Hybrid Nonlinear Control for Fighter with Center of Gravity Perturbation and Aerodynamic Parameter Uncertainty

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ABSTRACT In this paper, a hybrid nonlinear control scheme combines nonlinear dynamic inversion (NDI) and adaptive sliding mode (ASM) control is proposed for high angle of attack fighter system with center of gravity perturbation and aerodynamic parameter uncertainty. By introducing the affine nonlinear model of inner loop angular velocity with center of gravity perturbation, an adaptive sliding mode control scheme based on the online estimation of radial basis function (RBF) neural network is designed, which can reduce the dynamic inversion error of NDI control and rapidly compensate the multi-coupled channel oscillation. The aerodynamic force and moment coefficients are estimated via the iterative weighted least squares (IRLS) method. The outer loop integral sliding mode (ISM) NDI controller is designed, which can be robust to disturbances and mismatched uncertainties through adjusting adaptive gain, and the rudder yaw chattering caused by the uncertainty of aerodynamic parameters is reduced simultaneously. The Lyapunov stability theory and Barbarat lemma prove the stability of the designed control scheme, and the effectiveness of the hybrid nonlinear control scheme is verified based on F-16 model.

INDEX TERMS High angle of attack maneuver, Center of gravity perturbation, Aerodynamic parameter uncertainty, Adaptive control, Sliding mode control

I. INTRODUCTION
As a significant issue of the national army's air power, fighters have advantages such as wide flight envelope, fast flight speed, and close combat capability [1]. However, the nonlinearity, uncertainty, and fast temporal variability in the maneuvering process of fighters at high angles and large angular rates [2], increase the difficulty of controller design. In actual flight, fuel consumption and environmental factors will cause the fighter's mass and center of gravity to change continuously. And the aerodynamic parameters will change with the changes in flight speed, altitude and atmospheric density. Fighters usually adopt a statically unstable aerodynamic layout to achieve better maneuverability, such as F-16 fighters [3], which increases the fighter's sensitivity to parameter uncertainties. In order to enable the fighter to fly safely and reliably with good maneuverability under predictable parameter changes, eliminating the influence of uncertain parameters on system performance is an important research topic in the control of such uncertain nonlinear systems [4-7].

Traditional fighter modeling and controller design are mainly based on the assumption that the center of gravity is constant. This simplified method is robust to small changes in the center of gravity (e.g., fuel consumption, landing gear retraction, etc.). However, when the center of gravity encounters a wide range of changes, such as heavy airdrops (or in-flight structural damage to the fighters), it can severely affect the dynamics of the fighters and reduce the maneuverability [8-10]. The center of gravity perturbation has varying degrees of influence on the stability and maneuverability of the fighter, which manifests as the rise or fall of the flight trajectory, the sudden change of the flight speed, and the fighter maneuvering overdrive [11], where the fighters may deviate from its nominal dynamics. The control system has difficulty adapting to changes in dynamics and control characteristics, and robust stability and consistent performance cannot be guaranteed [12-13]. Jing Zhang et al

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used a dynamic inverse control based gravity center position estimation method to compensate for the controller, but the model they built could not reflect the dynamic process of center of gravity changes, and it is difficult to extend to the general cases [14]; sliding mode control as a common control method to adapt to uncertain system dynamics and center of gravity deviations through variable structural sliding mode, can compensate for external perturbations on the system, however it will introduce chattering to the control system, which requires additional compensation to eliminate it [15-17]; another simple and effective method is the robust flight control method based on the linear matrix of inequality (LMI), which can effectively guarantee the robustness against external disturbances and changes of internal parameters [18-19], but this method is based on a simple short-period approximate longitudinal model, which ignores the inherent nonlinear characteristics of the fighters, and the differences between the linear model and the actual nonlinear fighters affect the control performance and stability of the fighters.

In addition to the center of gravity perturbation, the complex aerodynamic characteristics of the fighters are difficult to accurately measure in wind tunnel tests, and its longitudinal and transverse stability changes from weak stability to unstable with the change of Mach number, angle of attack and sideslip angle. With large measurement errors and the aerodynamic parameters changing [20-21], the occurrence of uncertain aerodynamic parameters is inevitable. In order to ensure the flight performance and maneuverability of the fighters, many researchers have addressed the problem of aerodynamic parameter uncertainty in recent years [22-25]. Yun Y et al designed a smooth, fixed-time converging sliding-mode controller for missile flight systems with aerodynamic uncertainty to ensure that the system can track the desired instructions in a consistent finite time under different initial conditions [26]. Ma Y Y et al considered interference suppression and introduced an extended state observer to estimate online the aerodynamic parameter uncertainty term caused by modeling errors in the large-angle state, using feedback linearization and non-singular terminal slide control to ensure the controllability of 90° large-angle flight [27]; Tanaka K et al solved a linear matrix for a powered paraglider transverse lateral model with aerodynamic parameter uncertainty Inequalities are designed with robust nonlinear controllers to make the system stable over the operating domain [28]. However, the above-mentioned methods mostly aim at flight control systems with continuous aerodynamic parameter expressions, and need to control multiple parameters to compensate for the effects of aerodynamic parameter uncertainty, which has high complexity and is difficult to meet the requirements of high reliability of control laws for fighter systems with discrete blowing data.

Based on the above discussion, this paper takes a typical F-16 model as the research object, and designs a hybrid nonlinear control scheme based on adaptive sliding mode for the system with center of gravity perturbation, aerodynamic parameter uncertainty and external interference. First, in order to simulate the system parameter uncertainty, an affine nonlinear model with a center of gravity perturbation, uncertain aerodynamic parameters and external interference is established based on the F-16 nonlinear dynamics equations; then, considering the center of gravity perturbation that occurs during the large maneuver of the inner loop system, a hybrid adaptive sliding strategy is proposed based on the nominal NDI control law, and the center of gravity perturbation is estimated online through the RBF neural network, which is an effective method to estimate the unknown terms of nonlinear system [29]. The compensation of the adaptive gain of the sliding mode control system is based on the adaptive gain of the IRLS, and the robustness of the control system is ensured by adjusting the adaptive gain to achieve the stable tracking of the system for large-angle commands, which can reach the similar level of the robust control method [30]. The main work and contributions of this paper are presented as follows:

- An adaptive sliding mode control scheme based on the online estimation of RBF neural network can accurately estimate the center of gravity perturbation for systems with nonlinearity and parameterically uncertainty, which ensures small dynamic inversion errors while compensating for the multi-coupled channel oscillations caused by center of gravity perturbation.
- The design of the outer-loop adaptive ISM control scheme without the perturbation/uncertainty of the upper bound to be able to ensure the tracking performance of the control system, and the adaptive law only needs to update one parameter online to compensate for the influence of interference or non-matching aerodynamic parameter uncertainty, reducing the complexity of the calculation and improving the reliability of the controller.
- The estimation of aerodynamic and torque coefficients based on the IRLS algorithm improves the accuracy of the estimation and avoids the problem of low model accuracy caused by center of gravity perturbation and complex aerodynamic parameters in the F-16 fighter’s nonlinear model.

II. SYSTEM DESCRIPTION AND PROBLEM FORMULATION

This section introduces the nonlinear dynamics model of the F-16 fighters, the center of gravity perturbation model, and the aerodynamic parameter uncertainty model.

A. NONLINEAR DYNAMICS MODEL

F-16 is a complex nonlinear, strongly coupled system with a partial equation of state expressed as follows:
The uncertain aerodynamic coefficients can be expressed as:

\[ \Delta C_{ij} = \hat{C}_{ij}(1 + \Delta \hat{C}_{ij}) \]

where \( \Delta \hat{C}_{ij} \) is the estimated error of the aerodynamic parameter, \( \Delta \hat{C}_{ij} \) is the actuation range of the uncertain parameter, and the conditions: \( |\Delta \hat{C}_{ij}| \leq 0.4, |\Delta \hat{C}_{ij}| \leq 0.35, |\Delta \hat{C}_{ij}| \leq 0.4 \) are satisfied. Then the total aerodynamic parameter uncertainty is as follows:

\[ C_{m} = C_{m} + \Delta C_{m}(\alpha, \beta, \gamma) + \Delta C_{m}(\alpha, \beta, \gamma) + \Delta C_{m}(\alpha, \beta, \gamma) + \Delta C_{m}(\alpha, \beta, \gamma) + \Delta C_{m}(\alpha, \beta, \gamma) \]

\[ = q\theta (C_{m} + \Delta C_{m}(\alpha, \beta, \gamma) + \Delta C_{m}(\alpha, \beta, \gamma) + \Delta C_{m}(\alpha, \beta, \gamma) + \Delta C_{m}(\alpha, \beta, \gamma) + \Delta C_{m}(\alpha, \beta, \gamma)) \]

\[ + \frac{q}{2V} [C_{m} + \Delta C_{m}(\alpha, \beta, \gamma) + \Delta C_{m}(\alpha, \beta, \gamma)] + \Delta C_{m}(\alpha, \beta, \gamma) \]

\[ + \frac{q_{3}}{2V} [C_{m} + \Delta C_{m}(\alpha, \beta, \gamma) + \Delta C_{m}(\alpha, \beta, \gamma)] + \Delta C_{m}(\alpha, \beta, \gamma) \]

where \( \alpha, \beta, \gamma \) are the skid angle, pitch angle, and yaw angle, respectively.
Define the inner-loop state variable $\mathbf{x}_1 = [p \ q \ r]^T$ and the control input variable $\mathbf{u} = [\delta_\alpha \ \delta_v \ \delta_\delta \ \delta_e]^T$, where $\delta_\alpha$ is the aileron deflection angle, $\delta_v$ is the elevator deflection angle, $\delta_\delta$ is the directional rudder deflection angle, $\delta_e$ is the thrust vector lateral rudder deflection angle, and $\delta_\delta$ is the thrust vector longitudinal rudder deflection angle. Then an inner loop affine nonlinear model of the system with center of gravity perturbation can be written in the following form:

$$\begin{cases}
\dot{x}_1 = f_1(x_1, \bar{x}_{cgr}) + g_1(x_1)u \\
y_1 = x_1
\end{cases}$$

(6)

For the nonlinear system (6), the center of gravity perturbation $\bar{x}_{cgr}$ is replaced by its estimate $\hat{x}_{cgr}$, and the uncertainty $\Delta f_1$ consists of the center of gravity estimation error $\hat{x}_{cgr}$ and the modeling error. Then the inner loop affine nonlinear model (7) can be rewritten as follows:

$$\begin{cases}
\dot{x}_1 = f_1(x_1, \hat{x}_{cgr}) + g_1(x_1)u + \Delta f_1(x_1, u, \hat{x}_{cgr}) \\
y_1 = x_1
\end{cases}$$

(7)

**Assumption 1.** The uncertainty term $\Delta f_1(x_1, u, \hat{x}_{cgr})$ and its partial derivative to each variable is bounded.

Define the outer loop state variable $\mathbf{x}_2 = [\alpha \ \beta]^T$ and control input variable $\mathbf{x}_{id} = [\dot{p}_d \ \dot{q}_d \ \dot{r}_d]^T$, then the outer loop affine nonlinear model of the system with aerodynamic parameter uncertainty and perturbation is described as follows:

$$\begin{cases}
\dot{x}_2 = f_2(x_2) + g_2(x_2)(x_1 + d) + \Phi \\
y_2 = x_2
\end{cases}$$

(8)

where $\Phi = \Omega + Y$, $\Omega$ denotes aerodynamic parameter uncertainty due to estimation error, $Y$ denotes parameter uncertainty due to modeling error, $d \in \mathbb{R}^2$ denotes external perturbations to the system.

III. CONTROLLER DESIGN

In this section, a hybrid nonlinear control scheme is designed for F-16 six-degree-of-freedom nonlinear dynamics model to ensure that the system can quickly compensate for the effects of center of gravity perturbation, aerodynamic parameter uncertainty and external disturbance, and has a good tracking performance.

In order to achieve the above goals, the following key issues need to be resolved: (1) The effect of NDI control depends on the accuracy of the model, so the parameter estimation algorithms need to be designed to reduce the uncertainty caused by parameter estimation; (2) The perturbation of the center of gravity will affect the aerodynamic moment, and the multi-coupling channel oscillation caused by the perturbation needs to be compensated; (3) The uncertainty of aerodynamic parameters is specifically manifested as non-matching uncertainty, and it is necessary to design a control law to eliminate the impact of matching perturbation and non-matching uncertainty on the system.

Considering the obvious differences in the time scale of the flight state variables, the control system is divided into a fast loop and a slow loop, in which the outer loop is the slower attitude angle loop and the inner loop is the faster attitude angular velocity loop. The overall control block diagram of the system is shown in Fig.2.
Assumption 2. Both the disturbance $d$ and the uncertainty $\Phi$ are bounded, satisfying $\|d\| \leq d_0$, $\|\Phi\| \leq \phi_0$, $d_0$, $\phi_0$ is a known constant.

Remark 1. $g_1(x_1)$ and $g_2(x_2)$ are matrices of $3 \times 5$ and $2 \times 3$, respectively, whose elements are functions related to angle of attack, angle of slip, wing area, wing span, rotational inertia, aerodynamic parameters, etc.

Remark 2. Disturbance is generated by gusts of wind and turbulence and can be regarded as a sinusoidal signal with bounded amplitude\cite{31}. Since the uncertainty $\Delta f_1(x_1, u, \hat{x}_{cgr})$ is a function of bounded variables (including attitude angular velocity, rudder yaw angle and center of gravity) and the perturbation of the uncertain aerodynamic parameters is bounded, Assumptions 1 and 2 are reasonable in real cases.

A. INNER LOOP CONTROLLER DESIGN

In this section, a nonlinear fault-tolerant control scheme based on adaptive sliding mode is designed for an inner loop attitude angular velocity system with center of gravity perturbation. Based on the equilibrium equation, the center of gravity is estimated online through RBF neural network training, and the result is introduced into the control law to compensate for the perturbation.

1) ASM CONTROL LAW DESIGN FOR INNER LOOP

In order to compensate for errors in center of gravity estimation and fighter's modeling, the inner loop control law is designed based on NDI control as follows:

$$u = g_1^{-1}(x_1)(v_1 - f_1(x_1, \hat{x}_{cgr}))$$

where $v_1$ is the virtual control signal of the inner loop.

To design the control signal $v_1$, define the sliding mode surface as follows:

$$s_1 = e_1 + K_1 \int e_1 \, dt$$

where $e_1 = x_1 - x_{1id}$, $x_{1id}$ is the reference instruction, provided by the output of the outer loop subsystem, $K_1 = \text{diag}(k_{i1}, k_{i2}, k_{i3})$ is the diagonal gain matrix, and $k_{i1} > 0$, $i = 1, 2, 3$.

Taking the derivative of the sliding mode surface (10), we can get:

$$\dot{s}_1 = \dot{e}_1 + K_1 e_1 = f_1(x_1, \hat{x}_{cgr}) + g_1(x_1) u + \Delta f_1(x_1, u, \hat{x}_{cgr}) - \dot{x}_{1id} + K_1 e_1$$

Substituting Eq. (9) into Eq. (11):

$$\dot{s}_1 = v_1 + \Delta f_1(x_1, u, \hat{x}_{cgr}) - \dot{x}_{1id} + K_1 e_1$$

Define

$$\Delta F_1 = \Delta f_1(x_1, u, \hat{x}_{cgr})$$

then

$$\dot{s}_1 = v_1 + K_1 e_1 + \Delta F_1$$

where $\Delta F_1$ is the uncertainty term of the inner-loop system, which can be viewed as a nonlinear function about $x_1, u, \hat{x}_{cgr}$.

Assumption 3. The uncertainty term $\Delta F_1$ satisfies $\|\Delta F_1\| \leq \mu_1 \hat{\vartheta}_1$, where $\mu_1 = [\mu_{11} \mu_{12} \mu_{13}]^T$. $\hat{\vartheta}_1 = [1 \|x_1\| \|u\|]^T$, $\mu_{1i} > 0$, $i = 1, 2, 3$. Also, $s_i^1 \Delta F_1 = \|s_i\| \|\Delta F_1\|$ and $\sum_i \|s_i\| = 0$ are not established at the same time.

Remark 3. $\Delta F_1$ is the function of $x_1$ and $u$. In real cases, the sizes of these variables are limited by the rate, so they are bounded. This ensures that Assumption 3 is reasonable. And the unknown matrix $\mu_1$ can be estimated online by an adaptive algorithm.

Remark 4. In real cases, following conditions are almost never met at the same time due to the noise: the angle between the sI and the uncertainty term $\Delta F_1$ is 0, and the values of $s_{11} - s_{13}$ are all 0. So we can assume that $s_i^1 \Delta F_1 = \|s_i\| \|\Delta F_1\|$ and $\sum_i \|s_i\| = 0$ will not be established at the same time.

The control signal $v_1$ is designed as follows:

$$v_1 = \begin{cases} -K_1 e_1 - \epsilon_1 \text{sgn}(s_1) - \frac{s_1}{\|s_1\|} \hat{\mu}_1^T \hat{\vartheta}_1, & s_1 \neq 0 \\ -K_1 e_1, & s_1 = 0 \end{cases} \tag{15}$$

where $\epsilon_1 > 0$, $\hat{\mu}_1$ is the estimated value of $\mu_1$. $\hat{\mu}_1 = \mu_1 - \mu_1$ is the estimation error.

Combined with Eq.(9), the inner loop control law is obtained as follows:

$$u = \begin{cases} g_1^{-1}(-K_1 e_1 - \epsilon_1 \text{sgn}(s_1) - \frac{s_1}{\|s_1\|} \hat{\mu}_1^T \hat{\vartheta}_1 - f_1), & s_1 \neq 0 \\ g_1^{-1}(-K_1 e_1 - f_1), & s_1 = 0 \end{cases} \tag{16}$$

Definition 1 [32]. A continuous function $f(t)$ has bounded derivatives in the interval $I$, then $f(t)$ is consistently continuous over the interval $I$.

Theorem 1. [33] (Barbarat lemma). If the differentiable function $f(t)$ is bounded and $f(t)$ is consistently continuous at $t \to \infty$, then $f(t) \to 0$ at $t \to \infty$.

Theorem 2. When Assumption 3 holds, the adaptive control law (16) and the adaptive parameter adjustment law (17) on $\hat{\mu}_1$ can ensure that the system state reaches the sliding mode surface $s_1 = 0$ in finite time and that the state error $e_1$ converges to zero asymptotically.

$$\hat{\mu}_1 = \frac{s_1}{\|s_1\|} P^{-1} \hat{\vartheta}_1$$

where $P$ is the positive definite symmetric matrix of $3 \times 3$.

Proof. Choose the following Lyapunov function:

$$V = \frac{1}{2} s_1^T s_1 + \frac{1}{2} \hat{\mu}_1^T P \hat{\mu}_1 \tag{18}$$

Substituting Eq.(13), (14) into the derivative of Eq.(18):

$$\dot{V} = s_i^T \dot{s}_1 + \hat{\mu}_1^T P \hat{\mu}_1$$

$$= s_i^T (-\epsilon_1 \text{sgn}(s_1) - \frac{s_1}{\|s_1\|} \hat{\mu}_1^T \hat{\vartheta}_1 + \Delta F_1) + \hat{\mu}_1^T P \hat{\mu}_1$$

$$\leq -\epsilon_1 \sum_{i=1}^3 |s_i| - \|s_1\| \|s_1\| \|\Delta F_1\| + \|s_1\| \|s_1\| \|\Delta F_1\|$$

Substituting Assumption 3 and Eq.(17) into Eq.(19):
\[
\dot{V} \leq -\varepsilon_1 \sum_{i=1}^{3} |s_i| - \|s_i\| \|\dot{\mu}_i^T \mathcal{A}_i + \|s_i\| \|\mu_i^T \mathcal{B}_i + \mu_i^T \mathbb{P} \dot{\mu}_i\| \tag{20}
\]
\[
= -\varepsilon_1 \sum_{i=1}^{3} |s_i| - \|s_i\| \|\dot{\mu}_i^T \mathcal{A}_i + \|s_i\| \|\mu_i^T \mathcal{B}_i\|
\]
\[
= -\varepsilon_1 \sum_{i=1}^{3} |s_i| < 0
\]

It follows that \(V(t) \leq V(0)\) and \(V\) are bounded. Therefore, \(s_1, \dot{s}_1\) are bounded, and since \(x_1, u, \dot{x}_cgr, \ddot{x}_cgr\) are all bounded variables, combining Eq. (10) and (11), we can see that \(e_1, \dot{s}_1\) are also bounded.

Taking the derivative of (11):
\[
\ddot{s}_i = \frac{\partial f_i(x_1, \dot{x}_cgr)}{\partial x_1}(f_1 + g_i u + \Delta f_1) + \frac{\partial g_i(x_1)}{\partial x_1} \dot{x}_i u + g_i(x_1) \dot{u} + \frac{\partial \Delta f_i(x_1, u, \dot{x}_cgr)}{\partial x_1} + \frac{\partial \Delta f_i(x_1, u, \dot{x}_cgr)}{\partial \dot{u}}\dot{u} + \frac{\partial \Delta f_i(x_1, u, \dot{x}_cgr)}{\partial \dot{\dot{x}}_cgr}\dot{x}_cgr.
\]

From the fact that \(\dot{u}, \ddot{x}_cgr, \dddot{x}_cgr\) is bounded, \(\dot{s}_1\) is bounded. Similarly, \(\ddot{s}_1\) is also bounded.

Taking the derivative of \(V\):
\[
\dot{V} = s_1^T \ddot{s}_1 + s_1^T \dot{s}_1 + \mu_i^T \mathbb{P} \ddot{\mu}_i + \mu_i^T \mathbb{P} \dot{\mu}_i.
\]

Therefore, \(\dot{V}\) is bounded, and by Definition 1, \(\dot{V}\) is consistent and continuous. It follows from Theorem 1 that \(\dot{V} \rightarrow 0\) when \(t \rightarrow \infty\). Also from Eq. (20), \(\dot{V} \leq -\varepsilon_1 \sum_{i=1}^{3} |s_i|\), then \(s_1 \rightarrow 0\), when \(t \rightarrow \infty\). According to Lyapunov stability theory and Barbaret lemma, the sliding variable \(s_1\) converges to zero asymptotically and can guarantee that the state error \(e_1\) converges to zero asymptotically.

The inner-loop control law designed in this section takes into account the influence of gravity perturbation and uncertainty caused by estimation error/modeling error, solves the problem of large inverse error of NDI control law caused by low model accuracy, and realizes the compensation of the center of gravity perturbation and uncertainty.

2) ONLINE ESTIMATION OF CENTER OF GRAVITY

Based on the simulation experimental data, the RBF neural network structure is adopted, with the flight variable \((\alpha, \beta, \gamma, \delta_e, \delta_\ell)\) as the input layer, the hidden layer using the radial basis function for the nonlinear transformation of the input vector, and the output layer \(x_{cgr}\) for the linear combination calculation of the intermediate layer, the structure is shown in Fig. 3.

The radial basis function of a neural network can be expressed as:
\[
\hat{x}_{cgr} = f(\alpha, \beta, \gamma, \delta_e, \delta_\ell)
\]
where \(f(\alpha, \beta, \gamma, \delta_e, \delta_\ell)\) is the nonlinear function of the center of gravity on the above flight variables.

Based on the simulation experimental data, the RBF neural network structure is adopted, with the flight variable \((\alpha, \beta, \gamma, \delta_e, \delta_\ell)\) as the input layer, the hidden layer using the radial basis function for the nonlinear transformation of the input vector, and the output layer \(x_{cgr}\) for the linear combination calculation of the intermediate layer, the structure is shown in Fig. 3.
where \( X = \{a, \beta, \gamma, \delta_e, \delta_y\} = [x_1, x_2, x_3, x_4, x_5] \), \( x_i \) is the \( i \) th input sample, \( i = 1, 2, 3, 4, 5 \). \( X' \) is the \( j \) th centroid, \( j = 1, 2, \cdots, n \). \( n \) is the number of nodes in the hidden layer. The variance \( \sigma \) is solved by Eq. (27):

\[
\sigma_j = \frac{x_{\text{max}}}{\sqrt{2n}}
\]

where \( x_{\text{max}} \) is the maximum distance between the selected center points.

Then the output of the neural network is

\[
y = f(\alpha, \beta, \gamma, \delta_e, \delta_y) = \sum_{j=1}^{n} w_j e^{-\frac{1}{2\sigma_j} ||X-X'||^2}
\]

where \( w_j \) is the neuronal connection weights between the implicit and output layers, calculated directly by least squares, expressed as follows.

\[
w_j = e^{-\frac{1}{\sigma_{\text{max}}^2} ||X-X'||^2}
\]

The global approximation capability and nonlinear fitting ability of the RBF neural network are suitable for fast-temporal and aggressive maneuvers, ensuring that the control law (16) based on this estimation result can effectively compensate the center of gravity perturbation term.

**B. OUTER LOOP CONTROLLER DESIGN**

In this section, the aerodynamic parameters are estimated based on the IRLS algorithm to improve the estimation accuracy. At the same time, considering the estimation error as a part of the aerodynamic parameter uncertainty, a nonlinear adaptive ISM control law is designed for the outer loop. And the uncertainty of the aerodynamic parameters and external disturbances are compensated in the lateral slip angle channel.

1) ADAPTIVE ISM CONTROL LAW DESIGN FOR OUTER LOOP

The outer loop control law \( x_{1d} \) consists of two parts, which can be expressed as follows:

\[
x_{1d} = x_{1n} + x_{1a}
\]

where \( x_{1n} = g^{-1}_2(x_2)(\nu_2 - f_2(x_2)) \) is the nominal control law of NDI control design, \( \nu_2 = K_2 \epsilon_2 \) is the virtual control signal, \( \epsilon_2 = x_2 - x_{2d} \) is the state error, \( x_{2d} \) is the reference command, \( K_2 = \text{diag}(k_{21}, k_{22}) \) is the diagonal gain matrix and \( k_{2i} > 0, i = 1, 2 \). \( x_{1a} \) is the adaptive ISM control law to compensate for the effects of external interference and aerodynamic parameter uncertainty.

The sliding mode surface is defined as follows:

\[
\dot{s}_2 = G(x_2, x_{2d}) - Z(x_2, x_{2d})
\]

\[
G(x_2, x_{2d}) = \rho(x_2)(x_2 - x_{2d})
\]

\[
Z(x_2, x_{2d}) = \rho(x_2)^{\top}(f_2(x_2) + g_2(x_2)x_{1n} - x_{2d})d\tau
\]

where \( s_2 \in \mathbb{R}^3 \), \( x_{2d}(0), x_{2d}(0) \) is the initial value of the state variable, \( \rho(x_2) \) is the 3x2 matrix to be designed.

Substituting Eq.(7), (32), (33) into the derivative of \( s_2 \):

\[
\dot{s}_2 = \rho(x_2)g_2(x_2)(x_{1d} + d) + \rho(x_2)(\Phi + g_2(x_2)x_{1n})
\]

where \( \Phi = \rho(x_2)^{\top}(f_2(x_2) + g_2(x_2)x_{1n} + \rho(x_2)(\Phi + g_2(x_2)d)\)

**Assumption 4.** \( \rho(x_2)g_2(x_2) \) is reversible.

When \( s_2 = 0 \), the ISM control law is designed as follows:

\[
x_{1a} = -\kappa(\rho(x_2)g_2(x_2))^{-1}(\rho(x_2)g_2(x_2)d + \rho(x_2)\Phi)
\]

Substituting the ISM control law (35) into the derivative of the state error equation:

\[
\dot{e}_2 = f_2(x_2) + g_2(x_2)x_{1n} - g_2(x_2)(\rho(x_2)g_2(x_2))^{-1}x_2d
\]

\[
\rho(x_2)^{\top}(f_2(x_2) + g_2(x_2)x_{1n} - x_{2d} - g_2(x_2)d + d + \Phi - g_2(x_2)(\rho(x_2)g_2(x_2))^{-1}\rho(x_2)\Phi)
\]

\[
= f_2(x_2) + g_2(x_2)x_{1n} - x_{2d} - A\Phi
\]

where \( A = I_2 - g_2(x_2)(\rho(x_2)g_2(x_2))^{-1}\rho(x_2) \) is the projection operator.

From Eq. (36), the effect of matching perturbations can be fully compensated when \( s_2 = 0 \), but the projection operator \( A \) amplifies the effect of non-matching uncertainties \( \Phi \) on the system, i.e., \( ||A\Phi|| \geq ||\Phi|| \), and the optimal matrix \( \rho(x_2) \) needs to be chosen to minimize the non-matching uncertainty paradigm. Here \( \rho(x_2) = \Phi^T(x_2) \) is chosen to ensure that the projection operator \( A \) is symmetrically equal to the power. Thus, the effect of the non-matching uncertainty term is not amplified, i.e., \( ||A\Phi|| = ||\Phi|| \).

**Remark 5.** \( R \in \mathbb{R}^{n \times n} \) satisfies \( ||I - R(R^T R)^{-1}R^T|| = 1 \).

\( R = g_2(x_2) \), \( ||I - g_2(x_2)(g_2^T(x_2)g_2(x_2))^{-1}g_2^T(x_2)|| = 1 \). Then the projection operator \( A \) satisfies \( A^2 = A, ||A|| = 1 \).

When \( s_2 \neq 0 \), the design control law is as follows.

\[
x_{1a} = -\kappa(\rho(x_2)g_2(x_2))^{-1} \frac{s_2}{||s_2||}
\]

where \( \kappa \) is the modulation gain, which is usually greater than the parametric value of interference and uncertainty, but higher modulation gain will cause undesired chattering in the system, which is not feasible in practical engineering.

Therefore, the adaptive modulation gain function (38) is designed here to control the magnitude of the adaptive gain \( \kappa \).

\[
k = \left\{ \begin{array}{ll}
\kappa & \text{if } ||s_2|| > |x_2|, \kappa > \mu_2 \\
\mu_2 & \text{otherwise}
\end{array} \right.
\]

where \( \varepsilon > 0, \kappa > 0, \kappa^* > 0, \mu_2 \) is a known positive number.

**Theorem 3.** For the nonlinear system (8), the sliding mode surface (31) is defined, then the control law (30), (37) and the modulation gain (38) can ensure that the gain \( \kappa \) has an upper bound, i.e., there is a positive \( \kappa^* \) so that the modulation gain satisfies \( \kappa \leq \kappa^* \), and the sliding mode surface \( s_2 \) can converge to an equilibrium position, and the system state can asymptotically track the reference instruction.
Proof. Select the following positive definite Lyapunov function:
\[
V = \frac{1}{2} s_2^T s_2 + \frac{1}{2K} \tilde{\kappa}^2
\]  
(39)

where \( \tilde{\kappa} = \kappa - \kappa^* \).

When: \( \kappa > \mu_2, \|s_2\| > \varepsilon_2 \), derive for Eq.(39) and bring Eq.(34) and (38) into the obtainable
\[
\dot{V} = s_2^T s_2 + \frac{\tilde{\kappa} \kappa}{\kappa} = s_2^T [\Phi(x_2)g_2(x_2)(x_{1a} + d) + \rho(x_2)] \Theta + (\kappa - \kappa^*)\|s_2\| \geq 0
\]  
(40)

Substituting the control law (37) into (40), the following formula is obtained after simplification:
\[
\dot{V} = s_2^T \phi(x_2)g_2(x_2)x_{1a} + s_2^T \phi(x_2)g_2(x_2)d + s_2^T \phi(x_2)d + s_2^T \phi(x_2)\Phi + (\kappa - \kappa^*)\|s_2\| 
\]  
(41)

From Assumption 2, we can see that the interference \( d \) and the uncertainty \( \Phi \) are bounded and satisfy \( \|d\| \leq d_0, \|\Phi\| \leq \phi_0 \), then \( \|\phi(x_2)g_2(x_2)d\|, \|\phi(x_2)\| \) are also bounded, and by choosing the parameter \( \kappa^* \), satisfying \( \kappa^* \geq \|\phi(x_2)g_2(x_2)d\| + \|\phi(x_2)\| \), then \( \dot{V} < 0 \).

When \( \kappa > \mu_2 \), \( \|s_2\| \leq \varepsilon_2 \), \( \kappa \leq 0 \), it is clear that the model gain is guaranteed to decrease when \( \|s_2\| \leq \varepsilon_2 \).

When \( \kappa \leq \mu_2 \), it can be obtained from Eq.(38) that
\[
\dot{V} = s_2^T s_2 + \frac{\tilde{\kappa} \kappa}{\kappa} = s_2^T \phi(x_2)g_2(x_2)x_{1a} + d + \rho(x_2)d + s_2^T \phi(x_2)\Phi + (\kappa - \kappa^*)\|s_2\| 
\]  
(42)

Thus, the control laws (37) and (38) ensure that the modulation gain satisfies \( \kappa \leq \kappa^* \) and the closed-loop system (8) is asymptotically stable in the sliding mode.

Based on the above control law, the adaptive modulation gain \( \kappa \) is updated online without the need to know the upper bounds of the non-matching uncertainties and disturbances, which eliminates their influence on the system and ensures the tracking performance of the closed-loop system.

2) IRLS-BASED AERODYNAMIC PARAMETER ESTIMATION

The aerodynamic parameter estimation error \( \Omega \) is part of the uncertainty term \( \Phi \) of the aerodynamic parameters, in order to minimize \( \Omega \) and improve the estimation accuracy of the aerodynamic parameters, based on the discrete aerodynamic data of the F-16 model, the aerodynamic force and aerodynamic torque coefficients in the IRLS estimation are used, and the relevant aerodynamic coefficients are functions of the angle of attack \( \alpha \), the sideslip angle \( \beta \), and the angle of elevator deflection \( \delta \). To achieve the best fit to the aerodynamic parameters, a linear regression model with multiple input variables is introduced here:
\[
y = X\Theta + e
\]  
(43)

where \( X \in \mathbb{R}^{n \times m} \) is the matrix of input variables \( \alpha, \beta, \delta, y \) is an \( n \)-dimensional output vector of aerodynamic parameters. \( e \) is an \( n \)-dimensional random estimation error, and \( \Theta = [\theta_1, \theta_2, \ldots, \theta_m]^T \) is the matrix of unknown parameters to be estimated. Then the fitted regression expression is expressed as follows:
\[
y_i = \hat{\theta}_1 x_{i1} + \hat{\theta}_2 x_{i2} + \cdots + \hat{\theta}_m x_{im} + e_i = x_i^T \hat{\Theta} + e_i
\]  
(44)

where \( y_i \) is the \( i \) th row element of the vector \( y \), \( x_{ij} \) is the \( i \) th row, \( j \) th column element of the matrix \( X \), \( x_i = [x_{i1}, x_{i2}, \ldots, x_{im}]^T \), \( \hat{\Theta} \) is the estimated value of the matrix \( \Theta, e_i \) is the estimated error of the \( i \) th row, \( i = 1, 2, \ldots, n, j = 1, 2, \ldots, m \). The control objective is to estimate the unknown model coefficients \( \theta_j, j = 1, 2, \ldots, m \) such that the following least-squares function (45) obtains the minimum value.
\[
J_1 = \sum_{i=1}^{n} l(e_i) = \sum_{i=1}^{n} l(y_i - x_i^T \hat{\Theta})
\]  
(45)

where \( l(e_i) \) is a function of the weighting factor associated with the estimation error [36].

In Eq. (43), the partial derivative of the vector \( \hat{\Theta} \) is obtained by the following equation:
\[
\sum_{i=1}^{n} \sigma(y_i - x_i^T \hat{\Theta})x_i^T = 0
\]  
(46)

where \( \sigma(e_i) \) is the derivative of \( l(e_i) \).

Define the weight function as \( w(e_i) = \frac{\sigma(e_i)}{e_i} \) the weight factor as \( w_i = w(e_i) \) [36], then the estimated equation can be written as follows:
\[
\sum_{i=1}^{n} w_i (y_i - x_i^T \hat{\Theta})x_i^T = 0
\]  
(47)

Then the control objective is to make the least squares function (49) take the smallest value.
\[
J_2 = \sum_{i=1}^{n} w_i^2 e_i^2
\]  
(48)

To obtain maximum likelihood estimates, the nominal equation for IRLS is expressed as:
\[
X^T W X \hat{\Theta} = X^T W y
\]  
(49)

where \( W = \text{diag}(w_1, w_2, \cdots, w_n) \) is the weight matrix. Then the parameter estimates are designed as follows:
\[
\hat{\Theta}^{(t)} = [X^T W^{(t-1)} X]^T X^T W^{(t-1)} y
\]  
(50)

Remark 6. The initial value of the parameter \( \Theta^{(0)} \) is estimated by least squares, and the estimate at \( t \) is obtained by
calculating the estimation error $e_i^{(t-1)}$ and the weighting factor $w_i^{(t-1)} = w[e_i^{(t-1)}]$ at $t = 1$.

The IRLS algorithm improves the estimation accuracy of complex aerodynamic parameters by iteratively estimating the weighting factor $w_i$ and reduces the uncertainty caused by the estimation error of aerodynamic parameters $\Phi$, thus making the uncertainty term $\Phi$ of aerodynamic parameters as small as possible and ensuring the reliability of the outer loop control law (37) and (38).

### IV. NUMERICAL SIMULATION

This section is based on MATLAB/Simulink to simulate and analyze the proposed control scheme. In order to show the influence of perturbation and uncertainty on the system, the simulation analysis includes two parts: (1) Sliding mode control simulation for inner loop with center of gravity perturbation. (2) Adaptive ISM control simulation for outer loop with aerodynamic parameter uncertainty.

| Controlled Helm | Minimum | Maximum | Rate Limit |
|-----------------|---------|---------|------------|
| $\delta_e$     | -21.5°  | 21.5°   | ±80°/s     |
| $\delta_r$     | -25°    | 25°     | ±60°/s     |
| $\delta_i$     | -30°    | 30°     | ±120°/s    |
| $\delta_y$     | -15°    | 15°     | ±45°/s     |
| $\delta_d$     | -15°    | 15°     | ±45°/s     |

Set the initial flight state to $31^3$: $\alpha_0 = 10$ deg, $\beta_0 = 0$ deg, $V_0 = 125\text{m/s}$, $\rho_0 = \rho_0 = n_0 = 0$ deg/s. The angle of attack reference signal $\alpha_i$ is set by the command generator based on the first-order low-pass filter. The reference signal $\alpha_i$ changes rapidly from 10deg to 40deg, and the sideslip angle reference signal $\beta_i = 0$ deg. The simulation time is 40s, and the step size is 0.1s.

The system drives the rudder deflection through the control input $u$, and the actuator that generates the deflection command consists of a first-order low-pass filter, a deflection saturation limiter, and an angle rate limiter $3^{11}$, as shown in Table 1.

### A. SIMULATION OF SLIDING MODE CONTROL FOR INNER LOOP WITH CENTER OF GRAVITY PERTURBATION

F-16’s non-perturbation reference center of gravity position $x_{cgr} = 0.35c$, the average aerodynamic chord length $c = 3.25$m. Set the center of gravity to perturb at the 5th to 15th second, the perturbation range is ±0.1c, perturbation step length is 1 second. According to the simulation data, 10 samples are generated for RBF neural network training, and the number of hidden layer nodes $n = 10$, then a nonlinear approximation function with the flight variable $(\alpha, \beta, \gamma, \delta_e, \delta_i)$ as input is obtained. On this basis, the variable $(\alpha, \beta, \gamma, \delta_e, \delta_i)$ data from the measurement feedback is used to calculate $\delta_i$ to achieve the online estimation of the center of gravity, and the estimation results are shown in Table 2.

| Table 2 Estimation of neural network for center of gravity |
|-----------------|---------|---------|---------|---------|---------|---------|
| $\alpha(\circ)$ | $\beta(\circ)$ | $\gamma(\circ)$ | $\delta_e(\circ)$ | $\delta_i(\circ)$ | $\hat{x}_{cgr}(\text{c})$ | $x_{cgr}(\text{c})$ |
| 9.98            | 0.01    | -15.07  | -1.74   | -0.04   | 0.55    | /       |
| 11.35           | -0.03   | -15.67  | -13.22  | -0.48   | 0.36    | 0.3608  |
| 15.44           | -0.31   | -16.25  | -6.04   | -1.21   | 0.38    | 0.3784  |
| 24.64           | -1.11   | -17.17  | -1.42   | -2.41   | 0.42    | 0.4188  |
| 34.45           | -1.26   | -18.03  | 8.00    | -2.27   | 0.39    | 0.3912  |
| 39.42           | -0.37   | -18.83  | 1.31    | -0.78   | 0.36    | 0.3587  |
| 40.30           | 0.63    | -19.61  | 2.61    | 0.05    | 0.34    | 0.3415  |
| 40.45           | 1.26    | -20.38  | 1.54    | 0.58    | 0.33    | 0.3292  |
| 40.44           | 1.38    | -21.15  | 1.62    | 0.87    | 0.31    | 0.3113  |
| 38.54           | 1.25    | -21.91  | 1.16    | 1.03    | 0.32    | 0.3209  |
| 35.04           | 0.98    | -22.68  | 0.57    | 1.09    | 0.33    | 0.3308  |
| 32.03           | 0.49    | -23.29  | 0.22    | 0.40    | 0.35    | /       |

Then, the randomly generated center of gravity with perturbation and the center of gravity estimated online through neural network are fitted, and the results are shown in Fig.4. The estimated value of the center of gravity $\hat{x}_{cgr}$ is very close to its actual value $x_{cgr}$, and the estimation error is less than 1%, while the error of the traditional method is about 3-4% $10$. The RBF neural network algorithm can realize the online estimation of the perturbation center of gravity, which is also more accurate and faster than the traditional observer estimation method.

![FIGURE 4 Estimation curve of center of gravity](image-url)
adaptive controller, and the parameters of the controller are selected as $K_1=\text{diag}\{6,6,6\}$, $P=\text{diag}\{1.4317,2.6548,1.8266\}$, $\varepsilon_1=0.8$.

**FIGURE 5** Comparison curves of attitude angle velocity $p$, $q$, $r$ with center of gravity perturbation

**FIGURE 6** Comparison curves of rudder deflection $\delta_\alpha$, $\delta_\beta$, $\delta_\delta$, $\delta_\tau$ with center of gravity perturbation

**FIGURE 7** IRLS estimation results of aerodynamic parameters

on the attitude angular velocity channel and enable it to track the reference command stably. From Fig. 6, we can see that the NDI-controlled rudder deflection angle oscillates sharply at the initial moment and has obvious chattering, especially the elevating rudder deflection angle, and the longitudinal deflection angle of the thrust vector approaches its maximum deflection angle at the 8th second, which may cause saturation. While the system adopting adaptive sliding mode control has a small overshoot at the initial moment, and there is no obvious oscillation during the center of gravity perturbation, and the deflection angle is guaranteed to be within the maximum deflection angle limits.

**B. SIMULATION OF ADAPTIVE ISM CONTROL FOR OUTER LOOP WITH AERODYNAMIC PARAMETER UNCERTAINTY**

The uncertainties of aerodynamic parameters include: uncertainties caused by estimation errors and uncertainties caused by modeling errors. The estimated results of some typical aerodynamic parameters are shown in Fig. 7, from which we can see that the IRLS method can basically fit the discrete aerodynamic data, but due to the complexity and temporal variability of the aerodynamic parameters, there are still some data deviations from the fitting surface, which is regarded as the uncertainty of the aerodynamic parameters caused by the estimation error.
The uncertainty due to modelling errors is expressed in the following form.

\[
\begin{align*}
C_{pn} &= \hat{C}_p[1 + 0.4(2\eta - 1)] \\
C_{qn} &= \hat{C}_q[1 + 0.35(2\eta - 1)] \\
C_{ln} &= \hat{C}_l[1 + 0.4(2\eta - 1)]
\end{align*}
\]

(51)

Consider external interference to the system at 8th second to 9th second and 20th second to 21st second in the following manner.

\[
d = [d_p , d_q , d_r]_T
\]

(52)

\[
d_p = \begin{cases} 
1.2 \sin \pi t, & 8 < t \leq 9 \\
-2.5 \sin \pi t, & 20 < t \leq 21 \\
0, & \text{others}
\end{cases}
\]

(53)

\[
d_q = \begin{cases} 
2 \sin(0.5\pi t), & 8 < t \leq 9 \\
-\sin(0.5\pi t), & 20 < t \leq 21 \\
0, & \text{others}
\end{cases}
\]

(54)

\[
d_r = \begin{cases} 
0.4 \sin \pi t, & 8 < t \leq 9 \\
-0.8 \sin \pi t, & 20 < t \leq 21 \\
0, & \text{others}
\end{cases}
\]

(55)

Selecting controller parameters \( \hat{\kappa} = 0.025 \), \( \kappa(0) = 0.005 \), \( K_2 = \text{diag}[1.2,1.2,1.2] \), \( \varepsilon_2 = 0.025 \), \( \mu_2 = 0.001 \).

FIGURE 8 Comparison curves of attitude angle velocity \( p, q, r \) with aerodynamic parameter uncertainty

Fig.8 and Fig.9 compare the simulation results of attitude angular velocity and rudder deflection angle of the system under NDI control and adaptive ISM control when considering the uncertain aerodynamic parameters and external interference. It can be seen from the figure that uncertain aerodynamic parameters lead to varying degrees of chattering in each channel of the system, especially the chattering of roll angle speed, aileron deflection angle and rudder deflection angle. The system oscillates when external disturbance occurs, which makes the adjustment time longer. Under the adaptive ISM control, effect on the system can be largely compensated for, and when disturbances occur, the control inputs converge to a steady state within the permissible range, and the state response curve is stable enough to track the reference command. Moreover, compared with the robust control method, the method adopted in this paper can achieve approximate control effect [30], and the uncertainty can be effectively compensated.

FIGURE 9 Comparison curves of rudder deflection \( \delta_a, \delta_e, \delta_r, \delta_y, \delta_z \) with aerodynamic parameter uncertainty

FIGURE 10 Comparison curves of \( \alpha, \beta \) with aerodynamic parameter uncertainty

Fig.10 shows the outer loop state variables comparison curves of angle of attack and sideslip angle. From the figure, we can see that, under the influence of uncertain aerodynamic parameters and external interference, the aerodynamic...
parameters directly affect the aerodynamic lift and lateral force, and indirectly affect the outer loop channel. As a result, the NDI control angle tracking speed is slow, and the tracking error caused by external interference is large. There is a large time lag, limiting the rapidity required for large maneuver. The sideslip angle tracking is obvious, which affects stability of the system during large maneuvers. Under the adaptive ISM control, although the angle of attack channel still has a certain tracking time lag, the tracking error is significantly reduced and eventually tends to 0. This enables the angle of attack and the sideslip angle to accurately track the given reference commands, and the parameter uncertainty and interference can be effectively compensated, ensuring the speed and stability required for the rapid maneuvering of the fighter.

V. CONCLUSION

This paper proposes a hybrid adaptive sliding-mode nonlinear control scheme based on NDI for the F-16 aircraft system with center of gravity perturbation, uncertain aerodynamic parameters and external interference. The effectiveness of the proposed scheme is verified via MATLAB/Simulink simulation, and following conclusions are obtained:

1) The scheme based on the online estimation of RBF neural network shows a faster estimation speed and higher estimation accuracy than the general observer estimation. The scheme based on the adaptive sliding mode controller reduces the dynamic inversion error of the traditional NDI control, shortens the adjustment time of the closed-loop system, and reduces the overshoot caused by the perturbation.

2) The aerodynamic and torque coefficients are estimated by the IRLS algorithm to improve the accuracy of the model. The designed adaptive ISM scheme eliminates the influence of non-matching aerodynamic parameter uncertainty and matching interference by adjusting only one adaptive gain parameter, reduces the multi-channel tracking caused by uncertainty, and ensures the tracking accuracy of the angle of attack, sideslip angle, and attitude angular velocity.

VI. FUTURE RESEARCH PLAN

This paper only considers the effect of parameter uncertainty on the maneuverability of large angle of attack, without considering the occurrence of time delay in practical system, which may cause a serious decline in the effect of sliding mode control and should be handled in order to cater the engineering practice. Future work will focus on the control scheme design for nonlinear flight control systems with time delays, external disturbances and parameter uncertainties at the same time.

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