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Adaptive Non-Singular Terminal Sliding Mode Control for Structural Damaged Aircraft

JIA Peng[1], ZHANG Jing[2], YANG Lingyu[2]

1. School of Automation Science and Electrical Engineering, Beihang University, Beijing, 100191, China.
E-mail: 274410481@qq.com

2. School of Automation Science and Electrical Engineering, Beihang University, Beijing, 100191, China.
E-mail: jijizhj1982@163.com

Abstract:.Structural damage brings strong coupling effects and increases the difficulty of flight control. Aiming at this problem, a nonlinear model of aircraft is introduced firstly, and then an adaptive non-singular terminal sliding mode control method is presented. The approach can ensure the system to reach the sliding surface and converge to equilibrium point in finite time from any initial state without the singularity. In addition, the proposed adaptive non-singular terminal sliding mode controller can real-time estimate the uncertainties and external disturbances. The adaptive law is derived in the Lyapunov framework, which can guarantee the asymptotical stability of the closed loop system. Finally, simulations with Generic Transport Model (GTM) are conducted to demonstrate the effectiveness and advantages of the proposed approach.

Key Words: Structural damage, Adaptive non-singular terminal sliding mode, GTM

1 Introduction

Structural damage will cause a sudden variation of aircraft's quality, center of gravity and aerodynamic characteristics, as well as the destruction of longitudinal symmetry, thus bringing strong coupling effects, which greatly increases the difficulty of control and affects flight security seriously. It is of great significant to improve the robustness of the controller, which can keep the basic control capability of an aircraft and reduce the accident rate when structural damage occurred.

Extensive studies on control of the damaged aircraft have been conducted in the past years. In reference [1], a hybrid adaptive method is applied to solve the damage problem, but it needs the large learning parameter of neural network, which leads to saturation and affect the stability of the system. In reference [2], an adaptive algorithm combined dynamic inverse with neural network is proposed to solve the problem, but stability and control performance can't be ensured at the same time. In reference [3], a multivariable adaptive reconfigurable control method is applied to the damaged aircraft, which can compensate and stable attitude control of disturbance in the damaged moment, but its parameter adjustment is complex. And in reference [4], an active disturbance rejection control method for structural damaged aircraft is presented, but the response time is relatively long.

In this paper, an adaptive non-singular terminal sliding mode control is presented for the problem of structural damage with partial loss wing in aircraft, which can reach the sliding surface and converge to equilibrium point in a finite time from any initial state without the singularity. Meanwhile, the uncertainties and external disturbances can be estimated by the proposed adaptive non-singular terminal sliding mode controller. Finally, simulations with GTM are conducted to demonstrate the effectiveness and advantages of the proposed control approaches.

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2 Model Of Damaged Aircraft

2.1 Nonlinear Model of Damage Aircraft

Structural damage will cause a sudden change of aircraft's center of gravity and aerodynamic characteristics, as well as the destruction of symmetry. So the dynamics equations are necessary to be reconsidered and derived.

Assume that the earth is flat and stationary in the inertial space, and the aircraft is a rigid body. Then the dynamic equations with a fixed body axis can be derived by using the Newton's second law [5].

$$\begin{cases} F_x = m\left[\dot{u} - rv + qw - \left(q^2 + r^2\right)\Delta x + \left(pq - \dot{r}\right)\Delta y + \left(pr + \dot{q}\right)\Delta z\right] \\ F_y = m\left[\dot{v} + ru - pw + \left(pq + \dot{r}\right)\Delta x - \left(p^2 + q^2\right)\Delta y + \left(qr - \dot{p}\right)\Delta z\right] \\ F_z = m\left[\dot{w} + pv - qu + \left(pr - \dot{q}\right)\Delta x + \left(qr + \dot{p}\right)\Delta y - \left(p^2 + q^2\right)\Delta z\right] \end{cases} \tag{1}$$

$$\begin{cases} L = -I_{xy}\dot{q} + I_x\dot{p} - I_{xz}\dot{r} + I_{xy}pr - I_{xz}pq + \left(I_z - I_y\right)qr + m\left(\dot{w} - qu + pv\right)\Delta y + m\left(pw - \dot{v} - ru\right)\Delta z + I_{yz}\left(r^2 - q^2\right) \\ M = -I_{xy}\dot{p} + I_y\dot{q} - I_{yz}\dot{r} + I_{yz}pq - I_{xy}qr + \left(I_x - I_z\right)pr + m\left(qu - pv - \dot{w}\right)\Delta x + m\left(\dot{u} - rv - qw\right)\Delta z + I_{xz}\left(p^2 - r^2\right) \\ N = -I_{xz}\dot{p} + I_{yz}\dot{q} + I_z\dot{r} - I_{yz}pr + I_{xz}qr + \left(I_y - I_x\right)pq + m\left(\dot{v} + ru - pw\right)\Delta x + m\left(rv - \dot{u} - qw\right)\Delta y + I_{xy}\left(q^2 - p^2\right) \end{cases} \tag{2}$$

where m is the mass of aircraft ; $\left[F_x \;\; F_y \;\; F_z\right]^T$ are the body-axis components of the aircraft's combined force;

$[L \quad M \quad N]^T$ are the body-axis components of the aircraft's moments; $[u \quad v \quad w]^T$ are the body-axis components of the aircraft's speed; $[p \quad q \quad r]^T$ are the body-axis components of the aircrafts angular velocity; $[\Delta x \quad \Delta y \quad \Delta z]^T$ are the offset of in the body frame.; I_x 、I_y 、I_z are moments of inertia about aircraft around the body axis; I_{xy} 、I_{xz} 、I_{yz} are products of inertia.

As seen from the equations (1) and (2), the structural damage destroys the symmetry of the aircraft body, causing the offset of and additional moments of inertia, which will cause uncertain acceleration and moments and lead to the longitudinal and lateral couplings. All those will make the dynamic characteristic more complex.

2.2 Analysis of Aircraft Characteristics

The aerodynamic coefficients of GTM got from the reference[6] are shown in Table 1 and 2. Table 1 shows the aerodynamic coefficients of undamaged aircraft and Table 2 shows the aerodynamic coefficients of 33% wing damaged aircraft.

Table 1: Aerodynamic coefficients of undamaged aircraft

	C_X	C_Z	C_m	C_Y	C_l	C_n
0	-0.0284	-0.022	0.1556	0	0	0
α	0.0011	-0.0864	-0.0295	0	0	0
α^2	0.0015	0.0007	0.0011	0	0	0
q	-0.1929	-22.298	-44.889	0	0	0
q^2	523.073	1085.46	-19.726	0	0	0
δ_e	-0.0002	-0.0075	-0.0295	0	0	0
δ_e^2	0	-0.0001	-0.0003	0	0	0
β	0	0	0	-0.0176	-0.0024	0.0037
β^2	0	-0.0006	-0.0007	0	0	0
p	0	0	0	-0.0272	-0.3609	-0.0023
p^2	1.140	-0.0779	-0.4491	0	0	0
r	0	0	0	0.8274	0.0717	-0.3522
r^2	0.8199	-0.1518	-2.7175	0	0	0
δ_a	0	0	0	0.0003	0.0011	0
δ_a^2	0	0	0	0	0	0
δ_r	0	0	0	0.0067	0.0006	-0.0035
δ_r^2	0	0	0.0001	0	0	0

Table 2: Aerodynamic coefficients of 33% wing damaged aircraft

	C_X	C_Z	C_m	C_Y	C_l	C_n
0	-0.0284	-0.0125	0.1659	-0.0006	-0.0042	0.0001
α	0.0009	-0.0747	-0.0221	-0.0005	-0.0043	-0.0002
α^2	0.0013	0.0006	0.0009	0.0001	0	-0.0001
q	-0.2037	-20.859	-44.357	-0.0388	-0.3222	-0.016
q^2	515.058	-1085.4	-19.943	1.1059	-0.0725	-0.9065
δ_e	-0.0002	-0.0075	-0.0295	0	0	0
δ_e^2	0	-0.0001	-0.0003	0	0	0
β	0	-0.0009	-0.0006	-0.0176	-0.0024	0.0037
β^2	0	-0.0006	-0.0007	0	0	0
p	0.0066	-0.4644	-0.2741	-0.0251	-0.2433	-0.0026
p^2	0.7463	-0.0736	-0.3935	0.2437	-0.0004	-0.1947
r	0.0011	0.0245	-0.0156	0.8254	0.0669	-0.352
r^2	0.8297	-0.1373	-2.6944	0.0009	-0.0149	-0.0016
δ_a	0	0.0023	0.0019	0.0002	0.0006	0
δ_a^2	0	0	0	0	0	0
δ_r	0	0	0	0.0067	0.0006	-0.0035
δ_r^2	0	0	0.0001	0	0	0

The reference combine the result of CFD calculation with the wind tunnel measurements under the condition of attack angel is 5°.

We can find that the undamaged GTM model is similar to traditional aircraft. Both Longitudinal and transverse motion parameters have little effects on Lateral aerodynamic coefficients and longitudinal aerodynamic coefficients. The aircraft longitudinal motion is decoupled cross. From the coefficients of 33% wing tip damage, we can find that the aircraft longitudinal motion transverse coupling. The lift and drag generated by the wing is asymmetric, which will lead to additional roll and yaw moments.

Table 3: Variations of GTM's characteristic roots

category	mode	undamaged	33%damage
longitudinal	Long period	-0.0198 + 0.0270i	-0.0101 + 0.0342i
		-0.0198 - 0.0270i	-0.0101 - 0.0342i
	short period	-2.9803 + 5.7368i	-2.8767 + 5.3720i
		-2.9803 - 5.7368i	-2.8767 - 5.3720i
lateral	Roll damping	-5.1593	-3.8289
	spiral	-0.0012	-0.0006
	Holland roll	-0.7120 + 5.8202i	-0.7411 + 5.8303i
		-0.7120 - 5.8202i	-0.7411 - 5.8303i

As can be seen in Table 3, when the GTM is undamaged, all the modes of the longitudinal and lateral are stable, including the long period mode, the short period mode, the roll damping mode, the Holland rolling mode and the spiral mode. When the aircraft is 33% wing damaged, the entire characteristic roots shift right except the Holland roll mode, especially in the roll damping mode. So the reconfigurable control law is certainly demanded to ensure the aircraft performance.

3 Design Of Adaptive NTSM Control Law

The problem of sudden variations in aircraft's quality, center of gravity and aerodynamic characteristics, as well as the coupling between longitudinal and lateral are certainly caused by the wing damage. To solve these problems, an adaptive non-singular terminal sliding mode control is proposed, which can ensure the system to reach the sliding surface and converge to equilibrium point in finite time from any initial state without the singularity. In addition, adaptive control strategies can well update both the information of faults and disturbances. All those can improve the robustness and the rapidity of the control law so that the aircraft is better to be controlled.

The whole system contains adaptive law, the non-singular terminal sliding mode(NTSM) controller, control allocation, and the GTM. Adaptive law is designed to change the control law according to the extent of the damage and interference estimation, which can make the control more effective. The NTSM controller is designed to get the moments by inputting the attitude angle. And control allocation is designed to transform the moments into the deflections of control surface. The whole scheme is as follows:

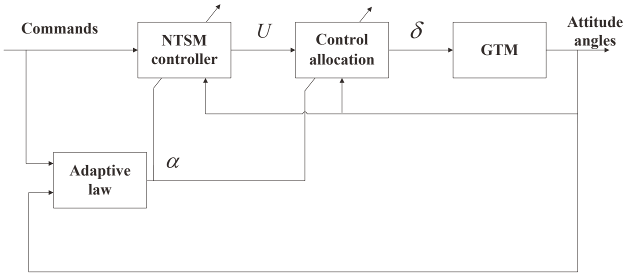

Fig. 1: The Control loop of damaged aircraft

3.1 NTSM Controller Design

When the wing tip is damaged, the center of gravity will change suddenly. The equations of angular motion can be described as follows:

$$\dot{\boldsymbol{W}} = \boldsymbol{I}^{-1}\boldsymbol{U} - \boldsymbol{I}^{-1}\left(\Delta_G\boldsymbol{\Omega}\right)\boldsymbol{V} - \boldsymbol{I}^{-1}\Delta_G\dot{\boldsymbol{V}} - \boldsymbol{I}^{-1}\left(\boldsymbol{\Omega}\boldsymbol{I}\right)\boldsymbol{W} \quad (3)$$

Where $\boldsymbol{W}=[p \quad q \quad r]^T$ are the body-axis components of the aircrafts angular velocity, and $\boldsymbol{X}=[\phi \quad \theta \quad \varphi]^T$ are the Euler angles, $\boldsymbol{U}=[L \quad M \quad N]^T$ are the body-axis components of the aircraft's moments. $\boldsymbol{W}$ satisfies:

$$\boldsymbol{W} = \boldsymbol{T_M} \cdot \dot{\boldsymbol{X}} \quad (4)$$

Where

$$\boldsymbol{T_M}=\begin{bmatrix} 1 & 0 & -sin\theta \\ 0 & cosf & cos\theta sinf \\ 0 & -sinf & cos\theta cosf \end{bmatrix} \boldsymbol{\Omega}=\begin{bmatrix} 0 & -r & q \\ r & 0 & -p \\ -q & p & 0 \end{bmatrix}$$

$$\boldsymbol{I}=\begin{bmatrix} I_x & -I_{xy} & -I_{xz} \\ -I_{xy} & I_y & -I_{yz} \\ -I_{xz} & -I_{yz} & I_z \end{bmatrix} \boldsymbol{V}=\begin{bmatrix} 0 & -w & v \\ w & 0 & -u \\ -v & u & 0 \end{bmatrix}$$

$$\Delta_G=\begin{bmatrix} 0 & -m\Delta z & m\Delta y \\ m\Delta z & 0 & -m\Delta x \\ -m\Delta y & m\Delta x & 0 \end{bmatrix}$$

The derivative of equation (4) is:

$$\dot{\boldsymbol{W}} = \boldsymbol{T}_{\dot{M}}\dot{\boldsymbol{X}} + \boldsymbol{T_M}\ddot{\boldsymbol{X}} \quad (5)$$

Taking (5) into (3)

$$\begin{aligned}\ddot{\boldsymbol{X}} &= \left(\boldsymbol{IT}_M\right)^{-1}\boldsymbol{U} - \left(\boldsymbol{IT}_M\right)^{-1}\left(\boldsymbol{IT}_{\dot{M}} + \boldsymbol{\Omega IT}_M\right)\dot{\boldsymbol{X}} \\ &\quad - \left(\boldsymbol{IT}_M\right)^{-1}\left(\Delta_G\boldsymbol{\Omega}\right)V - \left(\boldsymbol{IT}_M\right)^{-1}\Delta_G\dot{\boldsymbol{V}}\end{aligned} \quad (6)$$

Let

$$\boldsymbol{G} = \left(\boldsymbol{IT_M}\right)^{-1} \quad (7)$$

$$\begin{aligned}\boldsymbol{F}\left(\dot{\boldsymbol{X}},\Delta_G\right) &= -\left(\boldsymbol{IT}_M\right)^{-1}\left(\boldsymbol{IT}_{\dot{M}} + \boldsymbol{\Omega IT}_M\right)\dot{X} \\ &\quad - \left(\boldsymbol{IT}_M\right)^{-1}\left(\Delta_G\boldsymbol{\Omega}\right)V - \left(\boldsymbol{IT}_M\right)^{-1}\Delta_G\dot{\boldsymbol{V}}\end{aligned} \quad (8)$$

Equation (6) can be simplified as:

$$\ddot{\boldsymbol{X}} = \boldsymbol{GU} + \boldsymbol{F}\left(\dot{\boldsymbol{X}},\Delta_G\right) \quad (9)$$

Define the tracking error $\boldsymbol{e}$ as

$$\boldsymbol{e} = \boldsymbol{X} - \boldsymbol{X}_C \quad (10)$$

Then the nonlinear system can be written as

$$\ddot{\boldsymbol{e}} = \boldsymbol{GU} + \boldsymbol{F}\left(\dot{\boldsymbol{X}},\Delta_G\right) - \ddot{\boldsymbol{X}}_C \quad (11)$$

As above, the structure damaged aircraft can be considered as a MIMO nonlinear system. To apply NTSM to the damaged aircraft, the sliding surface is defined as

$$\boldsymbol{S} = \boldsymbol{e} + \frac{1}{\boldsymbol{\beta}}\dot{\boldsymbol{e}}^{g/h} \quad (12)$$

Where $\boldsymbol{\beta}$ is a positive definite matrix, g,h are the positive odd integers satisfying $1<g/h<2$.

The sliding mode trending law [7] is described as:

$$\dot{\boldsymbol{S}} = -\boldsymbol{Q}\left(\boldsymbol{kS} + \boldsymbol{yS}^{m/n}\right) \quad (13)$$

Where $\boldsymbol{k} = k_0 + \|x\|_1$, $\boldsymbol{y} = \dfrac{y_0}{1+\|x\|_1}$, $\boldsymbol{Q} = diag\left(\dot{\boldsymbol{e}}_i^{g/h-1}\right), i=1,2,3$, $k_0>0, y_0>1$ m,n are the positive odd integers satisfying $0<m/n<1$.

First order norm of state variables is introduced to this adaptive variable rate trending law, which can adaptively adjust the index approach velocity with the system state away from the equilibrium points.

From (10), (11) and (12), the NTSM control law is obtained

$$\boldsymbol{U} = -\boldsymbol{G}^{-1}\left[\boldsymbol{F}\left(\dot{\boldsymbol{X}},\Delta_G\right) + \frac{h}{g}\boldsymbol{\beta C}\dot{\boldsymbol{e}} - \ddot{\boldsymbol{X}}_c + \frac{h}{g}\boldsymbol{\beta}\left(\boldsymbol{kS} + \boldsymbol{yS}^{m/n}\right)\right] \quad (14)$$

Where

$$\boldsymbol{C} = diag(c_i) \qquad c_i = \begin{cases} \dfrac{1}{\dot{e}_i^{g/h-2}} & \dot{e}_i \neq 0 \\ 0 & \dot{e}_i = 0 \end{cases} \quad (15)$$

3.2 Adaptive NTSM Control

We take the system uncertainties and external disturbances as ΔF, to simplify the formula, $\ddot{X}_c$ is contained in ΔF, so the nonlinear system can be written as

$$\ddot{\boldsymbol{e}} = \boldsymbol{GU} + \boldsymbol{F}\left(\dot{\boldsymbol{X}},\Delta_G\right) - \Delta\boldsymbol{F} \quad (16)$$

If the control vector U does not contain the acceleration signal, the uncertainties ΔF is bounded by a positive function of the position and velocity measurements [8]

$$\|\Delta \boldsymbol{F}\| < b_0 + b_1\|\boldsymbol{e}\| + b_2\|\dot{\boldsymbol{e}}\|^2 \tag{17}$$

Where b_0, b_1, b_2 are positive numbers.

Then, we get the uncertain bound of each component

$$|\Delta \boldsymbol{F}_i| < \lambda_{i1} + \lambda_{i2}\|\boldsymbol{e}\| + \lambda_{i3}\|\dot{\boldsymbol{e}}\|^2, i = 1,2,3 \tag{18}$$

So

$$abs(\Delta F) = [|\Delta F_1| \ |\Delta F_2| \ |\Delta F_3|]^T \tag{19}$$

Assumption1. System uncertainty $\Delta \boldsymbol{F}$ satisfies

$$abs(\Delta \boldsymbol{F}) < \boldsymbol{\lambda X_K} \tag{20}$$

Where

$$\boldsymbol{X_K} = [1 \ \|\boldsymbol{e}\| \ \|\dot{\boldsymbol{e}}\|^2]^T, \boldsymbol{\lambda} = \begin{bmatrix} \lambda_{11} & \lambda_{12} & \lambda_{13} \\ \lambda_{21} & \lambda_{22} & \lambda_{23} \\ \lambda_{31} & \lambda_{32} & \lambda_{33} \end{bmatrix} \tag{21}$$

From Eq.(11),(12), (16), and (20), the adaptive non-singular terminal sliding mode (ANTSM) control law is obtained

$$\begin{aligned} \boldsymbol{U} = &-\boldsymbol{G}^{-1}\left[\boldsymbol{F}\left(\dot{\boldsymbol{X}}, \Delta_G\right) + \frac{h}{g}\boldsymbol{\beta C}\dot{\boldsymbol{e}} - \ddot{\boldsymbol{X}}_c\right] \\ &-\frac{h}{g}\boldsymbol{G}^{-1}\boldsymbol{\beta}\left(\boldsymbol{kS} + \boldsymbol{yS}^{m/n}\right) - \boldsymbol{G}^{-1}\boldsymbol{f_m} \end{aligned} \tag{22}$$

Where $\boldsymbol{f_m}$ is updated by the following adaptive law

$$\dot{\boldsymbol{f}}_m = \boldsymbol{M}\frac{g}{h\boldsymbol{\beta}}\boldsymbol{QS} \tag{23}$$

Where $\boldsymbol{M}$ is a relatively large matrix.

3.3 Convergence analysis

Theorem 1. For the system which satisfies the assumption 1, the control law is proposed as (23), the system states can reach the sliding surface in finite time.

Proof. Define the Lyapunov function candidate

$$V = \frac{1}{2}\boldsymbol{S}^T\boldsymbol{S} + \frac{1}{2}\tilde{\boldsymbol{f}}_m^{\,T}\boldsymbol{M}^{-1}\tilde{\boldsymbol{f}}_m \tag{24}$$

The derivative of V can be shown as follows

$$\begin{aligned} \dot{V} &= \boldsymbol{S}^T\dot{\boldsymbol{S}} + \tilde{\boldsymbol{f}}_m^{\,T}\boldsymbol{M}^{-1}\dot{\tilde{\boldsymbol{f}}}_m \\ &= -\boldsymbol{S}^T\boldsymbol{Q}\left(\boldsymbol{kS} + \boldsymbol{yS}^{m/n}\right) - \boldsymbol{S}^T\frac{g}{h\boldsymbol{\beta}}\boldsymbol{Q}\tilde{\boldsymbol{f}}_m + \tilde{\boldsymbol{f}}_m^{\,T}\boldsymbol{M}^{-1}\dot{\tilde{\boldsymbol{f}}}_m \\ &= -\tilde{\boldsymbol{f}}_m^{\,T}\left(\frac{g}{h\boldsymbol{\beta}}\boldsymbol{QS} - \boldsymbol{M}^{-1}\dot{\tilde{\boldsymbol{f}}}_m\right) - \boldsymbol{S}^T\boldsymbol{Q}\left(\boldsymbol{kS} + \boldsymbol{yS}^{m/n}\right) \end{aligned} \tag{25}$$

Where $\tilde{\boldsymbol{f}}_m = \boldsymbol{f}_m - \hat{\boldsymbol{f}}_m$

Supposing that the parameter variations and external disturbance change very slowly compared to the dynamics of disturbance, that is, the adaptation process of f_m is much faster than the changing rate of $\hat{\boldsymbol{f}}_m$ that could be achieved by choosing a large adaptation gain $\boldsymbol{M}$, we can obtain $\dot{\boldsymbol{f}}_m = \dot{\tilde{\boldsymbol{f}}}_m$ is updated by the following adaptation algorithm:

$$\dot{\tilde{\boldsymbol{f}}}_m = \boldsymbol{M}\frac{g}{h\boldsymbol{\beta}}\boldsymbol{QS} \tag{26}$$

Then

$$\dot{V} = -\boldsymbol{S}^T\boldsymbol{Q}\left(\boldsymbol{kS} + \boldsymbol{yS}^{m/n}\right) \le 0 \tag{27}$$

From Barbalat Lemma

$$-\boldsymbol{S}^T\boldsymbol{Q}\left(\boldsymbol{kS} + \boldsymbol{yS}^{m/n}\right) \to 0 \tag{28}$$

When $\boldsymbol{S} = 0$, the system states can reach the sliding surface in finite time[9], When $\boldsymbol{Q} = 0, \dot{\boldsymbol{e}}_i^{g/h-1} = 0, i = 1,2,3$

$$\begin{aligned} \frac{d\dot{\boldsymbol{e}}}{d\boldsymbol{e}} &= \frac{\ddot{\boldsymbol{e}}}{\dot{\boldsymbol{e}}} = \frac{\boldsymbol{GU} + \boldsymbol{F}\left(\dot{\boldsymbol{X}}, \Delta_G\right) - \Delta\boldsymbol{F}}{\dot{\boldsymbol{e}}} \\ &= \frac{-\frac{h}{g}\boldsymbol{\beta}\left(\boldsymbol{\varphi S} + \boldsymbol{yS}^{m/n}\right) - \tilde{\boldsymbol{f}}_m}{\dot{\boldsymbol{e}}} \\ &= \frac{-\frac{h}{g}\boldsymbol{\beta}\left(\boldsymbol{k}\left(\boldsymbol{e} + \frac{1}{\boldsymbol{\beta}}\dot{\boldsymbol{e}}^{\frac{g}{h}}\right) + \boldsymbol{y}\left(\boldsymbol{e} + \frac{1}{\boldsymbol{\beta}}\dot{\boldsymbol{e}}^{\frac{g}{h}}\right)^{m/n}\right) - \tilde{\boldsymbol{f}}_m}{\dot{\boldsymbol{e}}} \end{aligned} \tag{29}$$

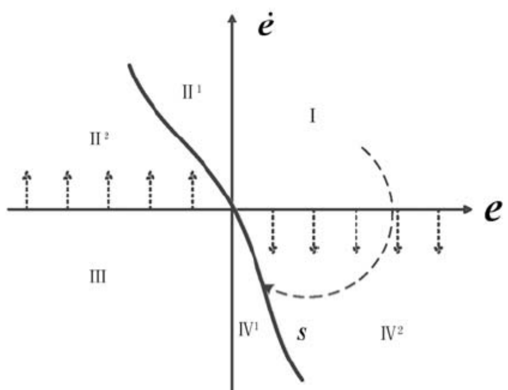

Fig. 2: Phase plane

1. when $\boldsymbol{e} > 0$,

$$\frac{d\dot{\boldsymbol{e}}}{d\boldsymbol{e}} = -\infty$$

As shown in Fig.2, when the phase path is in the vertical direction of axis $\dot{\boldsymbol{e}}$, the speed vector of $(e, \dot{e})$, which is nonzero points down. We can deduce that any state point that reaches the left half of axis $\dot{\boldsymbol{e}}$ can enter the area of $\dot{\boldsymbol{e}} \neq 0$. Thus, the system is stable.

2. when $\boldsymbol{e} < 0$, the system can also be derived to be stable using the same analysis method.

4 System Simulation

To verify the effectiveness of the adaptive non-singular terminal sliding mode controller, 5.5% of GTM shrinkage model is presented. The rectangular shape commands are given to the pitch and roll axes, as shown in Fig.3. Each rectangular signal ascends at 3s and descends at 7s, and 33% the structural damage happens at 5s. Simulation results are shown as Fig. 3 and Fig.4

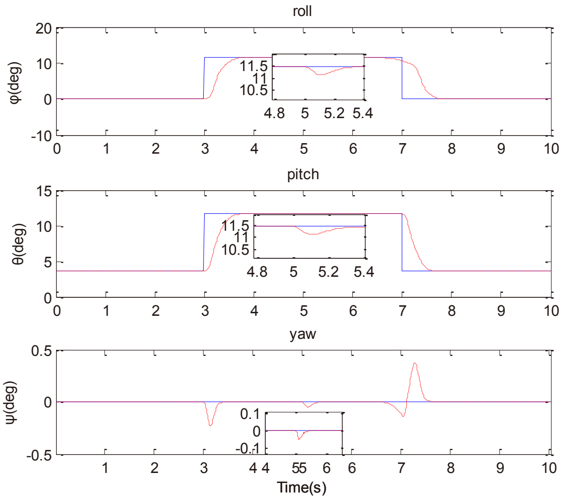

Fig.3 The attitude response

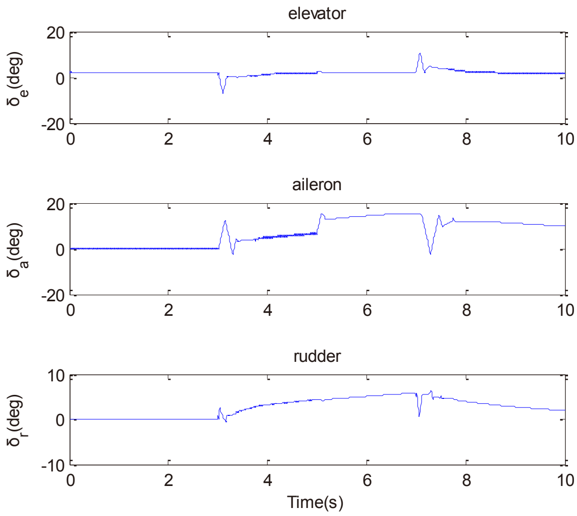

Fig.4 The actuator deflection

It easily can be seen from Fig.3 that both the responses of attitude angle rates can track the desired signals precisely when 33% damage occurred with right wing tip at 5s. And it tracks the command in 0.2s, which is faster than before. From Fig.4, the actuator deflections also meet the limits of the operating mechanism deflection and rate constraints, which is less than 20°. All those show the robustness and the rapidity of the control law.

5 Conclusion

In this paper, the ANTSM controller is developed to control the structural damaged aircraft. A three-channel attitude controller is designed for a wing damaged aircraft. The approach can ensure the system to reach the sliding surface and converge to equilibrium point in finite time from any initial state without the singularity. In addition, the proposed adaptive non-singular terminal sliding mode controller can real-time estimate the uncertainties and external disturbances. In order to verify effectiveness of the proposed algorithm, the algorithm is presented and illustrated through simulations on the GTM model. Simulation results show the proposed control method can solve the control problem in some extent after the aircraft damaged.

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