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AN ADAPTIVE DISTURBANCE SUPPRESSION BASED FAULT-TOLERANT CONTROL APPROACH AGAINST THE CONTROL SURFACE FAULTS

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Abstract. *This paper is concerned with the attitude control problem under the fault of aircraft control surface. Firstly, when the control surface suffers a structural missing fault, the aerodynamic characteristics will change significantly and the tracking performance of the pre-defined nominal controller will be seriously affected. An adaptive disturbance rejection control method based on nonlinear recursive least squares identification is proposed, which combines real-time model identification and nonlinear dynamic inverse to control the aircraft using the causeless force and moment models calculated from the identification model. Comparing the closed-loop simulation results of the adaptive disturbance suppression controller with the nominal controller shows that the control method in this paper improves the control quality, maneuvering stability and system response speed, and reduces the overshoot in the control process.*
Keywords: Structural missing fault of control surface, Recursive least squares, Nonlinear dynamic inverse, Adaptive control

1. **Introduction.** The fault of control surface means that the control surface is partially lost in the process of flight for various reasons, and the dynamics model previously got through wind tunnel experiments or computational fluid dynamics becomes inaccurate. The preset nominal controller is designed according to the dynamics model before the failure, so the control quality will deteriorate [1-6].

The control problems in the case of structural control surface failure have been fruitful research results in China and abroad. [7] proposed a method to make the redundant normal control surface give full play to its performance with coordinately assigned control commands in case of control surface fault, but this method can only be used in aircraft equipped with redundant control surfaces; [8] used the adaptive backstepping compensation control method for nonlinear tracking control with multiple inputs and outputs in case of control surface failure, but this method is only applicable to the control system that can be converted to the lower delta form control system [9], and since the traditional backstepping control method is used, it needs to derive the virtual control at each step, which will cause "differential explosion" if used for complex higher-order systems [10]; [11] addressed the differential explosion problem of the backstepping control method and proposes a dynamic surface control method, which introduces a first-order dynamic

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filter, but in the case of control surface fault, it is easy to cause the problem of control surface rate limitation and saturation; a model-referenced adaptive sliding-mode fault-tolerant controller based on neural network compensation was proposed for the rudder failure problem of multi-maneuvering surface aircraft [12]. The parametric adaptive law was obtained by designing adaptive estimation terms to compensate for the nonlinearity of unknown rudder faults based on the traditional model-referenced integral sliding-mode control law, respectively. This method has significant safety concerns in the transition phase from the original control law to the parametric adaptive law. [13] adopted a multi-model based parameter estimation method, designed an adaptive controller group, and proposed a control switching mechanism based on the optimal performance index function to select the best adaptive controller as the current controller, so as to achieve the desired fault compensation control. However, there is a real-time problem in its control method. When the aircraft is moving fast, the system parameters change quickly when a fault occurs, and the conventional online parameter identification method cannot meet the real-time requirement.

Currently there is not a stable method that is widely applicable and can quickly suppress the control problem under structural missing faults on the control surface. On the basis of the above work, this paper proposes an adaptive disturbance suppression control method based on recursive least squares real-time discrimination for the control problem. The method consists of three parts: real-time system identification, nonlinear dynamic inversion and adaptive disturbance suppression. First, the dynamics model, aerodynamic parameters and control derivatives of the control in the missing fault state are acquired by system identification, then the Euler angular command is converted to an attitude angular command by command decoupling under nonlinear dynamic inverse control, and then the attitude angular command is converted to an angular velocity command, while the adaptive interference suppression adjusts the control signal to suppress the angular velocity by the difference between the desired behavior and the measured behavior. It is applicable to most of the aircraft and various control systems, and can ensure the real-time control process, and finally achieve better control under the structural failure of the aircraft elevator control surface.

This paper is organized as follows: Section 2 clarifies the mathematical model of pitch attitude dynamics, the process and method of aerodynamic identification are presented in Section 3, the adaptive disturbance rejection control method is clarified in Section 4, and the results and discussions are shown in Sections 5 and 6.

2. **Dynamics Modeling under Consideration of Structural Failure of Control Surface.** When structural missing faults occur on the aircraft control surface, the rudder effect of the control surface will decrease, that is to say, the convergence speed of the aircraft attitude control slows down, that is, deteriorate the control stability. Assuming that the aircraft dynamics is a standard rigid body equation, this paper only studies the structural missing failure case of the elevator, so the longitudinal dynamics is modeled under the Soviet Union body coordinate axis as follows.

$$\begin{cases} \dot{V} = \dfrac{1}{m}\left(T\cos\alpha - D\right) \\ \dot{\alpha} = q + \dfrac{1}{mV}(-L + mg) \\ \dot{\theta} = q \\ \dot{q} = I^{-1}\left(M + M_\delta\right) \\ M_\delta = \bar{q}\bar{c}S\left(m_z^{\delta zl}\delta_{zl}f_l + m_z^{\delta zr}\delta_{zr}f_r\right) \end{cases} \tag{1}$$

where $\dot{}$ is time derivative; V is the current flying speed; m is the mass of the airplane; T is the thrust value; α is angle of attack; D is the resistance; q is body-axis pitch rates; L is the aerodynamic lift; g is gravitational acceleration; θ is the flight path inclination; I is pitch moments of inertial; M is pitch moments; M_δ is pitch moments due to control surface deflection; $\bar{q}$ is dynamic pressure; $\bar{c}$ is the characteristic length; S is the reference area; $m_z^{\delta zl}$ and $m_z^{\delta zr}$ are respectively pitch moment coefficients due to deflection of the left elevator and right elevator; δ_{zl} and δ_{zr} are respectively the deflection angle of the left elevator and right elevator; f_l and f_r are the elevator efficiency coefficients of the left elevator and right elevator, which are 1 when there is no structural loss of elevator surface. And the coefficients are different for different degrees of loss, which can be simulated in the mathematical simulation by modifying the values of the missing elevator surface failure.

3. **Recursive Least Squares Based Aerodynamic Identification of Control Surface in the Case of Structural Faults.** In this paper, the recursive least squares method [14] is used to achieve an efficient aerodynamic real-time identification task after a structural failure of the control surface of a aircraft. The "forgetting factor" is introduced in the recursive formula to reduce the computational effort and improve the computational efficiency, and the time-varying parameters can be estimated to ensure real time and accuracy. The flow chart of the algorithm is shown as Figure 1.

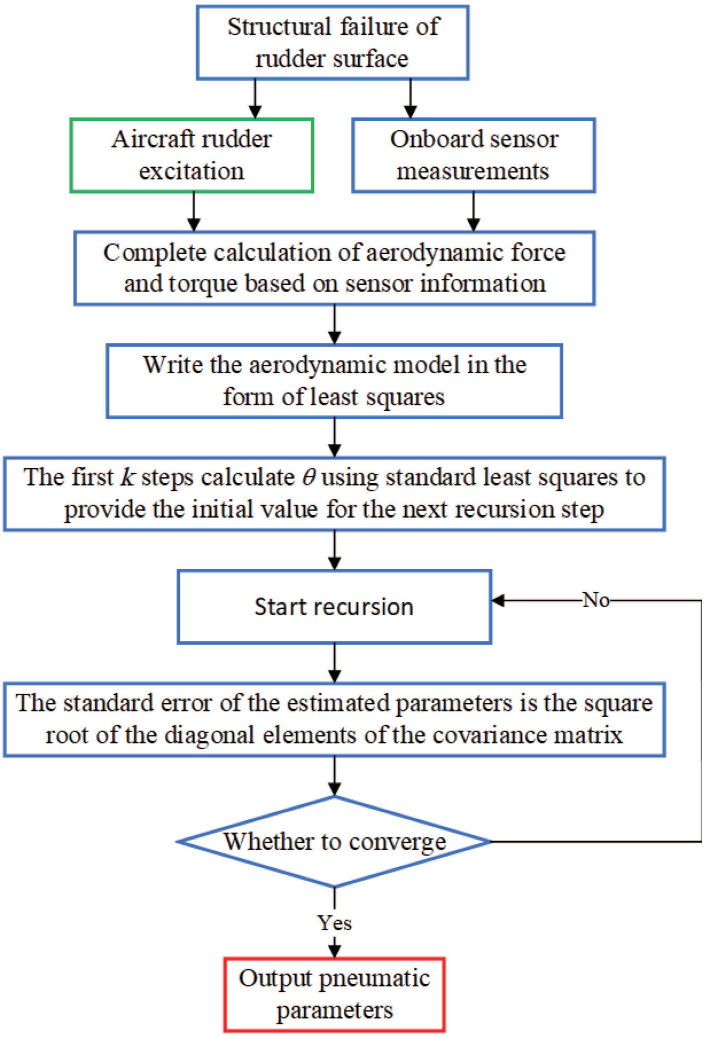


FIGURE 1. Recursive least squares identification

When the control surface of an aircraft fails, the optimal input signal should be applied to exciting the dynamic system of the aircraft to record the control surface signal and other flight data of the aircraft, and then the flight data recorded by the sensor should be used to calculate the aerodynamic force or torque coefficient as the observation of the least squares model. Secondly, the aerodynamic model is expanded into the form of matrix multiplication, and the recorded flight data is written into the observation matrix. The estimated values of unknown parameters are calculated using the data in the previous k steps, which are taken as the initial values of the next recursive algorithm. Finally, start recursive calculation, calculate the standard deviation of the estimated parameters and the output error of the observation model and the identification model, and judge whether it is convergent. If it is not convergent, continue recursive calculation, and if it is convergent, output the estimated parameters.

In the process of aerodynamic discrimination, it is first necessary to design the excitation signal of the control surface, which is summed with the actual control command as the actual signal of the aircraft control deflection at the current moment, and in order to make the flight state after excitation within the relative neighborhood of the current flight state, the mathematical expression is as Formula (2)

$$u = \sum_{k \in \{1,2,\ldots,M\}} A_k \sin\left(\frac{2\pi kt}{T} + \phi_k\right) \tag{2}$$

where k is the number of harmonics of the signal; ϕ_k is the phase of the k-th harmonic; the frequency range of the excitation signal is $\left[\frac{2\pi k_1}{T}, \frac{2\pi k_M}{T}\right]$; A_k is the amplitude of the harmonics, which is determined by $A_k = \frac{A}{\sqrt{n}}$, among them, A is the amplitude of the excitation signal, and n is the number of harmonics; $2\pi/T$ is the upper band of the excitation input.

In the event of a structural loss of the control surface, the flight state at the time of failure is first recorded, and aerodynamic modeling is performed using the flight data. The dimensionless force and moment coefficients in aerodynamics are used as dependent variables in the modeling problem, and since the values of the dimensionless aerodynamic forces and moment coefficients cannot be measured directly in flight, they must be calculated from the measured and known quantities using the following equation, because this paper studies aerodynamic discrimination under structural loss of the elevator, so only the pitch direction is identified and modeled as Formula (3)

$$\begin{cases} C_L = \dfrac{ma_L}{\bar{q}S} \\ m_z = \dfrac{I}{\bar{q}S\bar{c}}\dot{q} \end{cases} \tag{3}$$

where C_L is the body axis pitch direction causeless aerodynamic coefficient; a_L is the body axis pitch acceleration measure; m_z is the pitch moment coefficient.

In this method, the dependent variable is one of the dimensionless force or moment coefficients, and is modeled by an extension of the model term calculated from the explanatory variables (dimensionless aircraft state quantities and control surface deflections) that are nonlinear. Here, as an example, the pitch moment coefficient m_z after identifying a fault is used to express the equation error least squares problem for the pitch moment coefficient m_z using a specific model structure, setting the modeling function as Formula (4)

$$m_z = m_{z0} + m_z^{\alpha}\alpha + m_z^{q}\frac{q\bar{c}}{2V} + m_z^{\delta_z}\delta_z \tag{4}$$

where m_{z0} is the pitch moment coefficient at $\alpha = q = \delta = 0$; m_z^α is the pitch moment coefficient due to change in angle of attack; m_z^q is the pitch moment coefficient due to change in pitch angle velocity; $m_z^{\delta_z}$ is the pitch moment coefficient due to elevator deflection; δ_z is the elevator deflection angle.

The above equation can be written as Formula (5)

$$\mathbf{Z} = \mathbf{H}\boldsymbol{\theta} + \boldsymbol{\upsilon} \tag{5}$$

where $\mathbf{Z} = \begin{bmatrix} m_z(1) & m_z(2) & \cdots & m_z(N) \end{bmatrix}^T$ is the value calculated using the pneumatic moment formula; $\mathbf{H} = \begin{bmatrix} \alpha & \delta_z & 1 & \frac{q\bar{c}}{2V} \end{bmatrix}$ is the modeling function set; $\boldsymbol{\theta} = \begin{bmatrix} m_z^\alpha & m_z^{\delta_z} & m_z^q & m_{z0} \end{bmatrix}$ is the model identified; $\boldsymbol{\upsilon} = \begin{bmatrix} \upsilon(1) & \upsilon(2) & \cdots & \upsilon(N) \end{bmatrix}$ is the residuals.

When using a model in the least squares (LS) form, the uncertainty about $\boldsymbol{\theta}$ and $\boldsymbol{\upsilon}$ is not used, i.e., there is no probabilistic expression for $\boldsymbol{\theta}$ and $\boldsymbol{\upsilon}$, so the least squares estimate can be reached by the following corollary: for a given $\boldsymbol{\theta}$ and $\mathbf{Z}$, the best estimate can be derived from minimizing the weighted sum of the squared errors between the measured output and the model output, and the equation is as Formula (6)

$$\mathbf{J}(\theta) = \frac{1}{2}\left(\mathbf{Z} - \mathbf{H}\boldsymbol{\theta}\right)^T\left(\mathbf{Z} - \mathbf{H}\boldsymbol{\theta}\right) \tag{6}$$

This is one of the means to check the accuracy of the estimates, but at the same time the computational workload is increasing, so ρ as a forgetting factor is added in the recursive process to reduce the computational workload.

The initial value required for a recursion is first calculated from known observations and outputs, and when all observations and outputs before the moment k are known, note $\mathbf{B}(k) = \mathbf{H}^T(k)\mathbf{H}(k)$, then the parameter estimated at the moment before k is as Formula (7)

$$\boldsymbol{\theta}(k) = \mathbf{B}^{-1}(k)\mathbf{H}^T(k)\mathbf{Y}(k) \tag{7}$$

where $\mathbf{Y}(k)$ is the observed quantity.

After the initial value is known, the recursion begins, where $k = n$, and the recursion starts at the moment $n + 1$; the recursion is given by equation as Formulas (8)-(10)

$$\mathbf{K}(k) = \mathbf{P}(k)\mathbf{H}^T(k+1)\left[\rho^2 I + \mathbf{H}(k+1)\mathbf{P}(k)\mathbf{H}^T(k+1)\right]^{-1} \tag{8}$$

$$\hat{\boldsymbol{\theta}}(k+1) = \hat{\boldsymbol{\theta}}(k) + \mathbf{K}(k)\left[\mathbf{Z}(k+1) - \mathbf{H}(k+1)\hat{\boldsymbol{\theta}}(k)\right] \tag{9}$$

$$\mathbf{P}(k+1) = \frac{1}{\rho^2}[\mathbf{P}(k) - \mathbf{K}(k) * \mathbf{H}(k+1) * \mathbf{P}(k)] \tag{10}$$

where $\mathbf{K}(k)$ is the gain matrix at moment k; $\hat{\boldsymbol{\theta}}(k)$ is the estimator at moment k; $\mathbf{P}(k) = \mathbf{B}^{-1}(k)$ is the inverse of the information matrix at moment k.

When $\mathbf{K}(k)$ is a zero matrix, it means that the estimator $\hat{\boldsymbol{\theta}}(k+1)$ converges, at this point the post-fault pneumatic parameters are worked out and output m_{z0}, m_z^α, $m_z^{\delta_z}$ and m_z^q.

4. **Design Method of Adaptive Interference Suppression Control Based on Online Recognition.** In this paper, we design an adaptive disturbance suppression controller based on online recognition [15], which consists of two parts, taking the control problem under structural missing fault of elevator as an example: 1) The first part is a nonlinear dynamic inverse controller [16,17], which is divided into an outer-loop fast-loop control and an inner-loop slow-loop control, where the outer-loop control converts the pitch angle command into a pitch angle velocity command, then converts the pitch angle velocity command into a pitch angle acceleration command as an input to the inner-loop

control, and finally converts it into a torque command to be assigned to the elevator surface; 2) The second part is to design the adaptive disturbance suppression controller for the problem of model parameter uncertainty and external disturbance after the failure, so that it estimates the equivalent disturbance of the system, adjusts the control signal according to the difference between the expected and measured behaviors, suppresses the error caused by the structural missing fault on the control surface, and feeds the estimated value to the nonlinear dynamic inverse controller. The control structure is shown in Figure 2.

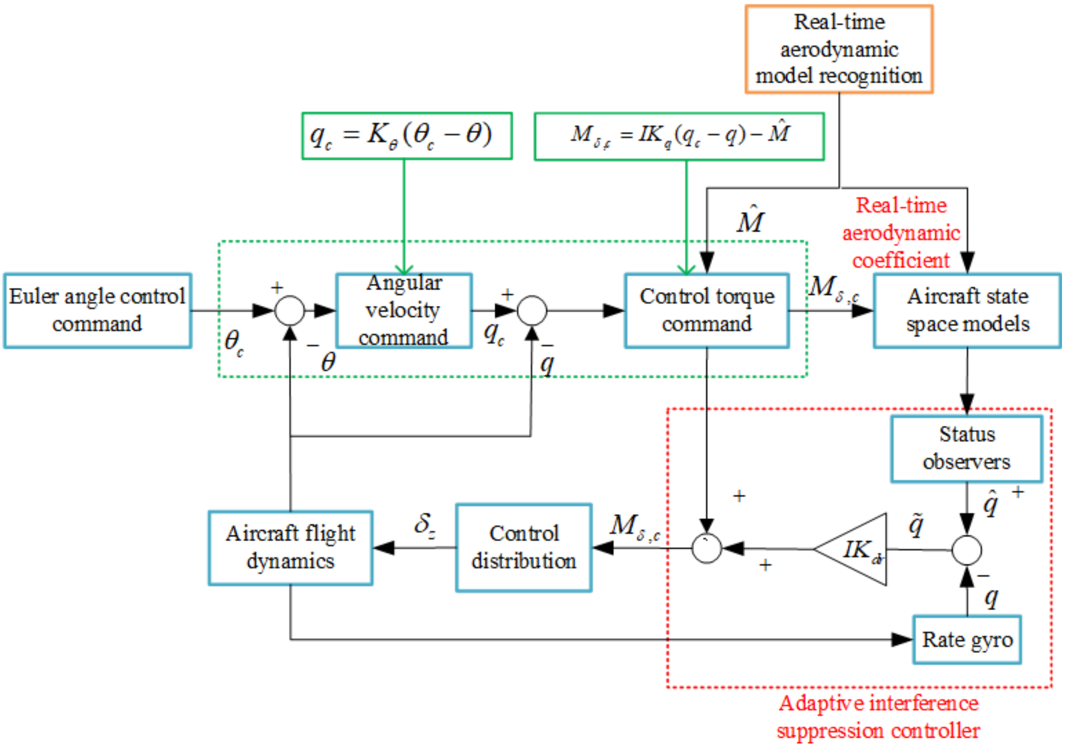


FIGURE 2. High-precision adaptive interference suppression control based on online model

4.1. **Nonlinear dynamic inverse controller design.** To track the commands for each variable, a nonlinear dynamic inverse is used in order to generate commands for faster variables using the time scale separation of the variables. The dynamic inverse of the outermost fast loop is the tracking pitch angle command θ_c and the command to solve for q based on θ_c, which according to the dynamical equations, has the following command as Formula (11)

$$q_c = K_\theta(\theta_c - \theta) \tag{11}$$

where q_c is the pitch angle speed command; K_θ is the control gain.

The dynamic inverse of the inner loop slow loop then solves the next dynamic inverse command, solving the inverse of the dynamical equations that track these angular rate commands inner loop to produce the desired angular acceleration proportional to the angular rate error, and the linearization equation is as Formula (12)

$$\dot{q}_c = K_q(q_c - q) \tag{12}$$

where $\dot{q}_c$ is the pitch angle acceleration command; K_q is the control gain.

According to Formula (1), the pitch moment command can be deduced from the pitch angle acceleration command as Formula (13)

$$M_{\delta,c} = IK_q(q_c - q) - \hat{M} \tag{13}$$

where $M_{\delta,c}$ is the pitch moment control command; $\hat{M}$ is the estimate of the current aircraft pitch moment, which is determined by the current discrimination model of the aircraft; K_q is the control gain.

4.2. **Adaptive interference suppressor design.** The adaptive disturbance suppression section is designed to assist the internal loop control in regulating the angular rate, with the main purpose of suppressing the effects of errors in the solution of the angular speed control command due to changes in the dynamics model caused by structural missing faults in the control surface. It works by comparing the measured response with an internal model of the desired dynamics. It then adjusts the control signal according to the difference between the desired and measured behavior, and the desired behavior is defined as a dynamic inversion component that successfully counteracts the nonlinearity and enhances the linear responsiveness.

$$\dot{q}_{\exp} = K_q\left(q_c - q_{\exp}\right) \tag{14}$$

where $\dot{q}_{\exp}$ is the observed pitch acceleration; $q_{\exp}$ is the observed pitch velocity.

Due to external disturbances and structural missing dynamics model changes on the control surface, the dynamic inverse will not be able to fully achieve its objective, so introducing d to describe the uncertainty error of the system and writing the system dynamics in a form similar to the desired behavior, so the actual differential equation of q is introducing uncertainty error in the above angular velocity tracking command.

$$\dot{q} = K_q(q_c - q) + A_{dr} + d \tag{15}$$

where $A_{dr} = I^{-1}\Delta M$, which is the adaptive part of the control input scale.

Therefore, the control internal model instruction required to define an adaptive input to suppress the effect of errors is as Formula (16)

$$\dot{\hat{q}} = K_q(q_c - q) + A_{dr} - K_{dr}\tilde{q} \tag{16}$$

where $\tilde{q} = \hat{q} - q$, we have $\dot{\tilde{q}} = \dot{\hat{q}} - \dot{q} = -K_{dr}\tilde{q} - d$; $\dot{\hat{q}}$ is an estimate of the pitch angle acceleration; K_{dr} is the control gain.

According to the method of solving the first-order non-simultaneous linear differential equation, the function with respect to time can be solved as Formula (17)

$$\tilde{q} = -\frac{d}{K_{dr}} + \left(\tilde{q}(0) + \frac{d}{K_{dr}}\right)e^{-K_{dr}t} \tag{17}$$

From the above equation, it follows that $\tilde{q}$ converges to $-\frac{d}{K_{dr}}$ at a certain time, during this time, $\dot{\tilde{q}}$ converges to 0, i.e., $-K_{dr}\tilde{q} - d$ converges to 0, and then $K_{dr}\tilde{q} + d$ similarly converges to 0. According to the actual q differential equation, such that $A_{dr} = K_{dr}\tilde{q}$, substituted into (15), we get

$$\dot{q} = K_q(q_c - q) \tag{18}$$

For a certain time, q converges to q_c, so $A_{dr} = K_{dr}\tilde{q}$ can suppress the effect of error d.

Based on the dynamic inverse of the above set of dynamics equations and adding the adaptive inputs to the nonlinear dynamic inverse control torque command, substituting Equation (18) into Equation (1) yields the following command for generating the total torque.

$$M_{\delta,c} = IK_q(q_c - q) - \hat{M} + IK_{dr}\tilde{q} \tag{19}$$

Finally, after passing the pitch moment command, the control distribution is carried out and the elevator commands are equally distributed to the elevator with the following elevator commands.

$$\delta_z = \frac{\delta_{zl} + \delta_{zr}}{2} = \frac{M_{\delta,c}}{\bar{q}S\bar{c}m_z^{\delta_z}} \tag{20}$$

5. **Simulation Results.** In this section, the adaptive disturbance suppression controller under structural missing fault of the aircraft elevator established in the previous section is simulated and verified and analyzed, and the simulation results are compared with the pre-defined PID controller. Some of the structural parameters required for the aircraft simulation are shown in Table 1.

Table 1. Simulation parameters

Description	Variable	Value
Mass	m	28560 kg
Gravitational acceleration	g	9.8065 m/s^2
Reference area	S	69 m^2
Wingspan length	l	15 m
Wing average aerodynamic chord length	$\bar{c}$	5.4 m

The simulation condition is that the aircraft keeps cruising at Mach 0.9 in the air at an altitude of 10 km. After flying for 20 s, a command is given to keep the pitch angle of 5° to climb, and in real-time recognition in the process of giving the elevator excitation, simulation conditions are as Table 2.

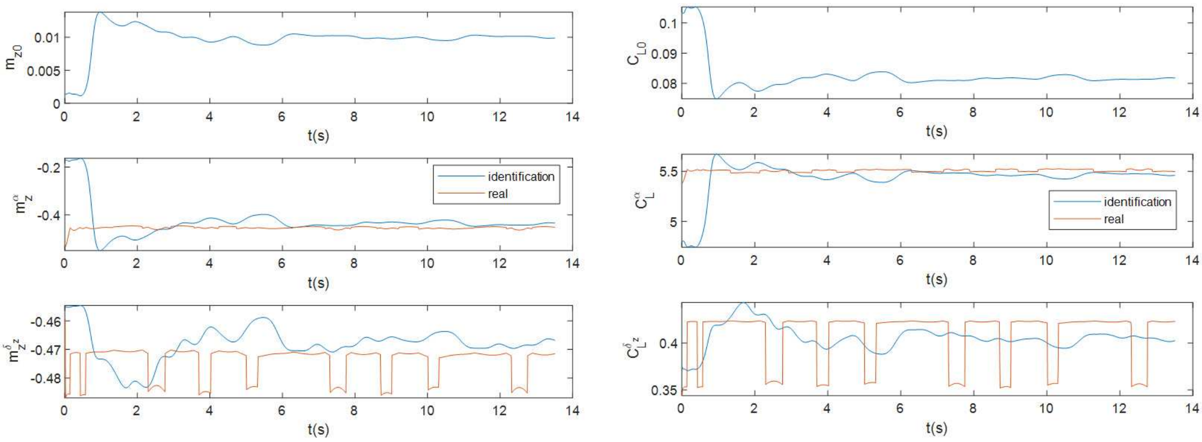
Table 2. Simulation conditions

Description	Variable	Value
Height	h	10 km
Speed	V	Ma0.9
Dynamic pressure	$\bar{q}$	4.14×10^{-7} kg/m^3
Flight time	t	40 s
Pitch angle command	θ_c	5°
Elevator excitation	u	$0.5 \times \left(\sin\left(\frac{4\pi t}{T} + 2.8274\right) + \sin\left(\frac{8\pi t}{T} + 2.1991\right) + \sin\left(\frac{12\pi t}{T}\right) + \sin\left(\frac{16\pi t}{T} + 1.8850\right)\right)$

Remark 5.1. *The overall parameters in Table 1 are the overall parameters of an actual aircraft. Aerodynamic data are obtained from wind tunnel data, and data in Table 2 are obtained from aerodynamic analysis.*

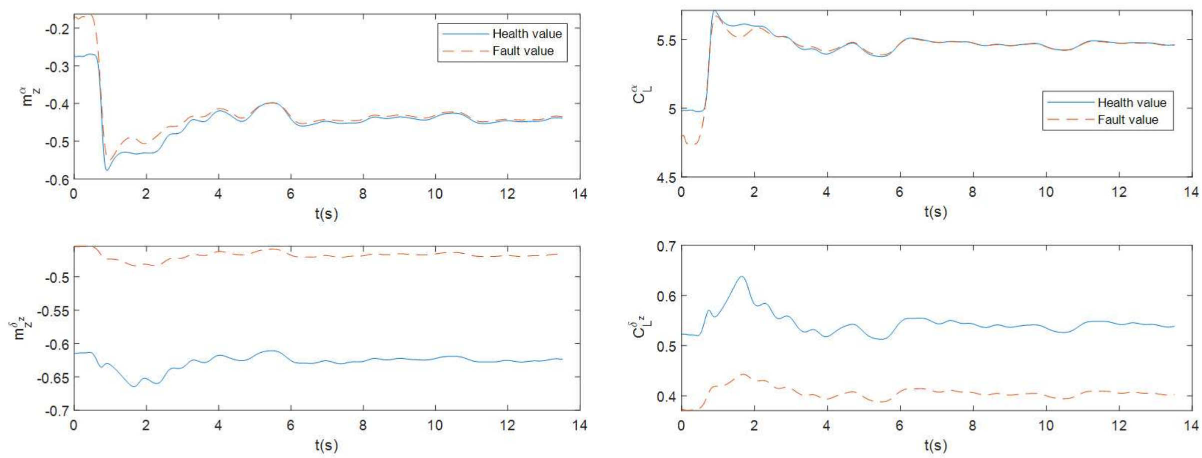
5.1. **Analysis of identification results.** Setting the left elevator damage 50%, carrying out online identification, and calculating the aerodynamic parameters, the flight status and identification results are shown in Figure 3.

From the identification results, it can be seen that the identification parameters converge to the true value of the interpolation calculation in about 1 second. The relative errors of m_z^α and $m_z^{\delta_z}$ identification are 3.5563% and 1.0305%, respectively, and the relative errors of C_L^α and $C_L^{\delta_z}$ identification are 0.7313% and 3.8191%, respectively, with a high accuracy.



(a) (b)

FIGURE 3. Identification error under 50% failure of structural deficiency of elevator: (a) Pitch moment coefficient identification error; (b) lift coefficient identification error



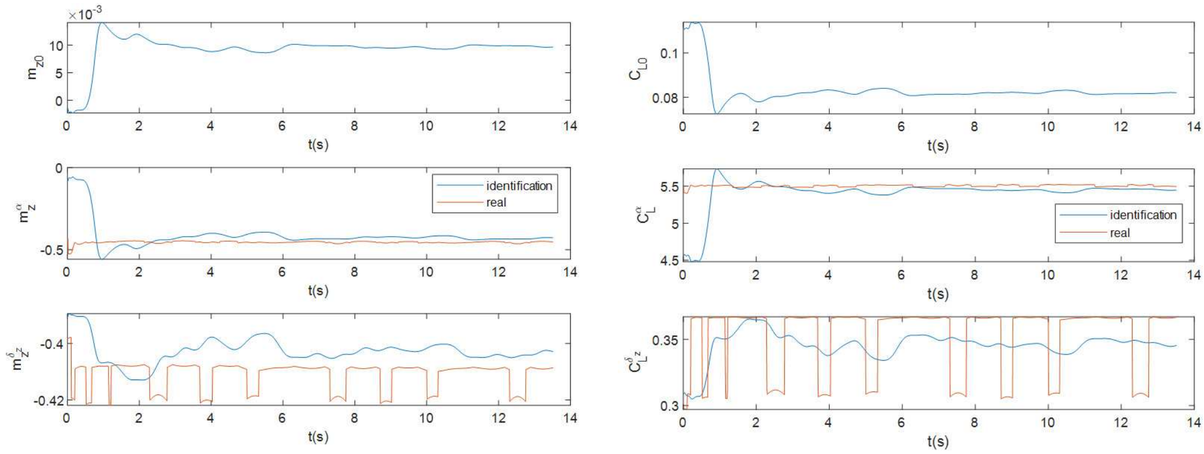
(a) (b)

FIGURE 4. Comparison of identification values between failure and healthy condition under 50% failure of structural deficiency of elevator: (a) Pitch moment coefficient; (b) lift coefficient

From the identification results, after 50% damage to the left elevator, there is almost no effect on the static stability coefficients C_L^α and m_z^α of the lift and pitch moment coefficients, and there is an effect on $C_L^{\delta_z}$ and $m_z^{\delta_z}$, and the whole process of identification is at about 75% of the healthy value; there is no effect on the static stability coefficients of the pitch moment coefficients and lift coefficients. In addition, the maneuverability coefficients $m_z^{\delta_z}$ and $C_L^{\delta_z}$ are reduced by about 25% compared with the healthy one, which is equivalent to about 25% reduction of the overall elevator effect.

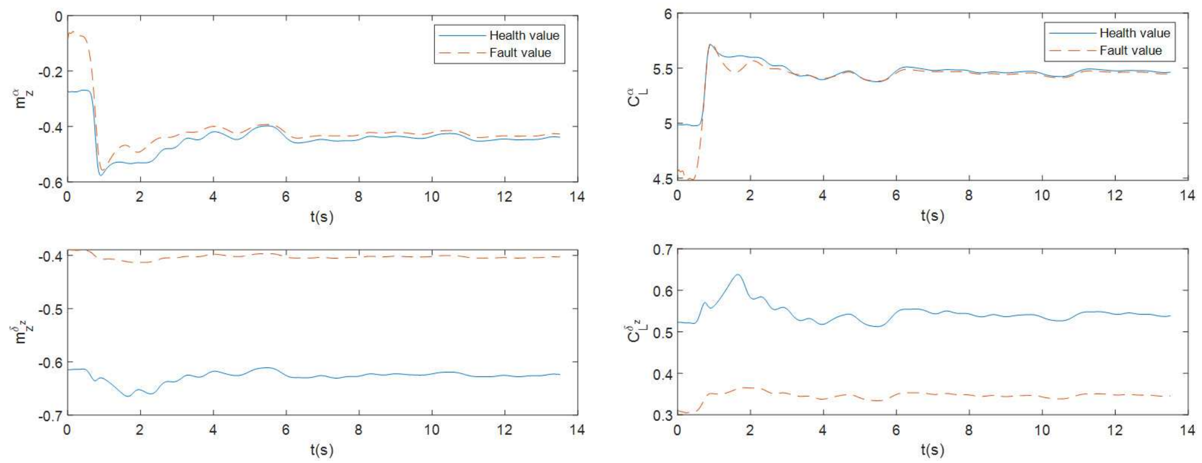
Setting the left elevator damage 70%, carrying out online identification, and calculating the aerodynamic parameters, the flight status and identification results are shown in Figure 5.

The identification results show that the identification parameters converge to the true value of the interpolation calculation in about 1 second. The relative errors of m_z^α and



(a) (b)

FIGURE 5. Identification error under 70% failure of structural deficiency of elevator: (a) Pitch moment coefficient identification error; (b) lift coefficient identification error



(a) (b)

FIGURE 6. Comparison of identification values between failure and healthy condition under 70% failure of structural deficiency of elevator: (a) Pitch moment coefficient; (b) lift coefficient

$m_z^{\delta_z}$ identification are 4.1640% and 1.4460%, respectively, and the relative errors of C_L^α and $C_L^{\delta_z}$ identification are 0.9371% and 4.7300%, respectively.

From the identification results, the static stability coefficients C_L^α and m_z^α of the lift and pitch moment coefficients have almost no effect after 70% damage to the left elevator, and there is an effect on $C_L^{\delta_z}$ and $m_z^{\delta_z}$, and the whole process of identification is around 65% of the healthy value; the static stability derivatives of the pitch moment coefficients and lift coefficients have no effect, and the maneuverability derivatives $m_z^{\delta_z}$ and $C_L^{\delta_z}$ are reduced by 35% compared to the healthy, which is equivalent to a 35% reduction in the overall elevator effect.

In summary, no matter the elevator has 50% or 70% missing fault, this paper based on recursive least squares method for real-time identification of aircraft model parameters converges fast, i.e., the required parameters can be identified in a very short time, and the computational efficiency is high, the identification error of the parameters is less than 5%, and the overall identification accuracy meets the requirements, which can realize the real-time identification of aircraft model parameters.

5.2. **Analysis of control results.** Comparison of the control effect between normal and elevator structural absence fault states. The simulations of the left elevator of the aircraft with 50% and 70% elevator efficiency loss are introduced to simulate the structural missing 50% and 70% failure of the left elevator surface of the aircraft, and the simulations of the PID controller in the normal and failure states are compared in Figure 7.

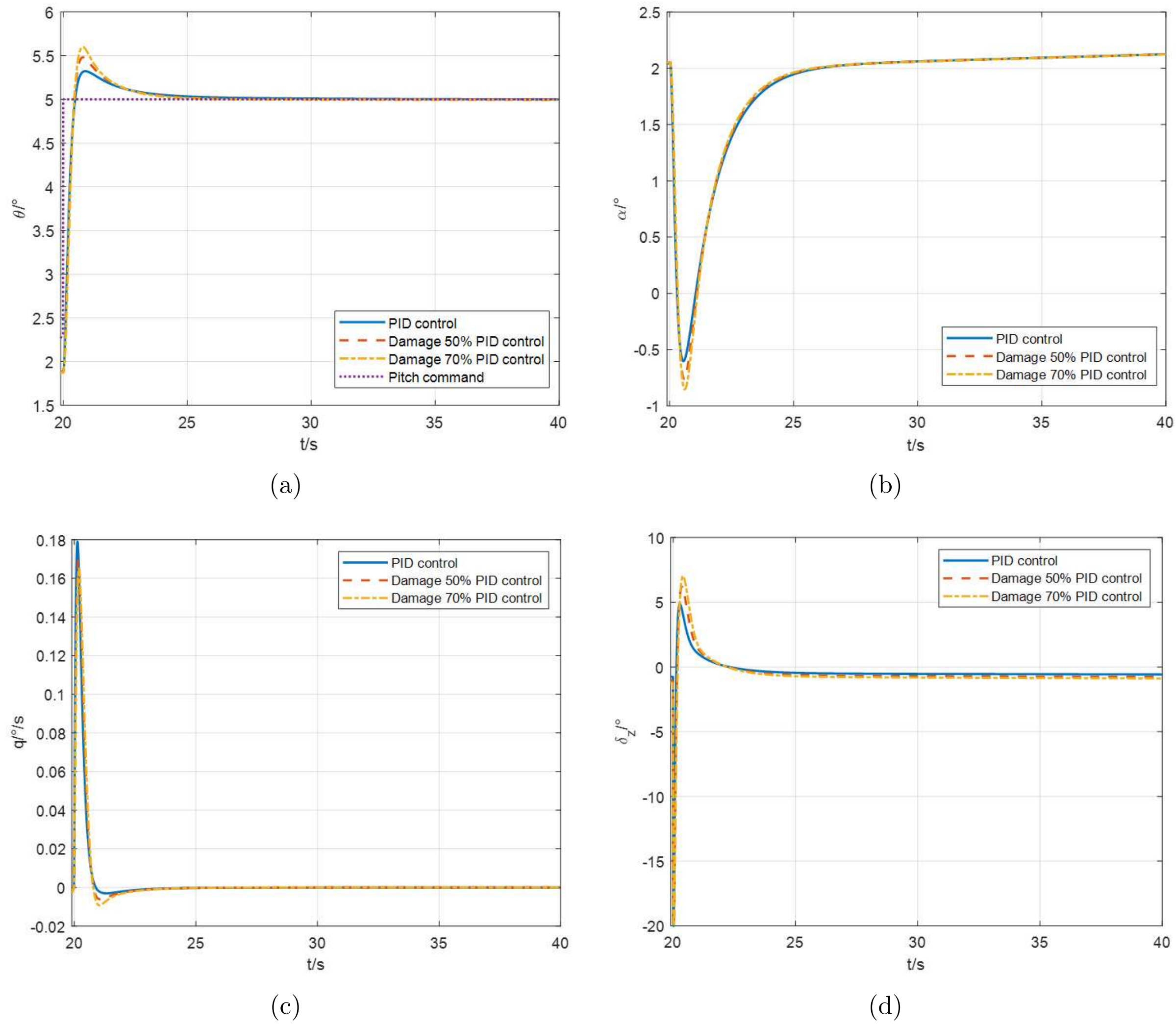


FIGURE 7. Simulation comparison of PID controller in normal state and fault state: (a) Pitch angle control curve; (b) angle of attack control curve; (c) pitch angle rate control curve; (d) elevator control curve

From the simulation results, it can be found that the overshoot is larger than the normal state of the rudder surface when PID controller is used to track the 5° pitch angle attitude command under the 50% and 70% structural loss of the left elevator surface, that is, convergence speed is slower, and adjustment speed becomes slower, in addition to more elevator out, indicating that in the elevator surface structural missing fault, the preset controller control becomes less effective.

Comparison of adaptive disturbance suppression control and PID control effects for a structural missing 50% fault condition of elevator. Simulation comparison of adaptive

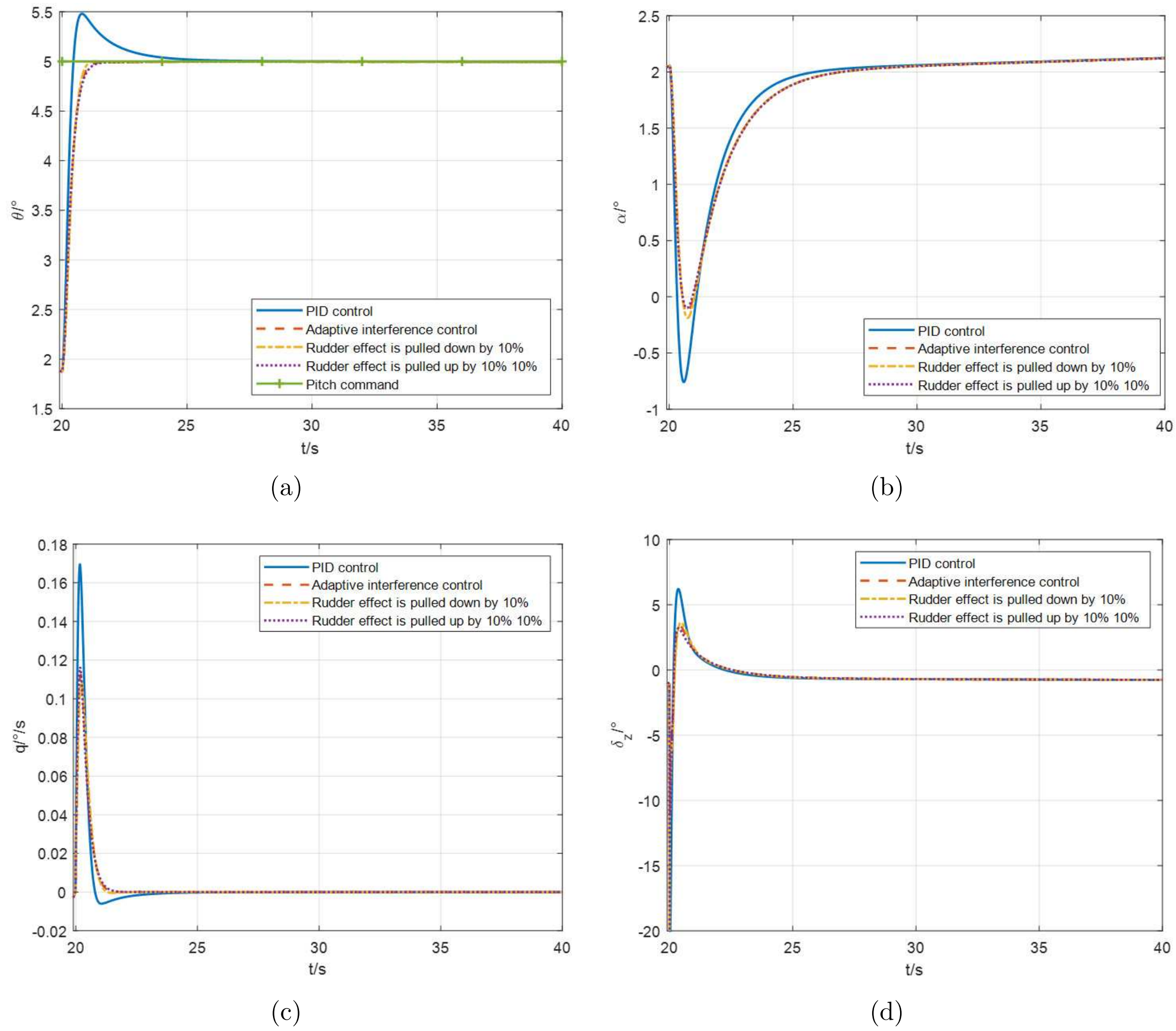


Figure 8. Simulation comparison under 50% failure condition with structural deficiency of elevator: (a) Pitch angle control curve; (b) angle of attack control curve; (c) pitch angle rate control curve; (d) elevator control curve

interference suppression control and PID control for a 50% loss of elevator effectiveness introduced to the left elevator of the aircraft simulating a 50% fault in the missing aerodynamic elevator surface of the aircraft is shown in Figure 8.

In the left elevator aircraft pneumatic actuator failure loss 50% cases tracking 5° pitch attitude command flight cases, compared with adaptive interference suppression control and traditional PID control simulation results can be found that the pitch angle convergence speed is fast 3.8 s and maximum pitching angle rate is lower 0.06°/s by using the adaptive interference suppression control. Besides, the largest amount of rudder is less 4°. And the pitch angle overshoot is smaller, decreased by about 10%. The angle of attack and pitch angle rate change radians are smaller and slower. The adjustment speed and convergence speed are faster. So the control quality and handling stability are better. Moreover, when the adaptive control is used, the control effect is still very excellent while the rudder is pulled off by 10%, which verifies the robustness of the adaptive interference suppression control.

Comparison of adaptive disturbance suppression control and PID control effects for a structural missing 70% fault state of elevator. Simulation comparison of adaptive interference suppression control and PID control for a 70% elevator efficiency loss introduced to the left elevator of the aircraft simulating a 70% fault in the missing aerodynamic control surface of the aircraft is shown in Figure 9.

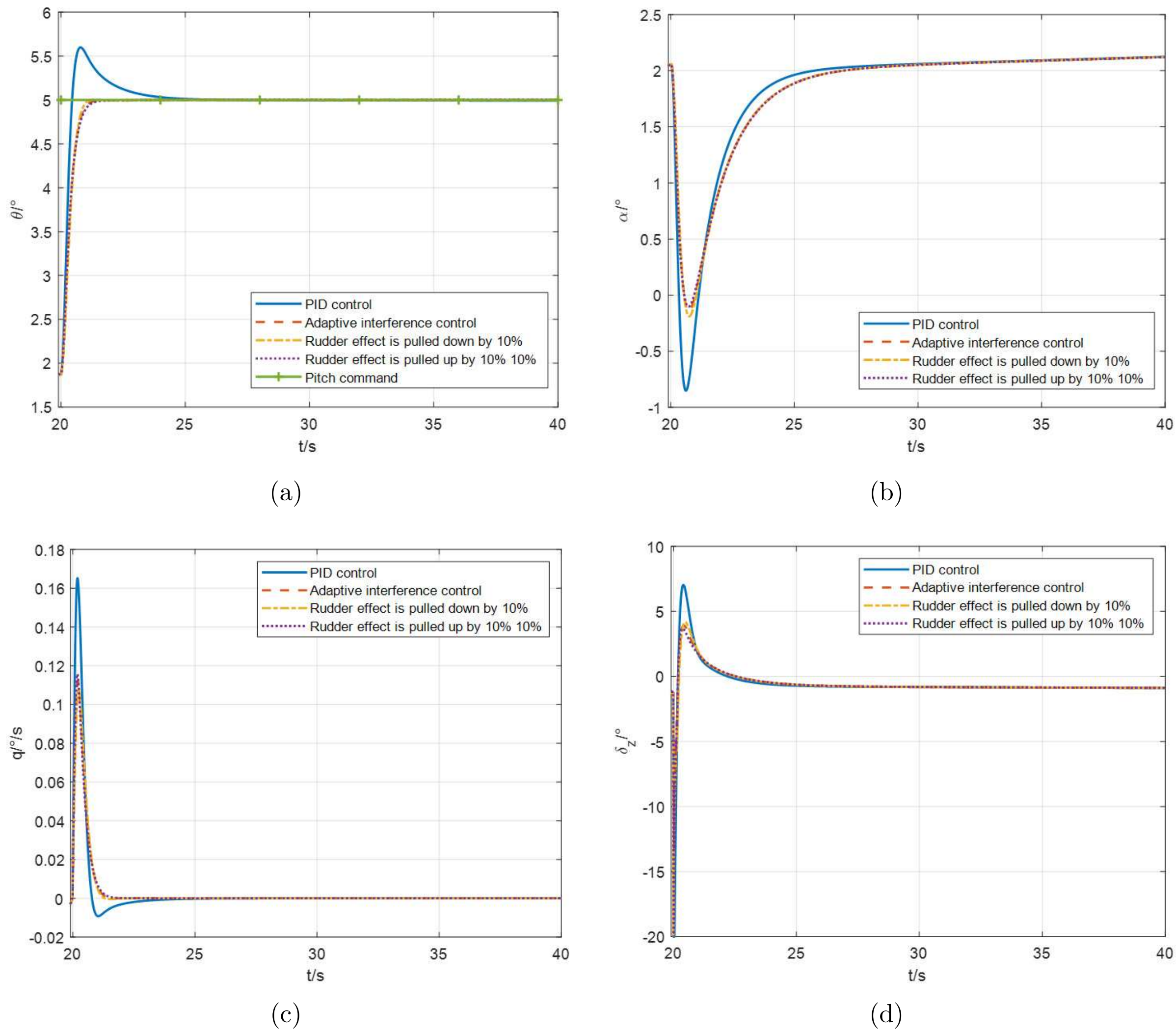


Figure 9. Simulation comparison under 70% failure condition with structural deficiency of elevator: (a) Pitch angle control curve; (b) angle of attack control curve; (c) pitch angle rate control curve; (d) elevator control curve

From the simulation results, it can be found that in the case of tracking 5° pitch angle attitude command flight with 70% loss of aerodynamic control surface failure of the left elevator of the aircraft, the adaptive disturbance suppression control has a smaller pitch angle overshoot compared with the traditional PID control, with a decrease of about 14%, less maximum out of the elevator, smaller and slower arc of angle of attack and pitch angle rate change, faster adjustment speed and convergence speed. And the pitch angle convergence speed is fast 4 s and maximum pitching angle rate is lower 0.05°/s by using the adaptive interference suppression control. So control quality and maneuvering stability of the adaptive disturbance suppression control are better. Besides, when the adaptive control is used, the control effect is still very excellent while the rudder is pulled off by 10%, which verifies the robustness of the adaptive interference suppression control.

6. **Conclusions.** In this paper, we have designed an adaptive disturbance suppression controller based on real-time recognition to solve the problem of poor control quality under the structural failure of control surface of the aircraft. Through comparative simulation analysis, it is found that the preset controller has the significantly worse control quality, larger overshoot and slower adjustment speed when the aircraft control surface is structurally missing; while the aerodynamic model used in real-time discrimination can show good prediction capability for the unused flight data in the modeling process regardless of whether the elevator is missing 50% or 70% of the time. The adaptive disturbance suppression controller using discriminative aerodynamic parameters also has a very good control quality with reduced pitch overshoot, faster command convergence, less elevator out, and less variation in angle of attack and pitch rate, verifying the effectiveness and advantages of the control method.

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REFERENCES

[1] J. G. Sun, S. M. Song and G. Q. Wu, Fault-tolerant track control of hypersonic vehicle based on fast terminal sliding mode, *Journal of Spacecraft and Rockets*, vol.54, no.6, pp.1304-1316, 2017.

[2] W. Cai, X. H. Liao and Y. D. Song, Indirect robust adaptive fault-tolerant control for attitude tracking of spacecraft, *Journal of Guidance, Control, and Dynamics*, vol.31, no.5, pp.1456-1463, 2008.

[3] B. Jiang, Z. H. Mao and H. Yang, *Fault Diagnosis and Fault Regulation of Control Systems*, National Defense Industry Press, Beijing, 2009.

[4] P. Chandler, M. Mears and M. Pachter, On-line optimizing networks for reconfigurable control, *Proc. of the 32nd IEEE Conference on Decision and Control*, vol.3, pp.2272-2277, DOI: 10.1109/CDC.1993.325602, 1993.

[5] J. M. Buffington, Tailless aircraft control allocation, *Proc. of AIAA Guidance, Navigation, and Control Conference*, New Orleans, LA, 1997.

[6] S. Juliana, Q. P. Chu and J. A. Mulder, The analytical derivation of nonlinear dynamic inversion control for parametric uncertain system, *AIAA Guidance, Navigation, and Control Conference and Exhibit*, San Francisco, California, 2005.

[7] J. H. Zhi, X. M. Dong and Y. Chen, Nonlinear finite-time fault-tolerant control assignment for multi-maneuvering surface aircraft, *Flight Dynamics*, vol.37, no.3, pp.32-37, 2019.

[8] X. D. Tang, G. Tao and S. M. Joshi, Adaptive actuator failure compensation for nonlinear MIMO systems with an aircraft control application, *Automatica*, vol.43, no.11, pp.1869-1883, 2007.

[9] J. Cai, C. Wen and H. Su, Adaptive inverse control for parametric strict feedback systems with unknown failures of hysteretic actuators, *International Journal of Robust and Nonlinear Control*, vol.25, no.6, pp.824-841, 2015.

[10] Z. T. Ding, *Nonlinear and Adaptive Control Systems*, Ph.D. Thesis, The Institution of Engineering and Technology, London, 2013.

[11] Z. B. Shi, D. Sun and T. Liu, Fault-tolerant reconfiguration design of pseudo-inverse optimal control-based flight control systems, *Journal of Civil Aviation University of China*, vol.32, no.6, pp.23-28, 2014.

[12] P. Zhu and W. H. Dong, Fault-tolerant control of multi-maneuvering surface aircraft rudder surfaces considering actuator nonlinearity, *Flight Dynamics*, vol.37, no.5, pp.51-56, 2019.

[13] L. Y. Wen, G. Tao, B. Jiang and J. Yang, Multi-model adaptive actuator fault compensation design for nonlinear dynamic mutating systems, *Acta Automatica Sinica*, vol.48, no.1, pp.207-222, 2022.

[14] H. D. Liu, Q. Z. Zhang, Y. H. Guo and J. W. Mao, Online identification of variant vehicle model parameters based on recursive least squares method, *Air & Space Defense*, vol.3, no.3, pp.103-110, 2020.

[15] S. M. Snyder, B. J. Bacon, E. A. Morelli, S. A. Frost, C. A. Teubert and W. A. Okolo, Online control design for learn-to-fly, *AIAA Atmospheric Flight Mechanics Conference*, DOI: 10.2514/6.2018-3311, 2018.

[16] M. Huang, X. Wang, Z.-M. Lu, L.-H. Ma, M. Xu and H.-Y. Su, Multiple model-based event-triggered adaptive control of a class of discrete-time nonlinear systems, *International Journal of Systems Science*, vol.50, pp.1353-1367, DOI: 10.1080/00207721.2019.1615569, 2019.
[17] M. Huang, Z. Hu and L. Wang, Optimal consensus control for heterogeneous nonlinear non-affine multi-agent systems with uncertain control directions, *ICIC Express Letters*, vol.16, no.2, pp.177-185, 2022.

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