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control surface failure, fault diagnosis, aircraft modeling, HILS

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Quick Diagnosis of UCAV Control Surface Failure and Its Hardware-in-the-loop Simulation

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Abstract: There are many problems in current detection methods such as conditional limitations and complexity of calculation which make them not easy to be put into practical application. *The relationship between the servo motor information and the failure of the control surface was analyzed, and a method of detecting the failure rate of the control surface by using the servo current information in combination with the aerodynamic model was proposed.* The result of its hardware-in-the-loop simulation shows that the detection system can quickly and accurately output the failure rate within 1 second which is a conservative estimate. That makes the reconfiguration system have a greater chance to regain the effective control of UCAV and increase the survivability of UCAVs that are damaged in the fights.

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无人机操纵面失效快速诊断及其半实物仿真

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摘要：目前的无人机操纵面失效检测技术很多存在着条件限制多、计算复杂不易投入实际应用等问题。*通过分析舵机信息与操纵面失效的联系，提出了结合气动模型通过舵机电流信息检测操纵面失效率的方法*。半实物仿真结果表明，检测系统能在控制姿态角稳态输入和大幅度变化的输入下，在 1 s 内迅速给出准确稳定的操纵面失效率检测输出。为重构系统能够有更大几率使飞控系统重新获得对飞机的控制打下基础，增加无人机的战后损伤状况下的生存能力。

关键词：操纵面失效；故障诊断；飞行器建模；半实物仿真
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Introduction

The flight action of Unmanned Combat Aerial Vehicles (UCAV) is more flexible than traditional fighters for there’s no need to consider the overload that human body withstand, which brings them huge advantages in air combat. With the rapid development of high-tech military technology, military departments in many countries put the development of UAV in the first place. Participating in even leading the competition for air supremacy is a key task for operational research of UCAV, which would probably replace the role of manned aircraft and become the pillar force in future warfare. The production cost of UCAVs is low and they are

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expendable, however, conventional war between the super powers is destined to be a war of attrition. Retaining a large number of UCAVs that are capable to fight continually will be the decisive factor to seize the battlefield initiative. In the course of the war, the UCAV control surface damage will lead directly to their crash. Therefore, it is of great practical significance to study on control surface failure diagnosis technology which provided vital information for the reconfiguration system so that these damaged UCAVs can return to base safely and quickly back into the battlefield after repairing, which would have loss in actions.

Fault detection of control surface is a very complex problem, and the common methods can be divided into the model based and the knowledge based detection algorithm. The model based ones use Calman filter or Berg observer[1-2], whose dimensions of residual vector are mainly restricted by the dimension of the system. Though there are many numerical methods to improve the accuracy of these methods, the tedious numerical calculation has some limitations in the application of fault detection for UCAVs. The knowledge based ones, including fault diagnosis method based on artificial neural network, pattern recognition or expert system[3-5], are all based on complex numerical calculation, which aggravate the burden of the flight control computer with poor timeliness, and low reliability. On the other hand, it's not difficult to discover the close relationship between control surface failure and actuator information. Therefore, the control surface fault detection algorithm based on the local information, which is the actuator information, has a wide application prospect because of its reliability and feasibility.

Based on the physical analysis of the hinge moment and actuator, an improved algorithm for failure rate of control surface is developed building on the local model of the control surfaces and the actuator information. A nonlinear aircraft model is established using Simulink which supports a large number of simulations in order to make further research on distinguishing the abnormal changes of index out of all. Then the hardware-in-the-loop simulations combined with close loop UCAV model are implemented to verify the real-time performance, efficiency and false alarm rate of the detection system. Meanwhile, a method of classification and output validity check is designed to provide failure rate directly and steadily without extra error results while the signal is changing in the condition of serious failure(>50%), which could largely improve the efficiency of reconfiguration system and improve the battlefield survival rate of damaged UCAVs.

1 Principle

UCAV control surface failure refers to its damage like holes, partially broken or completely off, cause of accidental factors such as foreign objects impact, hit by guided or unguided ammunition, which lead to the decrease of the ability of flight control system to control the plane. Usually the rest of the control surface still maintains part of the operating capacity except that it's completely fall off, and it can be proved that the control ability that remains can be measured by its hinge moment produced by unit angle of deflection.

The actuator loop is an indispensable part of the auto-control flight system which controls the control surfaces according to the electric signal from instruction units and realizes the automatic stability and control of angular motion and trajectory of flight. The control surface drive is the motor of the actuator loop whose outputs are torque and angular velocity.

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Therefore, the actuator has a close relationship with the control surface and these two are connected by hinge moment. The state of the control surface can be obtained by detecting the actuator information, so as to detect the extent of control surface failure effectively and reliably.

1.1 Approximate Calculation of Hinge Moment

The hinge moment is defined as the moment of the aerodynamic force on the control surface to the axis of the control surface which is formed by the air flow on the control surface. The actuator controls the deflection of the control surface by mechanical transmission. In order to deflect the control surface to the desired position, the hinge moment acting on the hinge shaft must be overcome. There are many factors that affect the hinge torque, mainly in the following:

a) Flight states: speed V, height H, angle of attack α, and side slip angle β;

b) The static stability, that is, $m_z^\alpha, m_y^\beta, m_x^\beta$;

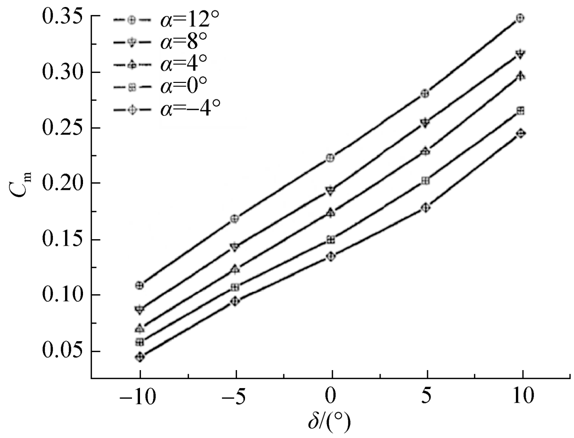
c) The maneuverability, that is, $m_x^\delta, m_y^\delta, m_z^\delta$;

d) The pressure center position of the control surface and the position of the hinge shaft;

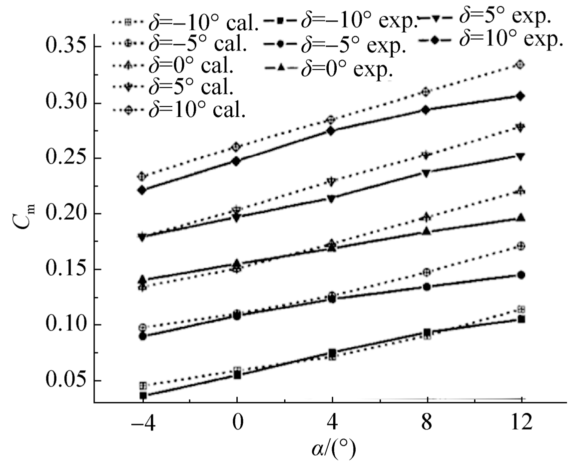
e) The dynamic characteristics of the control surface and its response to the deflection input[6];

And most of them are the values that have been determined when the aircraft was designed, such as the type of control surface and the wing geometry data. Therefore, in the actual calculation and wind tunnel test, we can draw a conclusion that under certain flight conditions, the deflection angle δ and the angle of attack α cause greater influence on the hinge moment of the control surface than other factors. In the Ref. [7], the actual wind tunnel test data and the engineering estimation results were used to test the hinge moments with multiple sets of single variables. According to the results shown in Fig. 1, the hinge moment changes caused by δ when the angle of attack α is fixed are much more violent than the changes caused by α when the δ is fixed. Thus in many factors, the rudder angle is the main decision variable of the hinge moment, which makes the hinge moment H_j can be approximately written as follow:

$$H_j = C_{hj}^{\delta} Q S_{\delta} \delta_j$$



(a) C_m changes caused by δ when α is fixed



(b) C_m changes caused by α when the δ is fixed

Fig. 1 C_m Changes caused by α and δ

1.2 Change of the hinge torque when the control surface failure occurs

From the analysis above we can describe the hinge moment as Fig. 2.

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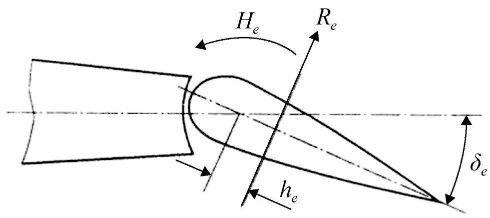

Fig. 2 Hinge moment of control surface

And the expression of the hinge torque can be written as:

$H_e = R_e \times h_e$

where H_e is the hinge moment, R_e is the aerodynamic force, which is related to Mach number, airflow angle, rudder deflection angle, control surface area. H_e is the distance between the shaft and the aerodynamic pressure center of the control surface. As a variety of aircraft control surfaces are not necessarily the same, this paper temporarily ignores the non-linear factors. We tentatively assume that H_e and the control effectiveness of control surface was proportional relationship under the same conditions.

2 Method

2.1 Detection of control surface failure

Actuators are widely used as the main components of the servo system of modern unmanned aerial vehicles. It is usually composed of motor, speed measuring device, position sensor, transmission and safety protection device, etc. Therefore, we use the electrical motor as the drive of control surface. Taking the DC torque motor as an example, the influence of the friction torque is neglected. The reverse electromotive force generated by the rotation of the torque motor during the deflection of the control surface and the change of the hinge moment when the control surface is deflected are both taken into consideration, the dynamics of the electric servo can be described as follows:

$$\Delta U = L\frac{\mathrm{d}\Delta I}{\mathrm{d}\Delta t} + \Delta IR - C_e\omega$$

$$\Delta M = C_{\mathrm{m}}\Delta I$$

$$\Delta M = J\frac{\mathrm{d}\Delta\omega}{\mathrm{d}t} + \Delta H_e$$

$$\Delta H_e = \Delta H(\alpha, \beta, \Delta\delta_j, H, V)$$

where I_a is the armature current; U is the voltage between the two ends of the actuator; R is the total resistance of the armature circuit; L is the total inductance of the armature circuit; M is the motor torque; C_{m} is the motor torque coefficient; J_{m} is the moment of inertia, ω is the angular velocity.

The hinge moment will reduce rapidly as a result of the sudden failure of the control surface due to external factors, and the torque on the control surface that the actuator needs to overcome will decrease rapidly as well, which makes the servo current lower than the normal state. Therefore, it is possible to find out the relationship between the current drop and the failure of the control surface by modeling and large amount of simulation, and obtain the failure degree of the control surface by detecting the change of the current.

In order to reduce the influence of the hinge moment acting on the actuator and meet the requirements of the flight control law, the actuator loop is used instead of the single servo in the flight control system. Fig. 3 shows the actuator loop structure with position feedback, where $C_{\mathrm{e}} = 0.487\,8$, $C_{\mathrm{m}} = 4.658\,2$, $R_a = 13\,\Omega$. $J_{\mathrm{m}} = 0.005\,9$. The parameters above are obtained by measuring the physical motor in the semi-physical simulation.

As Ref. [8] point out, when the aircraft reaches a steady state, the control surface angle position control command and the actual deflection angle remain unchanged, the actual hinge torque $H_e = M_a$ while the armature current is I_0. Assuming that the aerodynamic efficiency of the control surface becomes η when a damage occurs, the hinge moment will instantaneously change to ηM_a due to the change in the force of the control surface. But after the aircraft control system's adjustment, the steady state still become $M_a = H_e$ while the armature current change back to I_0. At this point the actual deflection angle approximately becomes $1/\eta$ of original and the hinge torque in the open loop estimator changes as the same, which leads to the armature current become ηI_0.

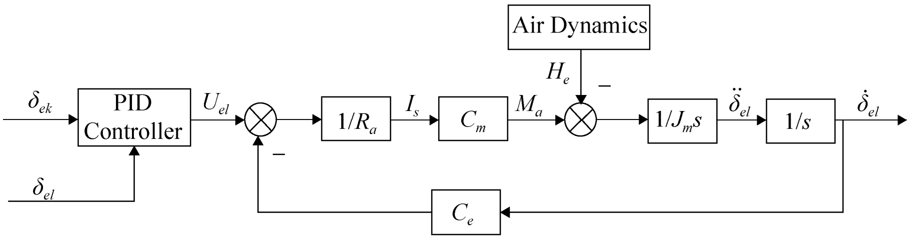

Fig. 3 Actuator loop structure with position feedback

However, during the simulation we found that, on the one hand, this method can detect the failure rate only when the damage is slight. In the test of more serious damage, the control surface cannot provide the torque required by the flight control system even if it is deflected to the limit angle; On the other hand, when the aerodynamic force on the control surface changes greatly due to normal operation, it will be misinterpreted as control surface failure, which is not caused by the inaccuracy of the actuator modeling. So the method cannot be put into practical applications of complex environments, but provide a way of thinking. So a more complete model detection algorithm is identified to solve these problems. By establishing the nonlinear aerodynamic model of the actuator part of the UCAV, the aerodynamic moments of the control surfaces under a full structural integrity are estimated by the atmospheric model with few current flight status (height, velocity, angle of attack, side slip angle) and deflection angle. Then add this moment into the ideal motor model to calculate the ideal standard armature current I_s as Fig. 3 shows, which can be written as:

$$I_s = (U_{el} + \frac{C_e H_e}{J_m s^2}) \Big/ (R_a + \frac{C_e C_m}{J_m s^2})$$

Meanwhile, by measuring the voltage U_{Rs} across the sampling resistance R_s, the actual current armature current is calculated as:

$$I_{act} = \frac{U_{Rs}}{R_s}$$

Then the control surface failure rate η can be calculated by the following formula:

$$\eta = 1 - \frac{I_{act}}{I_s}$$

2.2 Output once in place and grading

According to the requirements of the control reconfiguration system, the control surface failure detection system must give the correct failure rate output directly once in place. However, at the beginning of the impact, on the one hand, due to the sudden change in the aerodynamic torque of the control surface, the hinge moment H_e goes down rapidly which causes the sudden uncertain change of detection system output appearing in a very short time, on the other hand, the output changes have a process, although this process is often very short. So the detection system will output wrong test results until it reaches the correct one, which has a seriously affect the reconfiguration system. Through the observation of the change of the output signal, a judgment criterion of output stability is established.

This allows the detection system to output the correct signal one at a time after judging, which saves valuable time for the reconstructed system and is extremely important for the success of reconstruction before the aircraft is completely out of control. After a large number of tests and verification, we made the following definition: For the experimental object used in this paper, if the control surface failure detection output changes were both not more than 6% in the two consecutive sliding time period of 0.4 seconds, than the output can be regarded stable.

The criterion of success is that the UCAV can still have the most basic control in the condition of control surface failure. Since no more than 10% of the error is within acceptable limits of fault tolerance control[9]. Through a large number of simulation experiments, we recommend that the detection output be graded every 10% to obtain a sufficiently stable output to the reconfiguration system.

3 Simulation system building

3.1 6-DoF full scale plane modeling

To achieve a comprehensive closed-loop simulation, an aircraft model is established to simulate the dynamics of the aircraft. In this paper, the aircraft dynamics and kinematics are modeled using the conventional six-degree-of-freedom nonlinear full-scale equations, the derivation process refers to Ref. [10].

We make the following assumptions before establishing the airplane equation:

a) The aircraft is rigid body, and the quality is constant;

b) The ground coordinate system is the inertial system;

c) Ignore the ground curvature, that the ground is flat;

d) Aircraft has a geometric shape of symmetry and symmetrical mass distribution, inertial product meets the following conditions:

$$\begin{cases} I_{XY} = \int xydm = 0 \\ I_{ZY} = \int zydm = 0 \end{cases}$$

In this paper, the aircraft coordinate system used in the coordinate system such as the aircraft axis coordinate system, ground coordinate system are used in the US coordinate system, that is, the right hand orthogonal coordinate system, as shown in Fig. 4:

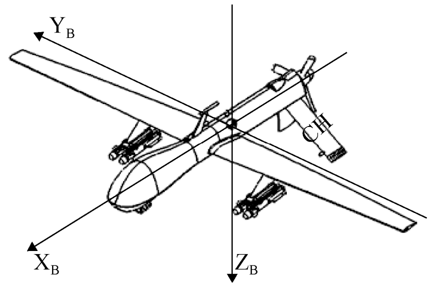

Fig. 4 Right hand orthogonal coordinate system

The dynamics equations of the aircraft are established in the body axis system as:

$$\dot{u} = wq - vr - g\sin\theta + F_X / m$$
$$\dot{v} = ur - wp + g\cos\theta\sin\phi + F_Y / m$$
$$\dot{w} = vp - uq + g\cos\theta\sin\phi + F_Z / m$$
$$L = \dot{q}I_X - \dot{r}I_{XZ} + qr(I_Z - I_Y) - pqI_{XZ}$$
$$M = \dot{q}I_Y + pr(I_X - I_Z) - pqI_{XZ}$$
$$N = \dot{r}I_Z - \dot{p}I_{XZ} + pq(I_Y - I_X) + qrI_{XZ}$$

where m is the aircraft mass, $V_B = [u, v, w]^T$ and $\omega_B = [p, q, r]^T$ are the linear and angular velocity vectors, $F_B = [F_X, F_Y, F_X]^T$ and $M_B = [L, M, N]$ are the external forces and moments vectors acting on aircraft.

Euler's angles (ϕ, θ, ψ) dynamics can be written as follows:

$$\dot{\phi} = p + (r\cos\phi + q\sin\phi)\tan\varphi$$
$$\dot{\theta} = q\cos\phi - r\sin\phi$$
$$\dot{\psi} = \frac{1}{\cos\theta}(r\cos\phi + q\sin\phi)$$

and the external forces and moments vectors acting on aircraft

$$R_X = QSC_X(\alpha,\beta,p,q,r,\delta...)$$
$$R_Y = QSC_Y(\alpha,\beta,p,q,r,\delta...)$$
$$R_Z = QSC_Z(\alpha,\beta,p,q,r,\delta...)$$
$$M_X = QSbC_{MX}(\alpha,\beta,p,q,r,\delta...)$$
$$M_Y = QSbC_{MY}(\alpha,\beta,p,q,r,\delta...)$$
$$M_Z = QSbC_{MZ}(\alpha,\beta,p,q,r,\delta...)$$
$$Q = \frac{1}{2}\rho V^2$$

where is the dynamic pressure, the air density is calculated according to the international standard atmospheric (ISA) model. The aerodynamic coefficients $C_X, C_Y, C_Z, C_{MX}, C_{MY}, C_{MZ}$ are usually given by wind tunnel test or flight test and are obtained by interpolation. The approximation of the ISA model is as follows:

$$T = \begin{cases} T_0 + \lambda h & \text{if } h \leqslant 11\,000 \text{ m} \\ T(h = 11\,000) & \text{if } h > 11\,000 \text{ m} \end{cases},$$

$$\Pr = \begin{cases} \Pr_0(1+\lambda\frac{h}{T_0})^{-\frac{g_0}{R\lambda}} \\ \quad \text{if } h \leqslant 11\,000 \text{ m} \\ \Pr(h=11\,000)e^{-\frac{g}{RT(h=11\,000)}(h-11\,000)} \\ \quad \text{if } h > 11\,000 \text{ m} \end{cases},$$

$$\rho = \Pr/RT, \quad a = \sqrt{\Gamma RT}.$$

where $T = 288.15$ K is the sea level temperature, $\Pr_0 = 10\,132$ N/m is the sea level pressure, $R = 287.05$ J/kg·K is the gas constant of air, $g = 9.806\,65$ m/s^2 is the gravitational acceleration on the sea level, $\lambda = \mathrm{d}T/\mathrm{d}h = -0.006\,5$ K/m is the gradient of temperature decreasing with height, $\Gamma = 1.41$ is the isentropic expansion factor of air.

Hinge moment can be calculated by the following formula:

$$H_e = QS_{el}C_{Hz}(\alpha,\beta,\delta_{el})$$

where S is the area of the control surface, and the aerodynamic coefficients C_{HZ} are obtained by interpolation.

The actuator loop of the left elevator in the nonlinear aircraft model in the simulation system is replaced by physical components such as a physical torque motor, a potentiometer, etc. The hinge torque is simulated by a load torque motor. The system inputs are the pitch angle and yaw angle. The latitudinal input is set as stable to observe the phenomenon of the elevator. The overall hardware-in-the-loop simulation system is shown in Fig. 5 where $\delta_i(i = e_l, e_r, a_l, a_r, r)$ and $x = [\alpha,\beta,\phi,\theta,\psi,p,q,r,H,V]^T$.

Since the aerodynamic force is simulated by the load torque motor, it also has a certain dynamic process. In order to test the principle better and exclude other interference, the dynamic model of it is also added.

3.2 Physical motor conditioning

The physical servo motor used in this paper is controlled by PID with position feedback. The position data is given by the goniometer. The motor control parameters need to be adjusted in order to ensure that the motor can quickly and accurately control the deflection of the virtual control surface. Since the detection system of the failure rate of the control surface does not require a very perfect dynamic characteristic of the actuator, considering the noise it may bring, the speed feedback is not obtained in the control of the motor. At the same time, we adjust the actuator loop model in the experiment making the dynamic characteristics of the model and the actual motor as close as possible.

We use the AD input of dSPACE to observe the response of the physical and model motor at the input of the square wave of δ_c that is the control angle (degree), and the results are shown in Fig. 6, the green curve for the input signal, the blue curve for the model estimated output, the red curve for the actual armature current of the physical motor. It can be seen that its stability and fast performance meet the experimental requirements well.

4 Results of simulation

In the experiment process, we first use Simulink to build the digital simulation model according to the principle mentioned above, and use the torque motor to build the physical model. The model built by Simulink is run by dSPACE and use its DA port to converse the control signal to voltage output to the motor, use AD port to collect the armature current and angle information. After the compilation, the Simulink model will run in dSPACE board connecting with the computer when we can set the operating parameters within the system on line so that the failure rate can be set in real time by means of using the Control Desk software provided by dSPACE.

The change of each variable in the system can be displayed in real time by the Control Desk where we designed a control/result panel.

4.1 Results under the constant input of deflection angle

The damages are set in various degrees to test the effectiveness of the detection system at multiple control surface failure efficiency. Aclinic flying state is set as the initial state of the plane. The test result is shown in Fig. 7. The red line represents the armature current collected by AD, the blue line is the remain rate $(1-\eta)$, the "Remain level" shows the classification output of the actual residual control surface efficiency, "Detection Time" window shows the detection time for the last detection. The first impact was set at near 7th second in the figure with a control surface failure efficiency of 20%, and the second impact at 24th with 70%. The detection system gives an accurate and stable failure rate detection output in 0.8 seconds. The conclusion can be drawn that the detection system can quickly and accurately detect vertical degree of failure when the deflection angle of actuator is constant.

4.2 Results under the Sinusoidal input of deflection angle

Same as in the test above, the faults are set for several times to show the results clearly. We adjust the aircraft input so that the deflection angle presents sinusoidal variation and has amplitude of 10 degrees. In this measure the effectiveness of the system can be tested in the condition of multiple deflection positions and when the angle changes constantly. The experimental results from Fig. 8 show that the amplitude of the control surface deflection is close to the full value, resulting in a large change in the armature current response. Since the accuracy of the model is not 100%, the detection system also appears to fluctuate, but the amplitude is much smaller than the current variation range, which makes slight impact on the output. The detection system gives the failure rate output exactly within 1 second. It can be seen that the detection method can guarantee the fast and effective detection of the control surface failure and provide the failure rate of the control surface accurately in the case of multiple deflection angle and continuous change of it. And the false alarm rate is very low.

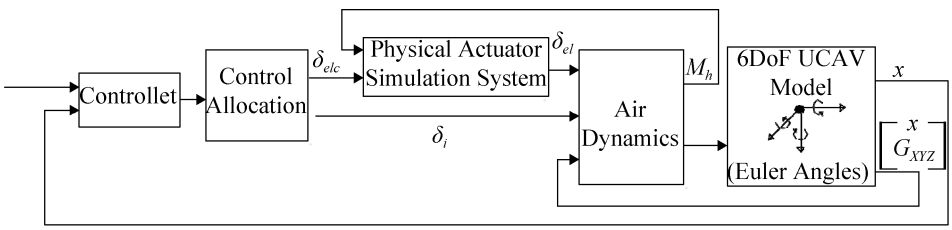

Fig. 5 Hardware-in-the-loop simulation system

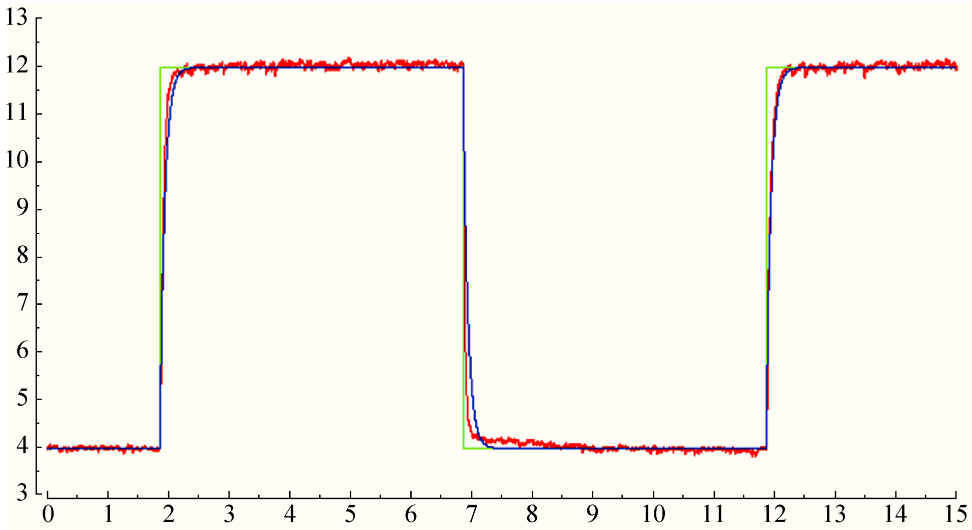

Fig. 6　Dynamic characteristic test for motors

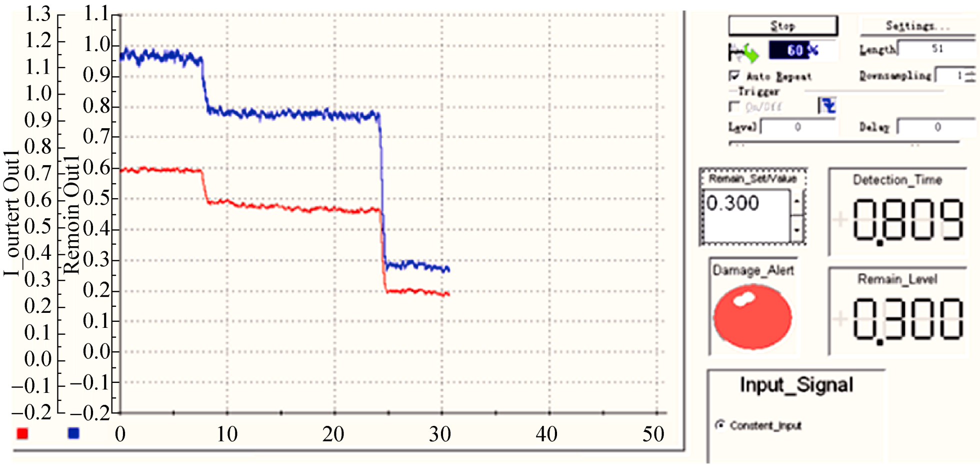

Fig. 7　Test result when deflection angle input is constant

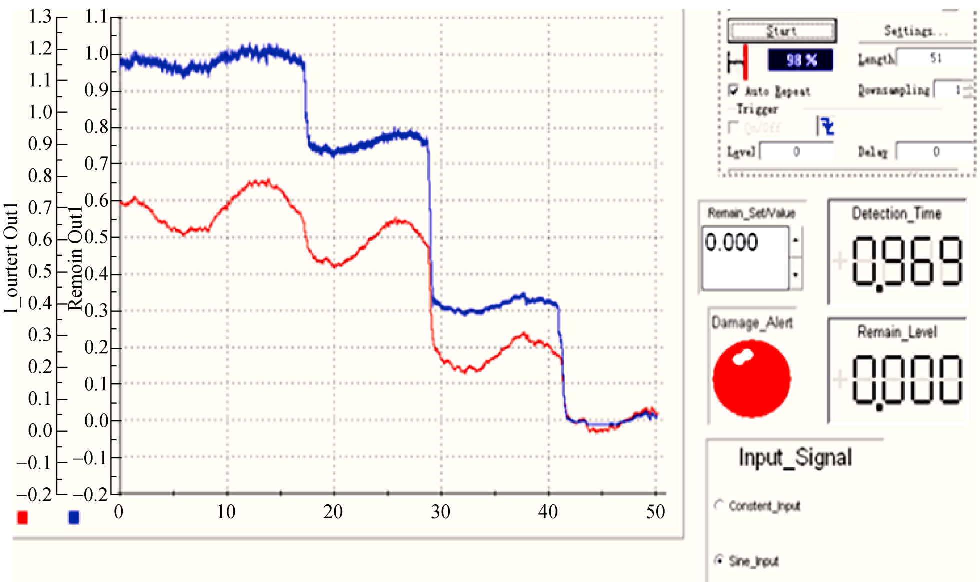

Fig. 8　Test result when deflection angle input is constant

5 Conclusion

In this paper, an effective UCAV control surface failure rate detection method is proposed based on the aerodynamic dynamic contrast model. In this method, the hinge moment of the control surface under normal conditions is accurately estimated by the well-established aircraft dynamic model, which creates the conditions for the detection system to output the detection results quickly and efficiently. The detection results are output after stabilization, despite sacrificing a small amount of time, but provide the accurate result once-in-place to reconfiguration system making it more efficient.

The hardware-in-the-loop simulation results show that the detection system can quickly and accurately detect the failure of the control surface and output the failure rate in the 1-second period conservatively estimate under the constant input of the attitude angle and the sinusoidal input. As a result, the reconfiguration system can have a greater chance to re-enable the flight control system to get control of the aircraft, increasing the survival of UCAV post-war damage conditions. The detection method proposed in this paper also has the advantages of small computation and universality applicability. For different UAVs, it can be used quickly and effectively without the need of modifying a large number of control parameters and training, and made a technical reserve for the future development of UCAV.

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