Transient Flight Controller of Vertical Take-Off and Landing UAV based on Multimode Integrated Design

Bin Wu, Xiaodong Song, Zhenchang Liu
School of Aerospace Engineering, Beijing Institute of Technology, Beijing 100081, China
wubinworks@163.com

Abstract: The controller is designed for the vertical take-off and landing mode, the transient flight mode and the fixed wing flight mode, respectively; a control signal fusion method is introduced in the transition flight control mode, which realizes the smooth switching of aerial vehicle in different flight modes; aiming at the problem of the control plane redundancy, the method of advancing amplitude of redundant control quantity is adopted to simplify the control distribution process. It is verified that the control method of Vertical Take-Off and Landing UAV can track the flight path well via six degrees of freedom simulation of the aerial vehicle.

1. Introduction
With the rapid development of related technologies for UAV, the functions that a single uav can accomplish are diversified. Compared with manned aircraft, UVA has the advantages of low production cost, safety, and relatively simple design, which suitable for replacing human beings to perform some high-risk tasks. At present, some of the main types of UAVs in the world mainly include fixed-wing UAVs and rotorcraft. The fixed-wing UAV adopts aerodynamic lift to offset the gravity, so it has high flight efficiency and is suitable for long-term and long-distance missions. However, it cannot achieve hover and ultra-low speed flight because the generation of lift depends on the flight speed; rotary-wing UAV can take off, land and hover with direct force generated by propeller, but it has a lower flight speed and a shorter cruising distance than the fixed-wing drone[1]. The above two types of UAVs have great application value in military and civilian fields, but their application range has been greatly limited due to their respective defects.

The UAV with vertical take-off and landing function can not rely on the take-off and landing site, and because of the existence of the vector engine, the aircraft has good low-speed maneuverability[2], and at the same time has the high-speed cruising ability of the fixed-wing aircraft. These features make the vertical takeoff and landing UAV a rising craze in the industry. At present, China has a vertical take-off and landing function. The research object in this paper adopts a vertical take-off and landing aircraft with serial wing layout[3], designs a controller for its three modes of vertical take-off and landing, transient flight and fixed-wing flight, and realizes the smooth switching of modes by means of control signal fusion. The research focuses on the low-speed maneuverability of the uav in transient flight.

2. Controlled Object
The purpose of controller design is to find a good mapping relationship between control quantity and state quantity. Vertical take-off and landing aircraft have different flight mechanics characteristics when flying in different modes. In order to design the controller more reasonably, the vertical coordinate mode
often uses the body coordinate system as the reference system, and the cruise mode often uses the track coordinate system as the reference system. This paper simplifies the control model under different reference systems and eliminates redundancy and coupling in the model.

2.1 Model of Vertical Take-off and Landing Mode

When the aircraft performs vertical take-off and landing flight, the thrust of the engine directly overcomes the gravity and tracks the flight path through the change of attitude. In this mode, the establishment of the control model under the coordinate system of the body requires the following three assumptions:

(a) During the vertical take-off and landing, the balance point between the pitch angle \( \theta \) and the roll angle \( \phi \) of the aircraft is \( 0^\circ \). When there is a slight change in \( \theta \) and \( \phi \), the attitude of the aircraft changes, causing a change in displacement. It is assumed that the aircraft does not make drastic maneuvers in this mode, and the attitude angle changes little, therefore, \( \phi \approx 0^\circ \) and \( \theta \approx 0^\circ \).

(b) During the vertical take-off and landing process, it is determined that the engine only produces thrust in the X-Z plane of the body, and no lateral force is generated.

(c) In the process of vertical take-off and landing, the UAV flies at a low speed and has weak aerodynamic force. Therefore, the aerodynamic action is ignored and the actual aerodynamic force is considered to be an external disturbance.

In the body coordinate system, the motion equation of the center of mass of the aircraft can be expressed as:

\[
\begin{bmatrix}
\dot{X} \\
\dot{Y} \\
\dot{Z}
\end{bmatrix} =
\begin{bmatrix}
\cos \psi \cos \theta & \sin \theta \sin \phi \cos \psi - \cos \phi \sin \psi & \sin \theta \cos \phi \cos \psi + \sin \phi \sin \psi \\
\sin \psi \cos \theta & \sin \theta \sin \phi \sin \psi + \cos \phi \cos \psi & \sin \theta \cos \phi \sin \psi - \sin \phi \cos \psi \\
-\sin \theta & \sin \phi \cos \theta & \cos \theta
\end{bmatrix} 
\begin{bmatrix}
u \\
v \\
w
\end{bmatrix}
\]

(1)

According to hypothesis (1), it can be simplified as

\[
\begin{bmatrix}
\dot{X} \\
\dot{Y} \\
\dot{Z}
\end{bmatrix} =
\begin{bmatrix}
\cos \psi & -\sin \psi & 0 \\
\sin \psi & \cos \psi & 0 \\
0 & 0 & 1
\end{bmatrix} 
\begin{bmatrix}
u \\
v \\
w
\end{bmatrix}
\]

(2)

According to hypothesis (2) and (3), and ignore the vector product of angular velocity and velocity, then:

\[
\begin{bmatrix}
\dot{u} \\
\dot{v} \\
\dot{w}
\end{bmatrix} =
\begin{bmatrix}
-g \sin \theta \\
g \sin \phi \cos \theta \\
g \cos \phi \cos \theta + T_z / m
\end{bmatrix}
\]

(3)

The kinematic equation of rotation around the center of mass can be expressed as:

\[
\begin{bmatrix}
\dot{\phi} \\
\dot{\theta} \\
\dot{\psi}
\end{bmatrix} =
\begin{bmatrix}
1 & 0 & 0 \\
0 & 1 & 0 \\
0 & 0 & 1
\end{bmatrix} 
\begin{bmatrix}
p \\
q \\
r
\end{bmatrix}
\]

(4)

The kinematics equation of center of mass rotation can be expressed as:

\[
\begin{bmatrix}
I_x & 0 & I_{xz} \\
0 & I_y & 0 \\
I_{za} & 0 & I_z
\end{bmatrix} 
\begin{bmatrix}
\dot{p} \\
\dot{q} \\
\dot{r}
\end{bmatrix} +
\begin{bmatrix}
p \\
q \\
r
\end{bmatrix} 
\begin{bmatrix}
I_x & 0 & I_{xz} \\
0 & I_y & 0 \\
I_{za} & 0 & I_z
\end{bmatrix} =
\begin{bmatrix}
l_t \\
m_t \\
n_t
\end{bmatrix}
\]

(5)

The control quantity mapping relationship can be established as shown in the equation via the above equation,
2.2 Cruise model in Flight Path Axis System

Unlike the vertical take-off and landing flight mode, the main control force of the UAV during cruise is aerodynamic. Through the change of the angle of attack and the angle of sideslip, the aerodynamic force fixed to the track coordinate system is changed indirectly, thus the flight path is changed. In this paper, the vertical take-off and landing UAV is modeled in the cruising state, and the controller of the cruising mode is designed by means of decoupling control in the vertical and horizontal direction. First, the following assumptions should be adopted:

(1) It is assumed that in the flight of the balance point, track drift angle $\chi=0$, sideslip angle $\beta=0$, velocity roll angle $\mu=0$, angle of attack $\alpha=0$ and track dip angle $\gamma=0$;

(2) It is assumed that the thrust of the UAV is along the axis $Ox_b$ of the body when flying at the equilibrium point, i.e. $T_y=0$, $T_z=0$;

(3) It is assumed that when flying at the balance point, track dip angle and track drift angle change rate are ignored, and there is approximately $\dot{\gamma}=0$, $\dot{\chi}=0$.

According to the hypothesis, the longitudinal and lateral kinematics equations of UAV in the track coordinate system can be established. The longitudinal motion equation of UAV can be expressed as:

$$
\begin{align*}
\dot{X} &= V \cos \gamma \\
\dot{Z} &= -V \sin \gamma \\
\dot{V} &= \frac{T_x - D}{m} - g \sin \gamma \\
\dot{\gamma} &= \frac{g \cos \gamma}{V} + \frac{L}{mV}
\end{align*}
$$

The transverse motion equation can be expressed as:

$$
\begin{align*}
\dot{Y} &= V \sin \chi \\
\dot{\chi} &= \frac{L \sin \mu}{mV}
\end{align*}
$$

According to hypothesis (3), the kinematic equation of center of mass rotation can be expressed as:

$$
\begin{pmatrix}
\dot{p} \\
\dot{q} \\
\dot{r}
\end{pmatrix} =
\begin{bmatrix}
1 & 0 & 0 \\
0 & 1 & 0 \\
0 & 0 & 1
\end{bmatrix}
\begin{pmatrix}
p \\
q \\
r
\end{pmatrix}
$$

Since the dynamic equations of rotation around the center of mass in both vertical take-off and landing state and cruising state are established in the coordinate system of the body, the simplified method is the same. The mapping relationship can be constructed in the cruise state:

$$
\begin{align*}
\mu &\rightarrow p \rightarrow l_s \\
\alpha &\rightarrow q \rightarrow m_s \\
\beta &\rightarrow r \rightarrow n_s
\end{align*}
$$

3. Design of Transient Flight Controller

In the previous section, the establishment of the control model has been completed and the mapping relationship between control quantity and state quantity has been constructed through simplification.
Based on the simplified model, this paper stratifies the input response speed according to the state quantity according to the time-scale separation method. In the vertical take-off and landing mode, cascade PID method is adopted for attitude and flight path tracking control. In cruise mode, the total energy method is used for flight path tracking control. Based on the above two methods, during the transition of vertical take-off and landing, the UAV can switch smoothly between three modes: vertical take-off and landing, low speed cruising and cruising through the fusion of design.

3.1 Controller for Vertical Take-off and Landing

According to the principle of time scale separation method, the control quantity is divided into internal and external loop. The external loop controls the position and the internal loop controls the attitude. By properly adjusting the control parameters of the internal loop, the interference into the internal loop can be effectively suppressed, the anti-disturbance ability of the system can be improved, and the dynamic characteristics of the system can be improved. In addition, cascade PID has the advantages of good model adaptability, strong anti-interference, fast adjustment speed and small oscillation period.

3.1.1 Controller for External Loop

In the vertical take-off and landing mode, similar to rotorcraft UAV, the attitude change and the track change are coupled. When the attitude changes, the thrust produces a component in the horizontal plane, causing a displacement motion in the horizontal direction. Therefore, the external loop is needed for position control. The desired attitude angle is calculated via the desired position solution and input into the cascade PID controller.

Firstly, the three-axis direction reference speed command can be obtained according to equation 2, namely:

\[
\begin{bmatrix}
  u^r \\
  v^r \\
  w^r \\
\end{bmatrix} =
\begin{bmatrix}
  \dot{X}^d \cos \psi + \dot{Y}^d \sin \psi \\
  Y^d \cos \psi - \dot{X}^d \sin \psi \\
  \dot{Z}^d \\
\end{bmatrix}
\]

(11)

In the formula, superscript \( r \) represents the reference value and superscript \( d \) represents the expected value. Where, the expected velocity information can be obtained according to the feedback error between the reference track coordinate and the actual position coordinate, namely:

\[
\begin{bmatrix}
  \dot{X}^d \\
  \dot{Y}^d \\
  \dot{Z}^d \\
\end{bmatrix} =
\begin{bmatrix}
  X^r \\
  Y^r \\
  Z^r \\
\end{bmatrix} -
\begin{bmatrix}
  X \\
  Y \\
  Z \\
\end{bmatrix}
\]

(12)

In the formula, \( k_x, k_y \) and \( k_z \) are controller gains.

Similarly, the control command of reference pitch angle and reference roll angle can be obtained according to formula 3:

\[
\begin{align*}
\theta' &= -\arcsin \left( \frac{\dot{u}^d}{g} \right) \\
\phi' &= \arcsin \left( \frac{\dot{v}^d}{g \cos \theta'} \right)
\end{align*}
\]

(13)

In the formula, \( \dot{u}^d \) is the \( O_x^b \) axial expected acceleration; \( \dot{v}^d \) is the \( O_y^b \) axial expected acceleration, and the calculation method is as follows:

\[
\begin{bmatrix}
  u^d \\
  v^d \\
\end{bmatrix} =
\begin{bmatrix}
  k_u \\
  k_v \\
\end{bmatrix}
\begin{bmatrix}
  \dot{u}^r \\
  \dot{v}^r \\
\end{bmatrix} -
\begin{bmatrix}
  u \\
  v \\
\end{bmatrix}
\]

(14)

In the formula, \( k_u \) and \( k_v \) are controller gains.
The reference axial $O_z$ thrust $T_z^*$ can be calculated by the height cascade PID controller. The external loop of the controller adopts P control, and the control output is obtained. The output of the external loop is the input of the internal loop, and the following is obtained:

$$T_z^* = k_{pw} \left( w' - w \right) + k_{dw} \frac{d}{dt} \left( w' - w \right) + k_{iw} \int \left( w' - w \right) dt$$  \hspace{1cm} (15)

In the formula, $k_{pw}$, $k_{dw}$ and $k_{iw}$ are the controller parameters.

### 3.1.2 Controller for Internal Loop

The output attitude information of the above position controller is used as an input of the internal loop attitude controller. The inner ring controls the three movements of roll, pitch and yaw.

#### 3.1.2.1 Roll and Pitch Control

In the roll and pitch angle control, the input of the external loop is the reference attitude angle control command obtained by the position control in equation (13).

$$\begin{align*}
p' &= k_{pp} \left( \dot{\phi} - \phi \right) + k_{dp} \frac{d}{dt} \left( \dot{\phi} - \phi \right) + k_{ip} \int \left( \dot{\phi} - \phi \right) dt \\
qu' &= k_{pq} \left( \dot{\theta} - \theta \right) + k_{dq} \frac{d}{dt} \left( \dot{\theta} - \theta \right) + k_{iq} \int \left( \dot{\theta} - \theta \right) dt
\end{align*}$$  \hspace{1cm} (16)

Where, $k_{pp}$, $k_{ip}$ and $k_{dp}$ are the PID parameters of roll angular control; $k_{pq}$, $k_{iq}$, and $k_{dq}$ are the PID parameters for pitch angular velocity control. The input of the inner loop is the output of the outer loop. The output of the inner loop is the reference torque control command $l_p^*$, $m_q^*$. The calculation method is as shown in Equation (17).

$$\begin{align*}
l_p^* &= k_{pp} \left( p' - p \right) + k_{dp} \frac{d}{dt} \left( p' - p \right) + k_{ip} \int \left( p' - p \right) dt \\
m_q^* &= k_{pq} \left( q' - q \right) + k_{dq} \frac{d}{dt} \left( q' - q \right) + k_{iq} \int \left( q' - q \right) dt
\end{align*}$$  \hspace{1cm} (17)

Where, $k_{pp}$, $k_{ip}$ and $k_{dp}$ are the PID parameters of roll angular velocity control; $k_{pq}$, $k_{iq}$, and $k_{dq}$ are the PID parameters for pitch angular velocity control.

#### 3.1.2.2 Yaw Control

It is found through the observation formula (11) and (13): in the calculation of the controller, the yaw angle will affect the value of the body's lateral velocity ($v$), and $v$ will affect the value of the body's roll angle. If $v$ remains large, the aircraft will continue to roll sideways. During the transient flight, with the gradual increase of flight speed, this maneuvering mode is not conducive to the efficient use of aerodynamic force by UAV. Therefore, in the design of yaw controller, it is necessary to determine the selection strategy first, so as to keep $v$ within a reasonable range, and to make the nose point to the incoming flow direction as far as possible during stable flight. The design yaw angle selection strategy is shown in equation (18):

$$\begin{align*}
\psi' &= 0, \hspace{1cm} |v'| \leq 10 \\
\psi' &= k_{\psi} \arctan \left( \frac{\dot{Y}^d}{\dot{X}^d} \right), \hspace{1cm} |v'| > 10
\end{align*}$$  \hspace{1cm} (18)

In the formula (18), $k_{\psi}$ is the coefficient of variation of the yaw angle. Its function is to make $\psi'$ continuous changes and avoid the shock of the control system caused by sudden changes. Its definition is shown in equation (19).

$$\begin{align*}
k_{\psi} &= \left( \frac{v' - 10}{10} \right)^2, \hspace{1cm} 10 < v' < 20 \\
k_{\psi} &= 1, \hspace{1cm} 20 \leq v'
\end{align*}$$  \hspace{1cm} (19)
In the position control, \( v^r \) is calculated according to the formula (11), if \(|v^r| \leq 10 \), the UAV performs lateral maneuvering by rolling, and \( \psi^r = 0 \); if \(|v^r| > 10 \), \( \psi^r = k_\psi \tan^{-1}(Y^d/X^d) \), the UAV changes the maneuvering direction by adjusting the yaw angle. In the attitude control, according to the yaw angle, and the input of the external loop is \( \psi^r \), and the output of the external loop is \( r^r \), and the calculation method is as shown in equation (20).

\[
r' = k_{pr} (\psi' - \psi) + k_{dr} \frac{d}{dt}(\psi' - \psi) + k_{dr} \int (\psi' - \psi) dt \tag{20}
\]

In formula (20): \( k_{pr}, k_{dr}, k_{dr} \) are the yaw angle control PID parameters. The input of the internal loop is the output of the external loop, and the output of the internal loop is the reference torque control command \( n^r_t \), and the calculation method is as shown in equation (3.21).

\[
n^r_t = k_{pr} (r' - r) + k_{dr} \frac{d}{dt}(r' - r) + k_{dr} \int (r' - r) dt \tag{21}
\]

As the drone ignores the aerodynamic force during vertical take-off and landing, \( \delta_a = 0, \delta_e = 0 \); since \( T_x = 0, \delta_L = \delta_R = -90^\circ \). Because \( T_F \) is larger, \( \delta_e \) only needs a smaller yaw angle to produce a sufficient expected yaw moment and therefore approximates \( \cos \delta_F = 1 \). Therefore, the power system model can be obtained:

\[
\begin{bmatrix}
T_z \\
l_t \\
m_r \\
T_L \\
T_R \\
T_F
\end{bmatrix} =
\begin{bmatrix}
-1 & -1 & -1 \\
d_{yx} & -d_{yx} & 0 \\
-d_{uz} & -d_{uz} & d_{sf} \\
0 & 0 & 0
\end{bmatrix}
\begin{bmatrix}
T_L \\
T_R \\
T_F
\end{bmatrix}
\tag{22}
\]

\[
n^r_t = T_F \sin \delta_F d_{sf} \tag{23}
\]

According to formula (22) and formula (23), the engine reference thrust (1) and the lift fan rudder angle (2) can be obtained as shown in equations (24) and (25)

\[
\begin{bmatrix}
T'_L \\
T'_R \\
T'_F
\end{bmatrix} =
\begin{bmatrix}
-1 & -1 & -1 \\
d_{yx} & -d_{yx} & 0 \\
-d_{uz} & -d_{uz} & d_{sf}
\end{bmatrix}
\begin{bmatrix}
T_z \\
l_t \\
m_r
\end{bmatrix}
\tag{24}
\]

\[
\delta'_F = \arcsin\left(\frac{n^r_t}{T'_F d_{sf}}\right) \tag{25}
\]

Finally, according to the actual performance of the power system and the actuator, the controller output signal can be obtained after limiting the engine thrust, the vector nozzle yaw angle, and the vector rudder angle.

### 3.2 Design of Cruise Control

During the design of cruise controller, both longitudinal and transverse flight path tracking requirements should be considered at the same time, so the controller is designed separately in the two motion planes. Based on the decoupling of the longitudinal and lateral motion of the six degrees of freedom model, the control of the axial thrust \( (T_x) \) and the elevator \( (\delta_e) \) along the \( \Delta_s \) is calculated by the Total Energy Control System (TECS) in the longitudinal control. The advantage of this method is that the total energy variation and transfer characteristics are used to effectively align the short-period attitude motion and the long-period particle motion characteristics of the aircraft. The yaw control torque \( (n_t) \) and the rudder control \( (\delta_a) \) control information are calculated by the PID method in the lateral control. Finally, the engine thrust \( (T_L, T_R) \) is comprehensively solved based on the outputs of the two controllers.

### 3.2.1 Longitudinal Control

The TECS method adopts the multi-input and multi-output control strategy, breaking through the structural constraints of the internal and external loop nested control loops, enabling the system to
provide a unified standard for all flight control modes, and the decoupling link can be directly constructed, which makes the analysis and design of the system easier\cite{3}.

In TECS method, the following assumptions are made:

(a) The flight aircraft inclination angle $\gamma$ is small, similar to $\sin \gamma \approx \gamma$.

(b) During cruising, the initial thrust ($T_x$) is used to offset the aircraft's resistance($D$) and to ignore the change in $D$ caused by the change in $T_x$.

According to the above hypothesis and equation (6), the thrust required by the aircraft during flight is:

$$T_x = mg \left( \gamma + \frac{\dot{V}}{g} \right) + D$$

(26)

After the aircraft is disturbed, the required thrust increment is

$$T_x - D = G \left( \gamma + \frac{\dot{V}}{g} \right)$$

(27)

The total energy of the aircraft can be expressed as:

$$E_T = \frac{1}{2} \frac{G}{g} V^2 + GH$$

(28)

In formula 28, $E_T$ is the total energy of the aircraft, $E_P$ is the kinetic energy of the aircraft, $E_P$ is the potential energy of the aircraft; $H$ is the altitude of the aircraft and $H = -Z$.

The total energy per unit weight can be expressed as:

$$E = \frac{V^2}{2g} + H$$

(29)

The total energy per unit weight can be differentiated and sorted out:

$$\frac{\dot{E}}{V} = \frac{\dot{V}}{g} + \gamma = \frac{T_x - D}{mg}$$

(30)

It can be seen from formula (30) that during the flight of the aircraft, the change in total energy is controlled by the change in thrust. According to the hypothesis (b), the incremental control effect of the thrust can be expressed as:

$$\Delta T_x = G \left( \left( \frac{\dot{V}}{g} \right) + \gamma_e \right)$$

(31)

In formula (31), $\Delta T_x$ represents thrust increment; Table $e$ below represents parametric deviations. According to this formula, It can be concluded from equation (31) that the change in aircraft thrust causes the sum of the acceleration and yaw angle of the aircraft to change. Based on this, the control relationship between thrust ($T$) and total energy change rate ($\dot{E}$) can be obtained as follows

$$T' = \left( k_p + \frac{k_i}{s} \right) \dot{E}_e$$

(32)

In formula (32), $k_p$ is the thrust proportional gain; $k_i$ is the thrust integral gain.

The deflection of the elevator mainly causes the change of pitching moment, changes the attitude of the aircraft, but has little influence on the thrust and resistance. Therefore, the elevator can be used as the controller to distribute the kinetic energy and potential energy of the aircraft. In order to make the kinetic energy and potential energy have the same control priority, the energy distribution rate ($L$) is defined as:

$$L = \gamma - \frac{\dot{V}}{g}$$

(33)
The control relationship between the elevator yaw angle $\delta_e$ and the energy distribution rate can be expressed as:

$$\delta'_e = \left( k_{pe} + \frac{k_{ie}}{s} \right) L_e$$

(34)

In the formula, $k_{pe}$ is proportional gain and $k_{ie}$ is elevator integral gain. The basic principles are as follows: control the rate of change of total energy with thrust, and control the rate of distribution of total energy with elevator. In order to improve the dynamic characteristics of the system, the proportion part of the core algorithm uses $\dot{E}$ and $\dot{L}$ instead of $E_e$ and $L_e$ to obtain the control rate:

$$\begin{align*}
T'_e &= k_{pe} \dot{E} + \frac{k_{ie}}{s} \dot{E}_{eh} \\
\delta'_e &= k_{pe} \dot{L} + \frac{k_{ie}}{s} \dot{L}_{eh}
\end{align*}$$

(35)

3.2.2 Lateral Controller

The aircraft lateral controller solves the control torque of the aircraft based on the target flight path information, the current position information, and the speed information. Let $L = [X' - X \ Y' - Y]$ be the direction vector of the x-y plane. The distance between the aircraft and the target flight path reference point at the current moment can be expressed as:

$$|L| = \sqrt{(X' - X)^2 + (Y' - Y)^2}$$

(36)

The angle $\tau$ between the projection of the aircraft velocity vector $v_g$ in the x-y plane and the direction vector $L$ can be expressed as:

$$\tau = \arccos \left( \frac{v_{gxy} L}{|v_{gxy}| \times |L|} \right)$$

(37)

The reference normal acceleration of the aircraft can be calculated as:

$$a'_y = k_a |L| \sin \tau$$

(38)

In the formula, $k_a$ is the artificially set normal acceleration control gain. When the target flight path reference point is on the right side of the aircraft speed direction, $k_a > 0$, and on the left side of the aircraft speed direction, $k_a < 0$. The geometric relationship between the speed of the aircraft, the normal velocity and the position of the reference point is shown in the figure.

![Fig. 1 Schematic diagram of lateral motion geometry](image)

The expression for calculating the reference roll angle $\phi'^r$ in accordance with $a_y$ is:

$$\phi'^r = \arctan \left( \frac{a'_y}{g} \right)$$

(39)

According to the state feedback of roll angle, the PID controller is designed, and the PID controller is designed to obtain the required control torque:

$$l' = k_{p\phi} (\phi' - \phi) + k_{d\phi} \frac{d(\phi' - \phi)}{dt} + k_{i\phi} \int (\phi' - \phi) \ dt$$

(40)

In formula (40), $k_{p\phi}$, $k_{d\phi}$ and $k_{i\phi}$ are PID parameters in roll angle control.
The UAV adopts the yaw static stability design, with $\beta$ as the static stability point, so there is no need to design the controller to stabilize in the yaw direction. Considering the stability of flight when receiving interference, the stability of $\Psi = \Psi^r$ or $\Psi = 0$ can be stabilized. According to the state feedback of $\Psi$ and $\Psi^r$, the yaw moment control command $n_t$ is generated by the PID controller.

When the UAV is cruising, since $T_z = 0$, $\delta L = 0$, $\delta R = 0$, and the lift fan does not work, so $T_F = 0$, $\delta F = 0$, since $l_s = l_c$, $m_s = m_c$, so $l_t = m_t = 0$. According to formula (35) the dynamics model of the UAV can be simplified to:

$$
\begin{align*}
T_z &= T_L + T_R, \\
n_t &= (T_L - T_k)d_y.
\end{align*}
$$

According to formula 41, the reference engine thrust can be obtained as shown in formula 42.

$$
\begin{align*}
T'_L &= 0.5 \left( T'_L + \frac{n_t}{d_y} \right), \\
T'_R &= 0.5 \left( T'_R - \frac{n_t}{d_y} \right).
\end{align*}
$$

### 3.3 Transient Process Control

In the transition stage, the thrust control quantity solved by the vertical take-off and landing controller can be expressed as $T_{vtol} = [0]_T^T$, and the torque control quantity can be expressed as $M_{vtol} = [L M N]^T$. The thrust control quantity solved by the cruise controller can be expressed as: $T_{cruise} = [T_x^r 0 0]^T$, the torque control quantity can be expressed as: $M_{cruise} = [l_s^r m_s^r 0]^T$. It is assumed that the transitional state is when the flying speed of the UAV is at from 10 m/s to 20 m/s. The design fusion system $k_{mix}$ is:

$$
k_{mix} = \frac{1}{2} \left( \sin \left( \frac{V - 15}{10} \pi \right) + 1 \right), 10 m/s \leq V \leq 20 m/s
$$

The desired control force and torque of transition flight state after fusion can be expressed as follows:

$$
\begin{align*}
T_{mix} &= k_{mix} T_{cruise} + (1 - k_{mix}) T_{vtol}, \\
M_{mix} &= k_{mix} M_{cruise} + (1 - k_{mix}) M_{vtol}.
\end{align*}
$$

In consideration of the two factors of $V$ and $T_z$ in the transient flight, the $M_{mix}$ is re-controlled and distributed to obtain the pneumatic control torque $M_{s, mix} = [l_s m_s 0]^T$, and thrust control torque $M_{t, mix} = [l_t m_t 0]^T$, in the transient flight are obtained. When $V$ is larger, the proportion of $l_s$ and $m_s$ is larger; the smaller $T_z$ is, the larger proportion of $l_t$ and $M_t$ is. When $T_z = 0$, $l_t = 0$ and $M_t = 0$. Therefore, the design distribution coefficients $k_v$ and $k_T$ are shown as in formula 45 and 46

$$
\begin{align*}
k_v &= \frac{1}{2} \left( -\sin \left( \frac{V - 15}{10} \pi \right) + 1 \right), \\
k_T &= \frac{|T_z|}{mg}, |T_z| < mg, \\
k_T &= 1, |T_z| \geq mg.
\end{align*}
$$

The control torques in the transient flight can be expressed as:
\[ M_{s,t,\text{mix}} = \begin{bmatrix} l_s \\ m_t \\ n_s \end{bmatrix}_{\text{mix}} = \begin{bmatrix} l_c (1 - k_v) \\ m_t (1 - k_v) \\ 0 \end{bmatrix}_{\text{mix}} \] (47)

\[ M_{t,m,\text{mix}} = \begin{bmatrix} l_t \\ m_t \\ n_t \end{bmatrix}_{\text{mix}} = \begin{bmatrix} k_v k_f l_c \\ k_v m_c \\ n_c \end{bmatrix}_{\text{mix}} \] (48)

In accordance with \( T_{t,m,\text{ix}} \) and \( M_{t,m,\text{ix}} \), the actuator control amount is solved in series. Since the UAV is in the transitional flight and the left and right vector nozzles are equivalent, the dynamics model can be expressed as:

\[
\begin{align*}
T_x &= T_L \cos \delta_L + T_R \cos \delta_L \\
T_z &= T_L \sin \delta_L + T_R \sin \delta_L - T_F \cos \delta_F \\
l_x &= -T_L \sin \delta_L d_{yL} + T_R \sin \delta_L d_{yR} \\
m_t &= (T_L \sin \delta_L + T_R \sin \delta_L) d_{xL} + T_F \cos \delta_F d_{sf} \\
n_t &= (T_L \cos \delta_L - T_R \cos \delta_L) d_{xL} + T_F \sin \delta_F d_{sf}
\end{align*}
\] (49)

3.4 Simulation

In order to verify the control method in this paper, the designed target flight path is shown in equation 50.

\[
v_x = 0.5t \quad 0 \leq t \leq 60 \\
v_y = \begin{cases} 1.5 \sin(2\pi t / 40 - 0.5\pi) & 0 \leq t < 20 \\ 3 \cos(2\pi(t-20)/80) & 20 \leq t < 60 \end{cases} \\
v_z = \begin{cases} \sin(2\pi t / 40 + 0.5\pi) - 1 & 0 \leq t < 40 \\ 0 & 40 \leq t \leq 60 \end{cases}
\] (50)

In the target flight path, uniform acceleration motion is performed at 0.5 m/s along the \( O_x \) axis direction; along the \( O_y \) axis direction, the UAV performs lateral motion with sinusoidal change in speed, and the maximum lateral velocity is 2 m/s; along the direction of \( O_z \), the UAV is climbing at the speed of 0s ~ 40s with sinusoidal variation. The maximum climbing speed is 2 m/s, and the flying height of UAV remains unchanged after 40s. In order to facilitate the analysis, in this example, \( v_x \) is used as the flight state to judge the speed, thereby obtaining the fusion coefficient \( k_{mix} \) and the velocity distribution coefficient \( k_v \). Therefore, in this example, 0s~20s is the vertical take-off state, 20s~40s is the transient flight state, and 40s~60s is the cruising state. This example fully demonstrates the continuous switching process of flight mode in the 6 degrees of freedom motion of UAV. The simulation step size was 0.01s, and the result is shown in the figures.
It can be obtained from the figure that the UAV has a good tracking effect on the target flight path in 0s~20s. Since $v_x < 10$, the UAV uses a vertical take-off and landing controller. In the initial stage, the UAV flies with its head down and its pitch angle is negative. Acceleration is performed using the component of $T_z$, the rolling torque is generated by the thrust of the left and right engines, the lateral maneuver is performed using the component force of the $T_z$ in the $O_y g$ axis direction and the crucible is always maintained at around zero.

In 20s to 40s, the UAV uses the transient flight controller to generate new control commands by merging the signals of the two controllers. The initial state is mainly controlled by the component of the thrust. As the speed increases, the proportion of aerodynamics and pneumatic control torque increases.
gradually, the cruise controller is in dominant position, and $\delta_l$ and $\delta_r$ are gradually increased to 0 degrees.

In the 40s to 60s, the UAV fully flew in the cruising mode, and the change trend of aerodynamic rudder angle and the change trend of control torque are the same, which indicates that the torque is completely generated by the aerodynamic rudder surface.

It can be concluded based on the above simulation results that both the vertical take-off and landing controllers and the cruise controller can better track the target flight path of the corresponding state. In addition, the transient flight controller can reasonably combine the output commands of the two controllers in accordance with the current speed and thrust information so that the UAV can realize the transient flight.

4. Conclusion

The multi-mode fusion transition flight control method for three flight modes respectively of fixed-wing vertical take-off and landing drone is proposed in this paper designs controllers. With this method, it can generate a transitional flight control command by integrating the vertical takeoff and landing control command with the cruise control command. We can draw the following conclusions:

1. The control method proposed in this chapter can successfully track the 6-degree-of-freedom track of small maneuver, and it has the advantages of small calculation amount and fast solution rate.

2. When the drone adopts multi-mode fusion control, it can fly smoothly in each mode and has certain mobility during the transition process. But there are still some shakes when there is a large angle maneuver.

References

[1] Zhang Bo, Zhu Xiaoping, Zhou Zhou, et al. Turbulence Mitigation of UAV Based on Longitudinal Direct Force Control [J]. Journal of Northwestern Polytechnical University, 2014 (5): 675-681.

[2] Faleiro L F, Lambregts A A. Analysis and Tuning of a 'Total Energy Control System' Control Law using Eigenstructure Assignment[J]. Aerospace Science and Technology, 1999,3(1999):127-140.

[3] Zhang Qingzhen. Application of QFT/TECS in automatic landing control of aircraft [D]. Northwestern Polytechnical University, 2014. (in Chinese).

[4] Tang Shengjing, Li Mengting, Liu Zhenchang, Guo Jie. Parallel wing vertical takeoff and landing drone transition maneuvering flight control[J]. Systems Engineering and Electronics, 2019, 41(06): 1342-1350. (in Chinese).

[5] Liu Zhenchang, Tang Shengjing, Li Mengting, Wang Xiao, Guo Jie. Study on Optimal Control Distribution of Transition Maneuvering of Fixed-Wing Vertical Landing and Unmanned Aerial Vehicle[J]. Acta Armamentarii, 2019,40(02):314-325. (in Chinese).

[6] Fang Zhenping, Chen Wanchun, Zhang Shuguang. Flight Dynamics of Aerospace Vehicles [M]. Beijing: Beijing University of Aeronautics and Astronautics Press, 2012

[7] Zhen Haichao. research on autonomous flight control and obstacle avoidance system of four-axis aircraft [D]. University of Electronic Science and Technology of China, 2016. (in Chinese).

[8] Stone R H. The T-wing tail-sitter research UAV[C]. 2002 Biennial International Powered Lift Conference and Exhibit. Williamsburg: AIAA 2002-5970