Active disturbance rejection attitude control for the dove flapping wing micro air vehicle in intermittent flapping and gliding flight

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Abstract
This paper proposes an attitude control scheme for the Dove flapping wing micro air vehicle in intermittent flapping and gliding flight. The Dove flapping wing micro air vehicle adopts intermittent flapping and gliding flight to make the wing movements more natural; this strategy also has the potential to reduce energy consumption. To implement this specific flight mode, this paper proposes a closed-loop active disturbance rejection control strategy to stabilize the attitude during the processes of flapping flight, transition and gliding flight. The active disturbance rejection control controller is composed of three parts: a tracking differentiator, a linear extended state observer and a nonlinear state error feedback controller. The tracking differentiator estimates the given target signal and the differential signal in real time. The extended state observer estimates the system states and system nonlinearity. Moreover, the bandwidth parameterization method is applied to determine the observer gains. The stability of the closed-loop system is verified using Lyapunov’s theorem. Several outdoor flight experiments have been conducted to verify the effectiveness of the proposed control method, and the results show that the proposed method can guarantee the stability of intermittent flapping and gliding flight.

Keywords
Active disturbance rejection control, attitude control, intermittent flapping and gliding flight, flapping-wing micro air vehicle, outdoor flight test

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Introduction
Inspired by the unmatched flying capabilities of birds in nature, researchers have developed many types of bird-like flapping wing air vehicles.1–4 The Dove flapping wing micro air vehicle (FWMAV) is one of them.4 Similar to large- and medium-sized birds, the Dove FWMAV often flies via a process consisting of low-frequency flapping flight, transition and gliding flight due to the high mass and large inertial force of the flapping wings. This combined flight mode integrating low-frequency flapping flight and gliding flight is often called the intermittent flapping and gliding flight mode.5,6 A simplified profile of this flight mode is shown in Figure 1. The most significant advantages of intermittent flapping and gliding flight are the resultant increases in flight range and endurance.7 However, this flight mode poses difficulties in control system design, especially in control of transitions between flapping and gliding.

First, the dynamic characteristics of the system exhibit large variations during the transitions between the two flight modes. In the flapping flight mode, the aerodynamic force and torque vary periodically, and
the characteristics are strongly nonlinear. By contrast, in the gliding flight mode, the aerodynamic force and torque vary linearly within certain ranges.

Second, a precise aerodynamic model of the transition process between the two flight modes is difficult to establish. The flapping frequency and flapping amplitude are time varying in the transition process. Moreover, the three-dimensional form of the flapping motion is complicated, and the flapping wings experience flexible deformations. Therefore, it is difficult to obtain a precise aerodynamic model. Although some approximation methods for aerodynamic force calculations have been proposed by researchers, such as the quasi-steady model, the thrust calculation accuracy is still not high.

Furthermore, FWMAVs are inevitably affected by unknown external disturbances in flight, such as wind disturbances and electromagnetic interference, which further complicates the control system design.

So far, significant progress has been made in flight control of FWMAVs in both low-frequency flapping flight and gliding flights. For low-frequency flapping mode, Baek et al. designed a closed-loop attitude control system using the proportional-integral-derivative (PID) control technique. Without the help of external measurement instruments, the system gives an ornithopter the ability to fly towards a target using only onboard sensing and computation. Roberts et al. modeled the diving behavior of a real FWMAV. They employed lookup tables with metamodels to control dive maneuver that is demonstrated on the RoboRaven FWMAV platform. Autonomous dives were executed successfully and reached within 6m of the goal location, which is very close to the model error. Dietl et al. tested and compared the control effects of several different controllers. The results of flight simulations showed that periodic time-varying controllers can effectively decrease tracking errors. He et al. constructed a full-state feedback controller. With measurement results from a Vicon composite motion capture system, the altitude control of the FWAMV was satisfactory. Because the flapping wings are fixed in a particular position in the gliding flight mode, FWMAVs operating in this mode have the same flight pattern as fixed-wing aircraft. As a result, the common control methods applied for the gliding flight of fixed-wing aircraft, such as adaptive control methods and robust control methods, can also be used for the gliding flight of FWMAVs with only slight changes. Notably, the above control system design methodologies mostly solve only the control problem associated with a single flight mode. Very few works address the control problem for the transition process between flapping and gliding flight.

The characteristics of the aerodynamic forces exhibit large variations during the transformations between the flapping and gliding modes, and the efficiency of the control surface gradually decreases with decreasing flight speed. Consequently, the control effect of the traditional PID method deteriorates, potentially even causing flight accidents. To ensure stable control during the transition process and while considering the needs of both the flapping and gliding processes, this paper presents an active disturbance rejection control (ADRC) method to address the problems encountered in intermittent flapping and gliding flight. This control strategy is independent of a precise mathematical model and has a strong anti-interference ability. It is able to estimate the total disturbance of the system, including changes in the system dynamics. The controller presented in this paper is composed of three parts: a tracking differentiator, a linear extended state observer (ESO) and a nonlinear state error feedback controller. The function of the tracking differentiator is to smooth the target signal and obtain the corresponding differential value. This approach can reduce system overshoot and response time. Therefore, the tracking differentiator can protect the actuator of an FWMAV from being damaged. The ESO is less dependent on an accurate model and takes the system nonlinearity as the extended state to be estimated. This observer has high efficiency in estimating nonlinear dynamics. Hence, it is appropriate for intermittent flapping and gliding flight. The nonlinear state error feedback controller is constructed with nonsmooth functions. Thus, small errors lead to
high gains, and large errors lead to small gains.\textsuperscript{25,26}

This feature guarantees effective control during intermittent flapping and gliding flight. Under the synergistic effects of these three system components, an ADRC controller is developed to overcome the problems related to large variations in the system dynamics, imprecise mathematical models and unknown external disturbances.

Compared with other works, the main contributions of this paper are as follows.

1. An ADRC strategy is applied to control the attitude of the Dove FWMAV during intermittent flapping and gliding flight, which involves a nonlinear time-varying system. Outdoor flight tests reveal good control effects with the proposed controller.

2. The convergence of the ESO is demonstrated. The stability of the closed-loop system is proven by using the Lyapunov stability theorem.

The remainder of this paper is organized as follows. The Dove FWMAV with flapping and gliding flight capabilities is introduced in the Dove FWMAV section. The dynamic model of the FWMAV system and the problem formulations are given in the Dynamic model of the Dove FWMAV section. An attitude controller design strategy and a stability analysis of the closed-loop system based on a Lyapunov function are presented in the Control design section. Then, several outdoor flapping and gliding intermittent flight experiments are conducted to verify the effectiveness of the attitude controller we proposed. The experimental results are presented in the Experimental results and analysis section, and concluding remarks are given in the Conclusion section.

**Dove FWMAV**

The Dove FWMAV considered in this paper is shown in Figure 2. The geometric parameters of the wing and fuselage are similar to those of adult pigeons. The planar dimensions of a single flapping wing are shown in Figure 3. The basic parameters of the flying platform are listed in Table 1. The lift force and thrust needed for flight are generated by a pair of flapping wings. The driving mechanism of the Dove FWMAV is shown in Figure 4. The design of this mechanism is based on the four-bar principle. When the Dove is operating in the gliding flight mode, the four-bar mechanism is simply held in the dead point position to ensure that the position of the flapping wings is fixed. The actuators of the Dove FWMAV are shown in Figure 5. The taileron is driven by two steering gears. A pitch control torque is generated by deflecting the control surfaces in the same direction. A roll control torque is generated by deflecting the control surfaces in opposite directions. This bird-like FWMAV has high structural efficiency\textsuperscript{27} and the ability to perform both flapping flight and gliding flight.\textsuperscript{28} Hence, this paper takes the Dove as the control object to study the problem of attitude control in intermittent flapping and gliding flight.

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**Table 1.** Basic parameters of the Dove FWMAV.

| Parameter | Physical meaning | Value |
|-----------|------------------|-------|
| $m$       | Total mass of the FWMAV | 220 g |
| $m_{\text{wing}}$ | Mass of a flapping wing | 11 g |
| $l$       | Wing span        | 500 mm |
| $I_y$     | Rotational inertia of the y-axis | $1.4 \times 10^{-3}$ kgm$^2$ |

FWMAV: Flapping wing micro air vehicle.

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**Figure 2.** Photograph of the Dove FWMAV.

**Figure 3.** The planar dimensions of a single flapping wing.

**Figure 4.** Driving mechanism of the Dove FWMAV.
The origin of the body-axis coordinate system is located at the centre of mass. The attitude of the aircraft is represented by the roll, pitch, and yaw angles.

Dynamic model of the Dove FWMAV

Based on rigid body dynamics theory, the following equations of angular motion from Liu\(^29\) are adopted to describe the motion characteristics during intermittent flapping and gliding flight

\[
\begin{align*}
L(t) & = \begin{bmatrix} I_x \dot{\phi}(t) - I_{xz} \dot{r}(t) \\ I_z \dot{q}(t) \\ I_z \dot{r}(t) - I_{xz} \dot{p}(t) \end{bmatrix} \\
M(t) & = \begin{bmatrix} (I_x \dot{p}(t) - I_{xz} \dot{r}(t))q(t) - I_z \dot{q}(t)r(t) \\ -\dot{\psi}(t)\cos(\theta(t))\sin(\phi(t)) - \dot{\psi}(t)\cos(\theta(t))\cos(\phi(t)) \\ I_p(t)q(t) - (I_p(t) - I_{xz}r(t))q(t) \end{bmatrix}
\end{align*}
\]

where \(p(t)\), \(q(t)\) and \(r(t)\) are the roll angular velocity, pitch angular velocity and yaw angular velocity, respectively; \(I_x\), \(I_z\) and \(I_r\) represent the moments of inertia about the \(x\), \(y\) and \(z\) axes; \(I_{xz}\) denotes the product of inertia in the \(xz\) plane; and \(L(t)\), \(M(t)\) and \(N(t)\) are the resultant moments of the roll, pitch, and yaw axes, respectively. In addition, the following equations can be used to describe the conversion relations for the three-axis angular velocity and attitude angles

\[
\begin{bmatrix} p(t) \\ q(t) \\ r(t) \end{bmatrix} = \begin{bmatrix} \dot{\phi}(t) - \dot{\psi}(t)\sin(\theta(t)) \\ \dot{\phi}(t)\cos(\phi(t)) + \dot{\psi}(t)\cos(\theta(t))\sin(\phi(t)) \\ -\dot{\phi}(t)\sin(\phi(t)) + \dot{\psi}(t)\cos(\phi(t))\cos(\theta(t)) \end{bmatrix}
\]

where \(\phi(t)\), \(\theta(t)\) and \(\psi(t)\) are the Euler angles in the inertial frame. By substituting equation (2) into equation (1), the angular motion equations can be written in the following form

\[
\begin{bmatrix} \dot{\phi}(t) \\ \dot{\theta}(t) \\ \dot{\psi}(t) \end{bmatrix} = \begin{bmatrix} 1 \\ \cos(\phi(t)) \\ \cos(\theta(t))\cos(\phi(t)) \end{bmatrix} \begin{bmatrix} \dot{p}(t) + A(t) \\ \dot{q}(t) + B(t) \\ \dot{r}(t) + C(t) \end{bmatrix}
\]

where

\[
A(t) = \ddot{\psi}(t)\sin(\theta(t)) + \dot{\psi}(t)\cos(\theta(t))\dot{\theta}(t)
\]

\[
B(t) = \tan(\phi(t))\dot{\phi}(t)\dot{\theta}(t) - \dot{\psi}(t)\cos(\theta(t))\tan(\phi(t))
\]

\[
- \dot{\psi}(t)\phi(t)\cos(\theta(t)) + \dot{\psi}(t)\sin(\theta(t))\dot{\theta}(t)\tan(\phi(t))
\]

\[
C(t) = \frac{\ddot{\theta}(t)\tan(\phi(t))}{\cos(\theta(t))} + \frac{\dot{\phi}(t)\dot{\theta}(t)}{\cos(\theta(t))} + \dot{\psi}(t)\tan(\theta(t))\dot{\theta}(t)
\]

As the flapping frequency and amplitude decrease, the changes in the three axial torques, \(L(t)\), \(M(t)\) and \(N(t)\), are nonlinear and continuous. As a result, the dynamics of the FWMAV system experience notable changes during the transition process between flapping and gliding. Moreover, the ADRC controller can be used to effectively predict the uncertainty in the system model and the external disturbances and to compensate for the error in advance to solve the relevant problems. Based on equations (3) to (6), the FWMAV system is typically regarded as having three control channels, namely \(\dot{\phi}(t)\), \(\dot{\theta}(t)\) and \(\dot{\psi}(t)\). There are obvious coupling effects among the three control channels, but the coupling among the channels is estimated and compensated for as a disturbance of a single control loop. Therefore, we can independently design a controller for each channel.

For simplicity, we take pitch control as an example in this paper. The other two control channels can be analyzed using the same method. The dynamics equation for the pitch channel can be expressed as follows

\[
f(t) = B(t) + \frac{1}{I_z\cos(\phi(t))}(I_{xz}\dot{r}(t)^2 - p^2(t)) + (I_z - I_r)\dot{r}(t)p(t) + \Delta u(t)
\]

\[
\frac{1}{I_z\cos(\phi(t))}M(t) = b_0u(t) + \Delta u(t)
\]
where \( u(t) \) is the control input signal, \( \Delta u(t) \) denotes the external disturbance, and \( f(t) \) represents the known and unknown nonlinear system dynamics, including all system acceleration and coupling terms. If we set \( y(t) = \theta(t) \), the longitudinal dynamics model of the FWMAV can be regarded as a second-order system, and the pitch dynamics in equation (3) can be written as follows

\[
\dot{y}(t) = f(t) + b_0 u(t)
\]

(9)

where \( b_0 \) is a gain coefficient that varies under different circumstances. Accordingly, \( b_0 \) should be adjusted to obtain reliable results in different practical applications. Additionally, we set \( x_1(t) = y(t) \) and \( x_2(t) = \dot{y}(t) \). Thus, the dynamics equation given in equation (9) is rewritten as follows

\[
\begin{align*}
\dot{x}_1(t) &= x_2(t) \\
\dot{x}_2(t) &= f(t) + b_0 u(t)
\end{align*}
\]

(10)

**Control design**

In consideration of the complicated nonlinear characteristics of the studied FWMAV during intermittent flapping and gliding flight, an ADRC strategy is designed to stably and accurately achieve attitude control. The block diagram of the ADRC is shown in Figure 6.

In Figure 6, the function of the tracking differentiator is to smooth the input signal \( v(t) \). The two outputs of the tracking differentiator are \( v_1(t) \) and \( v_2(t) \). \( v_2(t) \) is the differential value of \( v_1(t) \), i.e., \( \dot{v}_1(t) = v_2(t) \), which is tracked by the system output \( y(t) \). The ESO is designed to estimate the system states \( x_i(t) \) (\( i = 1,2 \)) and the total nonlinear system dynamics \( z_i(t) \). \( z_i(t) \) (\( i = 1,2 \)) denotes the observations of the system states. The nonlinear state error feedback controller utilizes the system state errors to determine the control inputs \( u_0(t) \), which can be compensated by the estimated value \( z_3(t) \) to obtain the final control variable \( u(t) \).

The design processes for the three components in Figure 6 and the meanings of the formulas in the figure are described below.

**Tracking differentiator design**

Since the input signal is occasionally abrupt, the differential signal is fairly large in magnitude. If the corresponding control law is designed by utilizing the large differential signal, system overshoot can easily occur. To avoid this problem, a transient process must be constructed. Note that the selection of a nonlinear function in the design of the tracking differentiator is not invariable. However, the most widely used form for tracking differentiators was proposed in Han,\(^2\) and this form has been proven to be correct and effective in various practical systems. On the basis of the form presented in Han,\(^2\), the tracking differentiator proposed in this paper is designed to obtain the continuous and smooth differential signal for the actual target signal in this paper. The tracking differentiator is designed as follows

\[
\begin{align*}
\dot{v}_1(t) &= v_2(t) \\
\dot{v}_2(t) &= f_{\text{han}}(v_1(t) - v(t), v_2(t), r_0, h_0)
\end{align*}
\]

(11)

where \( v(t) \) is the desired target signal. \( v_1(t) \) is the tracking signal of \( v(t) \), and \( v_2(t) \) is the tracking signal of \( v(t) \). \( f_{\text{han}}(\cdot) \) is an important nonlinear error function that is expressed as follows

\[
a = \begin{cases} 
\frac{d}{r_0 h_0} & \quad \text{if } |y| > d \\
\frac{v_2 + \frac{(a_0 - d)}{2} \text{sign}(y)}{h_0} & \quad \text{if } |y| \leq d \\
\frac{f_{\text{han}}(v_1 - v, v_2, r_0, h_0)}{h_0} & \quad \text{if } |y| > d \\
\frac{r_0 \text{sign}(a)}{d}, & \quad |a| > d \\
\frac{r_0 a}{d}, & \quad |a| \leq d 
\end{cases}
\]

(12)

where \( r_0 \) and \( h_0 \) are adjustable parameters that can be selected individually to obtain the desired tracking speed and smoothness. The parameter \( r_0 \) is related to
the tracking speed. From experience, we know that a higher value of $R_q$ usually results in a faster convergence speed until the physical limit is reached. The parameter $h_0$, which is called the filter factor of the tracking differentiator, can be adjusted in accordance with the desired tracking smoothness. Due to the complexity of the tracking differentiator, there is no suitable method for theoretically analysing the ranges of these two parameters. Therefore, the choice of these two values still depends on experience in actual applications.

**Linear ESO design**

To simplify the parameter adjustment process and controller structure for practical engineering applications, a linear ESO is introduced to address uncertain disturbances and the unknown nonlinear dynamics of the FWMAV system, as described in equation (10). For simplicity, the strongly nonlinear term $f(t)$, which represents both the internal dynamics and external disturbances, is assumed to be continuously differentiable and bounded. We extend the unknown nonlinear term $f(t)$ as an additional state $x_3$, i.e., $f(t) = x_3(t)$. Then, we assume that $w(t) = \dot{f}(t)$ is bounded in practice. The dynamics model of the FWMAV (10) is rewritten as follows

$$
\begin{align*}
\dot{x}_1(t) &= x_2(t) \\
\dot{x}_2(t) &= f(t) + b_0u(t) \\
\dot{x}_3(t) &= w(t) \\
y(t) &= x_1(t)
\end{align*}
$$

where $w(t)$ is the derivative of $x_3(t)$. By adopting the definition $e_1(t) = z_1(t) - y(t) = z_1(t) - x_1(t)$, a linear ESO for the system defined in equation (13) is constructed as follows

$$
\begin{align*}
\dot{z}_1(t) &= z_2(t) - \beta_1 e_1(t) \\
\dot{z}_2(t) &= z_3(t) - \beta_2 e_1(t) + b_0u(t) \\
\dot{z}_3(t) &= -\beta_3 e_1(t)
\end{align*}
$$

where $z_1(t)$, $z_2(t)$ and $z_3(t)$ are the observations of $x_1(t)$, $x_2(t)$ and $x_3(t)$, respectively, and $\beta_1$, $\beta_2$ and $\beta_3$ are the observer gains. The three observer gains can be parameterized as follows

$$
[ \beta_1, \beta_2, \beta_3 ] = [3w_0, 3w_0^2, w_0^3]
$$

where $w_0$ is the ESO bandwidth. As long as the tuning parameter $w_0$ is suitably set, the unknown nonlinear term $f(t)$ will be approximately tracked by the observer state $x_3(t)$. Therefore, this bandwidth parameterization method simplifies the tuning process and thus is widely used by engineers. Let $e_2(t) = z_2(t) - x_2(t)$ and $e_3(t) = z_3(t) - x_3(t)$ be the ESO tracking errors. Moreover, based on the system dynamics (equation (13)) and the constructed linear ESO (equation (14)), the error system is given by

$$
\begin{bmatrix}
\dot{e}_1(t) \\
\dot{e}_2(t) \\
\dot{e}_3(t)
\end{bmatrix} = A \begin{bmatrix}
e_1(t) \\
e_2(t) \\
e_3(t)
\end{bmatrix} + Bw(t)
$$

where $A = \begin{bmatrix} -\beta_1 & 1 & 0 \\
-\beta_2 & 0 & 1 \\
-\beta_3 & 0 & 0 \end{bmatrix}$ and $B = [0 0 -1]^T$.

We assume that $w(t)$ is bounded in practice. When the observer gains are selected through the tuning method described in equation (15), $A$ is a Hurwitz matrix. Furthermore, based on the previous analysis in Huang and Xue, the estimation errors $e_i(t)$ ($i = 1, 2, \text{and } 3$) of the ESO are bounded. Moreover, the larger the bandwidth $w_0$ of the ESO is, the faster the convergence speed. The system estimation errors can be made as small as possible by increasing the adjustable parameter $w_0$. Of course, in practice, the observer bandwidth is constrained by hardware limitations, such as the sampling frequency and sensor noise.

Notably, the constructed linear ESO (equation (14)) is able to observe the total disturbance, i.e., $f(t)$, since $x_3(t) = f(t)$, in addition to the state variables $x_1(t)$ and $x_2(t)$. Moreover, there is no need to obtain the information of $w(t)$; the observer requires only the system input $u(t)$ and output $y(t)$. The convergence of the closed-loop system will be verified later in this article.

**Nonlinear state error feedback controller design**

The different feedback forms of the system state variables significantly differ in restraining uncertain disturbances. The superiority of a nonlinear feedback form in this respect is obvious. The errors between the tracking differentiator and the linear ESO are given as follows

$$
\begin{align*}
\zeta_1(t) &= v_1(t) - z_1(t) \\
\zeta_2(t) &= v_2(t) - z_2(t)
\end{align*}
$$

where $v_1(t)$ and $v_2(t)$ are the given angle and angular velocity, respectively, and $z_1(t)$ and $z_2(t)$ are the observed signals of the corresponding system states. The nonlinear state error feedback control law $u(t)$ is defined as

$$
u(t) = \frac{x_1 \sin(\zeta_1(t), \zeta_1, \delta_0) + x_2 \sin(\zeta_2(t), \zeta_2, \delta_0) - z_3(t)}{b_0}$$

(18)
where $z_1$ and $z_2$ are two independent tuning parameters; $\lambda_1$ and $\lambda_2$ are two given constants, where $0 < \lambda_1 < 1 < \lambda_2$; and $\delta_0$ is a small given parameter representing the interval of the linear segment. The function $fal(\zeta, \lambda, \delta)$ is defined as follows

$$
fal(\zeta, \lambda, \delta) = \begin{cases} \frac{\zeta}{\delta^{1/2}}, & |\zeta| \leq \delta \\ |\zeta|^2 \text{sign}(\zeta), & |\zeta| > \delta \end{cases} \tag{19}
$$

For simplicity, $fal(\zeta, \lambda, \delta)$ is denoted by $fal(\zeta)$. The errors between the target signals and the system states are given as follows

$$
\begin{align*}
    s_1(t) &= v_1(t) - x_1(t) \\
    s_2(t) &= v_2(t) - x_2(t)
\end{align*} \tag{20}
$$

Then, the error equation (17) can be written in the following form

$$
\begin{align*}
    \zeta_1(t) &= s_1(t) - e_1(t) \\
    \zeta_2(t) &= s_2(t) - e_2(t)
\end{align*} \tag{21}
$$

Based on the above analysis, from equations (11), (17) and (20), we propose the following error system

$$
\begin{align*}
    \dot{s}_1(t) &= s_2(t) \\
    \dot{s}_2(t) &= \dot{v}_2(t) - f(t) - b_0 u(t)
\end{align*} \tag{22}
$$

Note that $\dot{v}_2(t)$ is the second derivative of the target signal $v_1(t)$, and in general, it is continuous and bounded. Moreover, from the error feedback control law (18) and error signal (20), it is easy to obtain the derivative of $s_2(t)$, which is given as follows

$$
\begin{align*}
    \dot{s}_2(t) &= \dot{v}_2(t) - f(t) - \alpha_3 fal(\zeta_1(t)) \\
                   &\quad - \alpha_2 fal(\zeta_2(t)) + z_3(t) \\
                   &= \dot{v}_2(t) - \alpha_1 fal(\zeta_1(t)) \\
                   &\quad - \alpha_2 fal(\zeta_2(t)) + e_3(t)
\end{align*} \tag{23}
$$

where $e_3(t) = z_3(t) - f(t)$ and is defined as the error between the estimated value and the system nonlinearity.

**Theorem:** Consider the closed-loop system (equation (22)) with the nonlinear state error feedback controller (equation (18)). If the controller gains $x_1$ and $x_2$ are chosen as appropriate positive values, the closed-loop system is stable. Thus, the system outputs $x_1(t)$ and $x_2(t)$ converge to the target inputs $v_1(t)$ and $v_2(t)$, respectively, based on control law (equation (18)).

**Proof:** A Lyapunov function is constructed as follows

$$
V(t) = \frac{\eta}{2} (s_1(t) - e_1(t))^2 + (s_2(t) - e_2(t))^2 \\
+ 2(s_1(t) - e_1(t))(s_2(t) - e_2(t)) \tag{24}
$$

where $\eta > 2$. It is easy to obtain that $V(t)$ is positive.

According to equations (18), (21) and (23), the derivative of $V(t)$ is obtained as follows

$$
\begin{align*}
    \dot{V}(t) &= \eta(s_1(t) - e_1(t))(\dot{s}_1(t) - \dot{e}_1(t)) \\
                   &\quad + 2(s_2(t) - e_2(t))(\dot{s}_2(t) - \dot{e}_2(t)) \\
                   &\quad + 2(s_1(t) - e_1(t))(\dot{s}_2(t) - \dot{e}_2(t)) \\
                   &\quad + 2(\dot{s}_1(t) - \dot{e}_1(t))(s_2(t) - e_2(t)) \\
                   &= (\eta(s_1(t) - e_1(t)) \\
                   &\quad + 2(s_2(t) - e_2(t))(s_2(t) - e_2(t) + \beta_1 e_1(t)) \\
                   &\quad - 2(s_1(t) - e_1(t) + s_2(t) - e_2(t))(\dot{s}_2(t) - \dot{e}_2(t)) \\
                   &\quad + 2(s_1(t) - e_1(t) + s_2(t) - e_2(t))(\dot{s}_2(t) - \dot{e}_2(t)) \\
                   &= -2(\zeta_1(t) + \zeta_2(t))(x_2 fal(\zeta_1(t)) \\
                   &\quad + x_2 fal(\zeta_2(t))) + H(t) \tag{25}
\end{align*}
$$

where

$$
H(t) = (\eta(s_1(t) - e_1(t)) + 2(s_2(t) - e_2(t))(s_2(t) - e_2(t)) \\
+ \beta_1 e_1(t)) - 2(s_1(t) - e_1(t)) \\
+ s_2(t) - e_2(t))(\dot{s}_2(t) - \dot{e}_2(t)) \\
+ 2(s_1(t) - e_1(t) + s_2(t) - e_2(t))(\dot{v}_2(t) + e_3(t)) \tag{26}
$$

According to the analysis above, we conclude that $e_1(t)$, $e_2(t)$ and $e_3(t)$ are bounded. Moreover, the errors $s_1(t)$ and $s_2(t)$ are also bounded in practice. Setting $S(t) = |H(t)|$, $S(t)$ is also bounded.

Letting $x_1 = x_2 = \lambda^*$, to simplify the analysis, we obtain

$$
\dot{V}(t) = -2\lambda^* G(\zeta_1(t), \zeta_2(t)) + H(t),
$$

where $G(\zeta_1(t), \zeta_2(t)) = (\zeta_1(t) + \zeta_2(t))(fal(\zeta_1(t)) + fal(\zeta_2(t)))$.

Note that the function $fal(\cdot)$ is a monotonically increasing odd function; therefore, the expression $(\zeta_1(t) + \zeta_2(t))$ has the same sign as the expression $(fal(\zeta_1(t)) + fal(\zeta_2(t)))$. Hence, we obtain $G(\zeta_1(t), \zeta_2(t)) \geq 0$. As a result, if the nonlinear gain coefficient $\lambda^*$ is sufficiently large, we obtain the following inequality

$$
\dot{V}(t) < -2\lambda G(\zeta_1(t), \zeta_2(t)) + S(t) \tag{27}
$$

That is, an appropriate coefficient $\lambda^*$ is sufficient to ensure that $\dot{V}(t) < 0$ always exists.

According to the analysis above, we conclude that if the coefficients $z_1$ and $z_2$ are appropriately chosen, then
the closed-loop system (equation (22)) is stable with the ADRC controller.

**Experimental results and analysis**

The flight test platform consisted of a Dove FWMAV, an autopilot, a ground control station, and a remote-control unit. An elevator was installed on the horizontal stabilizer to achieve attitude control. The microcontroller was a PX4 open-source autopilot running on Pixfalcon hardware. The QGroundControl software was used for control instruction generation and flight data surveillance. The flight target signal was generated by the autopilot. To verify the effectiveness and practicability of the proposed control method, we conducted several outdoor flight tests, including flapping flight tests, gliding flight tests and intermittent flapping and gliding flight tests. The experimental results presented here are based on real flight data. During testing, the data recording and the operation of the control algorithm were all performed online. Real flight data were recorded on the onboard memory card. In each of the three types of tests, the ADRC controller had the same controller gains. Moreover, the test results were analysed and compared with those of the proportional-derivative control method, which is one of the most universally used methods in aircraft attitude control applications. The controller structure of the proportional-derivative method is as follows:

$$u_{pd} = -K_p e_r - K_d \dot{e}_r$$

where $K_p = 0.1$ and $K_d = 0.08$. The parameters of the ADRC controller are listed in Table 2. The abbreviation “PitchSP” represents the pitch setpoint, which is the target signal to be followed. “PitchSpeed” represents the pitch rate. The term “Pitch” represents the system output “$y(t)$”.

**Experimental results in the flapping flight mode**

The target signal designed for the flapping flight tests is as follows

$$\theta(t) = -\frac{45}{180} \times \pi \times \left(0.1 \times \sin\left(\frac{2}{5} \times \pi t\right) - 0.15\right)$$  \hspace{1cm} (28)

It is a sinusoidal signal. The coefficients in the formula were determined based on the flight capability of the aircraft. The throttle signal was in the full position, that is, $r = 1$, during the flapping flight tests. The tracking effect of the ADRC controller is shown in Figure 7. Figure 8 shows the tracking effect of the proportional-derivative controller.

The flapping frequency was measured to be approximately 10 Hz in the flapping flight tests. Because of the influence of the wing motion, the pitch angle is characterized by periodic fluctuations within a small range. The pitch data is generally going to show up as a “double peak” in flapping flight. The reason for this behaviour is that there is a non-zero thrust component in addition to the lift that affects the attitude of the FWMAV. As shown in Figure 7, although the pitch angle fluctuates periodically, the overall trend tracks the target signal “PitchSP” with a small phase delay. Moreover, the observed value $z_1(t)$ and the pitch angle display nearly uniform variations. The experimental results are presented in Table 2.

### Table 2. Basic parameters of the ADRC controller.

| Parameter | Physical meaning | Value |
|-----------|------------------|-------|
| $r_0$     | Tracking speed factor of the tracking differentiator | 1000  |
| $h_0$     | Filtering factor of the tracking differentiator | 5     |
| $w_0$     | Bandwidth of the ESO | 40    |
| $x_1$     | Feedback factor of the nonlinear controller | 0.05  |
| $x_2$     | Feedback factor of the nonlinear controller | 0.1   |
| $b_0$     | Compensation factor of the nonlinear controller | 200   |

ADRC: Active disturbance rejection control.
results for tracking a sinusoidal signal with an amplitude of 4.5° and a frequency of 0.2 Hz using the proportional-derivative controller are shown in Figure 8. As shown in the graph, there is a long time delay of approximately 0.5 s in the pitch angle. The tracking error is approximately 3°. Compared with the results shown in Figure 7, although the proportional-derivative controller can address the attitude control problem in the flapping flight mode, the time delay and tracking accuracy are not satisfactory.

Experimental results in the gliding flight mode

The target signal designed for the gliding flight tests is as follows

\[
\theta(t) = \begin{cases} 
\frac{45}{180} \times \pi \times 0.1, & t < 0.5 \\
\frac{45}{180} \times \pi \times 0.25, & 0.5 < t < 2.5 
\end{cases}
\] (29)

It is a step signal. The coefficients in the formula were selected according to the actual variation range of the pitch angle. The throttle signal was in the zero position, that is, \( r = 0 \), during the gliding flight tests. The tracking effect of the ADRC controller is shown in Figure 9. Figure 10 shows the tracking effect of the proportional-derivative controller.

The efficiency of the control surface gradually decreases over time in the gliding flight mode. If the airspeed is too low, a flight stall may occur, which is a threat to flight safety. Therefore, the duration of gliding flight should not be too long, which requires a fast response time of the controller. In this study, the gliding time was designed to be 2.5 s. As shown in Figure 9, the pitch output can be controlled to track a step signal at 6.75° based on the ADRC method. The tracking differentiator constructs a transient process and smoothens the target signal. The time delay is approximately 1 s. The control results of the proportional-derivative controller are shown in Figure 10. As seen from the trend of the curves, as time continues to elapse, the pitch angle output may eventually be able to track the target signal. However, it is difficult to track the signal within 2.5 s. As shown in this figure, compared with the control results of the ADRC method, the proportional-derivative controller has a longer time delay and a larger tracking error.

Experimental results in the intermittent flapping and gliding flight mode

Birds in nature can dynamically adjust their intermittent flapping and gliding flight strategy in accordance with their flight states and the external environment. The energy costs are closely related to specific flight strategies. To verify the effect of the proposed controller, a fixed flight strategy was adopted in this study. A flow chart of the flight process during the intermittent flight tests is shown in Figure 11. Before each test, to ensure flight safety, it was necessary to check whether the altitude and attitude were in the appropriate ranges.

For the intermittent flapping and gliding flight tests, the target signal was designed as follows

\[
\theta(t) = \begin{cases} 
\frac{45}{180} \times \pi \times 0.3, & 0 < t < 0.5 \\
\frac{45}{180} \times \pi \times (-0.0937t + 0.3469), & 0.5 < t < 3.7 \\
\frac{45}{180} \times \pi \times (0.2t - 1.34), & 6.7 < t < 8.2 \\
\frac{45}{180} \times \pi \times 0.3, & 8.2 < t < 9.2
\end{cases}
\] (30)

Figure 9. Experimental results for tracking a step signal at 6.75° using the ADRC controller.

Figure 10. Experimental results for tracking a step signal at 6.75° using the proportional-derivative controller.

Figure 11. Flow chart of the flight process during the intermittent flight tests.
To achieve safe intermittent flapping and gliding flight, the throttle signal was designed as follows.

\[
    r = \begin{cases} 
    1.0 & t < 0.5 \\
    -0.2667t + 1.1333 & 0.5 < t < 2 \\
    -0.1429t + 0.8857 & 2 < t < 2.7 \\
    -0.5t + 1.85 & 2.7 < t < 3.7 \\
    0 & 3.7 < t < 5.7 \\
    0.6667t - 3.8, 5.7 < t < 7.2 
    \end{cases}
\]

(D31)

During intermittent flapping and gliding flight, the throttle signal was controlled based on formula (31). The result was that the flapping frequency decreased to zero in the transition mode. Snapshot images of the flapping mode and the gliding mode are shown in Figure 12. The tracking effect of the ADRC controller is shown in Figures 13 through 15.

Figure 13 shows the pitch angle fluctuations over time during intermittent flapping and gliding flight. As shown in Figure 13, the pitch angle fluctuates periodically around the target angle during the transition process from flapping flight to gliding flight. The tracking performance is good. In the gliding flight mode, the decrease in the control efficiency results in a poor control effect. The error between the peak pitch angle and the target angle is approximately 6°. During the transition from gliding flight to flapping flight, the start of flapping motion results in an abnormal spike in the pitch angle. After this spike, the pitch angle begins to track the target angle again. Throughout the whole transition process, the output value \( z_1(t) \) of the ESO varies with the pitch angle, thereby verifying the effectiveness of the ESO. Figure 14 shows the observed results for the pitch angle rate. The true pitch angle rate was measured by the onboard sensors. The output value \( z_2(t) \) of the ESO varies with the pitch angle rate. When the ADRC method is adopted, the number of sensors required may be reduced. Figure 15 shows the control input of the ADRC controller and the estimated value of the system nonlinearity. First, because of the decrease in the control efficiency during gliding flight, the estimated value of the system nonlinearity gradually decreases. After the calculation of the control law in equation (18), the control input gradually increases to compensate for the decrease in
the control efficiency. Subsequently, at the spike position, the estimated value of the system nonlinearity rapidly increases. Then, the input value decreases to counteract the negative influence caused by the start of the flapping motion. The estimated value of the system nonlinearity $z_3(t)$ and the input value exhibit opposite variational trends, embodying the characteristics of system nonlinearity estimation and compensation.

To further explain the characteristics of the ADRC strategy, the control effects of the proportional-derivative control method are compared. Figure 16 shows the corresponding experimental results. As shown in Figure 16, complete intermittent flapping and gliding flight can be achieved by the proportional-derivative controller. However, compared with the experimental results based on the ADRC method shown in Figure 13, the tracking error of the proportional-derivative controller is larger during flapping flight. The error between the peak pitch angle and the target angle is approximately 11° during gliding flight, which is greater than the 6° error of the ADRC controller. At the spike position, because the system nonlinearity is not estimated and compensated, the proportional-derivative controller can track the target signal only after two oscillations.

The reason for the above phenomenon is as follows. During intermittent flapping and gliding flight, the system nonlinearity exhibits large variations. The ADRC method performs real-time nonlinear estimation and compensation; therefore, it is a more appropriate method to be used in intermittent flapping and gliding flight.

Because of the complexity of the outdoor flight environment, the initial values of the system and the external disturbances could not be identical in each test. Therefore, the above analysis is only a qualitative analysis of the experimental results. However, the findings sufficiently indicate that the proposed ADRC method provides effective control during intermittent flapping and gliding flight.

**Conclusion**

To address the problems related to large variations in system dynamics, imprecise mathematical models and unknown external disturbances in intermittent flapping and gliding flight, an ADRC controller is successfully designed for the attitude stabilization of the Dove FWMAV operating in this intermittent flight mode. The results of outdoor flight experiments show that the controller proposed in this paper can achieve satisfactory control effects in terms of tracking precision and phase delay. Moreover, due to its real-time compensation capability, the controller can effectively counteract the adverse effects caused by the start of flapping motion and has good prospects for engineering applications.

Although the controller constructed in this paper is based on the Dove FWMAV, it can also be used in other bird-like FWMAVs with flapping and gliding flight capabilities. In the experiment section, the target motions introduced in this paper were revised repeatedly according to the flight data and flight experience. In general, different target trajectories will result in different levels of energy consumption. Therefore, there may be better target trajectories. In the future, we will optimize the flapping and gliding strategy to reduce the flight energy consumption.

**Declaration of conflicting interests**

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