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RESEARCH ON ATTITUDE CONTROL OF THE AIR-BREATHING HYPERSONIC VEHICLE BASED ON FADING MEMORY LEAST SQUARES

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Abstract. *The attitude control problem of the air-breathing hypersonic vehicle (AHSV) is studied. The paper illustrates the method flows of control gain adjustment strategy based on real-time aerodynamic parameter identification. Firstly, the flight dynamical model is established by considering the highly coupling relationship between the airframe and engine. Then, to estimate the aerodynamic uncertainty effected by the coupling relationship, a parameter identification method based on fading memory least squares is adopted. Furthermore, combined with the characteristic parameters of the aircraft attitude control closed-loop system, the control gain is adjusted on-line, where the low-pass filter is added to identify the data with sensor errors in real time. Finally, the comparison simulations are carried out, and the results show that the control gain adjustment strategy based on aerodynamic parameter identification can improve the attitude angle command tracking quality and the robustness of the system. In summary, for the AHSV with aerodynamic/thrust coupling characteristics, the strategy in this paper can effectively improve the quality of flight control and enhance the stability of the system.*
Keywords: The air-breathing hypersonic vehicle, Gain adjustment strategy, Real-time aerodynamic parameter identification, Closed-loop system

1. **Introduction.** The air-breathing hypersonic vehicle (AHSV) refers to an aircraft which flying Mach number is greater than 5, using a scramjet engine and its combined power, capable of maneuvering in the adjacent space at a height of 20-100 km from sea level and relying on a thin atmosphere to perform missions such as strikes, investigations, carrying and confrontation [1].

The scramjet engine has the characteristics of multi-variable, non-linear and time-varying. The main design difficulty of its control system stems from its own complex dynamic characteristics [2], that is, the aerodynamic force of AHSV, the working state of the engine, coupling and mutual influence of control system. Therefore, the coordinated control between the three aspects of attitude, speed, and thrust is particularly important. Due to the particularity of the research object itself, the methods often used in control system design are not suitable for application [3]. The aircraft body has complex cross-coupling and mutual influence in flight dynamics, aerodynamic, thermodynamics, control

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system frequency and mission requirements. Among them, [4] obtained the above characteristics by obtaining the structural modal of the finite element model and performing CFD calculations. [5,6] explained the influence of this characteristic from the control design and control performance.

The National Aeronautics and Space Administration (NASA) developed the "learn-to-fly" (L2F) method in the transforming aviation concept program (TACP), with the purpose of changing the development model of aircraft. The traditional aircraft development process includes wind tunnel testing and aero-fluid dynamics calculations (CFD), simulation development, control law design, and the final flight test sequence, and iterates continuously. Inevitably, during the flight test, the aerodynamic model needs to be continuously updated. Whenever the aerodynamic model needs to be updated, the development process will be iterated once. L2F combines real-time nonlinear aerodynamic modeling with autonomous control law design. The advantages of L2F include the use of aerodynamic models based on real-time identification of flight data in the control law design. Therefore, there is no need to modify the Reynolds number, blockage, boundary layer turbulence, etc. The control system design is developed based on the actual flight dynamics response rather than the simulation results. As L2F technology matures, new aircraft designs may not require ground testing to fly. The three pillars of L2F theory are the hot spots of research in recent years: 1) real-time nonlinear aerodynamic modeling; 2) "learning" control law design method; 3) guidance algorithm. In the L2F research, the flight test aerodynamic modeling method is combined with the real-time guidance and learning control law design method on the aircraft and applied in flight. These technologies enable the aircraft to learn how to fly [9-18].

The contribution of the work has three points:

- Complete nonlinear aerodynamic identification based on real-time flight data.
- Complete the control law design of control gain adjustment based on real-time identification results.
- Complete the comparative control simulation and got the method to solve the aerodynamic/thrust coupling problem of the AHSV.

This paper is organized as follows: Section 2 clarifies the mathematical model of AHSV pitch attitude dynamics, the process and method of aerodynamic identification and adaptive gain adjustment are in Section 3, and the results and discussions are in Sections 4 and 5.

2. **The AHSV Dynamics Model.**

The force of AHSV is divided into three categories: 1) vertical downward gravity, 2) thrust in the direction of the body axis, 3) aerodynamic force, the aerodynamic force is decomposed into a lift perpendicular to the speed, and the resistance is opposite to the speed direction. Combining the forces of AHSV in the direction of speed, combined with Newton's law, the following equations can be obtained. The system of equations describes the relationship between the force of AHSV and the displacement of the longitudinal channel and flight attitude. The control is based on this system of equations.

$$
\begin{aligned}
&mV\frac{d\theta}{dt} = T\sin\alpha + L - mg\sin\theta \\
&I_{yy}\frac{d\omega}{dt} = M \\
&\frac{d\vartheta}{dt} = \omega \\
&\alpha = \vartheta - \theta
\end{aligned}
$$

$$
\begin{aligned}
&M = M_T + M_A \\
&M_T = T\Delta z_T \\
&M_A = \bar{q}ScCm_0 + \bar{q}Sc\alpha Cm_\alpha + \bar{q}Sc\delta Cm_\delta + \bar{q}Sc\omega Cm_\omega
\end{aligned} \tag{1}
$$

where m is the mass of the AHSV; V is the current flying speed; θ is the flight path inclination; T is the thrust value; α is angle of attack; L is the aerodynamic lift received by the AHSV; g is gravitational acceleration; ω is the pitch rate; M is the total pitch moments; I_{yy} is pitch moments of inertial; M_A, M_T are aerodynamic pitch moment and thrust pitch moment; Δz_T is longitudinal eccentricity for thrust; $\bar{q}$ is dynamic pressure; S is the reference area; c is the characteristic length; Cm_α, Cm_δ, Cm_ω, Cm_0 are pitch static stability coefficient, handling coefficient, damping coefficient and natural coefficient.

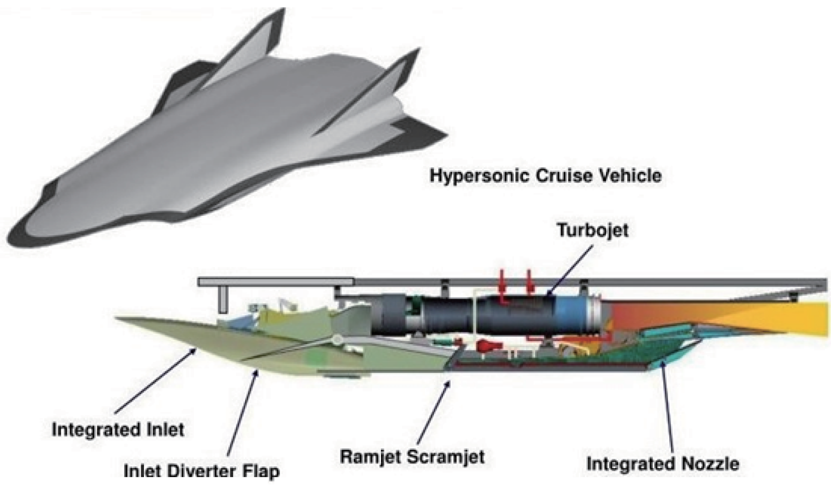

FIGURE 1. The ASHV schematic

The main influencing factors of the AHSV thrust are angle of attack, speed, and fuel equivalent ratio. Among them, when the angle of attack is between -4 and 6 degrees, the larger the angle of attack, the greater the air mass flow rate in the engine runner, so the thrust increases with the increase of the angle of attack. In the range of Mach 4-7, the thrust decreases with the increase of Mach. The change trend of thrust with fuel equivalent ratio is that when the fuel equivalent is relatively small, the thrust increases with the increase of the fuel equivalent ratio. When the fuel equivalent ratio is too large, the AHSV thermal congestion occurs and the thrust is zero. Its characteristics are detailed in [7,8].

Select the three state variables of pitch angle, pitch angle rate, and angle of attack to expand the motion equation of the aircraft's longitudinal channel with small disturbance linearization [19,20], and the linearization equation is as Formula (2)

$$
\begin{aligned}
&\Delta\dot{\vartheta} = \Delta\omega \\
&\Delta\dot{\omega} = \frac{M_\alpha}{I_{yy}}\Delta\alpha + \frac{M_\omega}{I_{yy}}\Delta\omega + \frac{M_\delta}{I_{yy}}\Delta\delta \\
&\Delta\dot{\alpha} = \Delta\omega + \left(\frac{T + L_\alpha}{mV} - \frac{g\sin\theta}{V}\right)\Delta\alpha - \frac{g\sin\theta}{V}\Delta\vartheta - \frac{L_\delta}{mV}\Delta\delta
\end{aligned} \tag{2}
$$

3. **The Adaptive Control Method Based on Fading Memory Least Squares.** Frist, the real-time parameter identification is introduced. In the process of aerodynamic identification, it is first necessary to design the excitation signal of the rudder surface. This signal is added to the actual control command as the actual signal of the aircraft

rudder surface command at the current moment, in order to make the excited flight state within the relative neighborhood of the current flight state, the following requirements are put forward for the disturbance signal: 1) the harmonic frequency range of the disturbance signal should cover the natural frequency of the open loop system of the aircraft movement; 2) the RPF value is the smallest; 3) the excitation signals acting on different channels (pitch, yaw, and roll) are orthogonal to each other. According to [9-14], the excitation signal adopts programmed test input (PTI) signal, and its mathematical expression is as Formula (3)

$$u_j = \sum_{k\in\{1,2,\ldots,M\}} A_k \sin\left(\frac{2\pi kt}{T} + \phi_k\right) \tag{3}$$

where j is the disturbance signal of the j-th channel, k is the number of harmonics of the signal, ϕ_k is the phase of the k-th harmonic, and 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. $\phi_k \in (-\pi, \pi]$. The RPF index for the j-th input signal is Equation (4)

$$RPF(u_j) = \frac{[\max(u_j) - \min(u_j)]}{2\sqrt{2}rms(u_j)} \tag{4}$$

For the identification of aerodynamic parameters, the identification equation needs to be set first, and the aerodynamic equation is established based on the aerodynamic characteristics obtained from the aircraft ground test, as Formula (5):

$$y = X\theta_m \tag{5}$$

In the formula, X is the flight state quantity matrix, y is the observation matrix, including the force and moment coefficients, and θ_m is the aerodynamic parameter matrix.

In the actual flight process, the observation matrix is usually not directly available. The solution of the observation matrix is mainly based on the data measured by the accelerometer and gyroscope. Taking the pitch moment coefficient as an example, the solution equation is as (6)

$$M = \frac{I_{yy}}{\bar{q}Sc}\left[\dot{q} + \frac{I_{xx} - I_{zz}}{I_{yy}}pr + \frac{I_{xz}}{I_{yy}}\left(p^2 - r^2\right)\right] \tag{6}$$

where p, q, r are the three-axis angular velocity in the European and American coordinate system, $\bar{q}$ is dynamic pressure, S is the reference area, c is the characteristic length, I_{xx}, I_{yy}, I_{zz}, I_{xz} are the moments of inertia corresponding to the coordinate axis.

According to the traditional least square method, the estimated value of the aerodynamic parameter matrix can be obtained, as Formula (7):

$$\hat{\theta} = \left(X^TX\right)^{-1}X^Ty \tag{7}$$

As the information continues to increase, the accuracy of the aerodynamic parameter estimates will become higher and higher and tend to be stable. This is also one of the means to check whether the estimated values are accurate. However, with the increase of observation information, the calculation workload also increases, and since each calculation requires all the amount of information, the previous calculation process is repeated continuously; in order to overcome this shortcoming, the recursive least squares method is introduced. Its basic form is as Equation (8)

$$\hat{\theta}_{j+1} = \hat{\theta}_j + K_j\left[y(j+1) - X(j+1)\hat{\theta}_j\right]$$
$$K_j = P_jX^T(j+1)\left[I + X(j+1)P_jX^T(j+1)\right]^{-1}$$

$$P_{j+1} = [I - K_j X(j+1)]P_j \tag{8}$$

The recursive method is used to replace the original X matrix inversion operation, which effectively reduces the amount of calculation.

In the actual situation, the continuous expansion of the information matrix leads to a gradual decrease in its positivity, which makes the new flight state have little effect on parameter estimation. If the aircraft or the flight state is extremely unstable, the identification error will increase. So, based on recursive least squares, adding a forgetting factor to reduce the weight of the previous information matrix, then the identification iteration equation is changed to Formula (9).

$$\begin{aligned} \hat{\theta}_{j+1} &= \hat{\theta}_j + K_j \left[y(j+1) - X(j+1)\hat{\theta}_j\right] \\ K_j &= P_j X^T(j+1) \left[\lambda I + X(j+1) P_j X^T(j+1)\right]^{-1} \\ P_{j+1} &= \frac{1}{\lambda}[I - K_j X(j+1)]P_j \end{aligned} \tag{9}$$

where λ is the forgetting factor.

In actual use, the dimension of the X matrix can also be limited. When the dimension overflows, a new set of data is removed, and a set of old data is removed. This not only saves space, but also maintains the calculation accuracy [21-25].

For the filtering of sensor error, a first-order low-pass filter is used, and its digital form is as Equation (10)

$$y(n) = t_0 \bar{y}(n) + (1 - t_0) y(n-1) \tag{10}$$

Among them, t_0 is the time constant, and $y(n)$ is the filtered value, and $\bar{y}(n)$ is the unfiltered value.

According to the cut-off frequency of the first-order low-pass filter, after the sampling time is determined, t_0 can be calculated according to the selection of the cut-off frequency. The criterion for the selection of the cut-off frequency is to ensure that the excitation signal is in the passband and the high-frequency error is in the stopband.

Then, the adaptive control method is explained. According to [26,27], the aircraft longitudinal channel usually adopts the PD control form. The addition of integral will affect the bandwidth of the closed-loop system. The parameter setting is mainly based on the aircraft linearization equation and the system characteristic equation.

Rewrite Equation (2) into the expression form of state space as Equation (11)

$$\begin{bmatrix} \Delta\dot{\vartheta} \\ \Delta\dot{\omega} \\ \Delta\dot{\alpha} \end{bmatrix} = \begin{bmatrix} 0 & 1 & 0 \\ 0 & \dfrac{M_\omega}{I_{yy}} & \dfrac{M_\alpha}{I_{yy}} \\ -\dfrac{g\sin\theta}{V} & 1 & \left(\dfrac{T + L_\alpha}{mV} - \dfrac{g\sin\theta}{V}\right) \end{bmatrix} \begin{bmatrix} \Delta\vartheta \\ \Delta\omega \\ \Delta\alpha \end{bmatrix} + \begin{bmatrix} 0 \\ \dfrac{M_\delta}{I_{yy}} \\ -\dfrac{L_\delta}{mV} \end{bmatrix} \Delta\delta \tag{11}$$

1) According to the small disturbance linearization method, linearized expansion is carried out for the state of motion in the level flight state, so the trajectory inclination is about 0;

2) According to the analysis of the AHSV aero data, it is found that when the aircraft is in hypersonic high-altitude flight, the cl_α is basically constant, and the magnitude of the elements in the matrix is very small, about 10^{-2}. The magnitude of other elements is about 10^1.

Based on the above two points, the following assumptions are given as (12):

$$\theta \approx 0, \quad \frac{T + L_\alpha}{mV} - \frac{g\sin\theta}{V} \approx 0 \tag{12}$$

Simplify the above equations to get the characteristic equation of aircraft attitude control as Equation (13):

$$s\left[s^2-\frac{M_\omega}{I_{yy}}s-\frac{M_\alpha}{I_{yy}}\right]=0 \tag{13}$$

According to the second-order system, Equation (14) is obtained.

$$\begin{aligned} 2\xi'\omega'_n &= -\frac{M_\omega}{I_{yy}} \\ \omega'^2_n &= -\frac{M_\alpha}{I_{yy}} \end{aligned} \tag{14}$$

The form of PD control law is as follows:

$$\Delta\delta = K_p\Delta\delta + K_d\Delta\omega \tag{15}$$

Assuming that the partial derivative of the rudder deflection to the lift is small, that is $\frac{L_\delta}{mV}\approx 0$, then the system characteristic matrix is as Formula (16)

$$A_c=\begin{bmatrix} 0 & 1 & 0 \\ \frac{M_\delta}{I_{yy}}K_p & \frac{M_\omega}{I_{yy}}+\frac{M_\delta}{I_{yy}}K_d & \frac{M_\alpha}{I_{yy}} \\ 0 & 1 & 0 \end{bmatrix} \tag{16}$$

The characteristic equation for closed-loop control of the aircraft is

$$s\left[s^2-\left(\frac{M_\omega}{I_{yy}}+\frac{M_\delta}{I_{yy}}K_d\right)s-\left(\frac{M_\delta}{I_{yy}}K_p+\frac{M_\alpha}{I_{yy}}\right)\right]=0 \tag{17}$$

The natural frequency and damping of the closed-loop system can be obtained as the following Formula (18).

$$\begin{aligned} \omega_n^2 &= -\left(\frac{M_\delta}{I_{yy}}K_p+\frac{M_\alpha}{I_{yy}}\right) \\ 2\xi\omega_n &= -\left(\frac{M_\omega}{I_{yy}}+\frac{M_\delta}{I_{yy}}K_d\right) \end{aligned} \tag{18}$$

K_p, K_d can be calculated by configuring the desired system characteristics.

Because the controlled object has a strong aerodynamic/thrust coupling, which is quite different from the characteristics of the aircraft, it is considered to add an integral term in the control law.

The gain adjustment method is based on the above-mentioned parameter calculation method, using the relevant aerodynamic parameters identified in real time and the expected system characteristics, and K_p, K_d can be iteratively updated in real time. The iterative process is as Figure 2.

In the figure, $\bar{\omega}$, $\bar{\xi}$ are the characteristic parameters of the closed-loop system, and λ is the iteration factor; ε is the small interval of iterative update. The main idea of the calculation is the inverse calculation method of the above-mentioned calculation of the characteristic parameters of the second-order system.

The process shown in Figure 2 can be explained as: first, perform aerodynamic identification based on the flight data at the current time. After the identification parameters are obtained, the expected error and differential gain are calculated based on the closed-loop characteristics of the secondary system. Then, enter the gain error judgment. The error gain and the differential gain are performed separately. If the difference between the current gain and the expected gain is less than a certain range, the current control

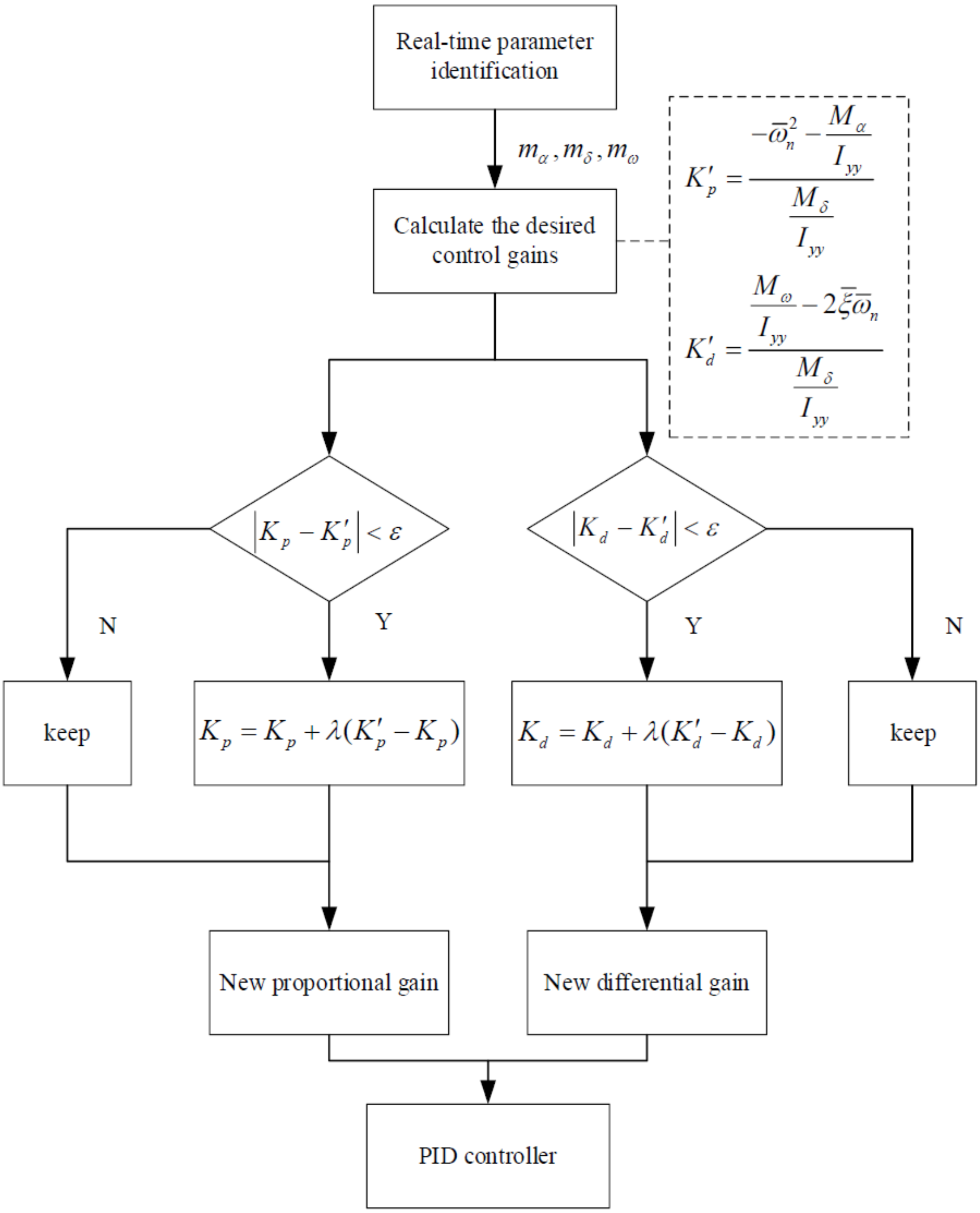

FIGURE 2. Gain update iteration flow chart

quality is considered acceptable and no parameter update is performed; if it exceeds the acceptable range, the first-order transition is used to update the parameters to obtain a new gain.

Remark 3.1. *According to the derivation of the transfer function and configuration method of the above-mentioned aircraft closed-loop system, the natural frequency and damping ratio of the second-order system are configured at a certain point, so the real part of the zero and poles of the system is less than 0, so it can be obtained that the closed-loop system is stable.*

4. **Simulation Results.** Based on the above model, the simulation is carried out. The dynamic parameters are shown in Table 1.

Table 1. The ontological parameters of AHSV

Variable	Description	Value
c	Characteristic length of AHSV	1.5 m
S	AHSV reference area	21 m^2
I_{yy}	Pitching moment of inertia	973470 Nm
m	Mass of AHSV	42094 kg

The following is calculated according to the method in Section 3.

Generally, the AHSV flight altitude is greater than 20 kilometers, and the flight speed is greater than Ma4. Through calculation, the initial simulation conditions are 22 km, Ma5. The calculated dynamic pressure is 70 Kpa. At this time, it is calculated $M_\alpha/I_{yy} = 13.6$. Therefore, it can be estimated that the natural frequency of the AHSV itself is 3.68 rad/s, so in the design of excitation signal excitation $T = 10$.

For aircraft with a natural frequency of 3.68 rad/s, it is necessary to design the number and phase of the excitation signal harmonics, choose four different frequency harmonics as the input excitation, and use Equation (4) to optimize the solution. In Table 2 phase optimization results can be obtained.

Table 2. Phase iterative solution results of PTI

Input	k	RPF/A	Phase
u_1	$2, 4, 6, 8$	0.9900	2.8274, 2.1991, 0, 1.8850
u_2	$3, 9, 12, 18$	1.0519	1.4137, 1.0996, 0.9425, −2.5133
u_3	$(5, 10, 15, 20)/2$	0.9946	−1.8850, 2.0420, −1.7279, −1.5708

According to [9-18], the injection of the excitation signal will not damage the control quality of the closed-loop feedback system. Similarly, the feedback will not affect the identification accuracy. At the same time, let the input excitation $A = 5$. With a simple filter design, a certain identification accuracy can also be maintained. The sampling time in the simulation is 0.01 s. According to the excitation frequency and the random error frequency of the sensor, the cut-off frequency of the first-order low-pass filter should be greater than 0.8 Hz and less than 100 Hz. After debugging, $t_0 = 0.07$ can get a relatively satisfactory filtering effect.

Analyzing the aerodynamic data of the AHSV to know that $M_\delta/I_{yy} = -8$, $M_\omega/I_{yy} \approx 0$, it can be determined according to Section 3 method, in this case which $\omega_n = 10$, $\xi = 0.8$. According to Formula (18), the gain can be calculated that $K_p = 10.8$, $K_i = 1.2$, $K_d = 2$ (The gain coefficient of the integral term is 1.2. In the case of less impact on the frequency domain characteristics of the system, it can meet the control accuracy requirements). Take the small interval of iterative update as $\varepsilon = 0.02$.

A −10% uncertainty effect of elevator effect is added to the control program to simulate the tracking ability of the attitude angle command. The pitch angle control command is as the following (19)

$$\vartheta_c = \begin{cases} 6^\circ & 0 < t < 25 \\ 2^\circ & t > 25 \end{cases} \tag{19}$$

The control result of adaptive gain adjustment (GA) is compared with the conventional control result as follows.

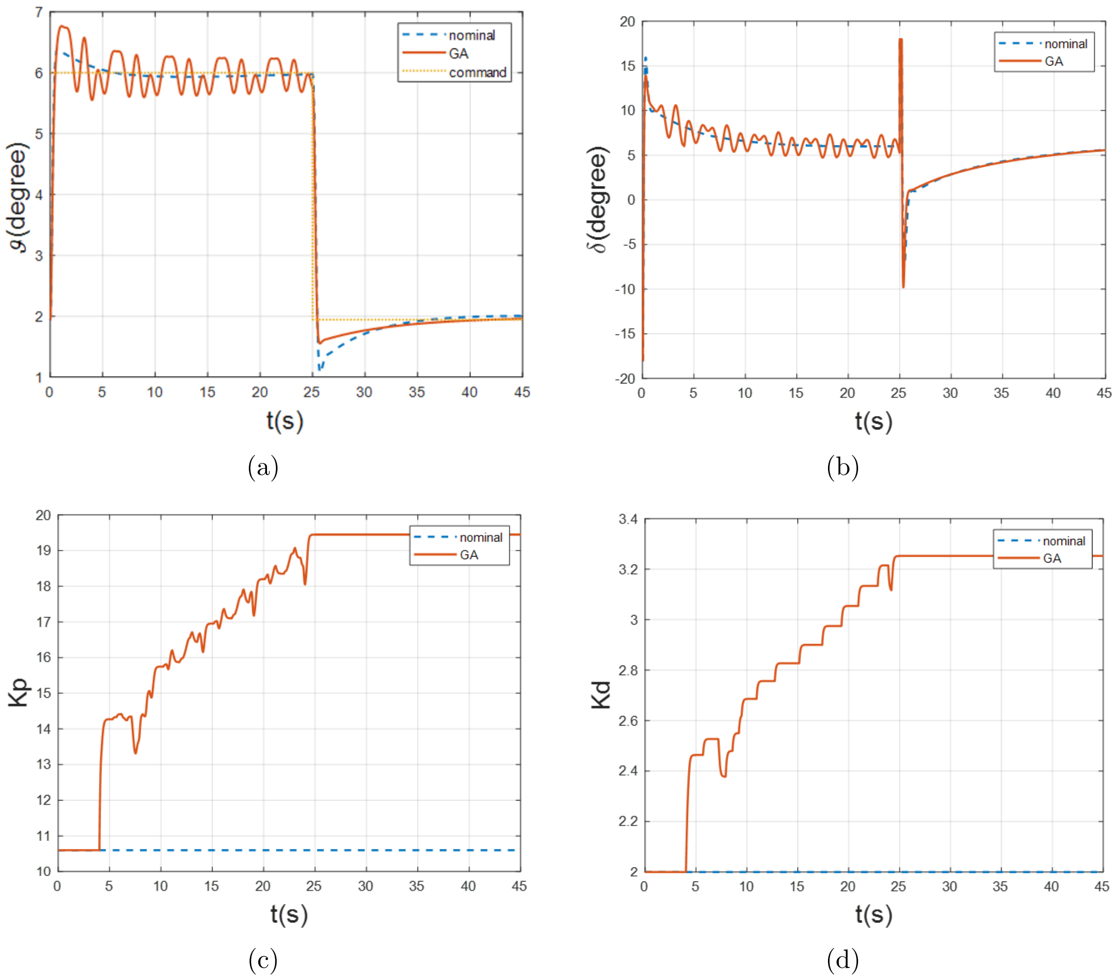

FIGURE 3. Adaptive gain adjustment control simulation curve of AHSV: (a) Pitch angle control curve; (b) elevator control curve; (c) proportional gain adjustment curve; (d) differential gain adjustment curve

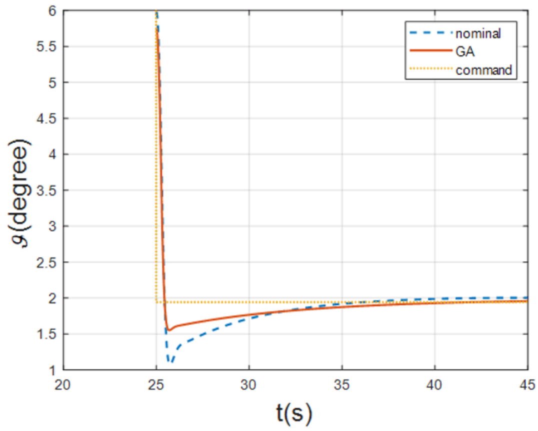

FIGURE 4. Pitch angle control curve after 25 seconds

From the simulation results in Figures 3 and 4, in the first 25 s, due to the addition of disturbances, the elevator deflection command oscillates, but due to the closed-loop control, the actual oscillation amplitude is less than 5. During the excitation process, aerodynamic parameter identification and gain adjustment are performed in real time. After 25 s, the excitation disappears, and then the identification stops. It can be seen from the pitch angle tracking curve that the static error and overshoot are reduced, the stabilization time is basically unchanged, and the control quality is significantly improved. This is because, due to the changes in flight status and AHSV attitude, the closed-loop system characteristics change greatly. The adaptive control gain adjustment method can effectively adjust the gain correction controller according to the current flight status and closed-loop characteristics.

Through calculation, the control quality after adjusting the gain, from the time domain analysis, the overshoot is 10%, the steady state error is 0, and the control effect of the unadjusted gain, the overshoot is 26%, and there is a small steady state error. According to the control theory, it can be calculated that the adjusted damping ratio is about 0.6, and the damping ratio before adjustment is 0.39. According to the elevator deflection curve, it can be known that the elevator deflection has reached the given limit when the system is adjusted. The introduction of the limiting function will inevitably cause the error between the system characteristics and the expected characteristics. This is the reason why the damping ratio of the actual control curve is not near 0.8.

Considering the random error of the $0.05°$ deflection signal, its control simulation is as Figure 5.

5. **Discussion and Conclusions.** This method uses the idea of learning to fly proposed by NASA, combines real-time aerodynamic identification with adaptive gain adjustment, and studies the AHSV attitude control with severe aerodynamic/thrust coupling. Currently, most of the controllers for AHSV attitude control are fixed, such as sliding mode controllers [28-31], and LQR controllers [32,33]. The method in this paper is based on the flight status and identification of the dynamic adjust controller gains. Future research directions can be divided into the following points.

1) Since the attitude control of the aircraft has strong real-time performance, and the aerodynamic recognition algorithm is a fitting algorithm, for this application environment, the optimization of the recognition algorithm should aim at real-time requirements. Under the premise of satisfying real-time performance, consider the accuracy of identification.

2) In terms of application objects, the method is applied to the research of wide-speed range aircraft attitude control.

3) More intelligent research, considers the use of intelligent methods, combined with aerodynamic parameter identification, to update the control gain and even the control configuration in real time.

The conclusion of this article is as follows.

In this paper, the PTI excitation signal is injected into the control surface to obtain real-time flight data. After the flight data is passed through a first-order low-pass filter, the fading memory least square method is used to identify the relevant parameters of the aerodynamic model. The aerodynamic parameters are substituted into the characteristic equation of the aircraft attitude control closed-loop system, and the control gain can be adjusted according to the desired system characteristic parameters to ensure the control performance of the controller. Meanwhile, the stability of the system is remarked simply. By using this method, the longitudinal attitude control simulation of the aircraft is completed. The results show that in practical applications, the identification method is used to obtain the characteristics of the system, and according to the idea of pole configuration, the way to achieve adaptive attitude control is feasible. This approach preliminarily solves the problem of serious thrust/aerodynamic coupling that causes difficulty in designing the control law.

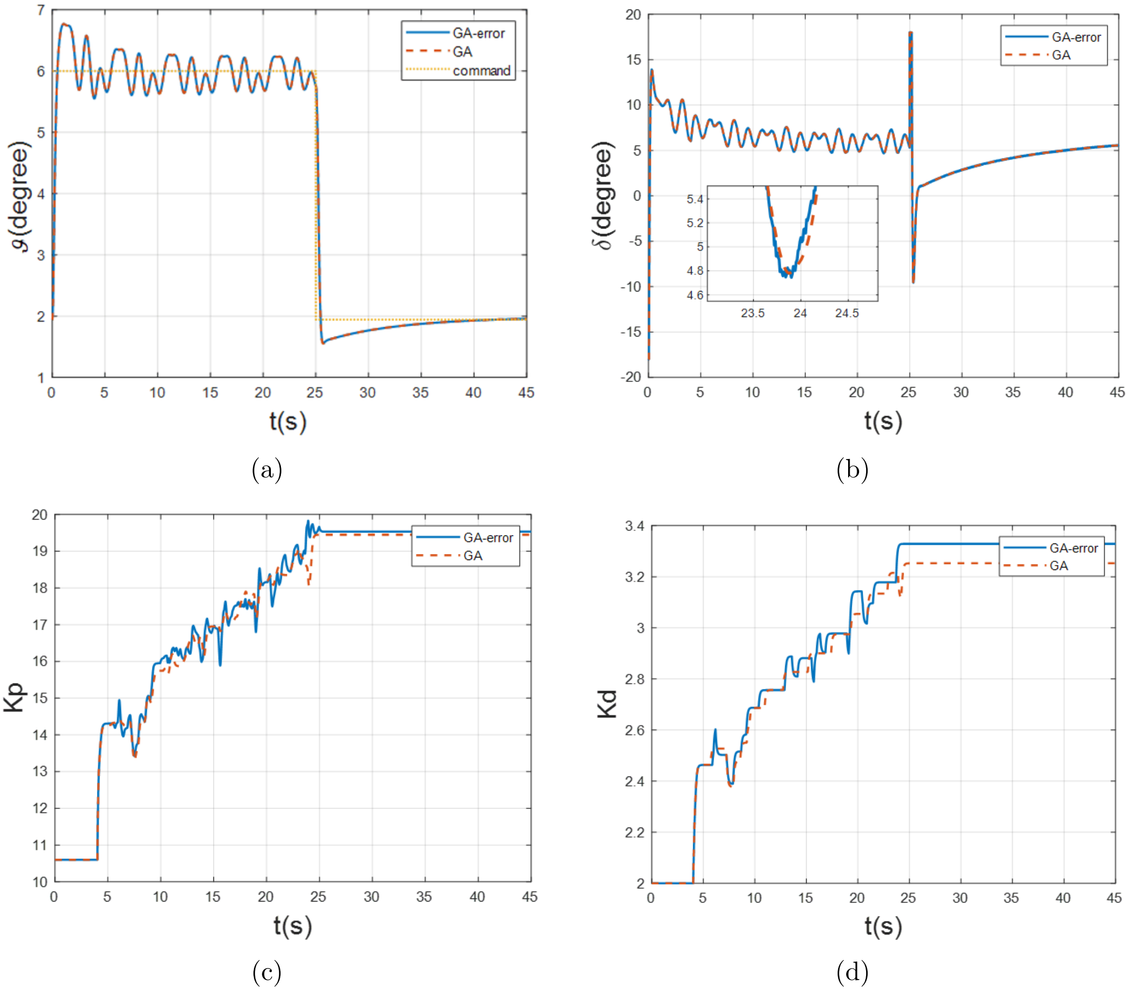

FIGURE 5. GA control simulation considering sensor error: (a) Pitch angle control curve; (b) elevator control curve; (c) proportional gain adjustment curve; (d) differential gain adjustment curve

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