Study on wind resistant performance of quad-rotor UAV

Xin Zheng[a]

Electricity Science Research Institute,Yunnan Power Grid Co.,Ltd, 650217 Kunming, Yunnan Province,China

Abstract. Quad-rotor UAV wind resistance problem persists in its flight, the wind disturbance in UAV attitude control stability of the effect. In this paper, quad-rotor UAV wind modeling and simulation performance issues, the first of building four-rotor UAV model; and then were in no wind, the posture angle control under sustained winds and gusts situation, observe and analyze the simulation curve; Finally, analyse the performance of quad-rotor UAV in different situations.

Keywords: UAV; wind performance; modeling and simulation.

1 Introduction

Quad-rotor UAV is a six degrees of freedom can be VTOL unmanned aircraft able to do hover and low speed flight, and indoor VTOL flying fixed-wing aircraft and other action can not be done with traditional helicopter compared to, but also has a simple structure and control, good stability, gyroscopic effect weaker advantage. Currently, the four-rotor UAV its many advantages and broad application prospects in the absence of investigation, traffic management, forest fire prevention, patrol and other areas of the city, and slowly become a hot topic internationally. But quad-rotor UAV dynamics model complexity, the model parameters and the environmental impact of uncertainty on the robustness of the controller and adaptive put forward higher requirements, especially in Yunnan, this large gust area, UAV flight stability is greatly reduced, which requires the UAV suffered gust disturbances UAV research and analysis to improve adaptability and robustness.

At present, for UAV control algorithm has a National Defense University Backstepping[1], ADRC[2], Nanjing University of Aeronautics and Astronautics synovial control [3] and so on. For Wind resistance of Nanchang Aeronautical University, Jiguang Li[4] The method of optimizing the use of direct force control feature robust structure configured to achieve the desired effect of wind landing; Yao Lei Graduate School of Chinese Academy of Sciences[5] analysis of the multi-rotor system design process has a low Reynolds number aerodynamic interaction between the rotor and the environment to control the impact model rotor system, combined with the aerodynamic characteristics of the double rotor unit completed the optimization Hex-rotor UAV rotor system and the formation of aerodynamic parameters of the final aerodynamic layout program design. In this paper, quad-rotor UAV fuselage dynamics modeling, based on the establishment of four-rotor aircraft model kinetic equation, according to the flight control system control algorithm for establishing control model; wind modeling, taking into account the duration and wind gusts of aircraft the impact will be very different, the different types of wind modeling and simulation, and simulation of wind speed in different

[a] Corresponding author : 283925003@qq.com

situations simulation testing; output by aircraft attitude simulation software, three-dimensional spatial location data and waveform, and then compare different environments the effect of the situation and control the plane disturbed and analyzed.

2 Quad-rotor UAV dynamics model

Before performing quad-rotor UAV model you will need to make certain conditions assumptions:

(1) the overall structure of quad-rotor UAV seen as a rigid body;

(2) corresponding to each section of the four-rotor UAV completely symmetrical;

(3) quad-rotor UAV's centroid located in the geometric center;

(4) during high speed rotation of the propeller is fixed non-deformable.

The flight control simulation test model is divided into three parts, namely, the plane body, attitude controller and position controller, and then use mathematical tools for mathematical modeling of the flight control. The aircraft can be simulated by the model tested for wind resistance can be tested for mathematical modeling of the wind, which was superimposed as a disturbance input, this method can not only affect the test wind on posture, also can be tested in sentinel hover wind on aircraft position, but also by changing the parameters of the plane body such as aircraft weight, the machine arm length and other parameters, and then test the wind resistance performance of different parameters of the aircraft, its ground coordinate system and the body coordinate system as Fig.1:

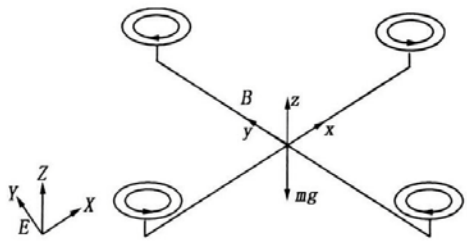

Figure 1. Earth Coordinate System and Body Coordinate System

2.1 Mathematical model

Simulation Model Selection and environment test in the same four-rotor model, a simulation model based on four-rotor is its mathematical model, so we must first four-rotor aircraft modeling and parameter identification, aircraft equations of motion created in the body coordinate system as follows:

$$\begin{bmatrix} \dot{x} \\ \dot{y} \\ \dot{z} \end{bmatrix} = \boldsymbol{R} \cdot \begin{bmatrix} u \\ v \\ w \end{bmatrix} \tag{1}$$

$$\begin{bmatrix} \dot{u} \\ \dot{v} \\ \dot{w} \end{bmatrix} = \begin{bmatrix} F_x \\ F_y \\ F_z \end{bmatrix} / m + g \begin{bmatrix} -\sin\theta \\ \sin\phi\cos\theta \\ \cos\phi\cos\theta \end{bmatrix} - \begin{bmatrix} qw - rv \\ ru - pw \\ pv - qu \end{bmatrix} \tag{1}$$

$$\begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} = \begin{bmatrix} 1 & \sin\phi\tan\theta & \cos\phi\tan\theta \\ 0 & \cos\phi & -\sin\phi \\ 0 & \sin\phi\sec\theta & \cos\phi\sec\theta \end{bmatrix} \cdot \begin{bmatrix} p \\ q \\ r \end{bmatrix} \tag{3}$$

$$\begin{bmatrix} \dot{p} \\ \dot{q} \\ \dot{r} \end{bmatrix} = \begin{bmatrix} I_x & 0 & 0 \\ 0 & I_y & 0 \\ 0 & 0 & I_z \end{bmatrix}^{-1} \cdot \begin{bmatrix} Q_x + (I_y - I_z)qr \\ Q_y + (I_z - I_x)rp \\ Q_z + (I_x - I_y)pq \end{bmatrix} \tag{4}$$

$$R = \begin{bmatrix} \cos\psi\cos\theta & \cos\psi\sin\theta\sin\phi - \sin\psi\cos\phi & \cos\psi\sin\theta\cos\phi + \sin\psi\sin\phi \\ \sin\psi\cos\theta & \sin\psi\sin\theta\sin\phi + \cos\psi\cos\phi & \sin\psi\sin\theta\cos\phi - \cos\psi\sin\phi \\ -\sin\theta & \cos\theta\sin\phi & \cos\theta\cos\phi \end{bmatrix} \tag{5}$$

Wherein, u, v, w of the linear velocity of the body coordinate system; θ, ψ, ϕ for the roll angle of the body coordinate system, pitch angle, heading angle; p, q, r is the attitude angular rate of the body coordinate system; x, y, z coordinates for the location of the aircraft navigation coordinates; m is vehicle mass;G is for local gravity; F_x, F_y, F_z in the aerodynamic body coordinate components; Q_x, Q_y, Q_z are moment together outside in the body coordinate system component; I_x, I_y, I_z are the axial moment of inertia;In the body coordinate system component R for the body to coordinate transformation matrix navigation coordinate system.

Four-rotor aircraft mechanical equations of the following formula:

$$\begin{cases} F_x = -\rho/2 \cdot S_{fx} \cdot u \cdot |u| \\ F_y = -\rho/2 \cdot S_{fy} \cdot v \cdot |v| \\ F_z = -k\_T \cdot \sum_{i=1}^{i} \Omega_i^2 - \rho/2 \cdot S_{fz} \cdot w \cdot |w| \end{cases} \tag{6}$$

$$\begin{cases} Q_x = J_r(-\Omega_1 + \Omega_2 - \Omega_3 + \Omega_4)q + \tau_\phi \\ Q_y = -J_r(-\Omega_1 + \Omega_2 - \Omega_3 + \Omega_4)p + \tau_\theta \\ Q_z = \tau_\psi \end{cases} \tag{7}$$

$$\begin{cases} \tau_\phi = \frac{\sqrt{2}}{2} lk\_T(\Omega_2^2 + \Omega_3^2 - \Omega_1^2 - \Omega_4^2) \\ \tau_\theta = \frac{\sqrt{2}}{2} lk\_T(\Omega_1^2 + \Omega_2^2 - \Omega_3^2 - \Omega_4^2) \\ \tau_\psi = k\_M(\Omega_1^2 - \Omega_2^2 + \Omega_3^2 - \Omega_4^2) \end{cases} \tag{8}$$

Wherein, τ_ϕ, τ_θ, τ_ψ respectively roll, pitch, heading in the direction of the fuselage moment; k_T pull factor for the motor; k_M trans torque coefficient; J_r is the rotor relative to the moment of inertia of the motor; ρ is the air density; S_{fx}, S_{fy}, S_{fz} is the four-rotor aircraft horizontal, vertical, vertical three directions of the body area; Ω_i is the rotor speed; l is the motor to the aircraft center of gravity distance.

Four-rotor power system is composed of four motors, small DC motor inertia is approximately as follows:

$$a\dot{\Omega} + \Omega = bU \tag{9}$$

Wherein, a, b is a coefficient, U is the PWM signal for input, Ω is the motor speed.

Through the establishment of the above four-rotor mathematical model to build Matlab/Simulink simulation model, the model for the four inputs $\lfloor \delta_{col} \quad \delta_{lon} \quad \delta_{lat} \quad \delta_{ped} \rfloor$ through pull distribution and limiter module outputs four motor desired voltage to control the rotational speed of the four motor, four four-rotor speed motors through their jointly constructed four-rotor power system to produce four torque $(T \quad \tau_{\phi} \quad \tau_{\theta} \quad \tau_{\psi})$ to achieve four-rotor flight movement.

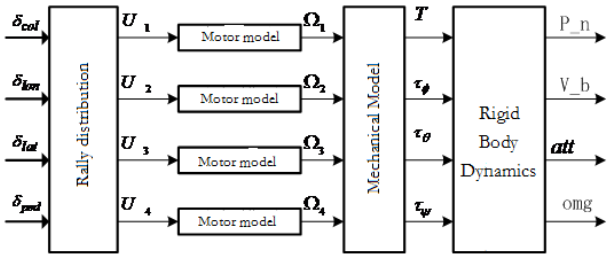

Figure 2. Four-rotor aircraft system composition structure

In figure 2 above represents to coordinate the position vector $P\_n = (x \quad y \quad z)^T$; $V\_b = (u \quad v \quad w)^T$ coordinate system inside the body represents the velocity vector; $att = (\phi \quad \theta \quad \psi)^T$ is the attitude angle vector unit rad; represented within the body coordinate system angular velocity; δ_{col} is the input of total distance is normalized steering and the input range is (0, 1); δ_{lon} standardized input pitch servo inputs, the range is (-1,1); δ_{lat} standardized input roll servo inputs, the range (-1, 1); δ_{ped} standardized input yaw steering input range (-1,1).

The above parameters used in the model parameters based on the actual aircraft set specific settings can be seen from Table 1:

Table 1. Four-rotor Aircraft Model Parameter Values

Parameter	Value	Unit
g	9.7643	m/s^2
m	0.875	K_g
l	0.225	m
k_T	$1.005*10^{-5}$	$N.s^2/rad^2$
k_M	$2.3992*10^{-7}$	$N.m.s^2/rad^2$
I_x	$9.5065*10^{-3}$	$K_g.m^2$
I_y	$1.000*10^{-2}$	$K_g.m^2$
I_z	$1.658*10^{-2}$	$K_g.m^2$
J_r	$6.0000*10^{-5}$	$K_g.m^2$
a	0.1950	s
b	1783	-
ρ	1.294	$K_g.m^3$
S_{fx}	0.00835	m^2
S_{fy}	0.0131	m^2
S_{fz}	0.017	m^2

3 Simulation model building

Using Matlab/Simulink tools to build a quad-rotor aircraft control loop structure, the initial position of the plane [0 0 0], give the position of the aircraft [8 8 -8], the response by observing the posture,

position angle response curve and curve analysis obtained aircraft wind performance. The overall framework of the simulation model shown in Fig.3, includes a control module (flight control), wind speed module, quad-rotor module (plane body), display memory module four parts. Wherein the control module is a flight control system simulation, the same control algorithms control algorithms and the actual use of the flight control system which is used to ensure that the simulation testing and environmental test results consistency.

In the actual aircraft flight, wind on the aircraft attitude and position can be seen as external interference, therefore simulation is also considered wind interference, its output port connected to the input port of the aircraft, and translate them into the body wind disturbance model coordinate system. Windless environment model, a model port *wnd* vacant, no disturbance input value; in windy environment model, wind disturbance model as a random noise source is connected to the input port and *wnd* as a disturbance input value. Taking into account the impact of sustained winds and gusts of aircraft is quite different, so the need for sustained winds and gusts of wind disturbance model were established.

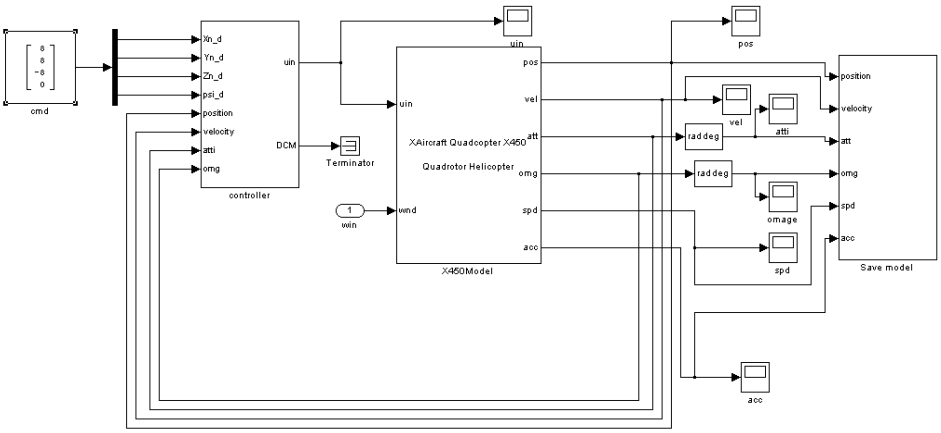

Figure 3. Overall block diagram in the case of no wind

Winds from the model in this paper were used *Wind Shear Model* and *Discrete Wind Gust Model* to simulate sustained wind gusts and the environment. Wherein, wind shear models belong sustained wind, the aircraft during operation are always there, enter the height and direction of the aircraft cosine matrix, the output for the next body to coordinate wind average speed; Discrete Wind Gust Model belongs gust, you can set the wind start time wind size and length, enter the aircraft airspeed, wind speed output produced by the body coordinate system. Both models will be following a brief introduction Winds:

Wind Shear Model is applied to the aircraft model wind shear model, which simulate wind speed to mean forward first obtained Winds Earth coordinates, followed by input DCM (direction cosine matrix) into the lower body of the coordinate system average wind speed. The application of a mathematical formula based on Military Specification MIL-F-8785C is. Wind shear model mathematical formula is as follows:

$$u_w = W_{20} \frac{\ln(\frac{h}{z_0})}{\ln(\frac{20}{z_0})}, 0.8m < h < 300m \tag{10}$$

In the formula, it represents the average wind speed at 6 meters height represents the measured wind speed, h represents height, z0 is a constant. wind model of the internal structure as shown in Fig.4:

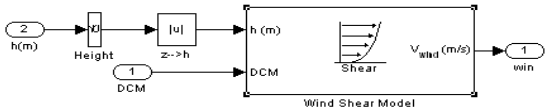

Figure 4. Wind shear wind speed model internal structure

Discrete Wind Gust Model is a typical model of "1-cosine" shape, the model can be time, length and size to define the wind began. Gusts can be independently applied to any axis or 3-axis. Specific mathematical formula is as follows:

$$V_{wind} = \begin{cases} 0 & x < 0 \\ \frac{V_m}{2}(1-\cos(\frac{\pi x}{d_m})) & 0 \le x \le d_m \\ V_m & x > d_m \end{cases} \tag{11}$$

Where, V_m represents the magnitude of the wind, D_m represents the length of the wind, x represents the distance traveled wind, V_{wind} represents the speed of the wind in the body coordinate system generated. When the wind advancing distance is less than zero, the wind speed is zero; when the wind advancing distance is greater than zero and less than the length of the wind, the wind speed to meet a certain relationship between the amplitude of "1-cosin" shape formula, wind speed and set; when when the wind advancing distance greater than the length of the wind speed in the set amplitude. Since the input discrete wind gust model for airspeed, but directly from the four-rotor model has been the speed of the aircraft triaxial velocity, it needs to be synthesized as a speed three-axis aircraft airspeed then enter the specific synthesis method is to first find triaxial velocity squared and then square root to get the aircraft airspeed,the internal structure as shown in Fig.5:

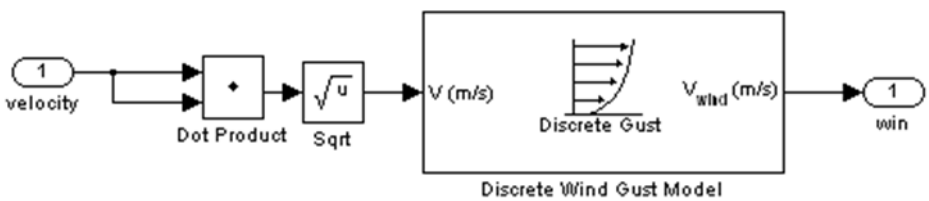

Figure 5. Winds from the added discrete wind Gust model after wind model

3.1 Windless environment simulation test

Due to the influence of wind on the four-rotor aircraft is mainly reflected in the attitude, speed and direction of the aircraft is not much change. So first change the attitude of the aircraft simulation to study dynamic response and steady state characteristics, performance verification control four-rotor model.

Added step signal in the attitude of the simulation model to simulate the throttle driven process, to observe changes in the attitude of the aircraft. First, given the attitude angle [roll pitch] = [0 0], when running 5s given step signal [roll pitch] = [0.13 0.1], after running 15s given step signal again [roll pitch] = [0.13 0.1], push the throttle to simulate the actual process of observation spacecraft attitude response, pitch angle and roll angle of the response curve shown in Fig.6:

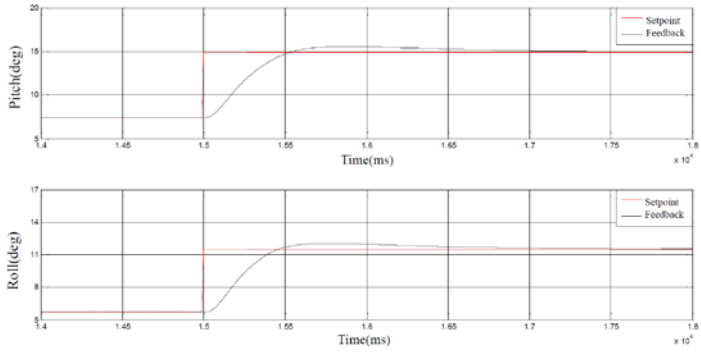

Figure 6. Part posture angular response curve

As can be seen from the simulation curve: 15s added after step signal, quad-rotor aircraft pitch angle overshoot 0.38%, transient response time of 0.9s, roll angle overshoot 0.25%, transient response time of 0.7s , both smoothly reaches the desired position, the corresponding change in the attitude angle within 3°, the change is not severe, a good attitude tracking performance of the validated model four-rotor flight control system.

By curve shows that the aircraft back to the positive process is very short, although the noise had an impact on the attitude, but the system can quickly stabilize posture traces largely unaffected in the 17s moment attitude into steady-state process, the entire process of the posture response once overshoot and lasted less than 3s, to meet rapid response requirements.

3.2 Windy environment simulation test

In order to verify the flight control system anti-jamming capability, adding wind speed model as input interference, a second, a third set of experiments, observing the control system model attitude angle curve. Through simulation and experiment results prove low wind speed, the aircraft attitude and no wind situation is basically the same, in order to improve test efficiency, highlighting the impact of wind speed aircraft flight performance, you can choose more than 5m/s wind speed for testing.

3.2.1 Add the wind shear model

The second set of experiments with wind shear model to simulate sustained wind, divided into two types: constant position, wind speed, wind speed and constant changes in the situation, circumstances change in position.

First, the default four-rotor aircraft in a hovering state, given a four-rotor aircraft position [X_n, Y_n, Z_n] = [8,8, -8], simulated wind speed was 5m/s, 6m/s, 7m/s , 8m/s, 9m/s, vehicle operating status 10m/s case, in which the direction north-uniformly sets 10°. Since the direction of the wind is not north, but showed a 10° angle, so get the wind speed is not exactly equal to the set value Fig.7 shows the wind speed curve for this model.

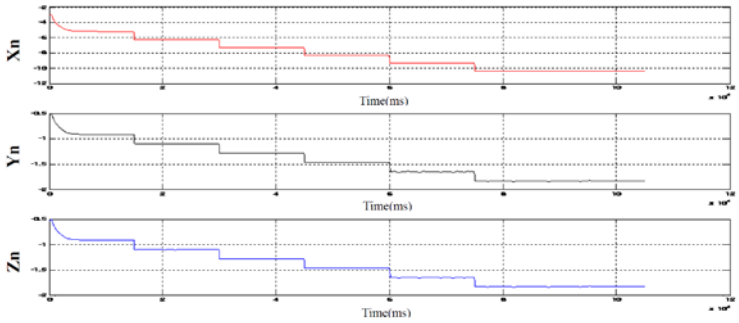

Figure 7. Winds response curve

Shown curve, the wind from the north direction is gradually increased from 5m/s, up to 10m/s. Since the direction of the wind is not north, but east of north 10°, thus increasing direction from the east by the velocity of 0.5m/s, up to 1.8m / s, from ground wind speed from 0 to 0.75m/s.

The default aircraft in hover state, when the attitude angle [φ, θ, ψ] = [0,0,0], added wind speed model (wind shear model), the spacecraft attitude angle change as shown in Fig.8.

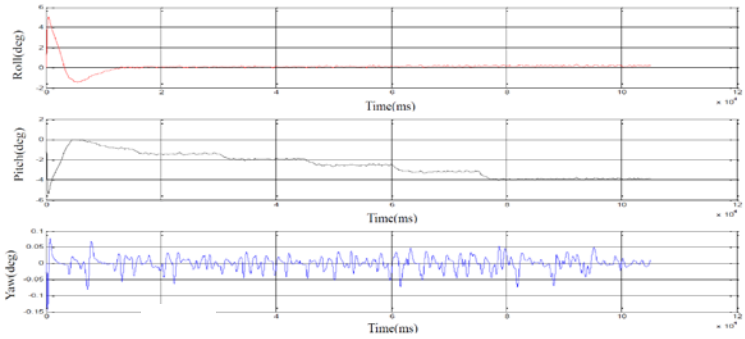

Figure 8. Winds from the constant position changes the attitude angle response curve

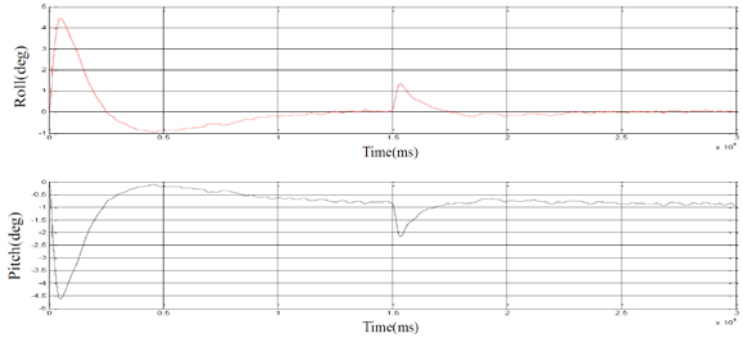

Figure 9. Winds from the constant change of position posture angular response curve

It can be seen from Fig.8: attitude angle response curve stabilized in the 15s, in the 11s after the roll angle reaches a steady state, eventually stabilized at 0°. The transient response time of the pitch angle is about 11s, after reaching steady state, but as the wind speed in the form of periodic step 15s is gradually increased, the pitch angle also step response cycle 15s is gradually increased, transient response time of 3s after holding 12s, this is because in the windy conditions, the aircraft will be inclined to resist wind interference, to achieve its own stability in the 105s time, pitch angle offset 3.9°. The yaw angle is almost free from the influence of wind, has been around 0 ° fluctuations, the error remains within 5%.

Fig.9 shows that the roll angle and pitch angle transient response time is about large 12s, overshoot in the 15s moment, the position added step signal changes from the original 5m to 6m, and therefore the pitch angle There was a small roll angle pulse overshoot for both is 153%, transient response times are 7s, after reaching a steady state. Steady-state error roll angle is maintained at less than 5%, the pitch angle of about -1° deviation from the original posture, steady-state error in the 30s always kept at 15%, with the change of time, the pitch angle of the steady-state error getting smaller and smaller. Due to the influence of wind on the spacecraft attitude is mainly reflected in the impact on the pitch, so that the pitch angle is maintained at -1° to resist wind interference.

Through the above analysis, when given sustained wind, constant wind speed, given the change of position, the control system and has taken certain parameters of anti-interference ability, good stability, meet the control requirements.

3.2.2 Add the discrete wind gust model

The third set of experiments using the discrete gust model (discrete wind gust model) as an interference input, to simulate wind gusts were simulated wind speed of 5m/s, 7m/s, aircraft operating status 10m/s case, which began to gust time unified to 5s, the length of the gust of unified [8 8 8]. After several simulation experiments, select gust length[8 8 8], in the 40s after the basic simulation to achieve the desired wind speed. Specifically related parameters in Table 2.

Table 2. Experimental parameters

Amplitude (m/s)	Gust length (m)	Gust start time	Simulation time	Simulation step
[5 6 3]	[8 8 8]	5s	40s	default
[7 7 3]	[8 8 8]	5s	40s	default
[10 10 3]	[8 8 8]	5s	40s	default

Referring to Table 1.8 of the relevant parameters set by the amplitude [X_W, Y_W, Z_W] = [5 6 3], wind speed curve shown in Fig.10.

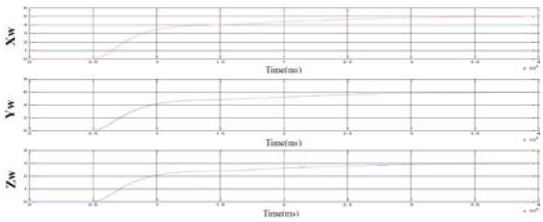

Figure 10. 5m/s wind speed response curve

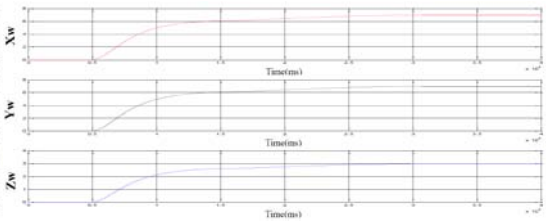

Figure 11. 7 m/s wind speed response curve

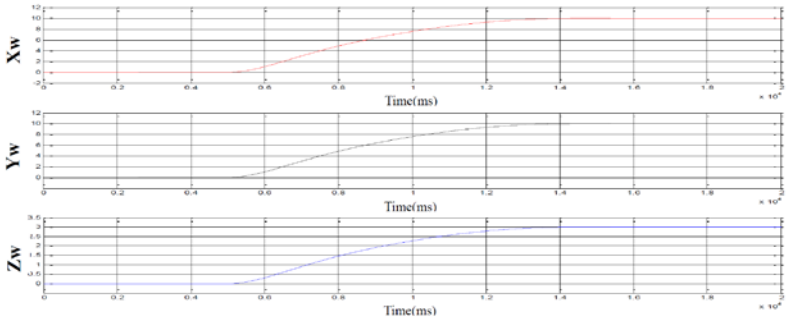

Figure 12. 10m/s wind speed response curve

Fig.10 can be seen, the first began 5s gusts, wind speed then gradually increases in accordance with the mathematical formulas described, reaches the set value in the 36S, considering this situation gust simulation consistent with the actual wind gusts, as Rafale aircraft model was added to the input. Fig.11,Fig.12 were given in response to changes in the curve 40s attitude angle.

The initial time, the aircraft in hover state, when the attitude angle [φ, θ, ψ] = [0,0,0], after the aircraft took off to reach a given position, the process in 5s added wind speed model attitude angle response curve shown in Fig.13.

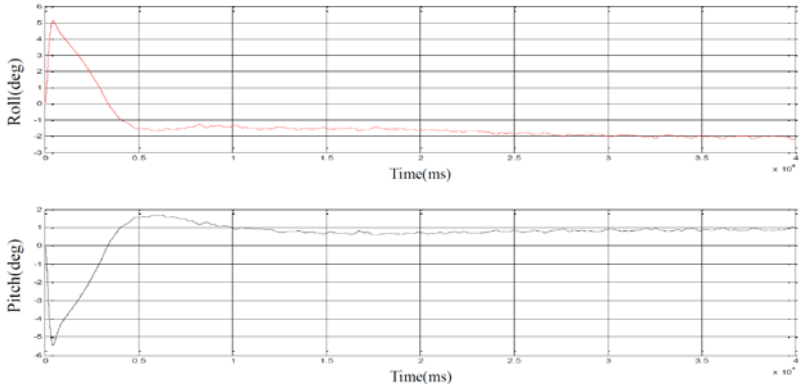

Figure 13. 5m/s posture angular response curve

As can be seen from Fig.13: roll angle transient response time of 5s, after reaching steady state, finally stabilizing at around -2°; pitch angle of about 10s time to reach steady-state, eventually stabilized at about 1°. Since wind speed throughout the simulation process has been changed, so the roll angle and pitch angle also requires dynamic response has been changing, with increasing wind speed, the angle changes roll angle increases. This is because in the case of adding gust, three axes of the aircraft are affected by gusts of wind aircraft is inclined to resist interference, to achieve its own stability.

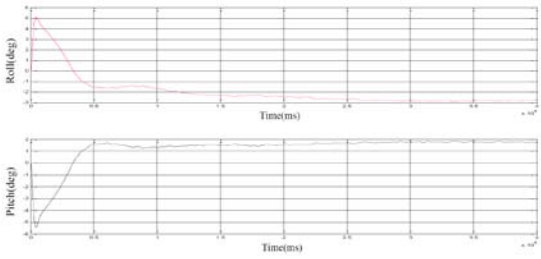

Figure 14. 7m/s posture angular response curve

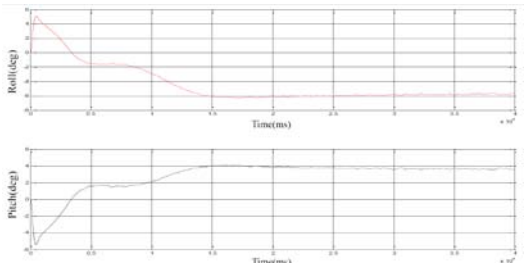

Figure 15. 10m/s posture angular response curve

As can be seen from the Figure: the roll angle in the 30s after reaching steady state, eventually stabilizing at about -2.86°; pitch angle of about 30s time to reach steady-state, eventually stabilize at around 1.74°. Since wind speed is constantly changing, so the roll angle and pitch angle also follow its dynamic response, with the increase of wind speed, change the angle of roll angle and pitch angle increases. Fig. 12 comparison can be found in Fig.13 in the roll angle of the transient response time is shorter, eventually stabilized at around -2°, and Fig.13 in the roll angle response time longer, and more angle Great;in FIG pitch angle of about 10s time to reach steady-state, eventually stabilize at 1°, and Fig.14 in the pitch angle of the response time is longer and eventually stabilize at around 1.74°. Causes of the difference we can see, the wind speed from the original [X_W, Y_W, Z_W] = [6 5 3] into the present [X_W, Y_W, Z_W] = [7 7 3], an aircraft needs to happen more inclined to resist wind interference, so roll angle and pitch angle than the original increase.

As can be seen from Figure: roll angle for the first time to reach steady state in the 5s, and kept at -2°, due to the added time 5s wind interference, the transient response of the roll angle occurs again, the response time is 10s, then 15s after reaching steady state, eventually stabilized at around -5.8°; pitch angle for the first time in 5s reaches a steady state, kept at 5°, after due 5s added wind disturbance, and therefore the pitch angle to follow the response again, the response time of 10s, in 15s time to reach steady state, eventually stabilize at around 1.74°. Comparison Fig. 13 and Fig.14, we can see that the greater the wind speed, roll angle and pitch angle of the aircraft greater, because the aircraft needs to happen more inclined to resist wind interference. The figure shows that the control system can withstand wind speed disturbance 10m/s, the system can quickly stabilized, good stability.

This simulation test model was tested four rotor position and no wind windy case, attitude change. No wind conditions were changed attitude and position, in response to the observing position and attitude angle; windy conditions were added sustained wind (wind shear model) and wind gusts (discrete wind gust model), and make wind speed varies from 5m/s to 10m/s, observe the quad-rotor attitude adjustment capability. Simulation results show that, under these two conditions, the quad-rotor can reach a desired position within a certain error range, good stability, pitch angle and roll angle can be varied according to changes in wind speed. Simulation results show that the control system and has taken certain parameters of wind capacity, high-precision cascade PID control system to meet the needs of wind resistance.

4 Conclusion

On aircraft attitude simulation experiments to test the wind:

1)No wind environment roll, pitch angle overshoot less than 0.5%, the response time of less than 1s, steady-state process less than 3s, test results and test results are substantially similar environmental testing, environmental simulation and experimental method described test results are consistent ;

2)Sustained wind environment, wind will reduce the response speed of the aircraft attitude, when the wind speed of 5m/s steady-state process becomes 13s, attitude angle overshoot becomes also increased by 50%, compared with no wind environment test results the increase, mainly because of the wind environment when the aircraft attitude changes require more power, flight control system to adjust the appropriate time longer; when sustained wind presence, position of the aircraft in the direction of the wind will have less than 2m of static error, the greater the wind speed position deviation increases, mainly due to the aircraft to offset the wind thrust must rely on positional deviation generated regulate the output quantity;

3)Environmental gust of wind on the airplane impact and sustained wind similar, but more random gust, wind speed faster, the aircraft's attitude is more significant, there appears in the attitude angle gust environment simulation test results volatility has increased, the fluctuation range of less than 2°, to meet the requirements of wind resistance.

Environmental experiments tested the effect of wind on the aircraft attitude simulation experiment tested the effect of wind on aircraft position and attitude, it can be proved by comparing that the flight control system has good wind indicators, using the occasion to meet the demand.

References

1. H. Zhu, J.Yue, Q.Zhang, “A Sigma-points Kalman Filterfor Micro Quadrotor Attitude Estimation,” *The 4th International Conferenceon Intelligent Information Technology Application*, pp. 494-497, (2010).
2. J. Wang, H. Ma, and W. Cai, “Research on Micro Quadrotor Control Base on ADRC,” *Journal of Projectiles, Rockets, Missiles and Guidance*, vol. **28**, no. 3, pp. 31-34, June (2008).
3. H. Shan, “Combined DI/QFT Fight Control for a Quad-rotor Unmanned Helicopter,” *Electionics Optics&Control*, vol. **15**, no. 12, pp. 68-71, December (2008).

4. J.Li and Y.Dong, "Counteracting Sidewind Control System by Directforce Control for UVA Landing," *Command Control & Simulation*, vol. **36**, no. 2, pp. 44-47, April (2014).
5. P.Castillo, A.Dzul, "Real-Time Stabilization and Tracking of a Four-Rotor Mini Rotorcraft," *On Control Systems Technology*, vol. **12**, no. 4, pp. 510-516, March (2004).
6. A.Tayebi, S.Mcgilvray, "Attitude Stabilization of a Four-rotor aerial robot," *IEEE Conference on Decision and Control*, vol. **12** pp. 14-17, (2004).
7. M.Elshafei, S.Akhtar,M.S.Ahmed, "Parametric models for ANN," *IEEE transactions on aerospace and electronic helicopter identification using systems*, vol. **36**, no. 4, pp. 1242-1252, (2000).
8. Besnard L., YB. Shtessel, et al."Disturbance observer. Collection Control Conference," *Hilton He ad,Control of a quad rotor vehicle using sliding mode of Technical Papers AIAA Guidance, Navi gation, and USA*, pp.58-77,January (2007).
9. Hoffmann GM., H. Huang, et al. "Quad rotor helicopter flight dynamics and control:Theory and experiment," *Collection of Technical Papers AIAA Guidance, Navigation,and Control Conference*," pp. 1670-1689, (2007)
10. Bouabdallah S., Noth, et al. "PID vs. LQ control techniques applied to an indoor micro quad rotor," *IEEE/RSJ International Conference on Intelligent Robots and Systems (IRO S)*, pp. 2541-2456, (2004).
11. Bouabdallah S., P. Murrieri, et al. "Towards autonomous indoor micro VTOL. Autonomous Robots, vol.**18**, no.2, pp.171-183, (2004).
12. Tayebi A., S. McGilvray. "Attitude stabilization of a VTOL quad rotor aircraft," *IEEE Transaction on Control Systems Technology*, vol. **14**, no. 3, pp. 562-571,(2006).
13. Calise A J, Rysdyk R T. "Nonlinear Adaptive Fight Control Using Neural Networks," *IEEE Control Systems Magazine*, vol. **18**, no. 6, pp. 14-25. (1998).
14. B. Song, C. Fan, Y. Liu, Mills J K, X. Cai. "Nonlinear System and Control of the Model-scale Helicopter," *In Proceedings of the IEEE International Conference on Robotics and Biomimetics*, Guilin, Guangxi, China, pp. 1081-1086, (2009).
15. Bouabdallah S., Noth, et al. "PID vs. LQ control techniques applied to an indoor micro quad rotor," *IEEE/RSJ International Conference on Intelligent Robots and Systems*(*IROS*),Sendai, Japan, pp. 2451-2456, (2004).