Design of a Fixed-Wing UAV Controller based on Adaptive Backstepping Sliding Mode Control method

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ABSTRACT In this paper, an advanced control method is proposed for a fixed-wing unmanned aerial vehicle (UAV) to maintain the stabilization of its altitude, attitude, and velocity. The mathematical model of a fixed-wing UAV is very complicated because of its characteristics of nonlinearity and large extent of multi-variable coupling. Thus, to design the relevant controller is also difficult. In addition, during the operation of a fixed-wing UAV, various uncertainties and disturbances will occur, which will make the control process harder to accomplish. To solve these problems, this study designs a variable-structure controller with multiple algorithm fusion. The design mainly adopts the backstepping sliding mode control method to simplify the complex nonlinear mathematical model, and an adaptive law is introduced to estimate the uncertainty and disturbance of the system. Subsequently, the tracking error of the controller is proved to converge to zero using Lyapunov's second method. Finally, it is verified that the controller has the ability to stably control a fixed-wing UAV by numerical simulation and can overcome the disturbance and uncertainty. The buffeting also can be eliminated by an adaptive law.

INDEX TERMS Fixed wing UAVs, Adaptive law, Backstepping sliding control, Uncertainty, Disturbance, Buffeting phenomenon

I. INTRODUCTION

Recently, the issue of unmanned aerial vehicles (UAVs) has become an increasingly popular topic. Until now, this technology has played an irreplaceable role in both military and civil applications [1]. Currently, UAVs are mainly divided into two categories: multirotor UAVs and fixed-wing UAVs. Both categories are widely used in different fields, according to their different characteristics. The multirotor UAVs mainly includes quadrotor and crossover twin-rotor UAVs. Quadrotor UAVs are mainly used to complete some tasks with high requirements for maintaining attitude, altitude and position, such as detailed terrain exploration and short-range transportation of small weight materials. This is because of their small size, flexible attitude, and the ability of hover. However, the limitations ofairspeed and altitude exist. Crossover twin-rotor UAVs are mostly used to accurately drop heavy loads because of their special physical structure. Their carrying capacity is much stronger than that of quad-rotor UAVs, but they also have flight speed and altitude limitations. Fixed-wing UAVs are aircraft driven by aerodynamic forces acting on a fixed surface, which are often used to accomplish tasks that require high altitude and high speed because of their fast flight speed, long flight distance and strong load capacity characteristics.

Although fixed-wing UAVs have many advantages, designing control systems to ensure good performance is not an easy task. This is because accurate models are needed in the design of controllers, but the models of fixed-wing UAVs are very complex owing to the uncertainty, nonlinearity, and coupling of multiple state quantities. Extensive research has been conducted on the modeling process of fixed-wing UAVs [2,3], but all of them can only work out approximate mathematical and dynamic models. When the fixed-wing UAV are analysed, it is necessary to make a lot of assumptions in advance, such as taking the UAV as rigid body and the constant ground acceleration, etc. This is that makes the difference between the actual model and the ideal model, which is also the main reason for the uncertainty of system parameters.

To date, the most common control method for fixed-wing UAVs is the traditional PID control. PID control algorithm has the superior characteristics of simple control principle, easy to modify parameters and low requirements for model accuracy. However, this algorithm has high requirements on the selection of control parameters, which requires a large amount
of data experimental testing. Due to the uncertainty in the system, the control parameters obtained by numerical simulation are not always feasible in the actual situation. Thus, a lot of practical experiments are inescapable to optimize the control parameters. This is a complicated and lengthy process. Furthermore, the system of fixed-wing UAVs is a complex nonlinear system, which cannot be directly controlled by traditional PID without a tedious linearization process.

Owing to the significant complexity of the linearization process, to control the fixed-wing UAV accurately, some robust nonlinear control methods can be introduced to avoid the linearization process. The current popular robust control methods include adaptive control [4,5], neural control [6], linear quadratic control (LQR) [7], linear tracking control [8], dynamic inversion control [9], nested saturation control [10,11], feedback linearization control [12,13], backstepping technique [14,15,16], Sliding mode variable structure control [16,17], etc.

Sliding mode variable structure control is currently one of the mainstream robust control laws, which is mainly aimed at solving the control problem of nonlinear models. The fundamental difference between it and conventional control lies in the discontinuity of control, which means that the switching characteristics of the system structure will change with time. Sliding mode motion can force the system to move up and down with a small amplitude and high frequency along the specified state trajectory under certain characteristics. However, this principle also directly brings the chattering to the system. This phenomenon refers to that the controlled system will vibrate around the sliding surface rather than directly stopping on the surface when it operates under the action of the control law. It poses risks to the stability of the system. At present, how to overcome the chattering problem has become the focus of many scholars.

Several studies have proposed variations in the related aircraft control. Chaoan et al. [18] introduced an adaptive gain sliding mode control scheme based on a multivariable finite-time observer for a fixed-wing unmanned aircraft (UAV) with unmeasured angular velocity and unknown mismatched disturbances. For the attitude subsystem, a multivariable finite-time observer (MFO) was constructed to estimate the unknown state. Using the estimates provided by the MFO, a new adaptive dual-layer continuous terminal sliding mode (ADL-CTSM) controller was proposed to track the reference attitude instructions in a limited time. The experimental results indicate that the controller can effectively eliminate chattering without a known interference boundary and has good control performance. Espinoza et al. [19] designed several backstepping sliding mode controllers, including a backstepping sliding mode controller, backstepping double sliding mode controller, and backstepping high-order sliding mode controller (HOSM), to solve the attitude control problem of fixed-wing UAVs and compared their control performance. The experimental data proved that the HOSM has the best performance and can effectively suppress chattering. Cantañeda et al. [20] provided robust second-order adaptive sliding mode controllers for the attitude and airspeed of fixed-wing UAVs. The controllers could approach the control targets without overestimating the control gain and the known upper bound of the disturbance. The good performance of resisting disturbances and overcoming uncertainty was verified experimentally. The experimental results showed that these controllers can complete the control work well and require less control effort in a limited time. Melkou et al. [21] improved the second-order sliding mode control and achieved maintenance control of the altitude and attitude of a fixed-wing UAV. The stability and effectiveness of the controller were also proved by experiments, and the related gain adaptation laws reduced the tremor. Qi et al. [22] presented a quality-driven fixed-wing UAV rather than a traditional aerodynamic-driven UAV and controlled it using a method based on adaptive sliding mode control. The designed controller is a type of adaptive sliding mode controller that integrates a fuzzy system, an RBF neural network, and sliding mode control. The experiments showed that the controller is robust and effective. Kun et al. [23] addressed the problem of fixed-wing UAV control by designing a nonsingular fast terminal sliding mode control law. Singularity was avoided by obtaining a modified saturation function near zero. Three different states of moving objects were used for numerical simulation. The simulation results verified the finite-time convergence and stability of the controller. Xiangke et al. [24] proposed a new distributed sliding mode control law to overcome the difficulty of master-slave formation flight with velocity constraints. Driven by the designed control law, the desired formation was achieved, and the linear and angular velocity constraints of the UAV were satisfied. Finally, the results were verified using a numerical simulation. Yang et al. [25] proposed a multivariable time-sliding mode interference observer for UAV attitude and airspeed control problems. A new vector control structure was proposed that extended the traditional three-channel vector control. The experiment proved the robustness and stability of this method. Zewei et al. [26] solved the control problem of the relative motion model of a fixed-wing UAV aircraft carrier landing using a backstepping sliding mode controller. According to the underactuated characteristics of the two vehicles, the six DOFs of the model were simplified to four DOFs for control. An adaptive backstepping sliding mode controller was used to control the UAV to track the expected landing international and maintain a constant support and roll angle. These experiments support the feasibility and effectiveness of the proposed method. Xueying et al. [27] applied an adaptive inverse sliding mode controller to accomplish the attitude and position control process of a six-degree-of-freedom quadrotor UAV. Internal and external disturbances were estimated using adaptive laws and compensated for the system. Fusion backstepping sliding mode control reduced the requirement of the traditional six-rotor controller design for an accurate model and upper bound of the disturbance. The experiment claimed that the method can effectively and stably complete the control process of a quadrotor UAV.
In this study, a control method combining adaptive method and backstepping sliding modes is proposed to control a fixed-wing UAV. Compared with the existing sliding mode control algorithms, this algorithm can solve the chattering problem while overcoming the uncertainties of model parameters and external disturbances. Because the fixed-wing UAV often works in a state of high moving speed, the precise control of its displacement is of little significance. This design mainly involves the control design of the UAV’s attitude, altitude, and flight speed. The dynamic and mathematical models of fixed-wing UAVs are extremely complex. To make it closer to a real situation, it is necessary to consider the coupling effect between different variables. In the controller design process, the part of strong coupling should be carefully analyzed and decoupled, and the controller should be designed for different state variables. The adaptive law is used to estimate the uncertainties and external disturbances of the system and compensate them to the system. In other words, the introduction of adaptive law can help the controller maintain good control performance even when the parameters of the controlled system vary owing to uncertainties. Furthermore, the adaptive law can also be used to eliminate the chattering. The backstepping design method decomposes the complex nonlinear system into subsystems that do not exceed the order of the system, and then introduces the Lyapunov function and the intermediate virtual control for the subsystems. This method is very effective in solving the complex model control problem of fixed-wing UAVs. In this study, the adaptive, backstepping, and sliding mode methods were combined to design the controllers for a fixed-wing UAV. The remainder of this paper is organized as follows: Section II introduces the kinematics model and dynamics model of fixed-wing UAVs. Section III presents the design of the adaptive backstepping sliding mode control laws and analyzes their stability. Section IV simulates the control system using the MATLAB software. The conclusions are presented in section V.

II. Establishment of UAV mathematical model
Owing to the complexity of the movement of the fixed-wing UAV, the following assumptions are required before the establishment of the dynamics and kinematics models [28,29].

1. The rotation of the earth is stagnant, and the curvature of the earth is zero.
2. The body of a fixed-wing UAV is considered rigid and does not deform or vibrate owing to changes in the forces.
3. The drone fuselage was perfectly symmetrical with respect to the central axis plane.

A. Multiple coordinate system establishment
A total of 12 state variables were required for the force and motion description of a fixed-wing UAV. There are three displacement state variables with their velocity state variables and three angle state variables with their angular velocity state variables. These 12 state quantities cannot be obtained from a single coordinate system; therefore, it is necessary to establish the following auxiliary coordinate system [28,29]:

1. Ground inertial coordinate system \( (O_GX_GY_GZ_G) \): The starting position of the UAV is the origin of coordinates, with the \( X_G \)-axis pointing to the north, the \( Y_G \)-axis pointing to the east, and the \( Z_G \)-axis pointing to the Earth’s core.
2. Body inertial coordinate system \( (O_BX_BY_BZ_B) \): The center of mass of the UAV, while the \( X_B \)-axis, \( Y_B \)-axis, and \( Z_B \)-axis are parallel to the ground inertial coordinate system and point to due north, due east, and the Earth’s core, respectively.
3. Body coordinate system \( (O_BX_BY_BZ_B) \): The center of mass of the UAV is the origin of the coordinate system, with the \( X_B \)-axis pointing to the nose, \( Y_B \)-axis pointing to the belly, and \( Z_B \)-axis pointing to the right wing.
4. Stable coordinate system \( (O_SX_SY_SZ_S) \): The center of mass of the UAV is the origin, while the \( X_S \)-axis points to the direction of the projection of the motion vector of the UAV in the longitudinal symmetric plane, the \( Z_S \)-axis points to the right wing of the UAV, and the \( Y_A \)-axis is determined by the right-hand rule.
5. Airflow coordinate system \( (O_AX_AY_AZ_A) \): The origin is the center of gravity of the UAV, and the \( X_A \)-axis points to the direction of the UAV’s motion speed. The \( Z_A \)-axis is vertically downward with the \( X_A \)-axis in the longitudinal symmetric plane of the UAV, and the \( Y_A \)-axis is determined by the right-hand rule.

By designating the transformation matrix according to \( \Psi \), \( \theta \), \( \Phi \), \( \alpha \), and \( \beta \), the relationship between the coordinate systems can be determined. \( \Psi \) is the yaw angle, \( \theta \) is the pitch angle, \( \Phi \) is the roll angle, \( \alpha \) is the attack angle, and \( \beta \) is the sideslip angle. According to these transformations, the transformation matrix can be established as follows [30]:

The transformation matrix between \( O_{BI}X_{BI}Y_{BI}Z_{BI} \) and \( O_{B}X_{B}Y_{B}Z_{B} \) is

\[
O_{B}X_{B}Y_{B}Z_{B} = \\
\begin{bmatrix}
\cos\Psi \cos\Theta & \cos\Psi \sin\Theta & -\sin\Psi \\
\sin\Theta \sin\Phi - \cos\Theta \cos\Phi \sin\Psi & \sin\Theta \sin\Phi + \cos\Theta \cos\Phi \sin\Psi & \cos\Theta \cos\Phi \sin\Psi \\
\cos\Theta \sin\Phi + \sin\Theta \sin\Phi \cos\Psi & \cos\Theta \sin\Phi - \sin\Theta \sin\Phi \cos\Psi & \sin\Theta \cos\Phi \cos\Psi
\end{bmatrix}
\]

(1)

The transformation matrix between \( O_{B}X_{B}Y_{B}Z_{B} \) and \( O_{A}X_{A}Y_{A}Z_{A} \) is

\[
O_{A}X_{A}Y_{A}Z_{A} = \\
\begin{bmatrix}
\cos\Theta \cos\Phi & \sin\Theta \cos\Phi & 0 \\
\sin\Theta \sin\Phi & \cos\Theta \sin\Phi & 0 \\
-\sin\Theta & \cos\Theta & 0
\end{bmatrix}
\]

(2)

B. Dynamics model and kinematics model
It is necessary to determine the definitions of the 12 state quantities of the UAVs. These 12 state variables were defined in terms of the ground coordinate system \( (O_GX_GY_GZ_G) \). \( P_x, P_y, P_z \) are the displacement states of the three axes. \( u, v, w \) represent the velocity of the displacement along the three axes. The Euler angle \( (\Psi, \theta, \Phi) \) is the angle at which the UAV body rotates on three axes, and their rate of change is denoted by \( p, q, r \).
Through the transformation matrix established in Section 2.2, the kinematics model of the fixed-wing UAV can be easily obtained [30].

\[
\begin{pmatrix}
    \dot{p}_x \\
    \dot{p}_y \\
    \dot{p}_z
\end{pmatrix} = 
\begin{pmatrix}
    \cos \Phi \cos \Theta & \sin \Phi & 0 \\
    -\sin \Phi \cos \Theta & -\cos \Phi & 0 \\
    -\sin \Theta & 0 & 1
\end{pmatrix}
\begin{pmatrix}
    u \\
    v \\
    w
\end{pmatrix}
\] (3)

\[
\begin{pmatrix}
    \dot{\psi} \\
    \dot{\theta} \\
    \dot{\phi}
\end{pmatrix} = 
\begin{pmatrix}
    1 & \sin \Phi \tan \Theta & \cos \Phi \tan \Theta \\
    0 & \cos \Phi & -\sin \Phi \\
    0 & \sin \Phi \sec \Theta & \cos \Phi \sec \Theta
\end{pmatrix}
\begin{pmatrix}
    p \\
    q \\
    r
\end{pmatrix}
\] (4)

In the process of establishing the dynamic model, the calculation is carried out mainly according to Newton’s second law. The core formulae are as follows:

\[ F = \frac{d}{dt} mV \] (5)

\[ M = \frac{d}{dt} H \] (6)

where \( F \) is the accumulation of external forces, \( m \) is the mass of UAV, \( M \) is the accumulation of torques, \( H \) is the angular momentum.

By expanding these two formulas according to the characteristics of the fixed-wing UAV, the dynamic model can be obtained as follows [29,30]:

\[ \begin{pmatrix}
    \dot{u} \\
    \dot{v} \\
    \dot{w}
\end{pmatrix} = 
\begin{pmatrix}
    -qw + rv \\
    -ru + pw + \frac{1}{m} f_y \\
    -pv + qu + \frac{1}{m} f_z
\end{pmatrix} \] (7)

where \( f_x, f_y, f_z \) are the forces of the UAV along the axis in the ground coordinate system.

\[ \begin{pmatrix}
    \dot{T}_1 \\
    \dot{T}_2 \\
    \dot{T}_3
\end{pmatrix} = 
\begin{pmatrix}
    T_1 p q - T_2 q r + T_3 r t_1 + T_4 t_3 \\
    T_2 p r - T_3 r q + \frac{1}{m} f_y t_2 \\
    T_3 p q - T_2 q r + T_3 r t_4 + T_4 t_3
\end{pmatrix} \] (8)

where \( t_1, t_2, t_3 \) are the projections of the torque on the three axes in the body coordinate system.

\[ j_x = \int (x^2 + y^2) dm, \quad j_y = \int (x^2 + z^2) dm, \quad j_z = \int (y^2 + z^2) dm, \quad j_{xy} = \int xyz dm, \quad j_{xz} = \int yxdm, \quad j_{yz} = \int yxdm, \quad j = \int j_{xy} \] (9)

\[ T = \frac{1}{2} \rho V_a^2 S C_D (\alpha, q, U_1) \] (10)

\[ t_2 = \frac{1}{2} \rho V_a^2 S C_m (\alpha, q, U_1) \] (11)

where \( c \) is the average aerodynamic chord length, \( \rho \) is the air density, \( S \) is the UAV wing area, \( C_L \) is the lift coefficient, \( C_D \) is the drag coefficient, \( C_m \) is the moment coefficient, \( U_1 \) is the elevator command signal.

In the control process of a fixed-wing UAV, the angle of attack is usually very small, so the aerodynamic parameters mentioned above can be regarded as linear. The following results were obtained using the first-order Taylor formula:

\[ f_{up} = \frac{1}{2} \rho V_a^2 S C_L (\alpha, \beta, U_2) \] (12)

\[ f_r = \frac{1}{2} \rho V_a^2 S C_D (\alpha, \beta, U_2) \] (13)

\[ t_2 = \frac{1}{2} \rho V_a^2 S C_m (\alpha, \beta, U_2) \] (14)

Where \( C_L (\alpha, \beta), C_D (\alpha, \beta), C_m (\alpha, \beta) \) are nonlinear equations related to \( \alpha \) with different coefficients, \( C_{Lq} = \frac{\partial C_L}{\partial q}, C_{Lu_1} = \frac{\partial C_L}{\partial u_1}, \)
\( C_{Dq} = \frac{\partial C_D}{\partial q}, C_{Du_1} = \frac{\partial C_D}{\partial u_1}, C_{mq} = \frac{\partial C_m}{\partial q}, C_{mu_1} = \frac{\partial C_m}{\partial u_1}. \)

The above results (\( f_{up} \) and \( f_r \)) are discussed in the stable coordinate system, and the transformation of these results to the body coordinate system is as follows:

\[ \begin{pmatrix}
    f_x \n    f_y \\
    f_z
\end{pmatrix} = \begin{pmatrix}
    \cos \alpha & -\sin \alpha & f_x \\
    \sin \alpha & \cos \alpha & f_y \\
    0 & 0 & f_z
\end{pmatrix} \begin{pmatrix}
    f_{up} \n    f_r \n    0
\end{pmatrix} \] (15)

According to the analysis of the lateral symmetry plane, the influence of aerodynamics is mainly related to the rudder \( (U_3) \), aileron steering gear \( (U_5) \), yaw angular velocity \( (\rho) \), roll angular velocity \( (\nu) \), and sideslip angle \( (\beta) \). Their relationship is as follows [31]:

\[ f_y = \frac{1}{2} \rho V_a^2 S C_L (\beta, p, r, U_2, U_3) \] (16)

\[ t_1 = \frac{1}{2} \rho V_a^2 S C_t (\beta, p, r, U_2, U_3) \] (17)

\[ t_3 = \frac{1}{2} \rho V_a^2 S C_t (\beta, p, r, U_2, U_3) \] (18)

where \( C_r \) is the dimensionless lateral force coefficient, \( C_t \) is the dimensionless rolling moment coefficient, \( C_m \) is the dimensionless yaw moment coefficient.

Using a method similar to the longitudinal aerodynamic study, the following results can be obtained by the first-order Taylor formula expansion:

\[ f_y = \frac{1}{2} \rho V_a^2 S C_{\alpha y} + C_{\alpha y} \beta \frac{p}{2V_a} + C_{\alpha y} \frac{r}{2V_a} - C_{\alpha y} U_2 - C_{\alpha y} U_3 \] (19)

\[ t_1 = \frac{1}{2} \rho V_a^2 S C_{\alpha t} (\beta, p, r, U_2, U_3) \] (20)

\[ t_3 = \frac{1}{2} \rho V_a^2 S C_{\alpha t} (\beta, p, r, U_2, U_3) \] (21)

2) ESTABLISHMENT OF THRUST MODEL

Through force analysis of the propeller used in a fixed-wing UAV [32], the expression of thrust can be obtained as

\[ F = \frac{1}{2} \rho S p [(m U_4)^2 - V_4^2] \] (22)
where $S_p$ is the area swept by the propeller, $C_p$ is the thrust-related parameter, $k_m$ is the propeller engine parameters, $U_s$ is the propeller engine acceleration.

**D. Coupling of kinematic and dynamic models with aerodynamic and thrust models**

By coupling the calculated aerodynamic model, torque model and thrust model with the state equations, the following twelve new state equations can be obtained:

\[ \dot{p} = \cos \psi \cos \theta u + (-\cos \phi \sin \psi + \sin \phi \sin \theta \cos \psi) v + (\sin \phi \sin \psi + \cos \phi \sin \theta \cos \psi) w \] \[ \dot{q} = \sin \psi \cos \theta u + (\cos \phi \cos \psi + \sin \phi \sin \theta \sin \psi) v + (-\sin \phi \cos \psi + \cos \phi \sin \theta \sin \psi) w \] \[ \dot{r} = \sin \phi \theta \cos \psi u - \sin \phi \theta \sin \psi w + \cos \phi \theta \sin \psi r + \cos \phi \theta \cos \psi \] \[ \dot{\psi} = \sin \phi \sin \psi u + \sin \phi \cos \psi v - \cos \phi \theta \sin \psi r - \cos \phi \theta \cos \psi \] \[ \dot{\theta} = pv - qw + \sin \phi \sin \theta \] \[ \dot{\phi} = p + \sin \phi \tan \theta + \cos \phi \tan \theta \] \[ \dot{u} = rv - qw + \sin \theta \] \[ \dot{\phi} = p + \sin \phi \tan \theta + \cos \phi \tan \theta \] \[ \dot{u} = rv - qw + \sin \theta \]

**III. Adaptive backstepping sliding mode control law design**

The control law designed based on adaptive backstepping sliding mode control can solve the unknown uncertain problem of fixed-wing UAVs, including the uncertainty of the system itself and external disturbances [33,34]. The adaptive law is used to estimate the uncertainty of the system itself and the external disturbance. Then, the estimated value is compensated to the controller to realize the process of compensation. The existing adaptive backstepping sliding mode control mainly uses self-use law to overcome uncertainty and external disturbance. In this paper, the adaptive law is proposed to accomplish the above task and overcome the impact of chattering at the same time.

Before introducing specific controllers, the control system of fixed-wing UAV is designed as shown in Fig. 1.

**FIGURE 1.** Fixed-wing UAV control system

The target attitude angle, airspeed and altitude are respectively fed to attitude controller, airspeed controller and altitude solver module at first. Attitude control is mainly accomplished by attitude backstepping sliding mode controller. The attitude adaptive module provides adaptive compensation for the attitude backstepping sliding mode controller by dealing with parameter uncertainty, external disturbance and the chattering caused by controller. The controller output the elevator command signal ($U_1$), rudder ($U_2$), and aileron steering gear ($U_3$) to actuators and collect the attitude feedback to accomplish attitude control. Airspeed control is mainly accomplished by airspeed inversion sliding mode controller. Its principle is similar to attitude control. The attitude control mainly relies on a PID controller coupled with attitude controller. The altitude controller obtains the desired attitude angles by processing the command height signal and outputs them to the attitude controller. This part will be explained detaily in II.C.

For the fixed-wing UAV controller, this section mentions five related control laws, which include three attitude angle control laws, a flight speed control law, and a flight altitude control law. Their stability are analyzed using the Lyapunov stability criterion.

**A. Attitude controller design**

The attitude control system includes three angles- $\theta, \psi, \phi$. 
According to Substitute the same time, the value of \( F \) are three input signals related to the attitude angle: the elevator command signal \((U_1)\), rudder \((U_2)\), and aileron steering gear \((U_3)\). To establish the required state-space functions, the kinematics and dynamics models need to further handle [35].

For angle \( \Psi \) control system,
\[
\Psi = \sin \Phi \sec \Theta + \cos \Phi \sec \Theta \tag{26}
\]
When the drone is flying, the values of \( \Phi \) and \( \Theta \) are usually extremely small. The change in angle \( \Psi \) is mainly related to \( r \). Therefore, when \( \Psi \) is analyzed, we can assume as follows:
\[
sin \Phi \approx 0, \cos \Phi \approx 1, \sec \Theta \approx 1 \]
View them as the uncertainty of the system.
So that the equation could be rewritten as
\[
\Psi = r + (\cos \Phi \sec \Theta - 1) r + \sin \Phi \sec \Theta q \tag{35}
\]
Define \( d_y \) is the external disturbance.
\[
d_y = (\cos \Phi \sec \Theta - 1) r + \sin \Phi \sec \Theta q \tag{36}
\]
Therefore
\[
\Psi = r + d_y \tag{37}
\]
Take the derivative of this with respect to \( t \):
\[
\dot{\Psi} = \dot{r} + \dot{d}_y \tag{38}
\]
Substitute (34) into (38):
\[
\dot{\Psi} = T_p q - T_t \dot{r} q + \frac{1}{2} p V_c^2 S_b (C_{13} + C_{14} \beta + C_{15} \frac{b}{2V_c} p)
+ C_{16} \frac{b}{2V_c} r + C_{17} U_2 + C_{18} U_3 \tag{39}
\]
Define \( w = (\Psi - d_y) \),
\[
\dot{w} = a_{y1} w + a_{y2} U_2 + d_y \tag{40}
\]
Where \( d_y \) is the total external disturbance,
\[
a_{y1} = \frac{T_p q - T_t \dot{r} q + \frac{1}{2} p V_c^2 S_b (C_{13} + C_{14} \beta + C_{15} \frac{b}{2V_c} p)
+ C_{16} \frac{b}{2V_c} d_y + C_{18} U_3} \tag{41}
\]
Therefore, the desired state-space vector of the attitude angle control system can be defined as
\[
X = (\Psi, \dot{\Psi}, \dot{\Theta}, \dot{\phi}, \dot{\psi})^T \in \mathbb{R}^6 \tag{47}
\]
The output vector of this system can be defined as
\[
Y = (\Psi, \dot{\psi})^T \in \mathbb{R}^3 \tag{48}
\]
According to (42), \( \alpha = \Theta - \varepsilon \), where \( \varepsilon \) is Climbing Angle.

It is already known that \( C_{14}(\alpha) \) is a nonlinear model related to torque that can only be obtained through wind tunnel flight tests. Therefore, it is feasible to replace it with a linear model
\[
C_{14}(\alpha) = C_{t20} + C_{t21} \alpha \tag{33}
\]
Therefore, the function could be written as
\[
\dot{\Theta} = T_p \dot{r} q - T_t (p^2 - r^2) + \frac{1}{2} p V_c^2 S_c (C_{t20} + C_{t21} (\Theta - \varepsilon))
+ C_{t2q} \frac{c}{2V_c} q + C_{t2u1} U_1 \tag{42}
\]
where \( d_y \) is the total external disturbance,
\[
a_{q1} = C_{t20} + C_{t21} \alpha \tag{43}
\]
Define \( d_y \) is the external disturbance of this system. Because of the close-to-zero values of \( \sin \Phi, \tan \theta \),
\[
\phi = p + d_\phi \tag{44}
\]
The derivation of \( \phi \) with respect to \( t \) is showed below:
\[
\dot{\phi} = \dot{p} + \dot{d}_\phi \tag{45}
\]
Substitute (32) into (45):
\[
\dot{\phi} = T_p \dot{r} q + T_t \dot{r} q + \frac{1}{2} p V_c^2 S_b (C_{t20} + C_{t21} \alpha)
+ C_{t2q} \frac{c}{2V_c} d_y + C_{t2u1} U_1 \tag{46}
\]
Therefore, the desired state-space vector of the attitude angle control system can be defined as
\[
X = (\Psi, \dot{\Psi}, \dot{\Theta}, \dot{\phi}, \dot{\psi})^T \in \mathbb{R}^6 \tag{47}
\]
The output vector of this system can be defined as
\[
Y = (\Psi, \dot{\psi})^T \in \mathbb{R}^3 \tag{48}
\]
Letting \( x_1 = \Psi, x_2 = \dot{\Psi}, x_3 = \dot{\Theta}, x_4 = \dot{\phi}, x_5 = \dot{\psi}, \phi = \dot{\phi} \). The related state-space function can be obtained as
\[
\dot{X} = A(X) + B(X) U = \begin{bmatrix}
\dot{\Psi} \\
\dot{x_2} \\
\dot{\Theta} \\
\dot{x_4}
\end{bmatrix} = \begin{bmatrix}
a_{q1} x_2 + a_{x2} x_3 + a_{x3} U_1 + d_\phi \\
0 & x_2 \\
0 & x_3 \\
0 & x_4
\end{bmatrix}
\]
\[
Y = C(X)
\]
Second, the adaptive backstepping sliding mode controller can be designed according to the above state-space function. The yaw angle controller design is based on below state space function group.
\[
\begin{align*}
\dot{x}_1 &= x_2 \\
\dot{x}_2 &= a_{\psi_1} x_1 + a_{\psi_2} U_2 + D_\psi \\
y_1 &= x_1
\end{align*}
\]  
(50)

where \(D_\psi\) is the total uncertainty of the yaw angle controller, which can be expressed as

\[
D_\psi = \Delta a_{\psi_1} x_2 + \Delta a_{\psi_2} U_2 + d_\psi
\]  
(51)

In the process of recursion, the sliding mode surface and adaptive law need to be introduced to determine the control effect of the controller. Adaptive estimation mainly focuses on the design of \(D_\psi\). There are three significant steps in the design of the adaptive backstepping sliding mode controller.

1. The construction of virtual control signal through Lyapunov function and definition of sliding surface.
2. The extension of the Lyapunov equation.
3. The introduction of adaptive law and control law.

So that, the adaptive backstepping sliding mode control law could be obtained.

Assume \(x_{1d}\) is the yaw angle command signal. The design details of the yaw angle controller are presented below.

Step 1:

Define tracking error is \(z_{\psi_1} = x_1 - x_{1d}\), so the derivative of tracking error is \(\dot{z}_{\psi_1} = \dot{x}_1 - \dot{x}_{1d}\).

Establish the Lyapunov function \((V_{\psi_1})\):

\[
V_{\psi_1} = \frac{1}{2} z_{\psi_1}^2
\]  
(52)

We define \(\tau_\psi = k_{\psi_1} z_{\psi_1} + z_{\psi_2}\), where \(c_{\psi_1}\) is a constant greater than zero and \(z_{\psi_2}\) is the virtual control signal.

Then the derivative of \(z_{\psi_1}\) can be expressed as

\[
z_{\psi_1} = \dot{x}_1 - x_{1d} = z_{\psi_2} - c_{\psi_1} z_{\psi_1}
\]  
(53)

The derivative of \(V_{\psi_1}\) can be expressed as

\[
V_{\psi_1} = z_{\psi_1} z_{\psi_2} - c_{\psi_1} z_{\psi_1}\n\]  
(54)

The sliding surface is defined below.

\[
\tau_\psi = k_{\psi_1} z_{\psi_1} + z_{\psi_2}
\]  
(55)

Where \(k_{\psi_1} > 0\).

Substitute (53) into (55):

\[
\tau_\psi = k_{\psi_1} z_{\psi_1} + z_{\psi_2} + c_{\psi_1} z_{\psi_1} = (k_{\psi_1} + c_{\psi_1}) z_{\psi_1} + z_{\psi_1}
\]  
(56)

It is obvious that \(k_{\psi_1} + c_{\psi_1} > 0\). If \(\tau_\psi = 0\), \(z_{\psi_1} = 0\) and \(z_{\psi_2} = 0\), which, in turn, will lead \(V_{\psi_1} \leq 0\). Therefore, it is necessary to introduce Step 2.

Step 2:

Establish another Lyapunov function \((V_{\psi_2})\):

\[
V_{\psi_2} = V_{\psi_1} + \frac{1}{2} \tau_\psi^2
\]  
(57)

\[
V_{\psi_2} = V_{\psi_1} + \tau_\psi \tau_\psi
\]  
(58)

\[
= z_{\psi_1} z_{\psi_2} - c_{\psi_1} z_{\psi_1}\n\]  
(59)

Because the uncertain part of the system appears in \(V_{\psi_2}\), Step 3 is required to determine whether \(V_{\psi_2}\) is greater than zero.

Step 3:

In the sliding mode control process, excellent stability and robustness require a precise upper bound on the total uncertainty. However, in the actual operation of a UAV, the accurate upper bound of the total uncertainty is difficult to obtain. Therefore, an adaptive law is introduced to estimate the upper bound of the total uncertainty of the system.

Establish the third Lyapunov function \((V_{\psi_3})\):

\[
V_{\psi_3} = V_{\psi_2} + \frac{1}{2 \delta_\psi} \overline{D_\psi}^2
\]  
(60)

where \(\overline{D_\psi} = D_\psi - \overline{D_\psi}\) is the error in the estimation of \(D_\psi\), \(\overline{D_\psi}\) is the value in the estimation of \(D_\psi\), \(\delta\) is a normal number.

\[
V_{\psi_3} = V_{\psi_2} + \frac{1}{2 \delta_\psi} \overline{D_\psi} \overline{D_\psi} = V_{\psi_2} + \frac{1}{2 \delta_\psi} (\overline{D_\psi}^2 - \overline{D_\psi})
\]  
(61)

Due to the insensible change of \(D_\psi\), assuming \(\overline{D_\psi} = 0\) is feasible.

\[
V_{\psi_3} = V_{\psi_2} + \frac{1}{2 \delta_\psi} \overline{D_\psi} \overline{D_\psi}
\]  
(62)

\[
= z_{\psi_1} z_{\psi_2} - c_{\psi_1} z_{\psi_1}\n\]  
(63)

\[
+ \tau_\psi (k_{\psi_1} (z_{\psi_2} - c_{\psi_1} z_{\psi_1}) + a_{\psi_1} (z_{\psi_2} + x_{1d} - c_{\psi_1} z_{\psi_1}) + a_{\psi_2} U_2 + D_\psi - x_{1d}^* + c_{\psi_1} z_{\psi_1})
\]  
(64)

\[
\frac{1}{2 \delta_\psi} \overline{D_\psi} \overline{D_\psi}
\]  
(65)

Select the yaw angle control law:

\[
U_2 = - \frac{1}{a_{\psi_2}} (-k_{\psi_1} (z_{\psi_2} - c_{\psi_1} z_{\psi_1}) - a_{\psi_1} (z_{\psi_2} + x_{1d} - c_{\psi_1} z_{\psi_1}) - \overline{D_\psi} + x_{1d}^* - c_{\psi_1} z_{\psi_1} - h_{\psi_1} (\tau_\psi + \epsilon_{\psi} sgn(\tau_\psi))
\]  
(66)

where \(h_{\psi_1}\) and \(\epsilon_{\psi}\) are positive constants.

Design the adaptive law of \(D_\psi\):

\[
\overline{D_\psi} = - \delta_\psi \tau_\psi
\]  
(67)

Similarly, the control laws and adaptive laws of the pitch and roll angles are as follows:

\[
U_1 = - \frac{1}{a_{\theta_3}} (-k_{\theta_1} (z_{\theta_2} - c_{\theta_1} z_{\theta_1}) - a_{\theta_1} (z_{\theta_2} + x_{3d} - c_{\theta_1} z_{\theta_1}) - a_{\theta_2} x_3 - \overline{D_\theta} + x_{3d}^* - c_{\theta_1} z_{\theta_1} - h_{\theta_1} (\tau_\theta + \epsilon_{\theta} sgn(\tau_\theta)))
\]  
(68)

\[
\overline{D_\theta} = - \delta_\theta \tau_\theta
\]  
(69)

\[
U_3 = - \frac{1}{a_{\phi_2}} (-k_{\phi_1} (z_{\phi_2} - c_{\phi_1} z_{\phi_1}) - a_{\phi_1} (z_{\phi_2} + x_{5d} - c_{\phi_1} z_{\phi_1}) - \overline{D_\phi} + x_{5d}^* - c_{\phi_1} z_{\phi_1} - h_{\phi_1} (\tau_\phi + \epsilon_{\phi} sgn(\tau_\phi))
\]  
(70)

\[
\overline{D_\phi} = - \delta_\phi \tau_\phi
\]  
(71)

where \(z_{\theta_1} = x_4 - x_{3d} = z_{\theta_2} - c_{\theta_1} z_{\theta_1}\),

\[
\overline{D_\theta} = x_6 - x_{5d} = z_{\theta_2} - c_{\theta_1} z_{\theta_1},
\]

\[
z_{\phi_1} = x_4 - x_{5d} + c_{\phi_1} z_{\phi_1}, \quad \overline{D_\phi} = x_6 - x_{5d} + c_{\phi_1} z_{\phi_1},
\]

\[
\tau_\theta = k_{\theta_1} z_{\theta_1} + \overline{D_\theta} + \tau_\theta, \quad \tau_\phi = k_{\phi_1} z_{\phi_1} + \overline{D_\phi} + \tau_\phi,
\]

\(c_{\phi_1}, k_{\phi_1}, h_{\phi_1}, \epsilon_{\phi}, c_{\theta_1}, k_{\theta_1}, h_{\theta_1}, \epsilon_{\theta}\) are positive constants.

**B. Airspeed controller design**
The airspeed of the UAV is the sum of the three-axis velocity vectors in the body coordinate system. 
\[ V_a = \sqrt{u^2 + v^2 + w^2} \]  

When airspeed control is carried out, the control process can be greatly simplified by only controlling the velocity vector in the direction of the X-axis and treating the velocity vectors in the other two axes as disturbances. Assume \( v = 0, w = 0 \). Therefore,
\[ V_a = u + d_{V_a1} \]  

Taking derivative \( V_a \):
\[ \dot{V}_a = \dot{u} + d_{V_a1} \]  

Substituting (29) into (70), it can be obtained as
\[ \dot{V}_a = rv - qw - g\sin\theta + \frac{1}{2m}\rho V_a^2 S \left[ C_1(\alpha) + C_2(\alpha) \frac{c}{v_a} + C_3(\alpha) U_1 + \frac{1}{2m}\rho S_P C_p \left( k_m U_4^2 - V_2^2 \right) \right] + d_{V_a1} \]  

According to (71), the conclusion that \( V_a \) can be controlled by both the pitch angle (\( \theta \)) and propeller engineaccelerograph (\( U_4 \)) can be obtained. Here, we choose to use \( U_4 \) to control airspeed and regard the influence of the pitch angle as a disturbance. Therefore, the (71) could be rewritten as
\[ \dot{V}_a = \frac{1}{m}\rho V_a^2 S C_1(\alpha) U_4 + \frac{1}{4m}\rho S_P C_2(\alpha) cq V_a + \frac{1}{2m}\rho S_P C_p \left( k_m U_4^2 - V_2^2 \right) + d_{V_a1} \]  

Taking derivative \( V_a \):
\[ V_a = \frac{1}{m}\rho S_P C_1(\alpha) V_a + \frac{1}{4m}\rho S_P C_2(\alpha) cq V_a + \frac{1}{2m}\rho S_P C_p \left( k_m U_4^2 - V_2^2 \right) + d_{V_a1} \]  

where \( d_{V_a1} = rv - qw - g\sin\theta \).

\[ \dot{V}_a = r v - q w - g \sin \theta + d_{V_a1} \]

Taking derivative \( V_a \):
\[ V_a = \frac{1}{m}\rho S_P C_1(\alpha) V_a + \frac{1}{4m}\rho S_P C_2(\alpha) cq V_a + \frac{1}{2m} S_P C_p \left( k_m U_4^2 - V_2^2 \right) + d_{V_a2} \]

\[ a_{V_a1} V_a + a_{V_a2} \dot{V}_a + a_{V_a3} U_4 + d_{V_a2} \]

where \( a_{V_a1} = \frac{1}{m}\rho S_P C_1(\alpha) + \frac{1}{4m}\rho S_P C_2(\alpha) U_4 - \frac{1}{2m} S_P C_p \), \( a_{V_a2} = \frac{1}{4m}\rho S_P C_2(\alpha) cq, a_{V_a3} = \frac{1}{2m} S_P C_p k_m^2 \).

Letting \( \dot{x}_7 = V_a, \dot{x}_8 = \dot{V}_a, \) The related state space function could be obtained as
\[ \begin{align*}
\dot{x}_7 &= V_a \\
\dot{x}_8 &= a_{V_a1} x_7 + a_{V_a2} x_8 + a_{V_a3} U_4 + D_{V_a}
\end{align*} \]

Where \( D_{V_a} = \Delta a_{V_a1} x_7 + \Delta a_{V_a2} x_8 + \Delta a_{V_a3} U_4 + d_{V_a2} \) is the total uncertainty.

Similar to the attitude controller design, the control law and adaptive law of airspeed can be obtained as follows:
\[ U_4 = -\frac{1}{a_{V_a3}} (-k_{V_a1} z_{V_a2} - c_{V_a1} \dot{z}_{V_a1}) - a_{V_a2} (z_{V_a2} + x_7 - c_{V_a1} \dot{z}_{V_a1}) \]

\[ D_{V_a} = -\delta_{V_a} v_a \]

\[ \tau_a = k_{V_a1} \dot{z}_{V_a1} + z_{V_a2}, \quad \tau_a = k_{V_a1} z_{V_a1} + z_{V_a2}, \quad c_{V_a1}, k_{V_a1}, h_{V_a1}, \epsilon_{V_a} \text{ are positive constants.} \]

C. Altitude controller design

The altitude control of a fixed-wing UAV is a relatively more complicated control process. To achieve this, it is necessary to first analyze the equation of state associated with \( P_z \).

\[ P_z = u \theta + \sin \theta u - u \theta - \cos \sin \phi \theta - \cos \phi \cos \theta \omega \]

In the airspeed control process under ideal conditions, the following assumptions can be made:
\[ v = 0, w = 0 \]

According to (25), the flight altitude is mainly related to the pitch angle and airspeed. To simplify the control process, the flight altitude state equation is modified as follows:
\[ P_z = u \theta + \sin \theta u - u \theta - \cos \sin \phi \theta - \cos \phi \cos \theta \omega + d_{P_z} \]

where \( d_{P_z} = \sin \theta u - u \theta - \cos \sin \phi \theta - \cos \phi \cos \theta \omega \) is defined as disturbance.

According to (77), it is known that \( P_z \) is determined by \( u \) and \( \theta \). Because it is complicated to control the flight altitude by using the attitude angle and airspeed at the same time, this section divides the flight process into different stages according to the altitude of the UAV to use different control methods. The altitude control process is mainly divided into four areas: taking off area, acceleration area, attitude holding area, and descent area (shown in Fig.2). During the UAV takeoff, the fixed throttle opening extent and pitch angle were set to ensure that the UAV stably entered the acceleration zone from the take-off zone. In the acceleration area, the drone's pitch angle is set to a fixed value \( (h_f) \), and the propeller engineaccelerograph was used to drive the drone up to the command speed and into the altitude-holding zone. Subsequently, in the altitude holding area, a controller coupled with PID control and sliding mode inversion adaptive control is used to adjust the height, which is also the main design target of this section.

Finally, if the height of the UAV exceeds the upper limit of the rated flight height \( (h_{max}) \), it enters the descent zone. Within this range, the throttle opening was set to 0, and the coupled altitude controller completed the descent process of the UAV until it reached the command height range.
The coupling controller based on PID control and sliding mode inversion adaptive control can be designed as shown in Fig. 3.

\[ O = \begin{pmatrix} c_{\psi_1} + h_{\psi_1} k_{\psi_1}^2 & h_{\psi_1} k_{\psi_1} - \frac{1}{2} \\ h_{\psi_1} k_{\psi_1} - \frac{1}{2} & h_{\psi_1} \end{pmatrix} \]  

\[ Z = \begin{pmatrix} z_{\psi_1} \\ z_{\psi_2} \end{pmatrix} \]

\[ Z^T O Z = \left( z_{\psi_1} z_{\psi_2} \right) \begin{pmatrix} c_{\psi_1} + h_{\psi_1} k_{\psi_1}^2 & h_{\psi_1} k_{\psi_1} - \frac{1}{2} \\ h_{\psi_1} k_{\psi_1} - \frac{1}{2} & h_{\psi_1} \end{pmatrix} \begin{pmatrix} z_{\psi_1} \\ z_{\psi_2} \end{pmatrix} = c_{\psi_1} z_{\psi_1}^2 - z_{\psi_1} z_{\psi_2} + h_{\psi_1} k_{\psi_1} z_{\psi_1}^2 + 2 h_{\psi_1} k_{\psi_1} z_{\psi_1} z_{\psi_2} + h_{\psi_1} z_{\psi_2}^2 \]

\[ = c_{\psi_1} z_{\psi_2}^2 - z_{\psi_2} \left( c_{\psi_1} + h_{\psi_1} k_{\psi_1}^2 \right) + h_{\psi_1} k_{\psi_1} z_{\psi_1}^2 + h_{\psi_1} \left( c_{\psi_2} - h_{\psi_2} \right)^2 \]

\[ = c_{\psi_1} z_{\psi_2}^2 - z_{\psi_2} \left( c_{\psi_1} + h_{\psi_1} k_{\psi_1}^2 \right) + h_{\psi_1} k_{\psi_1} z_{\psi_1}^2 + 2 h_{\psi_1} k_{\psi_1} z_{\psi_1} z_{\psi_2} + h_{\psi_1} z_{\psi_2}^2 \]

\[ = c_{\psi_1} z_{\psi_2}^2 - z_{\psi_2} \left( c_{\psi_1} + h_{\psi_1} k_{\psi_1}^2 \right) + h_{\psi_1} k_{\psi_1} z_{\psi_1}^2 + 2 h_{\psi_1} k_{\psi_1} z_{\psi_1} z_{\psi_2} + h_{\psi_1} z_{\psi_2}^2 \]

Therefore, \( V_{\psi_3} = -Z^T O Z - h_{\psi_1} c_{\psi_1} [T_{\psi_1}] \).

If \( O \) is positive definite, \( V_{\psi_3} \) will be negative definite. If \( |O| = h_{\psi_1} (c_{\psi_1} + h_{\psi_1} k_{\psi_1}^2) - (h_{\psi_1} k_{\psi_1} - \frac{1}{2})^2 = h_{\psi_1} (c_{\psi_1} + k_{\psi_1}) - \frac{1}{4} \) (84)

Because \( O \) is positive definite if \( |O| \geq 0 \),

\[ h_{\psi_1} (c_{\psi_1} + k_{\psi_1}) - \frac{1}{4} \geq 0 \]

must be satisfied to ensure that \( V_{\psi_3} \) is negative definite when taking the values of \( h, h, k \) and \( \epsilon \) are system stable. The stability judgment method of other controllers is similar to this one, as long as the same conditions are satisfied when taking the values of \( h, h, k \) for the systems will be stable.

### IV. Simulation result and analysis

This section verifies the effectiveness and superiority of the proposed controllers. The effectiveness of a control system is often the most objective criteria for evaluating a controller. To verify the effectiveness of the proposed controller, the proposed adaptive backstepping sliding mode controller was applied to a simulated fixed-wing UAV and compared with the traditional PID control and backstepping slide mode control. By comparing the results of the adaptive backstepping sliding mode control and traditional PID control, a better control performance of the former is achieved. Furthermore, the necessity of adaptive law is reflected in the comparison between the adaptive backstepping sliding mode control and backstepping sliding mode control. The performance comparisons in this section are mainly based on the tracking process of the control signals, control input signals, and adaptive signals.

All controllers should be combined in the simulation process because of the interaction between multiple state variables. Ailerons and rudders, for example, act on roll and yaw motions, and changes in pitch angle also affect altitude and airspeed. The coupling process is so complex that the step simulation is difficult, which, in turn, lead to a significant amount of time spent on parameter selection and adjustment.
However, to compare the control effect more clearly, the simulation process was divided into three parts: attitude control, airspeed control, and altitude holding. In each part, the simulation results were compared with the traditional PID control method and the backstepping sliding mode control method.

A. Attitude Controller Simulation
This part presents the control simulation results for the three attitude angles of the fixed-wing UAV. In the simulation of attitude control, it is assumed that the initial state airspeed and altitude of the UAV are 60m/s and 2500m, respectively. The effects of attitude control procedures on airspeed and altitude are not discussed in this part. The initial state of the attitude angle ($\Psi, \theta, \phi$) is (0°, 0°, 0°), and the target status is (10°, 10°, 10°).

The parameters of the controller laws in (62), (64), and (66), and the adaptive laws in (63), (65), and (67) are selected as follows:

$$ k_{\Psi 1} = 60, c_{\Psi 1} = 50, h_{\Psi 1} = 65, \varepsilon_{\Psi} = 1.1, $$

$$ k_{\phi 1} = 45, c_{\phi 1} = 38, h_{\phi 1} = 55, \varepsilon_{\phi} = 1.2, $$

$$ k_{\theta 1} = 45, c_{\theta 1} = 40, h_{\theta 1} = 50, \varepsilon_{\theta} = 1.5, $$

$$ \delta_{\Psi} = 10, \delta_{\phi} = 28, \delta_{\theta} = 30. $$

For Backstepping Slide Mode control laws:

$$ k_{\Psi BSM} = 60, c_{\Psi BSM} = 50, h_{\Psi BSM} = 65, \varepsilon_{\Psi BSM} = 1.1, $$

$$ k_{\phi BSM} = 45, c_{\phi BSM} = 38, h_{\phi BSM} = 55, \varepsilon_{\phi BSM} = 1.2, $$

$$ k_{\theta BSM} = 45, c_{\theta BSM} = 40, h_{\theta BSM} = 50, \varepsilon_{\theta BSM} = 1.5, $$

And for PID control laws:

$$ k_{\Psi} = 8, k_{\Psi} = 1.7, k_{\Psi} = 0.5, $$

$$ k_{\phi} = 6, k_{\phi} = 1.4, k_{\phi} = 0.2, $$

$$ k_{\theta} = 5, k_{\theta} = 1.8, k_{\theta} = 0.4. $$

To achieve a more realistic reaction control effect, the uncertainty data of the system must be obtained from the wind tunnel experiment. According to the wind tunnel test results of the UAV in [28], the total uncertain is taken as

$$ D_{\Psi} = -40 \sin(0.1t), $$

$$ D_{\phi} = -30 \sin(0.1t), $$

$$ D_{\theta} = -35 \sin(0.1t). $$

The comparison results of the attitude angle simulation control are shown in Figs.4-12. There are three control performance indexes recorded in Table I-III to help analyse. These indexes are steady error, overshoot and react time. From these figures and tables, it is obvious that all three control laws can achieve a stable state for the attitude control of fixed-wing UAVs. However, different control laws exhibit different performances for different attitude angle control processes. Next, the performances of these control laws were analyzed according to different attitude angles.

FIGURE 4. Roll Angle tracking comparison

A comparison of the input signals of the three control laws is shown in Fig.6. The input signal of the ABSM can reach the steady state smoothly, whereas that of the BSM produces a negative-6.5-overshoot before reaching a steady state. When a large overshoot occurs at the control input signal, it may directly damage the equipment. Therefore, such a situation should be avoided as much as possible during the control process. In the roll control process, the ABSM controller has better control performance than the BSM and PID controllers.

FIGURE 5. Roll Angle controllers’ input signal comparison
FIGURE 6. Roll Angle ABSM controller’s adaptive uncertainty estimation signal

TABLE I

| Control Performance Indexes of Different Roll Angle Controllers |
|-----------------------------------|
| PID       | BSM       | ABSM      |
| Stead error | 0.39°    | -0.03°   | 0.05° |
| Overshoot | 0°        | 0.32°    | 0°    |
| React time | 1.18s    | 0.19s    | 0.24s |

For the yaw controller, the situation was similar to that of the roll angle simulation. A more obvious oscillation occurred at the beginning of the BSM tracking signal (shown in Fig. 7 and Table II). The PID control law performs poorly as it does in the roll angle control simulation. The control input signal was roughly similar to the roll-angle case (Fig. 8). It is worth noting that the control input signal of the BSM occurs buffeting after 1.5s (Fig. 9). Buffeting is a common phenomenon in sliding mode control and can significantly affect the performance of the controller. The introduction of an adaptive law can not only estimate the uncertainty but also reduce the chattering of the system. Based on the absence of chattering generated by the ABSM in Fig. 8, it can be inferred that the design of the chattering reduction system introduced by the adaptive law is effective. Therefore, the performance superiority of the ABSM is reflected.
For the control process of the pitch angle, PID steady-state error still exists, and the control speed and accuracy of ABSM and BSM are close (Fig. 11 and Table III). However, it is worth noting that the known upper limit of the disturbance is a necessary condition when designing a BSM controller, while an ABSM controller can be obtained without this condition. In practice, the upper limit of the disturbance is often unpredictable; therefore, ABSM is still the better choice.

The input signal is similar to that of the previous two control processes. Although slight chattering also occurs in the ABSM between 1.9s and 2.1s, the effect of using the adaptive law to reduce chattering is significant when compared with the BSM input signal (Fig. 12 and 13).

TABLE II
CONTROL PERFORMANCE INDEXES OF DIFFERENT YAW ANGLE CONTROLLERS

|       | PID   | BSM   | ABSM  |
|-------|-------|-------|-------|
| Stead error | 0.29° | -0.05°| 0°    |
| Overshoot  | 0°    | 1.18° | 0°    |
| React time | 1.16s | 0.49s | 0.18s |

FIGURE 10. Yaw Angle ABSM controller’s adaptive uncertainty estimation signal

FIGURE 11. Pitch Angle tracking comparison

FIGURE 12. Pitch Angle controllers’ input signal comparison

FIGURE 13. Part of Pitch Angle controllers’ input signal comparison

FIGURE 14. Pitch Angle ABSM controller’s adaptive uncertainty estimation signal

TABLE III
CONTROL PERFORMANCE INDEXES OF DIFFERENT PITCH ANGLE CONTROLLERS
Table II

| PID     | BSM | ABSM |
|---------|-----|------|
| Stead error | -0.41° | -0.01° | 0° |
| Overshoot  | 0°   | 0.02° | 0° |
| React time | 1.19s | 0.17s | 0.14s |

The signals of the adaptive law are shown in Fig.6, 10, and 14. It can be observed that the adaptive signals will try to follow the disturbance when the system inputs the step signal, and then it can follow the sinusoidal uncertain signal periodically after entering the steady state.

### B. Airspeed control Simulation

This part presents the simulation results of the airspeed control. During the airspeed simulation, to reduce the influence of system disturbance on airspeed control, the attitude angles were fixed. Both the roll and yaw angles were set to 0 rad, and the pitch angle was fixed at 10 rad. The initial value of the airspeed was 0 m/s, and the target value was 100 m/s. The effect of airspeed control on altitude was not the focus of this part of study.

The total uncertainty of the airspeed control system must be selected prior to the simulation. The method of uncertainty cited was the same as that used in Section IV.A, where the uncertainty is selected as

\[ D_v = -45\sin(0.1t) \]

Other parameters related to the airspeed controllers are selected as follow:

For Adaptive Backstepping Slide Mode

\[ k_{A1} = 40, c_{A1} = 30, h_{A1} = 45, \varepsilon_A = 1.5, \delta_A = 35. \]

For Backstepping Slide Mode control laws:

\[ k_{BSM} = 40, c_{BSM} = 30, h_{BSM} = 45, \varepsilon_{BSM} = 1.5. \]

And for PID control laws:

\[ k_P = 4, k_I = 0.6, k_D = 0.1. \]

A performance comparison of the three controllers in the airspeed control process is shown in Fig.15 and Table IV. The tracking speed of ABSM and BSM is still faster than that of PID, and there is no steady-state error after reaching a steady state. Comparing the ABSM with the BSM control, the BSM has a faster response time as well as a small overshoot. According to the stability principle, the ABSM exhibits better performance and does not need to determine the upper limit of the disturbance in advance.

Even though the airspeed-following performance of the BSM controller can be regarded as good, the input signal has buffeting, as shown in Fig.16, which poses a risk to the stability of the system. However, there is no chattering in the input signal of the ABSM, and its adaptive signal periodically follows the uncertainty of the system after the system reaches a steady state (Fig.17). Therefore, in the process of airspeed control, ABSM is still the controller with the best control performance.

| PID     | BSM | ABSM |
|---------|-----|------|
| Stead error | 0.02° | -0.03° | 0.03° |
| Overshoot  | 0°   | 1.11° | 0°   |
| React time | 1.18s | 0.14s | 0.27s |
C. Altitude Holding Control Simulation

This section mainly analyzes the simulation results of the coupled height controller. It mainly simulates the process of controlling the height with the pitch angle in the height-holding area. The airspeed was kept constant at 60 m/s, and the roll angle and yaw angle were set to 0°. The initial height was 2500m, and the coupling controller controlled the altitude of the UAV to 2600m. The airspeed was kept constant; therefore, there was no additional perturbation to the altitude. Because the height-keeping controller is a coupling controller, the control performance of different control laws is slightly different from that of the previous control laws. The parameters of the primary PID controller are selected as follows:

\[ k_p = 15, \quad k_i = 3.5, \quad k_d = 1.6. \]

The second controller adopted the pitch angle controllers designed in the previous attitude control simulation. The initial value was 2500m. After the steady state was reached, the command signal was changed to 2600m. As shown in Fig. 18 and Table V, the coupling controller of the BSM produces an irredeemable steady-state error (-47.19m) and a small shock caused by the buffeting. Compared with PID-BSM, the control effects of the lower cascade PID controller and the PID-ABSM coupling controller are better. Both the PID-PID controller and PID-ABSM controller have good control performance, although the former has a state error within ±5m. However, the PID-PID control needs to know the precise parameters and disturbances of the controlled system, and the PID-ABSM coupling controller can overcome this limitation with an adaptive law. Therefore, the performance of the PID-ABSM coupling controller still performs better than the other two controllers in the height-keeping control process.

| TABLE V  |
| --- | --- | --- |
| Control Performance Indexes of Different Altitude Coupled Controllers |
| PID | BSM | ABSM |
| Stead error | 4.17m | -47.19m | 0.43m |
| Overshoot | 0m | 31.77m | 0m |
| React time | 0.23s | 0.19s | 0.62s |

V. Conclusion

In this study, a fixed-wing-UAV controller that combines adaptive method and backstepping sliding mode was designed and simulated. Compared with previous sliding mode controllers, this controller can overcome the uncertainty of system parameters as well as the external disturbances and eliminate chattering at the same time by adaptive law. The fixed-wing UAV model is a nonlinear, multi-state coupling and extremely complex model. Furthermore, owing to the influence of various disturbances and uncertainties, it is difficult to control a fixed-wing UAV. Therefore, it is recommended to use this designer to improve control performance. Its stability has been proved using Lyapunov's second method. Moreover, how to select relevant parameters to ensure stability has been specified. The simulation results verified the effectiveness and stability of the controllers. The advantages of ABSM control method are demonstrated in comparison with the PID and BSM control methods. Although the system is multi-variable and multi-coupled, it is feasible to design the controller separately for a single variable by decoupling analysis. The adaptive system is used to suppress the buffeting and estimate the uncertainty of the system and external disturbances. The effectiveness and traceability of the adaptive laws were verified using simulation results. In future work, we plan to apply the designed controller to a real fixed-wing UAV. The principle is to input the control algorithm into the STM32 chip and design the related hardware system. Planning ground station software to complete specific flight missions.
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