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A Multivariable Adaptive Reconfigurable Control Method Applied to the Wing Damaged Aircraft

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Abstract—**Focusing on the abrupt parameter variation of the aircraft dynamic characteristics brought by the wing damage, a multivariable adaptive reconfigurable control method has been studied. First the nonlinear 6DOF model of structural damaged aircraft is presented. The linearization process and the coupling between the longitudinal and lateral dynamics are studied, and the damaged aircraft dynamic model is denoted as a linear form with uncertain variations. And then the Multivariable Model Reference Adaptive Control (M-MRAC) method is introduced to compensate the abrupt variations in the state matrix, control matrix and the constant uncertainty. Finally NASA Generic Transport Model (GTM) is taken as an example and a typical case of left wing tip with 15% damage is considered. Three-channel attitude simulations are presented through comparing with PID control method. The results illustrate that the impact due to the parameter variations are significantly reduced, and the output tracks the desired trajectories rapidly and stably under uncertain damage.**

I. INTRODUCTION

Structural damage, due to structural fatigue, combat injuries or some sudden accidents, will result in the asymmetry in structure, more complicated dynamics, and strong coupling between longitudinal and lateral dynamics. The characteristic of the damaged aircraft has significantly deviated from the original design state of the flight control system. The inappropriate control system may cause terrible accidents. So it's eager to design a stable and resilient control system to compensate the abrupt variations and maintain a desirable flight performance when the aircraft suffers an abrupt injury.

Adaptive control of structural damaged aircraft has become an important subject in the field of flight safety. From 1999 to 2004, NASA launched IFCS research program which applied the neural network technique to the reconfigurable flight control system. A neural network-based direct adaptive controller has applied to an F-15 testbed first [1] [2] and then the test flight of X-36 [3] , and the controller was verified to effect well. In 2006, NASA launched IRAC plan, its main purpose is to improve the stability of the structural damaged aircraft and the possibility of safe landing. This project considered some typical structural damage situations such as wing-tip damage, wing perforation and horizontal or vertical tail damage, and the adaptive control method was adopted to design reconfigurable control law [4] [5]. In 2010, an F-18 TN 853 with active aero-elastic wings was used as the platform for a series of verification on a number of adaptive control systems, which achieved satisfactory results.

In order to adapt to the structure and parameter variations caused by damage, the Multivariable Model Reference Adaptive Control (M-MRAC) method has gradually become a focus of research. In [6], an MRAC design based on the LDS decomposition of high-frequency gain matrix is introduced for the control of aircraft with multiple wing damage. The method bases on two preconditions, one is the interactor matrix is uniform, the other is the leading principal minors of high-frequency gain matrices should be nonzero and their signs are changeless before and after the damage. In [7], the linearization of the nonlinear aircraft model with damage is studied and an MRAC controller is designed. However the controller does not requires maintaining signs of the high-frequency gain matrix unchanged. But only two channels (pitch and yaw) are considered in the paper above, the channel of roll has not take into consideration which is supposed to be affected strongly. In this paper, the coupling between the longitudinal and lateral dynamics is studied at first. Then the controller in views of three channels is designed on the basis of full study on the MRAC, and its effectiveness is verified.

The paper is organized as follows. Section II contains the damaged aircraft model, its linearization process, and the analysis of the coupling dynamic terms. In Section III, a state-feedback M-MRAC controller method, which is based on the LDS decomposition of high-frequency gain matrix, is introduced. In Section IV, a simulation study is presented based on the Generic Transport Model (GTM) [8] both under normal and damaged condition. Results illustrate that the M-MRAC controller has a desirable performance.

II. MODEL OF DAMAGED AIRCRAFT

A. Nonlinear Model of Damaged Aircraft

To the structural damaged aircraft, its aerodynamic characteristics, mass and C.G. (center of gravity) position will change abruptly. And the damaged aircraft model will be quite different from the normal one, so the dynamics equations need to be reconsidered and derived.

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Adopting the infinitesimal method for deduction and combining with Newton's laws of motion, we can derive out the three-axis force and moment equations contain C.G. offset as[8]

$$\begin{bmatrix} F_x \\ F_y \\ F_z \end{bmatrix} = \begin{bmatrix} \frac{d}{dt} & -r & q \\ r & \frac{d}{dt} & -p \\ -q & p & \frac{d}{dt} \end{bmatrix} \begin{bmatrix} u \\ v \\ w \end{bmatrix} + \begin{bmatrix} -q^2-r^2 & pq-\dot{r} & pr+\dot{q} \\ pq+\dot{r} & -p^2-r^2 & qr-\dot{p} \\ pr-\dot{q} & qr+\dot{p} & -p^2-q^2 \end{bmatrix} \begin{bmatrix} \Delta x \\ \Delta y \\ \Delta z \end{bmatrix} \quad (1)$$

$$\begin{aligned} \begin{bmatrix} L \\ M \\ N \end{bmatrix} = & \begin{bmatrix} I_x & I_{xy} & I_{xz} \\ I_{xy} & I_y & I_{yz} \\ I_{xz} & I_{yz} & I_z \end{bmatrix} \begin{bmatrix} \dot{p} \\ \dot{q} \\ \dot{r} \end{bmatrix} + \begin{bmatrix} 0 & -r & q \\ r & 0 & -p \\ -q & p & 0 \end{bmatrix} \begin{bmatrix} I_x & I_{xy} & I_{xz} \\ I_{xy} & I_y & I_{yz} \\ I_{xz} & I_{yz} & I_z \end{bmatrix} \begin{bmatrix} p \\ q \\ r \end{bmatrix} + \\ & m \begin{bmatrix} 0 & \Delta z & -\Delta y \\ -\Delta z & 0 & \Delta x \\ \Delta y & -\Delta x & 0 \end{bmatrix} \begin{bmatrix} 0 & -r & q \\ r & 0 & -p \\ -q & p & 0 \end{bmatrix} \begin{bmatrix} u \\ v \\ w \end{bmatrix} + m \begin{bmatrix} 0 & -\Delta z & \Delta y \\ \Delta z & 0 & -\Delta x \\ -\Delta y & \Delta x & 0 \end{bmatrix} \begin{bmatrix} \dot{u} \\ \dot{v} \\ \dot{w} \end{bmatrix} \end{aligned} \quad (2)$$

where m is the mass of aircraft; $\Delta x, \Delta y, \Delta z$ are the offset of C.G. in the body frame, u, v, w are the body-axis components of the aircraft's speed, p, q, r are the body-axis components of the angular velocity.

From (1) and (2) it can be concluded that when the aircraft model is in the normal condition, the structure is symmetric. The moments of inertia I_{yz}, I_{xy} and the offset $\Delta x, \Delta y, \Delta z$ are zero. Once the body is asymmetric damaged, the mass and aerodynamic force in the body-axis will change uncertainly, the aircraft's structure will be not symmetric any more. A nonzero offset of C.G. will be produced, and the corresponding moments of inertia change to be a nonzero value. So that some modified terms will be added to the equation of linear motion, and the angular acceleration term $\dot{p}, \dot{q}, \dot{r}$ and the nonzero offset of C.G. $\Delta x, \Delta y, \Delta z$ will appear. In the angular motion equation, some modified terms will also be added as I_{yz}, I_{xy} and $\Delta x, \Delta y, \Delta z$ are no longer to be zero, causing the coupling between the longitudinal and lateral dynamics. As a result, the dynamic characteristics will become more complicated.

According to the relationship between the attitude angles and the angular velocities, we can derive out the motion equations

$$\begin{cases} \dot{\phi} = p + \tan\theta(q\sin\phi + r\cos\phi) \\ \dot{\theta} = q\cos\phi - r\sin\phi \\ \dot{\psi} = (q\sin\phi + r\cos\phi)/\cos\theta \end{cases} \quad (3)$$

Where ϕ, θ, ψ represent the Euler roll, pitch and yaw angles. The above equations can describe the motion state of the aircraft at any time.

B. *Linearization*

A nonlinear aircraft system can be described as

$$\begin{aligned} \dot{x}(t) &= f(x(t), u(t)) \\ y(t) &= Cx(t) \end{aligned} \quad (4)$$

where u, y represent the input and output, and

$$\begin{aligned} x(t) &= [V, \alpha, q_b, \theta, \beta, r_b, p_b, \phi, \psi] \\ u(t) &= [\delta_e, \delta_r, \delta_a]^T \end{aligned} \quad (5)$$

A nonlinear aircraft system with damage can be described similarly as

$$\begin{aligned} \dot{x}(t) &= f_d(x(t), u(t)) \\ y(t) &= Cx(t) \end{aligned} \quad (6)$$

Linearization is a common method for nonlinear systems to develop a control design. First, we can get an exactly equilibrium point (x_0, u_0) to the normal system (4). But because of the uncertainty of the structural damage, damaged system (6) can't keep balance in this point. We linearize the system (4) and damaged system (6) at (x_0, u_0) and can obtain the linearized normal aircraft system

$$\begin{aligned} \dot{x}(t) &= A_n \Delta x(t) + B_n \Delta u(t) + f(x_0, u_0) \\ \Delta y(t) &= C\Delta x(t) \end{aligned} \quad (7)$$

and the linearized damaged system

$$\begin{aligned} \dot{x}(t) &= A_d \Delta x(t) + B_d \Delta u(t) + f_d(x_0, u_0) \\ \Delta y(t) &= C\Delta x(t) \end{aligned} \quad (8)$$

where $\Delta x(t) = x(t) - x_0, \Delta u(t) = u(t) - u_0, \Delta y(t) = y(t) - Cx_0$,

$A_n = \left.\frac{\partial f}{\partial x}\right|_{(x,u)=(x_0,u_0)}, B_n = \left.\frac{\partial f}{\partial u}\right|_{(x,u)=(x_0,u_0)}, A_d = \left.\frac{\partial f_d}{\partial x}\right|_{(x,u)=(x_0,u_0)}, B_d = \left.\frac{\partial f_d}{\partial u}\right|_{(x,u)=(x_0,u_0)}$

and $f(x_0, u_0) = 0$, $f_d(x_0, u_0)$ is an unknown constant disturbance.

According to (7) and (8), the structural damaged aircraft model can be described with the following type as

$$\begin{aligned} \dot{x} &= (A + \Delta A)x + (B + \Delta B)u + f \\ y &= Cx \end{aligned} \quad (9)$$

where $A \in R^{n\times n}, B \in R^{n\times m}, C \in R^{m\times n}, f \in R^{n\times 1}$. $\Delta A, \Delta B, f$ are unknown constant matrices represent the uncertain variation after the structural damage which should be zero in normal condition.

Then, we linearize the GTM model in the normal condition and then linearize the model with 15% left wing tip damage and obtain the results as

$$A = \begin{bmatrix} -0.0315 & 0.1945 & -0.1710 & -0.1187 & 0 & 0 & 0 & 0 & 0 \\ -0.4679 & -2.7643 & 0 & 55.3501 & 0 & 0 & 0 & 0.0001 & 0 \\ 0 & 0 & 0 & 1.0000 & 0 & 0 & 0 & 0 & 0 \\ 0 & -0.5943 & 0 & -3.2042 & 0 & 0 & 0 & -0.0022 & -0.0002 \\ 0 & 0 & 0 & 0 & -0.5939 & 0 & 0.2001 & 3.5763 & -56.6397 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1.0020 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1.0000 & 0.0629 \\ 0.0001 & 0.0004 & 0 & 0.0004 & -1.4642 & 0 & 0 & -4.9169 & 0.5786 \\ 0 & 0 & 0 & 0 & 0.4842 & 0 & 0 & -0.4150 & -1.0737 \end{bmatrix}$$

$$A_d=\begin{bmatrix} -0.0166 & 0.1932 & -0.1710 & -0.1165 & -0.0005 & 0 & 0 & -0.0058 & -0.0002 \\ -0.4529 & -2.6051 & 0 & 55.4044 & -0.0123 & 0 & 0 & -0.1330 & 0.0067 \\ 0 & 0 & 0 & 1.0000 & 0 & 0 & 0 & 0 & 0 \\ 0.0173 & -0.5212 & -0.0001 & -3.1517 & -0.0064 & 0 & 0 & -0.2274 & -0.0072 \\ -0.0003 & 0.0035 & 0 & -0.0015 & -0.6004 & 0 & 0.2001 & 3.5774 & -56.6396 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1.0020 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1.0000 & 0.0629 \\ -0.2538 & -1.3654 & 0 & -0.2420 & -1.3816 & 0 & -0.0002 & -4.2340 & 0.5553 \\ -0.0269 & -0.1808 & -0.0004 & -0.0240 & 0.4924 & 0 & -0.0001 & -0.3463 & -1.0477 \end{bmatrix}$$

$$B=\begin{bmatrix} -0.0200 & 0 & 0 \\ -0.2663 & 0 & 0 \\ 0 & 0 & 0 \\ -0.8470 & 0 & 0 \\ 0 & 0.2260 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \\ 0.0001 & 0.2166 & 0.7695 \\ 0 & -0.5549 & 0.0638 \end{bmatrix} \quad B_d=\begin{bmatrix} -0.0201 & -0.0005 & 0.0019 \\ -0.2667 & -0.0002 & 0.0343 \\ 0 & 0 & 0 \\ -0.8473 & -0.0003 & 0.0232 \\ 0 & 0.2263 & -0.0001 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \\ 0.0044 & 0.2179 & 0.6176 \\ 0.0001 & -0.5562 & 0.0514 \end{bmatrix}$$

From these linearization results, we make the define

$$A=\begin{bmatrix} A_{4\times4} & A_{4\times5} \\ A_{5\times4} & A_{5\times5} \end{bmatrix}, B=\begin{bmatrix} B_{4\times1} & B_{4\times2} \\ B_{5\times1} & B_{5\times2} \end{bmatrix} \tag{10}$$

In the normal condition, we have

$$A_{4\times5}\approx 0, A_{5\times4}\approx 0, B_{4\times2}\approx 0, B_{5\times1}\approx 0 \tag{11}$$

where the longitudinal and lateral dynamics are decoupled.

Once the aircraft is structural damaged, these items in (11) are not zero any more. It means that the longitudinal and lateral dynamics are coupled and dynamic characteristics will be more complicated as a result. It is necessary to design a new control law to guarantee the stability for the structural damaged aircraft.

III. Multivariable MRAC Control Law Design

The close-loop overall structure of M-MRAC control system is shown in Fig. 1.

The whole system contains the reference model, the parameter calculator, the adaptive servo and the main controller. The control objective is to make the output $y(t)$ track a specified given reference signal:

$$y_m(t)=W_m(s)[r](t)\in R^m \tag{12}$$

Then each part of the system will be introduced next.

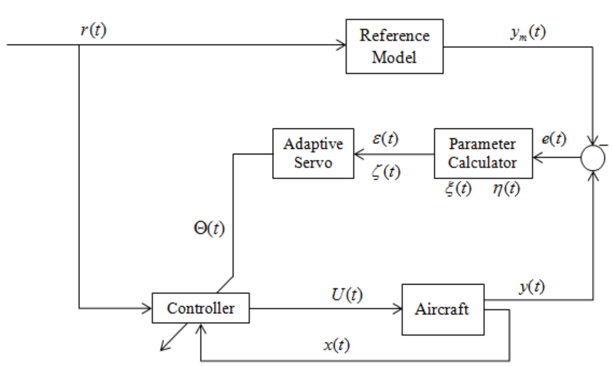

Figure 1. Overall structure of M-MRAC control system

A. Controller

To compensate the constant uncertainty in (9), we choose the state feedback controller structure as

$$U(t)=K_1^T(t)x(t)+K_2(t)r(t)+K_3(t) \tag{13}$$

where K_1, K_2 are the estimates of the nominal parameter matrices K_1^*, K_2^* which satisfy the matching equations

$$C(sI-A-BK_1^{*T})^{-1}BK_2^*=W_m(s), K_2^{*-1}=K_p \tag{14}$$

K_3 is selected by using Laplace final-value theorem and is used to compensate the disturbance.

We define a parameter matrix while designing the control law

$$\Theta(t)=[K_1^T(t),K_2(t),K_3(t)]^T \tag{15}$$

B. Adaptive servo

We choose the adaptive laws as

$$\begin{cases} \dot{\lambda}_i(t)=-\dfrac{\Gamma_{\lambda i}\varepsilon_i(t)\eta_i(t)}{m^2(t)} & \Gamma_{\lambda i}=\Gamma_{\lambda i}^T>0, i=2,3,\cdots,m \\ \dot{\Theta}^T(t)=-\dfrac{D_s\varepsilon(t)\zeta^T(t)}{m^2(t)} & \\ \dot{\rho}(t)=-\dfrac{\Gamma\varepsilon(t)\xi^T(t)}{m^2(t)} & \Gamma=\Gamma^T>0 \end{cases} \tag{16}$$

where a series of parameters are obtained through the parameter calculator unit.

C. Parameter calculator

This unit is mainly designed to process the error signal and to obtain the corresponding parameters for the adaptive servo.

First we should present two important parameter matrices, i.e. the interactor matrix $\xi_m(s)$ and the high-frequency gain matrix Kp, and we have

$$\lim_{s\to\infty}\xi_m(s)G(s)=Kp \tag{17}$$

where $G(s)$ is the transfer function matrix, $\xi_m(s)$ is a stable matrix and should be reversible.

Then, to design the M-MRAC controller, two key conditions must be satisfied. First, the interactor matrix $\xi_m(s)$ for each $G(s)$ could be the same both before and after the damage. Second, all leading principal minors of the high-frequency matrix Kp are finite and nonsingular, and their signs do not change when the damage occurs.

Other specific equations applied in this unit are listed as follows:

$$m^2(t)=1+\zeta^T(t)\zeta(t)+\xi^T(t)\xi(t)+\sum_{i=2}^{m}\eta_i^T(t)\eta_i(t) \tag{18}$$

$$\xi(t)=\Theta^T(t)\zeta(t)-h(s)[\Theta^T w](t) \tag{19}$$

$$\zeta(t)=h(s)[w](t) \tag{20}$$

$$w(t)=[x^T(t),r^T(t),1]^T \tag{21}$$

$$\varepsilon(t)=\bar{e}(t)+[0,\lambda_2^T\eta_2(t),\lambda_3^T\eta_3(t),\cdots,\lambda_m^T\eta_m(t)]^T+\rho(t)\xi(t) \tag{22}$$

$$\eta_i(t) = [\bar{e}_1(t), \ldots, \bar{e}_{i-1}(t)]^T \in R^{i-1}, i = 2, \ldots, m \quad (23)$$

$$\bar{e}(t) = \xi_m(s)h(s)[e](t) = [\bar{e}_1(t), \cdots, \bar{e}_m(t)]^T \quad (24)$$

$$e(t) = y(t) - y_m(t) \quad (25)$$

$$h(s) = \frac{1}{f(s)} \quad (26)$$

where $f(s)$ is a stable and monic polynominal of degree equals to the degree of $\xi_m(s)$. D_s is obtained through the LDS decomposition of the high-frequency gain matrix according to [9].

The stability properties of the control law (13) and adaptive law (16) have been verified in [10]. Next, we will apply the controller to the GTM model and present the simulation results.

IV. System Simulation

The linearized model is described as (9)

$$\begin{cases} \dot{x} = (A + \Delta A)x + (B + \Delta B)u + f \\ x = [V, \alpha, q, \theta, \beta, r, p, \phi, \psi]^T \\ u = [\delta_e, \delta_r, \delta_a]^T \\ y = C_i x = [\theta, \psi, \phi]^T \end{cases} \quad (27)$$

Single wing-tip broken is a typical asymmetric structural damage condition. In this paper, we choose the damage case with the loss of outboard left wing tip and the damage rate is taken as 15%. We can obtain the uncertain constant variation $f_1 = [0.33 \quad 0.65 \quad 0.43 \quad 0 \quad -0.01 \quad -0.66 \quad -1.2 \quad 0 \quad 0]^T$ through calculation. It can be verified that both in normal and damaged conditions, the transfer function matrices are strict proper and have full rank, and all the zeros have negative real-part. The interactor matrix is selected as

$$\zeta_m(s) = \begin{bmatrix} (s+1)^2 & 0 & 0 \\ 0 & (s+1)^2 & 0 \\ 0 & 0 & (s+1)^2 \end{bmatrix} \quad (28)$$

and is changeless after damage.

Then we can obtain the high-frequency gain matrix for the normal case and the damaged case

$$K_{p1}(s) = \begin{bmatrix} -0.8470 & 0 & 0 \\ 0 & -0.5560 & 0.0639 \\ 0.0001 & 0.1817 & 0.7735 \end{bmatrix} \quad (29)$$

$$K_{p2}(s) = \begin{bmatrix} -0.8473 & -0.0003 & 0.0232 \\ 0.0001 & -0.5573 & 0.0515 \\ 0.0044 & 0.1829 & 0.6208 \end{bmatrix} \quad (30)$$

obviously they are finite and nonsingular.

The signs of first leading principal minor of them are

$$\begin{aligned} sign(\Delta_{1Kp1}) &= sign(\Delta_{1Kp2}) = -1 \\ sign(\Delta_{2Kp1}) &= sign(\Delta_{2Kp2}) = 1 \\ sign(\Delta_{3Kp1}) &= sign(\Delta_{3Kp2}) = 1 \end{aligned} \quad (31)$$

there is no sign change. So that the two key conditions mentioned in III are satisfied.

For the control law (13) and adaptive law (16), we choose $f(s) = (s+8)^2$, $\Gamma_{\lambda 2} = 10$, $\Gamma_{\lambda 3} = [2,1]$, $\Gamma = diag\{10,10,10\}$, $D_s = diag\{-100, -50, 40\}$.

To make a reasonable flying trajectory, the reference input is selected as $r(t) = [5^\circ, 4^\circ, 3^\circ]^T$ which generates the reference pitch, yaw and roll angles of 5, 4and 3 deg. Damage occurs in 200 seconds. The simulation results are shown in Fig. 2-7.

From Fig. 2 and Fig. 3 we can see that the three-channel attitude angles approach to the reference input after a small wave, and the tracking errors are converge to zero. When the damage occurs at 200s, the errors converge to zero again after a small-scope fluctuation within a short period of time. The responses of each rudder bias are given in Fig. 4.

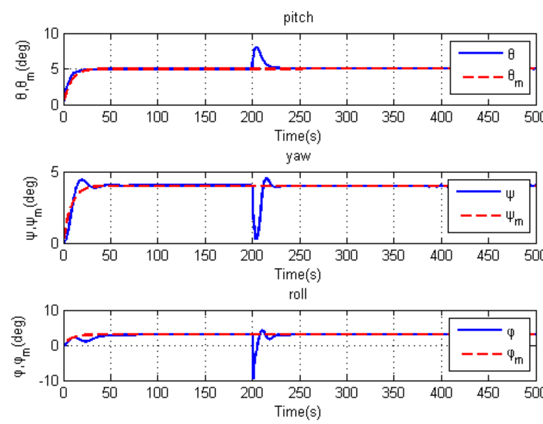

Figure 2. Response curves of pitch, yaw and roll under M-MRAC

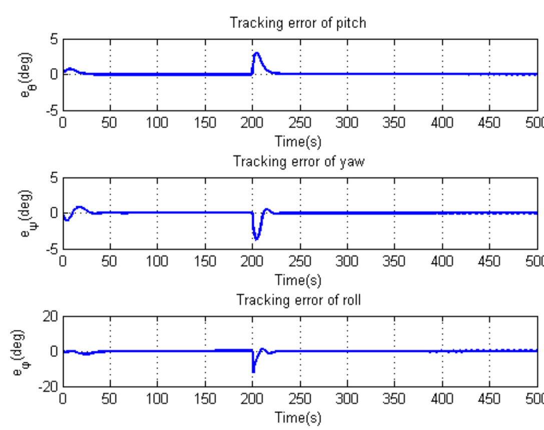

Figure 3. Tracking errors of pitch, yaw and roll

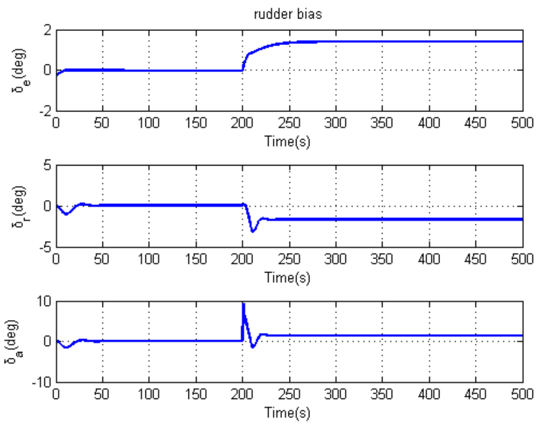

Figure 4. Response curves of elevator, rudder and aileron

Fig. 5-7 depict the adaption of parameter matrices $\Theta(t)$, $\lambda(t)$, $\rho(t)$. The parameters achieve a steady-state value after a short period of adaptive adjustment at the beginning. And each parameter reaches a new steady-state value after a small wave when damage occurs, which ensures tracking precision.

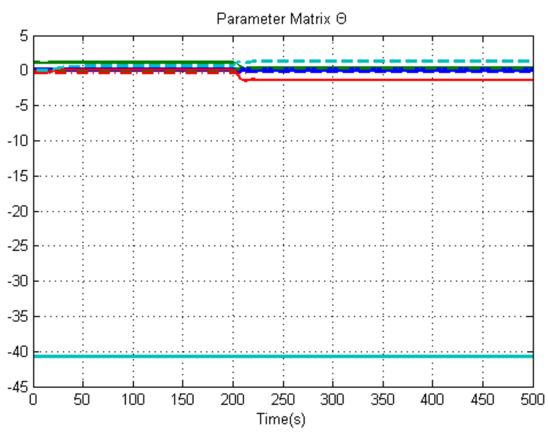

Figure 5. Adaptation of $\Theta(t)$

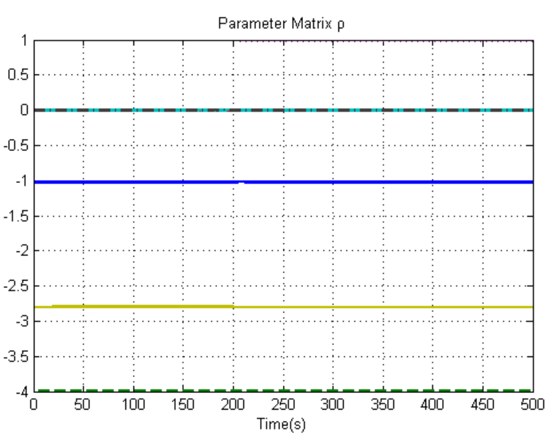

Figure 6. Adaptation of $\rho(t)$

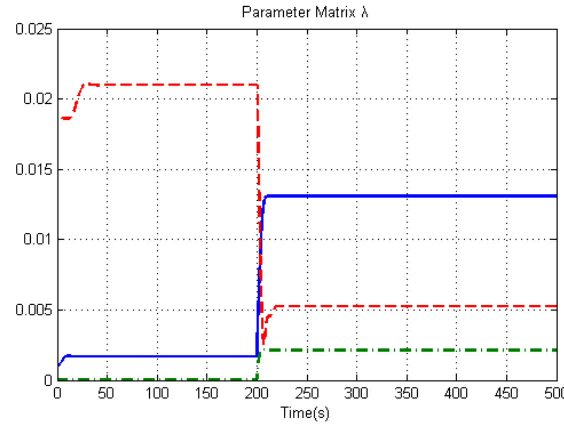

Figure 7. Adaptation of $\lambda(t)$

Then, we use the classical PID controller for the same model with the same reference input, the simulation results are shown in Fig. 8-9.

As it can be seen, although the attitude angles can track the reference input successfully under normal conditions with the PID method, an obvious steady-state error appear after the damage. It means that the PID controller can't guarantee the normal attitude control of the damaged aircraft any more.

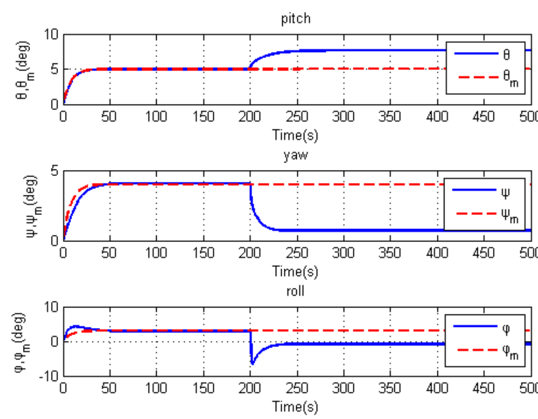

Figure 8. Response curves of pitch, yaw and roll under PID

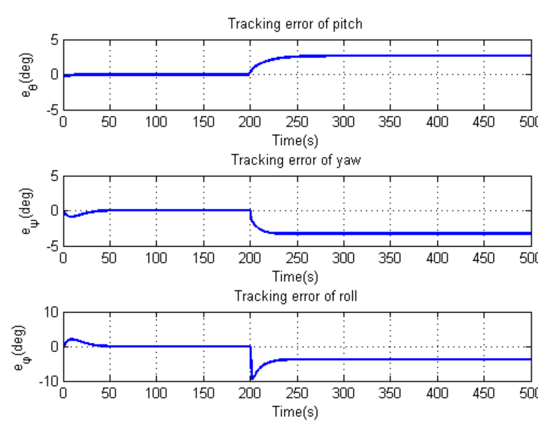

Figure 9. Tracking errors of pitch, yaw and roll under PID

Comparing the two methods according to all the simulation results above, we can see that the M-MRAC method can make the attitude angles track the reference input rapidly within a small range of tracking error when the damage occurs, while the PID method can't guarantee the normal control of the damaged aircraft's attitude. So a conclusion could be made that the designed controller has a more desirable tracking performance when the aircraft is structural damaged.

Then directly apply the M-MRAC controller on the original nonlinear aircraft model with the simulation conditions as the height of 304 m, the speed of 48.87 m/s and the attack angle of 3.5965 deg. The simulation results are shown in Fig. 10-11. It can be seen that the designed controller applied to the original nonlinear model still has a good tracking performance with desirable tracking precision.

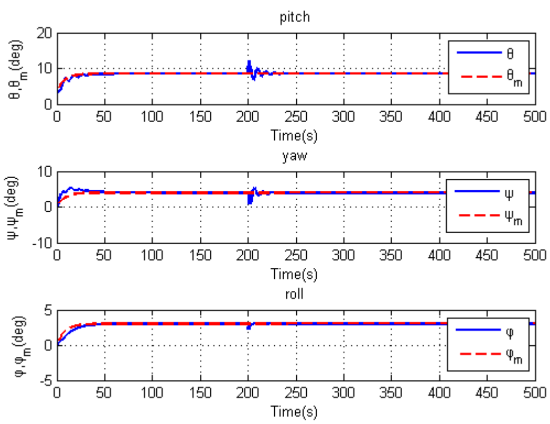

Figure 10. Response curves for the original nonlinear model

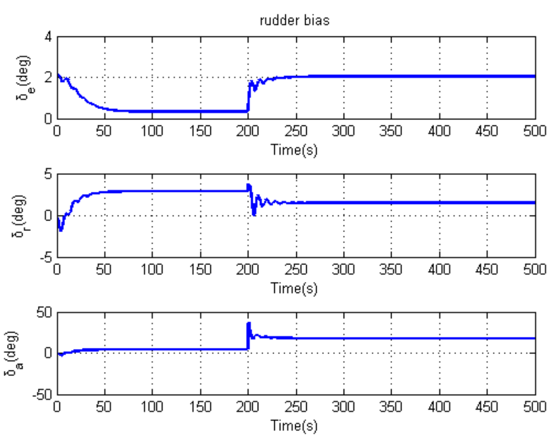

Figure 11. Response curves of elevator, rudder and aileron

V. CONCLUSION

In this paper, the M-MRAC method is applied to the reconfigurable control of the structural damaged aircraft. A three-channel attitude controller is designed for a wing damaged aircraft. By comparison with the PID controller, it is verified that the M-MRAC controller can compensate the abrupt variation caused by the structural damage, and the system has a good tracking accuracy and response speed. Further research will focus on the application of nonlinear systems and the combination of the M-MRAC method and the Active Disturbance Rejection Control (ADRC) method.

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