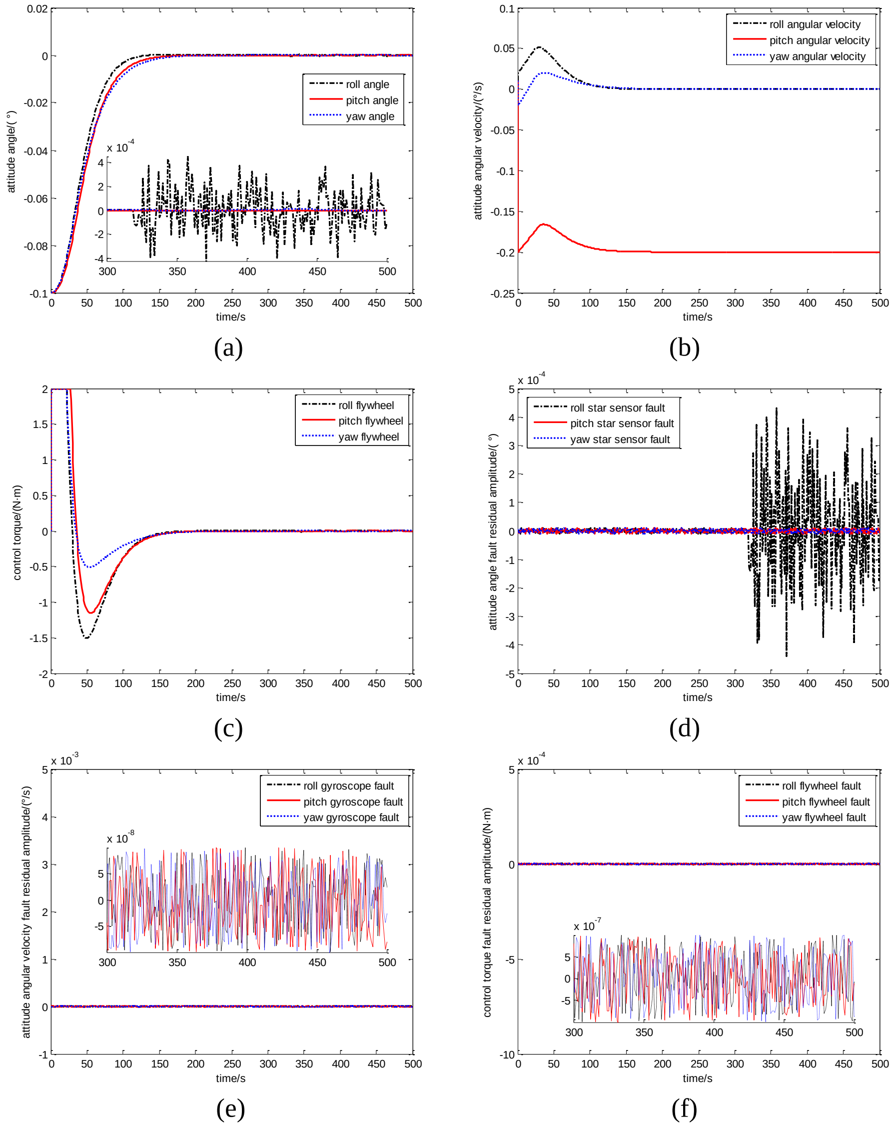

Fig. 4.1.2. Simulation experiment results in case of large random noise fault of the roll axis star sensor. (a) satellite attitude angle, (b) satellite attitude angular velocity, (c) satellite control torque, (d) attitude angle fault residual amplitude of star sensor, (e) attitude angular velocity fault residual amplitude of gyroscope and (f) control torque fault residual amplitude of reaction flywheel

If the pitch axis gyroscope occurs stuck fault in 360 seconds (output value is 0), the satellite attitude angle, attitude angular velocity, control torque, fault residual amplitude of star sensor, fault residual amplitude of gyroscope and fault residual amplitude of reaction flywheel.

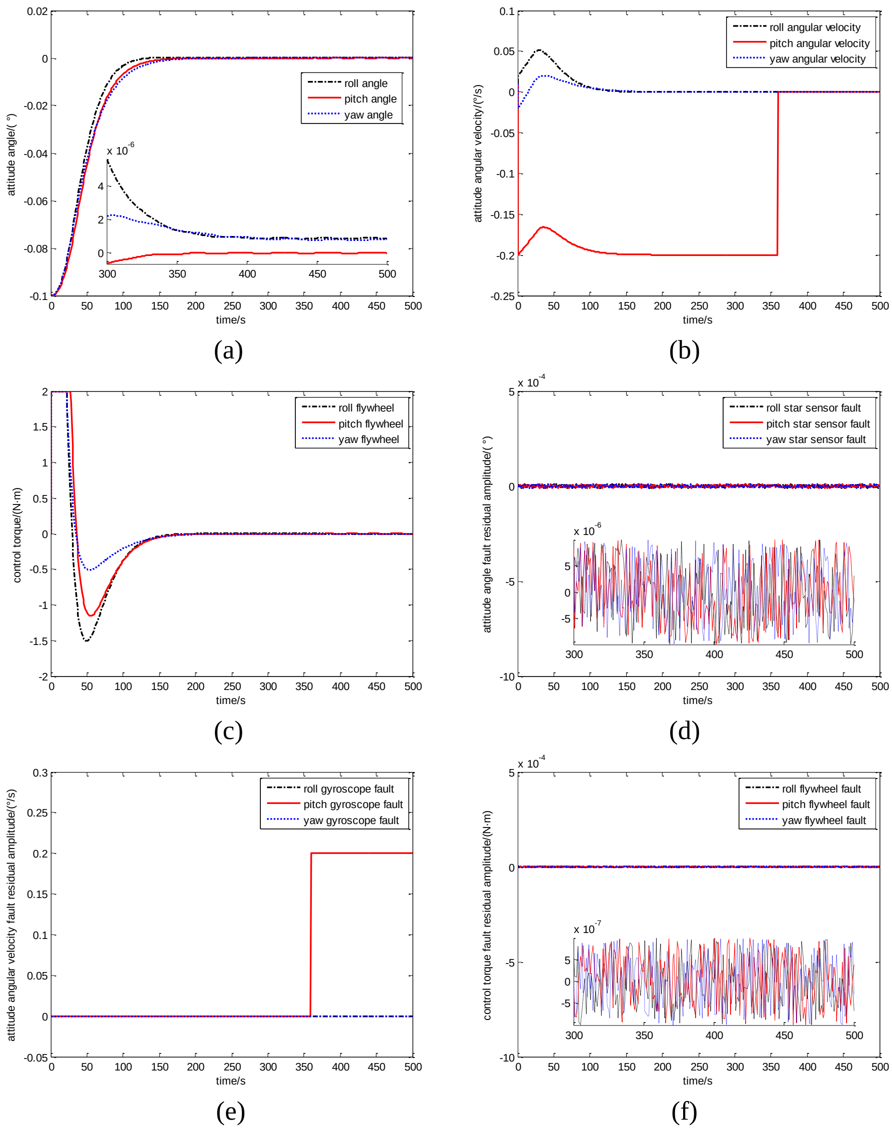

Fig. 4.1.3. Simulation experiment results in case of stuck fault of the pitch axis gyroscope. (a) satellite attitude angle, (b) satellite attitude angular velocity, (c) satellite control torque, (d) attitude angle fault residual amplitude of star sensor, (e) attitude angular velocity fault residual amplitude of gyroscope and (f) control torque fault residual amplitude of reaction flywheel

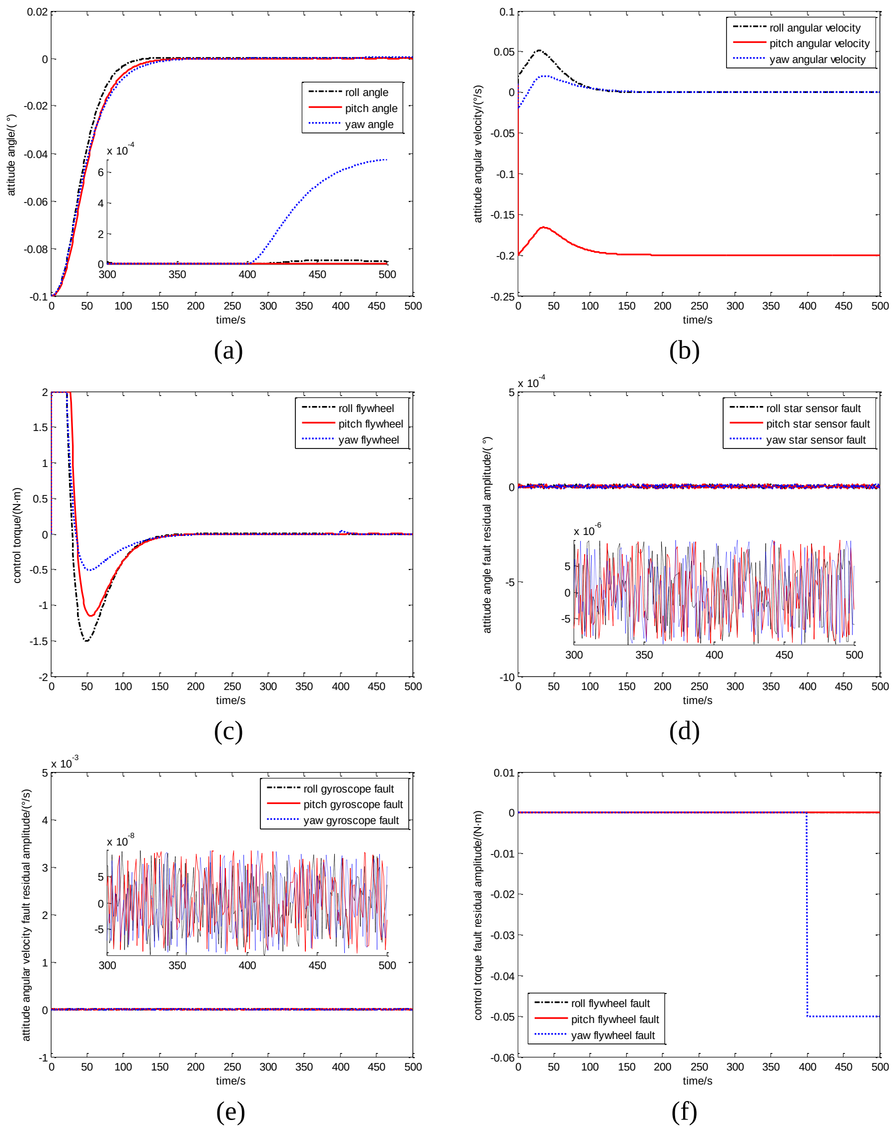

Fig. 4.1.4. Simulation experiment results in case of constant deviation fault of the yaw axis flywheel. (a) satellite attitude angle, (b) satellite attitude angular velocity, (c) satellite control torque, (d) attitude angle fault residual amplitude of star sensor, (e) attitude angular velocity fault residual amplitude of gyroscope and (f) control torque fault residual amplitude of reaction flywheel

If the yaw axis flywheel occurs constant deviation fault in 400 seconds (deviation value is 0.05), the satellite attitude angle, attitude angular velocity, control torque, fault residual amplitude of star sensor, fault residual amplitude of gyroscope and fault residual amplitude of reaction flywheel are shown in Fig. 4.1.4.

4.2. Results analysis and performance evaluation

When no fault of satellite occurs, the fault residual amplitude of star sensor and gyroscope as well as the fault residual amplitude of flywheel is maintained at a small magnitudes level, which indicates that no fault of attitude sensors and actuators occur. When some fault of star sensor, gyroscope and flywheel occurs, the corresponding fault residual amplitude drastically changes. What is more, not only the drastically changed fault residual is consistent with the fault component, but also the time of drastic change is consistent with the time of failure occurrence.

The simulation results show that the proposed approach can achieve accurate real-time fault diagnosis for satellite attitude control sensor and actuator. The proposed approach can realize various typical fault diagnoses for a variety of sensors and actuators and indicates better completeness.

5. Conclusions

In this paper, a model reference fault diagnosis approach for on-orbit satellite attitude control system was proposed based on measurement and calculation of environmental torques. On the basis of precise calculation for space environmental torques suffered by on-orbit satellite, the ESO is used to estimate the system disturbance to achieve to achieve fault detection and separation for attitude control actuator. The calculation of environmental torques is employed to correct the constructed reference model of attitude control system, and then the difference between the model and the actual system output is obtained to detect and isolate the fault of attitude sensor. Several simulation experiments with a variety of typical fault are carried out, and the results show that the proposed approach can effectively realize real-time fault diagnosis for on-orbit satellite attitude control system.

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