

Neural Network-Based Incremental Backstepping Sliding Mode Control for Flying-wing Aircraft

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The nonlinear trajectory tracking control problem is studied for a flying-wing aircraft. Starting from a nonlinear dynamics model of the flying-wing aircraft, the trajectory tracking control is decomposed into multiple loops of position control, flight path control and attitude control. An incremental backstepping sliding mode control is proposed to implement attitude control, while an incremental nonlinear dynamic inversion and a nonlinear dynamic inversion design are used to deal with the nonlinear system model for the flight path and position control respectively. In addition, a radial basis function neural network based extended state disturbance observer is proposed to deal with model uncertainties, gust disturbances and unknown faults of the aircraft. The closed-loop control system is proved to be stable using Lyapunov theory. The performance of the proposed disturbance observer based incremental backstepping sliding mode control is demonstrated in simulation through a set of three dimensional tracking scenarios. Compared with both backstepping control and backstepping sliding mode control, tracking performance measured by settling time, tracking error and overshoot are improved by the proposed design when realistic trajectory tracking missions are executed.

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Nomenclature

A_x, A_y, A_z	= specific acceleration along the body axis, m/s^2
$b, \bar{c}$	= wing span and mean aerodynamic chord length of the wing
C_l, C_m, C_n	= aerodynamic moment coefficients as referred in [31]
C_{la}, C_{ma}, C_{na}	= aerodynamic moment coefficients without control inputs along x, y and z-axis respectively,
$C_{l\delta_a}, C_{m\delta_e}, C_{n\delta_r}$	= aerodynamic derivative coefficients of the aileron, elevator and rudder respectively.
C_x, C_y, C_z	= aerodynamic force coefficients as referred in [31]
D_v, D_ω	= flight path disturbance and angular rate disturbance, respectively
$\hat{D}_v, \hat{D}_\omega$	= estimation value of D_v and D_ω
F_A^V, F_A^B	= aerodynamic force in the velocity and the body-fixed reference frame, respectively
$h(\cdot)$	= the basis function of neural network
$H(\cdot)$	= the filter function of state and command
m, I	= mass and inertial matrix of the aircraft
$\bar{q}$	= dynamic pressure
S	= reference area of the wing, m^2
T_c	= thrust input or power level angle
T_s	= sample period of the ESO
$T_{\delta_e}, T_{\delta_r}, T_{\delta_a}, T_{\delta_r}$	= the actuator time constants of elevator, thrust, aileron and rudder, respectively
V, γ, χ	= flight speed, flight path angle and flight azimuth(or heading) angle, respectively.
W^*	= the optimal weight matrix of the RBF-NN
y_{Eng}	= distance between engine and center of gravity along y axis
$\hat{z}_3, \hat{z}_4, \hat{z}_5$	= states of the ESO associated with the state $x_{3\Theta}, x_4, x_5$, respectively
$\beta_{01}, \beta_{02}, \beta_{03}$	= design parameters of the ESO
ξ	= position vectors of $[x_e, y_e, z_e]^T$ in the earth frame, m
$\delta_e, \delta_a, \delta_r$	= elevator, aileron, rudder inputs, respectively, rad
v	= virtual control input
μ, α, β	= bank angle, angle of attack (AOA) and sideslip angle, respectively
ω	= angular rate vectors of $[p, q, r]^T$
<i>Subscripts</i>	
d	= desired value
δ	= control surface
ref	= the reference input

I. Introduction

THE Flying-Wing (FW) aircraft, whereby there is minimal distinction between wing-fuselage and fuselage-tail, is one of the most promising configurations for future aircraft. Compared with the stretched cylindrical shape of conventional aircraft, it has potential advantages in greater internal volume, higher aerodynamic and structural efficiency, reduction of noise, and lower carbon emissions, so it is favorable for transporting a large number of passengers and long range flight [1]. However, because the aircraft has no empennage, there is a short arm for both pitch and yaw axes as well as increased coupling between the control channels. Therefore, the control design for a FW configuration is considerably more challenging than for a conventional aircraft [2].

Cook and de Castro studied stability margin and stability augmentation systems to assess flying qualities of a generic flying-wing or blended-wing-body (BWB) transport aircraft [3]. Rahman and Whidborne proposed an integrated design of flight control and propulsion for the BWB aircraft [4]. To realize a 50% reduced fuel consumption and related CO₂ emissions per passenger-kilometer, the European Union funded an Active Control for Flexible 2020 Aircraft (ACFA) 2020 research project. For the FW aircraft of ACFA, Hanis *et al.* proposed a low order H_∞ optimal control with prescribed structure for multi-model uncertainty [5]. Schirrer *et al.* proposed a Doyle, Glover, and Khargonekar (DGK)-iteration mixed- μ -synthesis design to control the lateral motion and the first wing bending mode of the BWB aircraft by utilizing parameterized Linear Fractional Representations (LFR) [6]. Liu *et al.* proposed a disturbance observer based linear parameter varying (LPV) control to improve robust trajectory tracking performance [7]. Since there are multiple control surfaces of the FW aircraft, control allocation has been studied by some researchers. Qu *et al.* proposed a biased-differential deflection of the split drag rudder for a FW UAV to suppress the control reversal phenomenon arising at higher angle-of-attack in the low speed regime [8]. Vile *et al.* developed a sliding mode controller with priority weighted control allocation to implement fault tolerant control of the actuator [9]. Schmidt *et al.* developed a model, design and flight testing of flutter control for a FW drone [10]. Zou *et al.* presented a time-delay control to suppress body-freedom flutter for a flexible FW [11]. Most of these regard the FW aircraft system as a linear system, which easily makes the system performance deterioration when the aircraft flight deviates far from the designed trim point.

To overcome this shortcoming for the aircraft with nonlinear models, some nonlinear control methods have been proposed. Abnous *et al.* developed and implemented a Lyapunov method based non-linear control strategy to control attitude and altitude under the hovering state for a FW transitioning UAV [12]. Chen *et al.* proposed a Disturbance Observer (DO) based dynamic inversion control to test the directional static unstable FW aircraft [13]. Chen *et al.* also presented a barrier Lyapunov function based nonsingular fast Terminal Sliding Mode Control (TSMC) with fixed-time non-recursive DO for the FW aircraft, thus mismatched disturbances and output constraints are overcome [14]. McFarland and Calise proposed neural network (NN) method to improve on approximate dynamic inversion for control of uncertain nonlinear systems and applied into antiair missiles [15]. Kim and Calise proposed a direct adaptive tracking control architecture using NN, and the model inversion is approximated by the NN, and illustrated the performance of the on-line neural network based adaptation algorithm [16]. Rysdyk and Calise applied neural-network augmented model inversion control to provide a civilian tilt-rotor aircraft with consistent response

characteristics throughout its operating envelope, and Lyapunov analysis guaranteed the boundedness of tracking errors and network parameters [17]. Fu *et al.* applied NNs to design a bank-to-turn missile autopilot, and a stable adaptive law was determined by using the Lyapunov theory [18]. Famularo *et al.* presented an observer based Nonlinear Dynamic Inversion (NDI) to adaptive control of hypersonic vehicles [19]. Quintin *et al.* proposed a Neural Network (NN) adaptive controller with approximate dynamic inversion to control lateral motion of the Cessna Citation X [20]. A recent review of NDI and Sliding Mode Control (SMC) can be found in [21]. This SMC is essentially based on NDI, where the equivalent control is a model-based estimation. However, NDI design depends on a precise model. To reduce the model dependency of the NDI design, Contreras Torres *et al.* proposed an adaptive backstepping (BS) controller for cruise flight of a FW aircraft with unknown parameters [22]. Another method is to use an incremental nonlinear dynamic inversion (INDI) [23]. Tal and Karaman presented an INDI with differential flatness for agile maneuvering control of a tailsitter FW [24]. And INDI have been developed for fault tolerant control [25] and flexible aircraft control [26]. Steffensen *et al.* derived a sensor-based NDI with actuator dynamics, and drove the INDI to approximate the corresponding NDI control law arbitrarily well under constraints [27]. INDI can be used for generic uncertain nonlinear systems without time-scale separation assumption [28]. Therefore, it can reduce model dependency and SMC chattering by a combination of SMC with INDI [21].

Because the FW aircraft is sensitive to unknown disturbances such as wind gusts and turbulences, a disturbance observer is an effective method to reject the disturbances. Chen *et al.* have provided a recent survey of a nonlinear disturbance observer-based control (DOBC) [29]. For control of a quadrotor aircraft under uncertainties and disturbances, Mechali *et al.* presented a fixed-time continuous nonsingular terminal sliding mode control with a DOBC to stabilize the attitude control and an active disturbance rejection control (ADRC) to estimate and cancel the lumped disturbance [30]. It is found that the control conservatism can be reduced by the DOBC design. Liu *et al.* proposed an Incremental Backstepping Sliding Mode (IBSM) control with neural network disturbance observer to improve stability and attitude control [31], but this DO is only used for unknown determined disturbance in the attitude loop and calculation of the DO gain is time consuming.

Motivated by the above work, and considering that the flying wing aircraft is sensitive to unknown wind and faults, this paper considers trajectory tracking control for a FW aircraft with unknown disturbances, model uncertainties and actuator faults. The main contributions of this paper can be summarized as follows: 1) This paper formulates a novel incremental backstepping sliding mode control law taking account stochastic model

using the time scale separation principle, the state variable vector can be divided into four component sub-vectors, i.e., the position vector in the earth frame $\mathbf{x}_1 = [x_e \ y_e \ z_e]^T$, the flight path vector $\mathbf{x}_2 = [V \ \chi \ \gamma]^T$, the attitude vector $\mathbf{x}_3 = [\mu \ \alpha \ \beta]^T$, the angular rate vector $\mathbf{x}_4 = [p \ q \ r]^T$. Let $\mathbf{V} = [V \ 0 \ 0]^T$, the angular speed vector $\boldsymbol{\Omega}_E^V = [-\sin \gamma \cdot \dot{\chi} \ \dot{\gamma} \ \cos \gamma \cdot \dot{\chi}]^T$ denotes the rotational rate of the kinematic reference frame with respect to the earth fixed reference frame.

The translational dynamics model of the FW aircraft is established according to the second Newton's law [32]

$$\begin{cases} \dot{V} = \frac{1}{m} F_{Ax} - g \sin \gamma \\ \dot{\chi} = \frac{1}{mV \cos \gamma} (\cos \mu \cdot F_{Ay} - \sin \mu \cdot F_{Az}) \\ \dot{\gamma} = \frac{1}{-mV} (\sin \mu \cdot F_{Ay} + \cos \mu \cdot F_{Az}) + \frac{g \cos \gamma}{V} \end{cases} \quad (1)$$

The aerodynamic forces in the velocity reference frame can be obtained by transformation matrix from the body fixed reference frame to the aerodynamic reference frame,

$$\mathbf{F}_A^V = \begin{bmatrix} \cos \alpha & 0 & \sin \alpha \\ 0 & 1 & 0 \\ -\sin \alpha & 0 & \cos \alpha \end{bmatrix} \begin{bmatrix} \cos \beta & \sin \beta & \sin \alpha \\ -\sin \beta & \cos \beta & 0 \\ 0 & 0 & 1 \end{bmatrix} \mathbf{F}_A^B \quad (2)$$

that is,

$$\begin{bmatrix} F_{Ax} \\ F_{Ay} \\ F_{Az} \end{bmatrix} = \bar{\mathbf{q}} \mathbf{S} \begin{pmatrix} \cos \alpha \cos \beta \cdot C_X + \cos \alpha \sin \beta \cdot C_Y + \sin \alpha \cdot C_Z \\ -\sin \beta \cdot C_X + \cos \beta \cdot C_Y \\ -\sin \alpha \cos \beta \cdot C_X - \sin \alpha \sin \beta \cdot C_Y + \cos \alpha \cdot C_Z \end{pmatrix} \quad (3)$$

and the aerodynamic derivatives are

$$\begin{aligned} C_X &= C_{X_0} + C_{X_\alpha} \alpha + C_{X_q} \frac{\bar{q}c}{V} + C_{X_{\delta e}} \delta_e + C_{X_T} T_c \\ C_Y &= C_{Y_0} + C_{Y_\beta} \beta + C_{Y_p} \frac{pb}{V} + C_{Y_r} \frac{rb}{V} + C_{Y_{\delta a}} \delta_a + C_{Y_{\delta r}} \delta_r \\ C_Z &= C_{Z_0} + C_{Z_\alpha} \alpha + C_{Z_q} \frac{\bar{q}c}{V} + C_{Z_{\delta e}} \delta_e + C_{Z_T} T_c \end{aligned} \quad (4)$$

where C_{X_0} , C_{X_α} , C_{X_q} , $C_{X_{\delta e}}$, C_{X_T} , C_{Y_0} , C_{Y_β} , C_{Y_p} , C_{Y_r} , $C_{Y_{\delta a}}$, $C_{Y_{\delta r}}$, C_{Z_0} , C_{Z_α} , C_{Z_q} , $C_{Z_{\delta e}}$, and C_{Z_T} are the associated aerodynamic coefficients.

Remark 1 In the attitude control loop, β should be kept at zero, because it will be uncomfortable for the passengers when there is side-slipping motion, especially in coordinated turn state. Therefore, it is reasonable to assume that β is close to zero.

Let $F_{A,y}^B = mA_y$, $F_{A,x}^B = mA_x$, $F_{A,z}^B = mA_z$, and substitute them into Eq. (1) with assumption of zero sideslip, yields

$$\dot{\chi} = \frac{1}{V \cos \gamma} (A_x \sin \alpha \sin \mu + A_y \cos \mu - A_z \cos \alpha \sin \mu) \quad (5)$$

$$\dot{\gamma} = \frac{1}{V} (A_x \sin \alpha \cos \mu - A_y \sin \mu - A_z \cos \alpha \cos \mu - g \cos \gamma) \quad (6)$$

By Eqs (1), (5) and (6), we get

$$\begin{bmatrix} \dot{V} \\ \dot{\chi} \\ \dot{\gamma} \end{bmatrix} = \begin{bmatrix} \frac{1}{m} & 0 & 0 \\ 0 & \frac{1}{mV \cos \gamma} & 0 \\ 0 & 0 & \frac{-1}{mV} \end{bmatrix} \left(L_B^V \begin{bmatrix} F_{Ax} \\ F_{Ay} \\ F_{Az} \end{bmatrix} + L_E^V \begin{bmatrix} 0 \\ 0 \\ mg \end{bmatrix} \right) \quad (7)$$

where the transformation matrix from the body-fixed reference frame to the velocity reference frame is

$$L_B^V = (L_V^B)^T = \begin{bmatrix} \cos \alpha \cos \beta & \sin \beta & \sin \alpha \cos \beta \\ -\cos \alpha \sin \beta \cos \mu + \sin \alpha \sin \mu & \cos \beta \cos \mu & -\sin \alpha \sin \beta \cos \mu - \cos \alpha \sin \mu \\ -\cos \alpha \sin \beta \sin \mu - \sin \alpha \cos \mu & \cos \beta \sin \mu & -\sin \alpha \sin \beta \sin \mu + \cos \alpha \cos \mu \end{bmatrix} \quad (8)$$

and the transformation matrix from the Earth-fixed reference frame to the velocity reference frame is

$$L_E^V = \begin{bmatrix} \cos \chi \cos \gamma & \sin \chi \cos \gamma & -\sin \gamma \\ -\sin \chi & \cos \chi & 0 \\ \cos \chi \sin \gamma & \sin \chi \sin \gamma & \cos \gamma \end{bmatrix} \quad (9)$$

From Eqs (4) and (7), we obtain

$$\begin{aligned} \begin{bmatrix} \dot{V} \\ \dot{\gamma} \end{bmatrix} &= \begin{bmatrix} -g \sin \gamma + \frac{\bar{q}S}{m} (\bar{C}_{x_0} \cos \alpha + \bar{C}_{z_0} \sin \alpha) \\ -\frac{A_y \sin \mu}{V} - \frac{g \cos \gamma}{V} - \frac{\bar{q}S}{mV} (-\bar{C}_{x_0} \sin \alpha \cos \mu + \bar{C}_{z_0} \cos \alpha \cos \mu) \end{bmatrix} \\ &+ \frac{\bar{q}S}{mV} \begin{bmatrix} V(C_{x_T} \cos \alpha + C_{z_T} \sin \alpha) & V(C_{x_a} \cos \alpha + C_{z_a} \sin \alpha) \\ C_{x_T} \sin \alpha \cos \mu - C_{z_T} \cos \alpha \cos \mu & C_{x_a} \sin \alpha \cos \mu - C_{z_a} \cos \alpha \cos \mu \end{bmatrix} \begin{bmatrix} T_c \\ \alpha \end{bmatrix} \end{aligned} \quad (10)$$

where $\bar{C}_{x_0} = C_X - C_{x_a} \alpha - C_{x_T} T_c$, $\bar{C}_{z_0} = C_Z - C_{z_a} \alpha - C_{z_T} T_c$, and the desired bank angle can be designed as

$$\mu = \arctan \left(\frac{\dot{\chi} \cos \gamma - A_y \cos \mu / V}{\dot{\gamma} + (A_y \sin \mu + g \cos \gamma) / V} \right) \quad (11)$$

According to Remark 1, assume A_y is small due to small sideslip angle, then

$$\mu = \arctan \left(\frac{\dot{\chi} \cos \gamma}{\dot{\gamma} + g \cos \gamma / V} \right) \quad (12)$$

And we can get [33],

$$\begin{bmatrix} \dot{\mu} \\ \dot{\alpha} \\ \dot{\beta} \end{bmatrix} = \begin{bmatrix} \cos \alpha \cos \beta & 0 & \sin \alpha \\ 0 & 1 & 0 \\ \sin \alpha \cos \beta & 0 & -\cos \alpha \end{bmatrix}^{-1} \left(\begin{bmatrix} p \\ q \\ r \end{bmatrix} - L_v^B \begin{bmatrix} -\dot{\chi} \sin \gamma \\ \dot{\gamma} \\ \dot{\chi} \cos \gamma \end{bmatrix} \right) \quad (13)$$

The rotational dynamics model of the FW aircraft is

$$\begin{bmatrix} \dot{p} \\ \dot{q} \\ \dot{r} \end{bmatrix} = -\bar{I}^{-1} \begin{bmatrix} p \\ q \\ r \end{bmatrix} \times \left(\bar{I} \begin{bmatrix} p \\ q \\ r \end{bmatrix} \right) + \bar{I}^{-1} \begin{bmatrix} \bar{q} S b C_{la} \\ \bar{q} S \bar{c} C_{ma} \\ \bar{q} S b C_{na} \end{bmatrix} + \bar{I}^{-1} \begin{bmatrix} \bar{q} S b C_{l\delta} \\ \bar{q} S \bar{c} C_{m\delta} \\ \bar{q} S b C_{n\delta} \end{bmatrix} \quad (14)$$

where subscripts of a and δ denote aerodynamic terms of surfaces without control and of control surfaces respectively. The aircraft inertial moment matrix is

$$\bar{I} = \begin{bmatrix} I_{xx} & -I_{xy} & -I_{xz} \\ -I_{xy} & I_{yy} & -I_{yz} \\ -I_{xz} & -I_{yz} & I_{zz} \end{bmatrix} \quad (15)$$

B. The fault-tolerant trajectory tracking control problem

Consider the FW aircraft models of (1) ~ (14). Let $\xi_d(t): [0, \infty) \rightarrow R^3$ be a given, sufficiently smooth, time-varying reference position with bounded time-derivatives of $(\dot{\xi}_d(t), \ddot{\xi}_d(t))$. The control objective is to design the nonlinear controller such that the FW aircraft closed-loop system meets the following requirements:

- 1) The closed-loop system is stable under external disturbances;
- 2) The three dimensional trajectory ξ_d is accurately tracked under specified model uncertainty, external disturbances and actuator faults.

$$x_{2d} = \begin{bmatrix} V_d \\ \mathcal{X}_d \\ \gamma_d \end{bmatrix} = \begin{bmatrix} \sqrt{(v_x)^2 + (v_y)^2 + (v_z)^2} \\ -\arcsin\left(\frac{v_y - \hat{v}_w}{V_d \cos \gamma_d}\right) \\ -\arcsin\left(\frac{v_z - \hat{w}_w}{V_d}\right) \end{bmatrix} \quad (17)$$

where v_x , v_y and v_z are the virtual speed inputs along x, y and z- axis , since there is the initial speed value V_0 , then usually $V_d = V_0 + v_V$, the $\hat{v}_w$, $\hat{w}_w$ are the wind speed estimation on the y- and z-axis respectively, and then the virtual control input is designed as

$$v_V = K_\xi (\xi_{ref} - \xi) \quad (18)$$

where $v_V = [v_x \ v_y \ v_z]^T$, K_ξ is the control gain matrix.

B. Flight Path Control

The objective of the flight path control is to drive V, χ, γ to their desired values, and to achieve the desired attitude $\Theta_{ref} = [\mu_{ref} \ \alpha_{ref} \ \beta_{ref}]^T$. Since the aircraft keeps the sideslip angle on zero during the aircraft cruising, then β is assumed to be zero. Then from Eq.(10) with disturbances

$$\begin{bmatrix} \dot{V} \\ \dot{\gamma} \end{bmatrix} = f_3 + g_3 \begin{bmatrix} T_c \\ \alpha \end{bmatrix} + D_V \quad (19)$$

where

$$f_3 = \begin{bmatrix} -g \sin \gamma + \frac{\bar{q}S}{m} (\bar{C}_{x_0} \cos \alpha + \bar{C}_{z_0} \sin \alpha) \\ -\frac{A_y \sin \mu}{V} - \frac{g \cos \gamma}{V} - \frac{\bar{q}S}{mV} (-\bar{C}_{x_0} \sin \alpha \cos \mu + \bar{C}_{z_0} \cos \alpha \cos \mu) \end{bmatrix}, \quad (20)$$

$$g_3 = \frac{\bar{q}S}{mV} \begin{bmatrix} V(C_{x_T} \cos \alpha + C_{z_T} \sin \alpha) & V(C_{x_\alpha} \cos \alpha + C_{z_\alpha} \sin \alpha) \\ C_{x_T} \sin \alpha \cos \mu - C_{z_T} \cos \alpha \cos \mu & C_{x_\alpha} \sin \alpha \cos \mu - C_{z_\alpha} \cos \alpha \cos \mu \end{bmatrix}. \quad (21)$$

Assume D_V meeting $|D_V| < D_{VU}$, D_{VU} is a real value. An incremental sliding mode control law is designed to achieve desired thrust and AOA is designed as

$$\begin{bmatrix} T_d \\ \alpha_d \end{bmatrix} = g_3^{-1}(\bar{x}_2, \bar{x}_3) \left([v_V \ v_\gamma]^T - (v_{F_{dNN}} + \hat{D}_V) - \dot{\bar{x}}_{2,0} \right) \quad (22)$$

where $\bar{x}_2 = [V \ \gamma]^T$, $\bar{x}_3 = [T \ \alpha]^T$, $\dot{\bar{x}}_{2,0} = H(\omega_n)x_2$, the filter $H(w_n)$ refers to Eq.(36); the virtual input is obtained by Proportional Integrative Derivative (PID) control, i.e., $[v_V \ v_\gamma] = K_{x_2} (x_{2d} - x_2)$, K_{x_2} is the gain matrix, $v_{F_{dNN}}$ is a robust virtual control for unobservable disturbance, uncertainty and fault tolerant, which can be approximated by the RBF neural network, i.e., $v_{F_{dNN}} = \hat{W}^T h$, and is designed next (Eq.(34)). The INDI control of the flight path control

loop is designed as in Fig.3, so the desired control input is $\bar{x}_{3d} = [T_d \quad \alpha_d]^T = \Delta\bar{x}_3 + \bar{x}_{3,0}$.

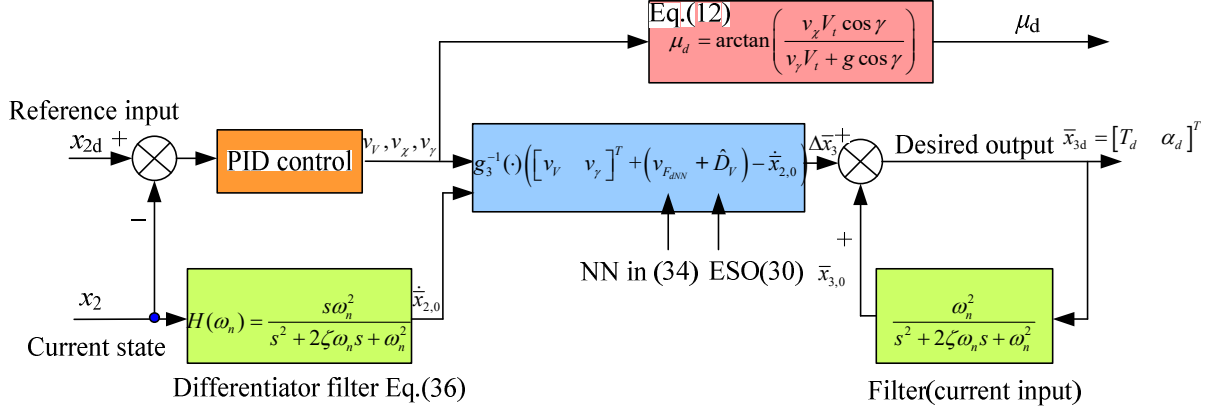


Fig. 3 The INDI design of the flight path control loop.

C. Attitude loop control

In the attitude control loop, the control objective is to track the reference signal $x_{3d} = x_{3,ref}$ with derivative $\dot{x}_{3d}$, where $x_{3,ref} = [\mu_{ref} \quad \alpha_{ref} \quad \beta_{ref}]^T$ is generated by the flight path control loop. Thus the tracking error vector of the attitude is defined as.

$$z_1 = x_3 - x_{3d} \quad (23)$$

From the kinematics of the aircraft of Eq.(13), we get

$$\dot{x}_3 = \bar{f}_3(x_3) + \bar{G}_3(x_3)x_4 \quad (24)$$

where

$$\bar{f}_3 = - \begin{bmatrix} \cos \alpha \cos \beta & 0 & \sin \alpha \\ 0 & 1 & 0 \\ \sin \alpha \cos \beta & 0 & -\cos \alpha \end{bmatrix}^{-1} \left(L_v^B \begin{bmatrix} -\dot{\chi} \sin \gamma \\ \dot{\gamma} \\ \dot{\chi} \cos \gamma \end{bmatrix} \right), \bar{G}_3 = \begin{bmatrix} \cos \alpha \cos \beta & 0 & \sin \alpha \\ 0 & 1 & 0 \\ \sin \alpha \cos \beta & 0 & -\cos \alpha \end{bmatrix}^{-1}. \quad (25)$$

Denoting the angular rate $x_4 = \omega = [p \quad q \quad r]^T$ as the state vector in the angular rate control loop, the attitude motion model of the FW aircraft can be rewritten as follows with bounded disturbances [31],

$$\dot{x}_4(t) = \dot{\omega} = f_4 + G_4 u + D_\omega \quad (26)$$

where $f_4(\cdot)$ and $G_4(\cdot)$ are continuous mapping functions as

$$f_4 = -\bar{I}^{-1} \begin{bmatrix} p \\ q \\ r \end{bmatrix} \times \left(\bar{I} \begin{bmatrix} p \\ q \\ r \end{bmatrix} \right) + \bar{I}^{-1} \begin{bmatrix} \bar{q}SbC_{la} \\ \bar{q}S\bar{c}C_{ma} \\ \bar{q}SbC_{na} \end{bmatrix}, \quad (27)$$

$$G_4 = \bar{I}^{-1} (\mathbf{M}_c)_\delta, \mathbf{M}_c = \begin{bmatrix} \bar{q}Sb(C_{l_{\delta a}} \delta_a + C_{l_{\delta r}} \delta_r) \\ \bar{q}S\bar{c}C_{m_{\delta e}} \delta_e \\ \bar{q}Sb(C_{n_{\delta r}} \delta_r + C_{n_{\delta a}} \delta_a) \end{bmatrix}, (\mathbf{M}_c)_\delta = \frac{\partial \mathbf{M}_c}{\partial \delta} = \bar{q}S \begin{bmatrix} C_{l_{\delta a}} & 0 & C_{l_{\delta r}} \\ 0 & C_{m_{\delta e}} & 0 \\ C_{n_{\delta a}} & 0 & C_{n_{\delta r}} \end{bmatrix} \begin{bmatrix} b \\ \bar{c} \\ b \end{bmatrix} \quad (28)$$

Remark 2 The disturbances $\mathbf{D}_V$ and $\mathbf{D}_\omega$ are assumed to be bounded, because the external winds often occur when the aircraft flight in the air, which are bounded. And the model uncertainties due to varying aerodynamic coefficients are bounded, and actuator faults are also bounded.

Now consider unknown disturbances of D_ω for the attitude system (26), which can be rewritten as

$$\begin{cases} \dot{\mathbf{x}}_{3\Theta} = \mathbf{x}_4 \\ \dot{\mathbf{x}}_4(t) = \dot{\omega} = \mathbf{x}_5 + \mathbf{f}_4 + G_4 \mathbf{u} \\ \dot{\mathbf{x}}_5 = \dot{D}_\omega \\ y = \mathbf{x}_{3\Theta} \end{cases} \quad (29)$$

where $\mathbf{x}_{3\Theta} = [\phi \quad \theta \quad \psi]^T$. A nonlinear extended state observer is designed as follows

$$\begin{cases} \varepsilon_3 = \hat{z}_3 - y \\ \dot{\hat{z}}_3 = \hat{z}_4 - \beta_{01} \varepsilon_3 \\ \dot{\hat{z}}_4 = \hat{z}_5 - \beta_{02} fal(\varepsilon_3, 0.5, \delta_w) + \mathbf{f}_4 + G_4 \mathbf{u} \\ \dot{\hat{z}}_5 = -\beta_{03} fal(\varepsilon_3, 0.25, \delta_w) \end{cases} \quad (30)$$

where ε_3 is the estimation error, $\hat{z}_3$, $\hat{z}_4$, $\hat{z}_5$ are estimator states, β_{0i} and δ_w are design parameters, $i=1,2,3$. $fal(x, a, \delta_w)$ is a nonlinear function defined as follows

$$fal(x, a, \delta_w) = \begin{cases} \frac{x}{\delta_w^{(1-a)}}, & |x| \leq \delta_w \\ \text{sign}(x) |x|^a, & |x| > \delta_w \end{cases} \quad (31)$$

For easy numerical calculation, the ESO of (29)~(31) can be discretized with sample period T_s [34]. And the RBF NNs are used to approximate estimation error, model uncertainties and actuator faults, by Eq (26), we get

$$\dot{\mathbf{x}}_4(t) = \dot{\omega} = \mathbf{f}_4 + G_4 \mathbf{u} + (\hat{z}_5 + W_i^{*T} \cdot h_i(x) + \varepsilon_i) \quad (32)$$

where ε is an arbitrarily small approximation error by regulating the weight and nodes of the RBF-NN, so we assume that $|\varepsilon| \leq \bar{\varepsilon}$, $\bar{\varepsilon} > 0$. The output of the NN is $h(x) = [h_j(x)]^T$, $h(x)$ is chosen as the following Gaussian function

$$h_j(x) = \exp\left(\frac{-\|x - c_j\|^2}{2b_j^2}\right) \quad (33)$$

where c_j is the centre value of neural net j , b_j is the width value of the Gaussian function of net j , and j is the node number of the hidden layer.

Similarly, for the flight path control loop, we get

$$\begin{bmatrix} \dot{V} \\ \dot{\gamma} \end{bmatrix} = f_3 + g_3 \begin{bmatrix} T_c \\ \alpha \end{bmatrix} + (\hat{z}_{4V} + W_o^{*T} \cdot h_o(x) + \varepsilon_o) \quad (34)$$

where $\hat{z}_{4V} = \hat{D}_V$ is the disturbance estimation of D_V by the ESO.

Now consider the kinematics model (24) and the dynamics model (26). The attitude controller is derived in two steps, and the detailed design is presented in Appendix .So the incremental control input is obtained as follows

$$\Delta u_{INBSM} = \bar{G}_4^{-1} \left(-\lambda_1 (-c_3 z_1 + G_3 z_2) - \dot{x}_{4,0} + \dot{x}_{4d} - (\hat{D}_\omega + \hat{W}_R^T h) - G_3^T z_1 - (\bar{h}s + \varsigma \operatorname{sgn}(s)) \right) \quad (35)$$

where $\hat{W}_R^T h = \hat{W}^T h + R$, $R = \frac{\partial f_4(x_3, x_4)}{\partial x} \Big|_0 \Delta x + \Delta D_\omega + O(x, u)$, $\bar{G}_4$ is the estimation of the control effectiveness

matrix $\frac{\partial G_4(x_3, x_4)}{\partial u} \Big|_0$. $\bar{h}$ and ς are sliding mode surface parameters with $\bar{h} > 0$, and $\varsigma > 0$. $\dot{x}_{4,0}$ can be obtained by the

filter

$$\dot{x}_{4,0} = H(\omega_n) \cdot s\omega = \frac{\omega_n^2}{s^2 + 2\zeta\omega_n s + \omega_n^2} s\omega. \quad (36)$$

In order to achieve the reference input $\dot{x}_{4d} = \dot{x}_{4,r} = \dot{\omega}_{ref}$, the dynamic inversion design is applied, and the virtual control input for the attitude control loop is designed as

$$v_3 = K_3 (x_{3,ref} - x_3) \quad (37)$$

where $v_3 = [v_\mu \ v_\alpha \ v_\beta]^T$, K_3 is the control gain matrix. By using Eq.(13) with inversion dynamics, the corresponding $x_{4,r} = \omega_{ref}$ is obtained, and its derivative is obtained by the filter

$$\dot{\mathbf{x}}_{4,r} = H(\omega_n) \cdot s \omega_{ref} = \frac{\omega_n^2}{s^2 + 2\zeta\omega_n s + \omega_n^2} s \omega_{ref}. \quad (38)$$

For the BWB aircraft, the elevator, aileron, rudder and engine actuator dynamics can be simplified as 1st order models,

$$G_{\delta_i}(s) = \frac{T_{\delta_i}}{s + T_{\delta_i}} \quad (39)$$

where $\delta_i = \delta_a, \delta_e, \delta_r, \delta_p$.

For the active disturbance rejection control, a tracking differentiator (TD) and a nonlinear state error feedback (NLSEF) control are usually applied with the ESO. Because the virtual control input in the proposed IBSM design are obtained by the PID method, tracking control in ESO can be replaced by the virtual control design, while the differentiator can be replaced by the filter (36). The NLSEF control is replaced by the backstepping sliding mode control. Therefore, the TD and NLSEF of the ADRC design are indirectly replaced by the IBSMC here.

IV. Simulation and Analysis

The studied aircraft model has a 80m wing span FW configuration, whose aerodynamics coefficients are as in Ref.[31], and the structure parameters are listed in Table 1.

Table 1 Parameters for the studied FW aircraft

Parameter	Value	Coefficient	Value
m	3.7×10^5 , kg	T_{δ_a}	15,sec
S	841.7, m ²	T_{δ_e}	15,sec
$\bar{c}$	12.31, m	T_{δ_r}	15,sec
b	80, m	T_{δ_p}	1,sec
x_G	30.4, m	z_G	0, m
y_{Eng}	5m		
I_{xx}	4.703×10^7 , kg·m ²	I_{yy}	2.507×10^7 , kg·m ²
I_{zz}	9.973×10^7 , kg·m ²	I_{xz}	0.0, kg·m ²

The aircraft flies at 20000ft (6096m) altitude with the speed of 375kts (191.9m/s), the trim state

is $\mathbf{x}_e = [u \ w \ q \ \theta \ h \ x \ v \ p \ r \ \phi \ \psi \ y]^T = [191.92 \ 18.31 \ 1.31 \ 0.095 \ 6096 \ 0 \ 0 \ 0 \ 0 \ 0 \ 0]^T$, unit: $[u \ v \ w]$

(m/s), $[p \ q \ r]$ (rad/s), $[x \ y \ h]$ (m), $[\phi, \theta, \psi]$ (rad). $\mathbf{u}_e = [\delta_e \ \delta_p \ \delta_a \ \delta_r]^T = [-0.11 \ 0.33 \ 0 \ 0]^T$, unit: $[\delta_a \ \delta_e \ \delta_r]$ (rad).

The position range for the actuators are $\delta_e \in [-0.35, 0.26]$ (rad), $\delta_p \in [0, 1]$, $\delta_a \in [-0.65, 0.65]$ (rad), $\delta_r \in [-0.38, 0.38]$ (rad). The initial position is set to $\xi_0 = [0, 0, -6096\text{m}]^T$, with initial body velocity $V_0 = [192 \text{ m/s}, 0, 0]^T$, initial attitude $\eta_0 = [0, 5.5^\circ, 0]^T$, and initial angular velocity $\omega_0 = [0, 0, 0]^T$. For the command filter (38), the damping ratio is chosen as $\zeta=0.9$, and natural frequency is selected as $\omega_n = 10 \text{ rad/s}$ to avoid large maneuver inputs. For the state filter (36), the damping ratio is chosen as $\zeta=0.8$, and natural frequency is selected as $\omega_n = 25 \text{ rad/s}$.

Scenario I: trajectory tracking control with gust disturbances

In this scenario consider unknown gust acting on the FW aircraft, and the turbulence is a stochastic process defined by velocity spectra. The Dryden wind turbulence model is selected here and its longitudinal, lateral, and vertical component spectra functions are as follows by according to MIL-F-8785C standards,

$$\Phi_u(\omega) = \frac{2L_u\sigma_u^2}{\pi V} \frac{1}{1 + \left(\frac{L_u\omega}{V}\right)^2}, \Phi_v(\omega) = \frac{L_v\sigma_v^2}{\pi V} \frac{1 + 3\left(\frac{L_v\omega}{V}\right)^2}{\left(1 + \left(\frac{L_v\omega}{V}\right)^2\right)^2}, \Phi_w(\omega) = \frac{L_w\sigma_w^2}{\pi V} \frac{1 + 3\left(\frac{L_w\omega}{V}\right)^2}{\left(1 + \left(\frac{L_w\omega}{V}\right)^2\right)^2}, \quad (40)$$

and the turbulence angular rate spectrum are[35]

$$\Phi_{p_g}(\omega) = \frac{\sigma_w^2}{VL_w} \frac{0.8\left(\frac{L_w\pi}{4b}\right)^{1/3}}{1 + \left(\frac{4b\omega}{\pi V}\right)^2}, \Phi_{q_g}(\omega) = \frac{\pm\left(\frac{\omega}{V}\right)^2}{1 + \left(\frac{4b\omega}{\pi V}\right)^2} \cdot \Phi_w(\omega), \Phi_{r_g}(\omega) = \frac{\mp\left(\frac{\omega}{V}\right)^2}{1 + \left(\frac{3b\omega}{\pi V}\right)^2} \cdot \Phi_v(\omega), \quad (41)$$

where b represents the aircraft wingspan, L_u, L_v, L_w represent the turbulence scale lengths, and $\sigma_u, \sigma_v, \sigma_w$ denote the turbulence intensities along x, y and z axis respectively. Consider the vertical wind gust, it can be transformed into a state space model as

$$\begin{bmatrix} \dot{\hat{w}}_{g1} \\ \dot{\hat{w}}_{g2} \end{bmatrix} = \begin{bmatrix} 0 & 1 \\ -1/\bar{L}^2 & -2/\bar{L} \end{bmatrix} \begin{bmatrix} \hat{w}_{g1} \\ \hat{w}_{g2} \end{bmatrix} + \begin{bmatrix} 0 \\ 1 \end{bmatrix} w \quad (42)$$

$$w_g = w_\sigma \begin{bmatrix} 1/(\bar{L}\sqrt{3}) & 1 \end{bmatrix} \begin{bmatrix} \hat{w}_{g1} \\ \hat{w}_{g2} \end{bmatrix} \quad (43)$$

where $\hat{w}_{g1}, \hat{w}_{g2}$ are gust state variants, w_g denote gust speed, $w_\sigma = \sigma_w / \sqrt{6/\bar{L}}$, $\bar{L} = L_w / V$, w denotes white noise. The longitudinal and lateral gust speeds can also be estimated by the associated state transformation. Here the

wind speed σ is 6m/s with zero wind direction, wind scale length L is 533.4m, wing span is 40m, band limited noise sample time is set as 0.002sec, noise seeds of $[u_g, v_g, w_g, p_g] = [23341 \ 23342 \ 23343 \ 23344]$.

The force and moment resulted from the gust disturbances can be calculated by aerodynamics, and the disturbance vector D is achieved. To illustrate improved performance of the proposed Incremental Backstepping Sliding Mode Control (IBSMC), a Backstepping Sliding Mode Control (BSMC) and a BackStepping (BS) control are applied for comparison, where the backstepping control is designed as

$$\mathbf{u}_{BS} = G_4^{-1} \left(-c_4 \mathbf{z}_2 - \mathbf{f}_4 + \dot{\mathbf{x}}_{4d} - G_3^T \mathbf{z}_1 \right). \quad (44)$$

The controller parameters are designed as follows after several design iterations: for IBSMC, BSMC, and BS, the gains are

$$\begin{aligned} K_{\xi,p} &= \text{diag}(0.001, 0.4, 0.4), K_{\xi,I} = \text{diag}(7e-5, 0.02, 0.08), K_{\xi,D} = \text{diag}(0.0005, 2.5, 0.25) \text{ in the position control loop;} \\ K_{2,p} &= \text{diag}(1.5, 0.5, 2), K_{2,I} = \text{diag}(0.2, 0.05, 0.2), K_{2,D} = \text{diag}(0.3, 0.15, 0.4) \text{ in the flight path control loop;} \\ K_{3,p} &= \text{diag}(0.8, 24, 0.3), K_{3,I} = \text{diag}(0.04, 1.0, 0.05), K_{3,D} = \text{diag}(0.02, 0.5, 0.05) \text{ in the attitude control loop.} \end{aligned} \quad (45)$$

The sliding mode control parameters for IBSMC and BSMC are designed as

$$\mathbf{h} = \text{diag}(8, 4, 0.7), \mathbf{\zeta} = \text{diag}(2.5, 1, 1), \varphi_s = 0.4, \lambda_l = \text{diag}(0.5, 0.5, 0.5), \quad (46)$$

and the backstepping parameters are design as

$$c_3 = \text{diag}(0.1, 5, 3) \text{ for IBSMC and BSMC, } c_4 = \text{diag}(1, 2.5, 0.1) \text{ for backstepping control.} \quad (47)$$

A desired trajectory is predefined to verify the tracking performance of the IBSMC. The aircraft responses using BS, BSM and IBSM are shown in Figs 4-5.

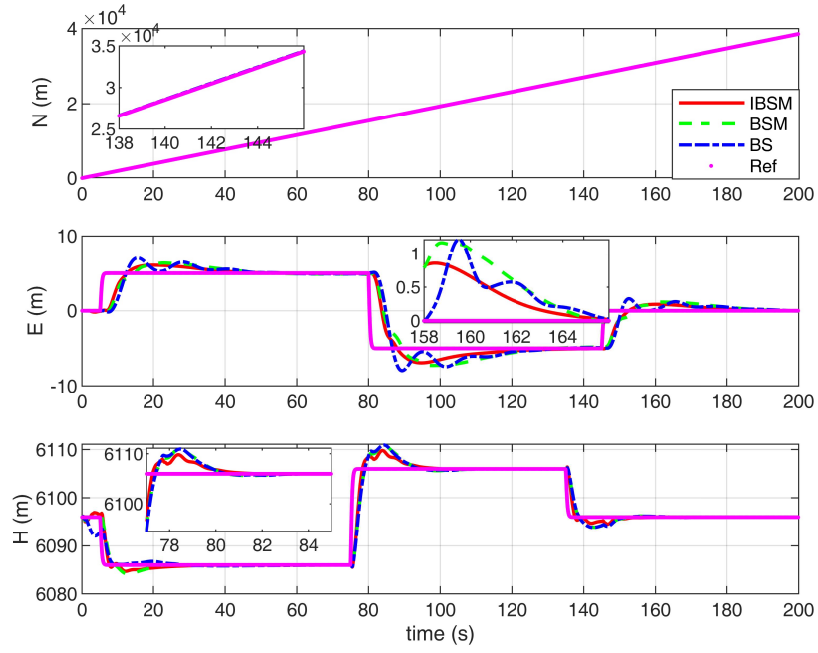


Fig. 4 Position tracking responses under wind gust disturbances.

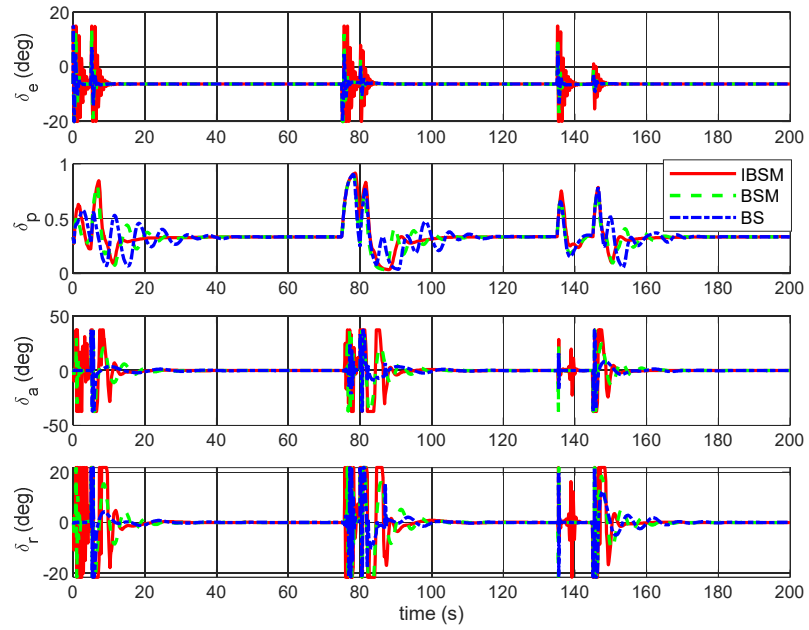


Fig. 5 Control input responses under wind gust disturbances.

From Fig.4 and Fig.5, it can be seen that IBSM has the best tracking accuracy, and the tracking performance using BS and BSM are notable deteriorated. Note also that the responses from the backstepping method have larger fluctuations than those by the IBSM and BSM design. Compared with BSM method, the position tracking responses have lower overshoots and smaller tracking errors from the IBSM, which shows more robustness of the IBSM design. However, the tracking error in the north direction from the IBSM is greater than that from the BSM design during the transition period, this is because that the trajectory tracking control in the north direction from the IBSM needs a larger thrust input.

To improve the trajectory tracking performance, the RBF neural network ESO of Eqs (29) ~ (33) are applied and combined with the incremental backstepping sliding mode control, a backstepping sliding mode control and a backstepping control, where NNESO based backstepping control is designed as

$$\mathbf{u}_{BS} = G_4^{-1} \left(-c_4 \mathbf{z}_2 - \mathbf{f}_4 + \dot{\mathbf{x}}_{4d} - \hat{D}_w - \hat{W}^T \mathbf{h} - G_3^T \mathbf{z}_1 \right). \quad (48)$$

The parameters of the RBF-NNs in the flight path control loop and the attitude control loop are set as

$$\gamma_w=0.1, c = [-1 \ 0 \ 1; -1 \ 0 \ 1; -1 \ 0 \ 1]; b_j = 0.01; h(0)=[0.01,0.01,0.01]^T. \quad (49)$$

The parameters of the ESO in the flight path control loop and the attitude control loop are set as

$$\begin{cases} \beta_{01} = 10, \beta_{02} = 3, \beta_{03} = 4, & \text{for } \mu, \alpha, \beta \text{ in the attitude loop} \\ \beta_{01} = 1, \beta_{02} = 0.6, \beta_{03} = 0.11, & \text{for } V, \gamma \text{ in the flight path} \end{cases}, \quad \delta_w=0.5, T_s = 0.01s, \gamma_D=1. \quad (50)$$

A desired trajectory is predefined to verify the tracking performance of the NNESO-IBSMC. The aircraft responses using NNESO-BS, NNESO-BSM and NNESO-IBSM are shown in Figs 6-7.

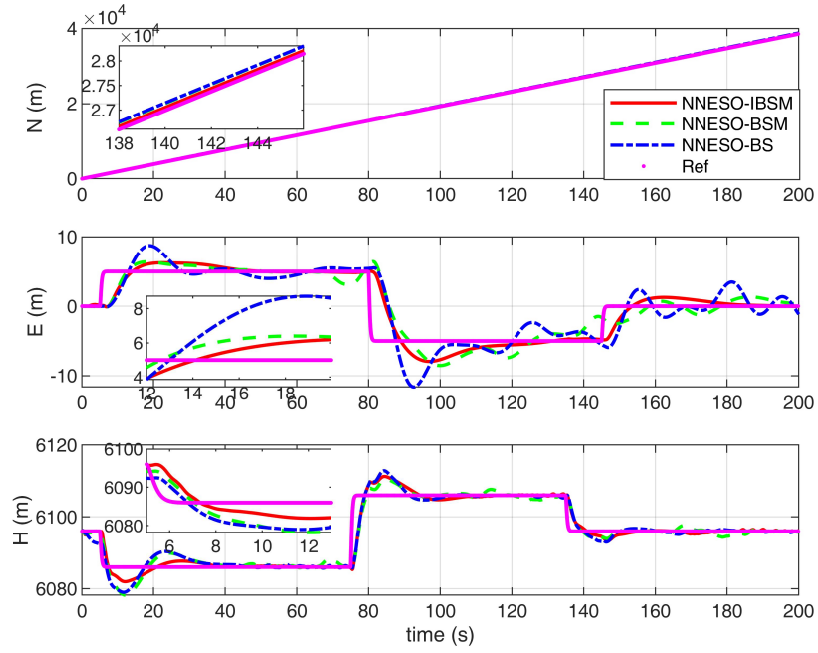


Fig. 6 Position tracking responses under wind gust disturbances.

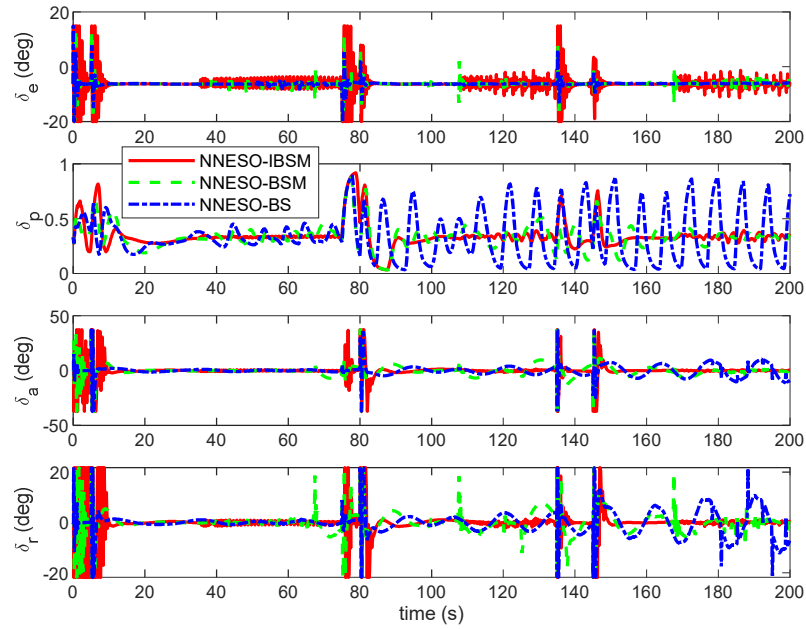


Fig. 7 Control input responses under wind gust disturbances.

From Fig.6 and Fig.7, it can be seen that IBSM has the best tracking accuracy. The responses by the NNESO-BS method have larger fluctuations than those by the NNESO-IBSM and the NNESO-BSM design. Compared with the NNESO-BSM method, the position tracking responses have lower overshoots and smaller tracking errors by the NNESO-IBSM, this demonstrates that the NNESO-IBSM has better robustness for the unknown wind gust disturbances. Compared with the IBSM, the control inputs of the NNESO-IBSM are quickly stable after oscillation and are less than those of the IBSM, which shows the NNESO can reduce its control conservation, see Fig.5 and Fig.7. And there is small oscillation of the elevator, which results from oscillation of pitch motion. The attitude disturbances and flight path disturbances with actuator faults are estimated by the NNESO of Eqs (29)~(33), which are shown in Fig.8~ Fig.11.

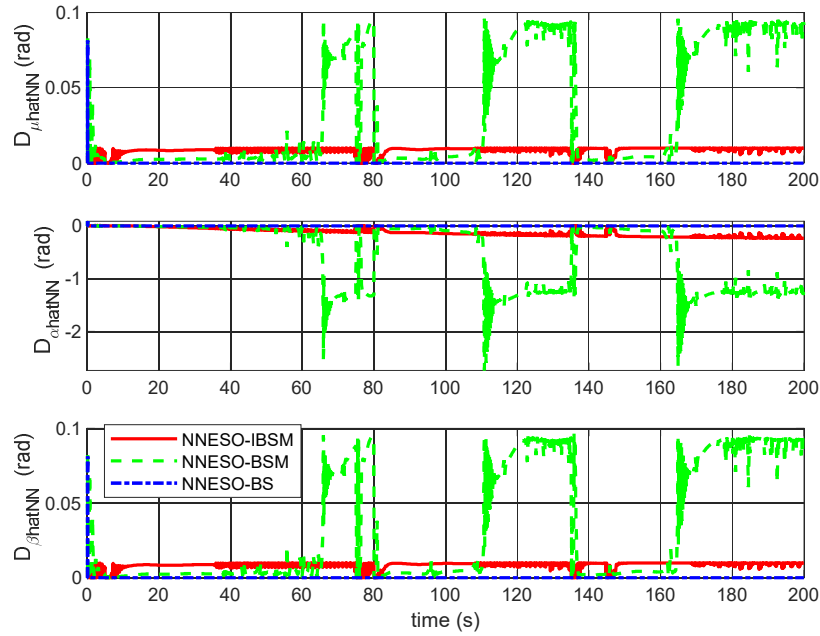


Fig. 8 Attitude disturbance estimation by NNs under Scenario I.

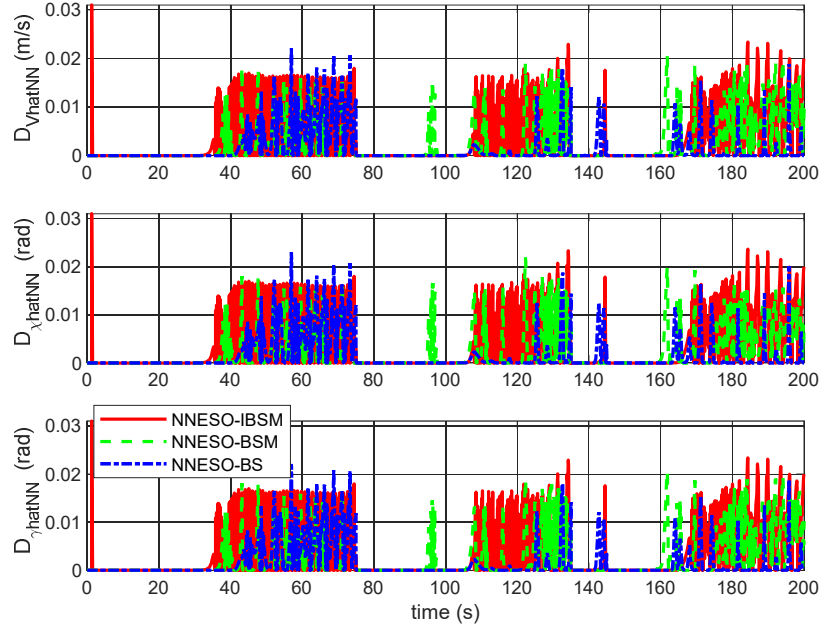


Fig. 9 Flight path disturbance estimation by NNs under Scenario I.

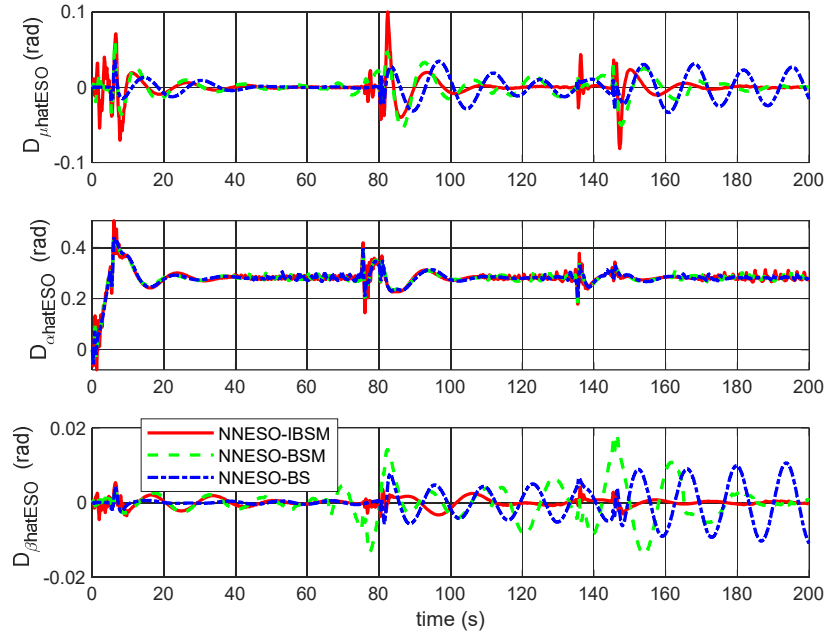


Fig. 10 Attitude disturbance estimation by the ESO under Scenario I.

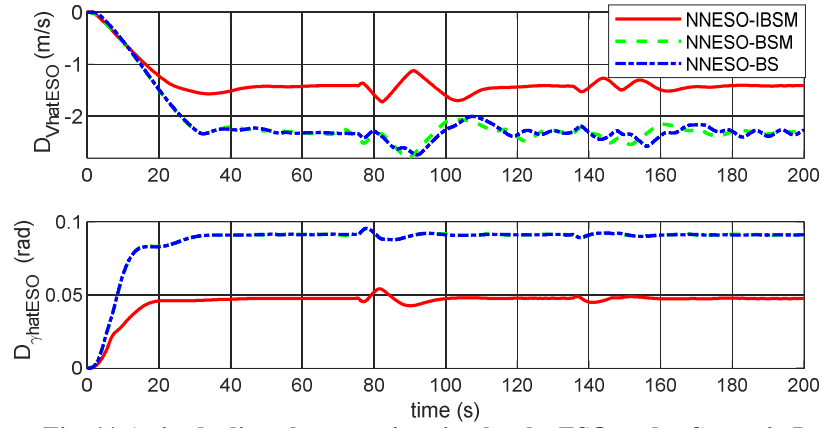


Fig. 11 Attitude disturbance estimation by the ESO under Scenario I.

From Fig.8 - Fig.11, it can be seen that the disturbance estimations of attitude disturbances and flight path disturbances are more stable by the NNESO-IBSM than those by the NNESO-BSM and the NNESO-BS. The disturbance estimations of flight velocity by the ESO are negative, see Fig.11, which means the disturbance velocity direction is reversible on the flight speed.

To implement a big change motion, there are two methods. One is to use a step input or a square wave input with big amplitude. For example, altitude command is from 6066m to 6126m, y command in the east direction is from 5m to -5m, and x command is from 2m to -2m, the tracking responses are shown as follows,

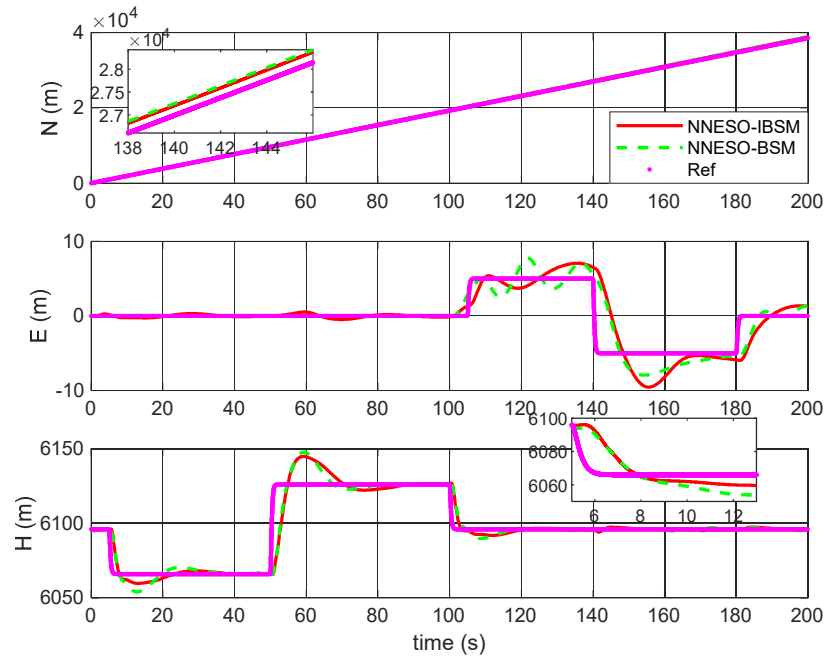


Fig. 12 Position tracking responses under big command inputs

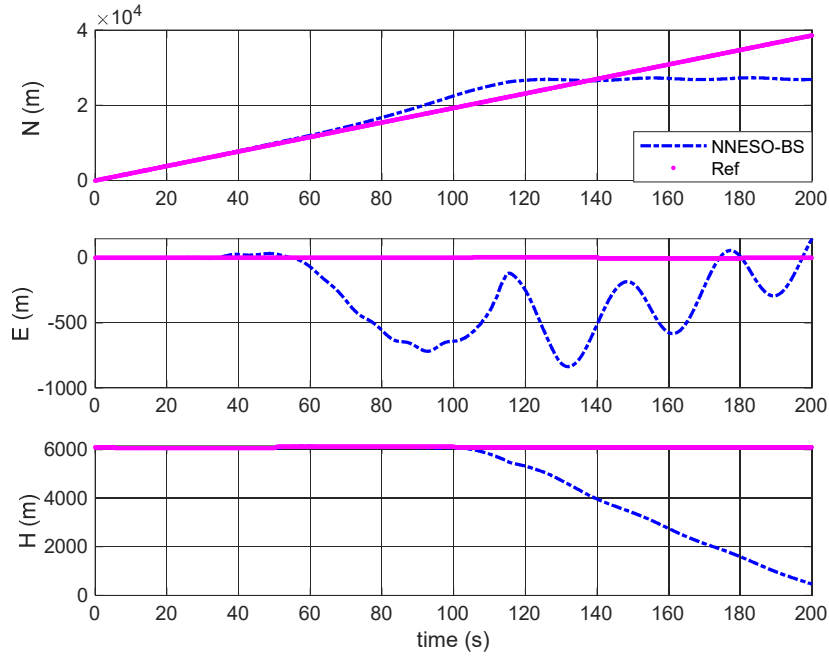


Fig. 13 Position tracking responses under big command inputs by BS control

The other way is to apply the ramp command. For flight velocity tracking, we can set a ramp command changing from 191.9m/s to 218.49m/s, and then from 218.49m/s to 167.08m/s, i.e., from 425 kts to 325 kts, and the tracking response is shown in Fig.14.

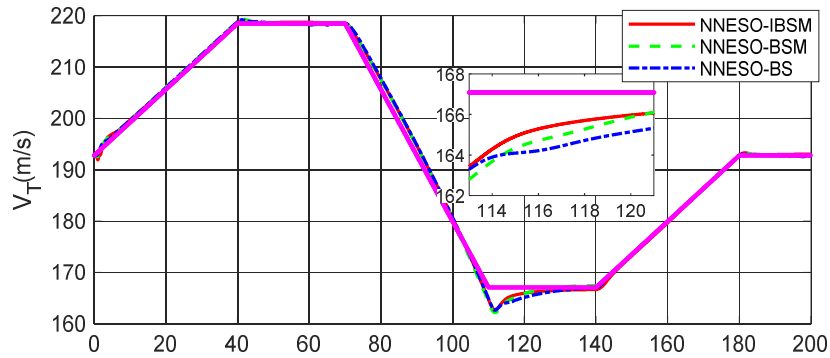


Fig. 14 Flight velocity tracking responses under ramp command with wind gust disturbances

From Figs12-14, it can be seen that IBSM has the best tracking accuracy in the square wave commands of positions and the ramp command tracking of flight velocity. Compared with the NNESO-BSM and the NNESO-BS method, the position tracking responses have lower overshoots and lower settling time by the NNESO-IBSM. The

position tracking responses by the NNESO-BS deviate from the desired positions when the big command are input, the results are resulted from the saturation of the actuators.

Scenario II: trajectory tracking control in the presence of wind gust disturbances and actuator faults

This scenario considers actuator faults due to damages and loss of effectiveness. The actuator faults are set as follows

$$\delta_i = (1 - \rho_{f_i}) \times \delta_i^0 + \delta_{\text{offset}} \quad (i=e, a, r), \quad (51)$$

where ρ_{f_i} is the loss effectiveness factor of each actuator meeting $\rho_{f_i} = 0.1$, δ_{offset} denotes the offset fault of the actuator meeting $\delta_{\text{offset}} = 0.1 \text{ rad}$, and the faults of the elevator, aileron and rudder occur at times of 50s, 80s, and 120s and remain until the end. The position references and wind gust disturbances are the same as scenario I. The parameters of the controllers are as in Eqs.(45)-(47), (49) and (50). The aircraft responses in presence of actuator faults and wind gust disturbances are shown in Figs 15 and Table2.

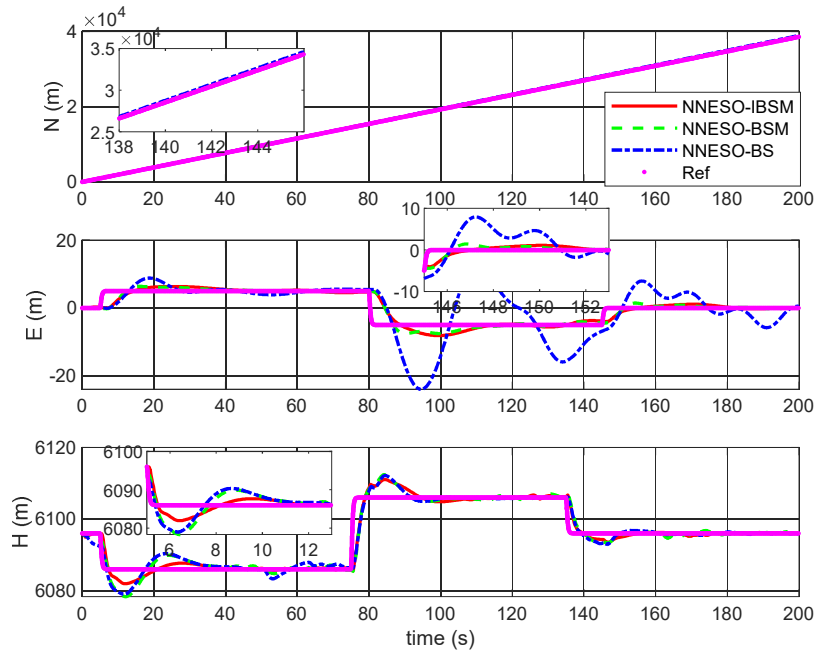
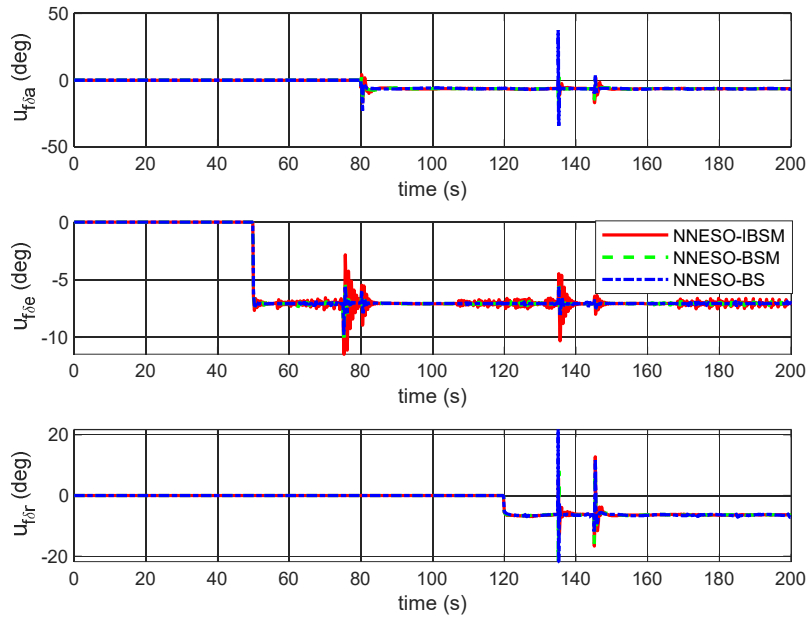


Fig. 15 Position tracking responses under actuator faults and wind disturbances.

Table 2 Performance comparative data under actuator faults and wind gusts

Controller	Variable	Settling time, s	Overshoots, m	RMSE, m
NNESO-BS	x	76.5	0.9231	201.0149
	y	64.1	-100	6.3772
	z	39.4	0.1010	2.5637
NNESO-BSM	x	56.3	0.4942	123.5856
	y	35.5	-100	1.7528
	z	39.1	0.1010	2.6295
NNESO-IBSM	x	53.0	0.2122	61.4143
	y	30.1	-100	1.7265
	z	35.1	0.0823	2.1663

Fault inputs of the actuators and the attitude response are shown in Figs 16-17,

**Fig. 16 Fault inputs of the actuators.**

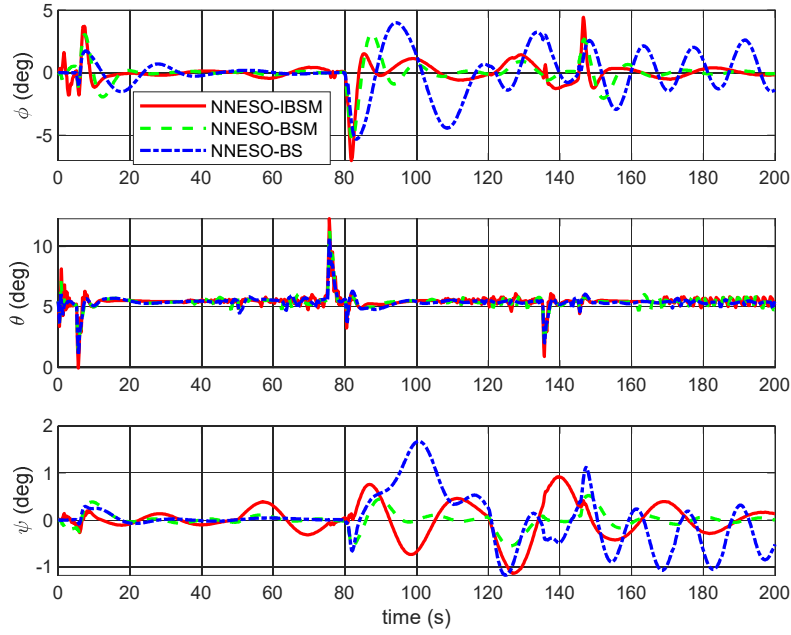


Fig. 17 Attitude tracking responses under actuator faults and wind gust disturbances.

From Figs 16-17, it can be seen that when the faults of the elevator, aileron and rudder occur at times of 50s, 80s, and 120s, the attitude tracking responses initially fluctuate, however, the closed-loop tracking systems are eventually converge, although the faults remain until the end. Among these responses, see Figs 15 and Table 2, the NNESO-IBSM has the root mean square error (RMSE) $E = [61.4143, 1.7265, 2.1663]$ (m), while the tracking errors using the NNESO-BS are largest although they are ultimately bounded, NNESO-BS control itself shows poor robust performance in this scenario. Meanwhile, the settling time of the tracking responses are smallest by using NNESO-IBSMC. Compared with NNESO-BSM method, the position tracking responses have lower overshoots from the NNESO-IBSM, which shows better robust performance than the NNESO-BSM design.

Scenario III: trajectory tracking under actuator faults with model uncertainties and gust disturbances

This scenario considers parameter uncertainties such as aerodynamic derivatives changing and so the robustness of the controllers to uncertainties is evaluated. The inertia parameters I_{xx} , I_{yy} , I_{zz} , I_{xz} , and the aerodynamic coefficients $C_x(V, \alpha, \beta, \delta_{act})$, $C_y(V, \alpha, \beta, \delta_{act})$, $C_z(V, \alpha, \beta, \delta_{act})$, $C_l(V, \alpha, \beta, \delta_{act})$, $C_m(V, \alpha, \beta, \delta_{act})$, $C_n(V, \alpha, \beta, \delta_{act})$ are multiplied with random combinations of scaling factors, which are in the range of [0.8, 1.2]. The parameter uncertainty actuator faults and gust disturbances are considered simultaneously. Actuator faults and wind gust disturbances are also

considered as for Scenario II. The parameters of the controllers are as in Eqs.(45)-(47), (49) and (50). By using the proposed NNESO-IBSM design, the simulation results are shown in Figs 18-19 and Table 3.

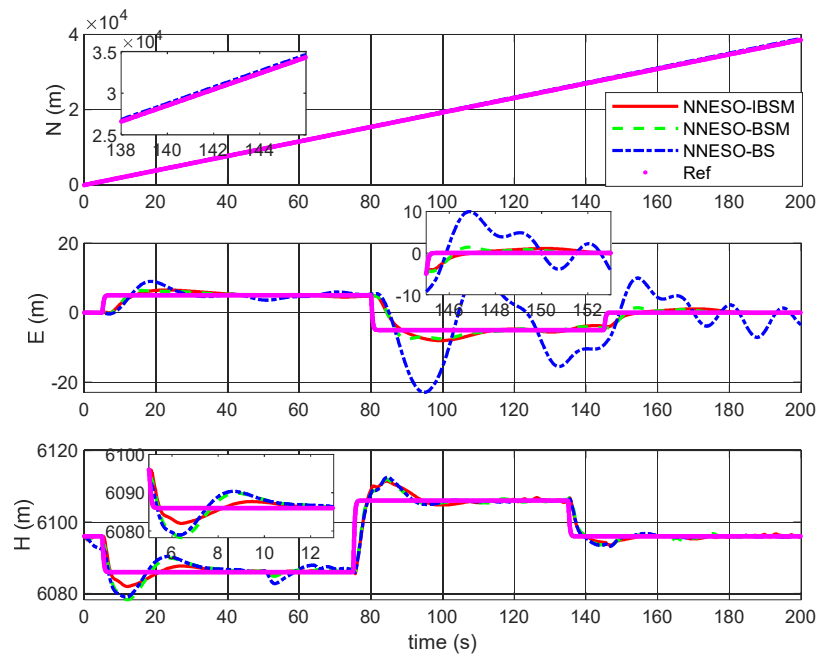


Fig. 18 Position tracking responses under faults and model uncertainties and gusts.

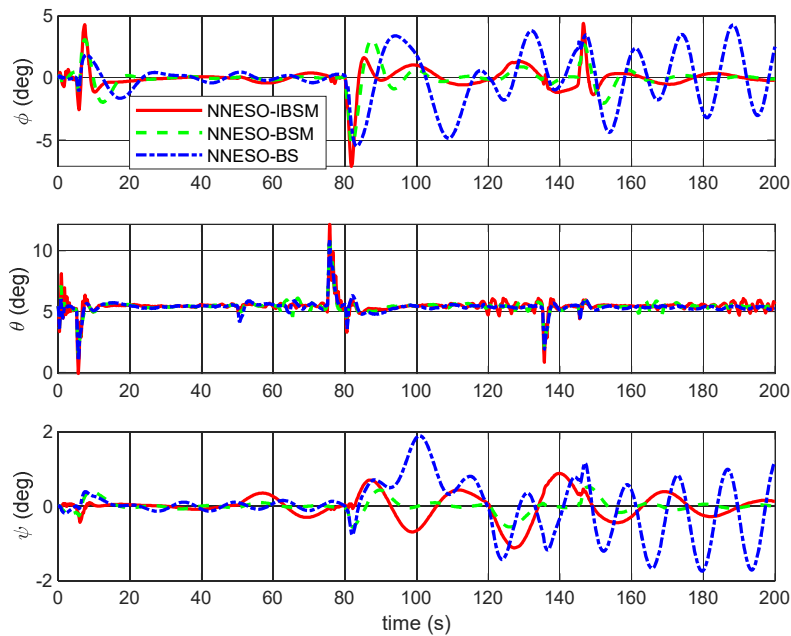


Fig. 19 Attitude tracking responses under faults and model uncertainties and gusts.

Table 3 Performance comparative data under faults and model uncertainties and gusts

Controller	Variable	Settling time, s	Overshoots,m	RMSE,m
NNESO-BS	x	77.98	1.0428	213.2881
	y	27.72	-100	6.4320
	z	39.70	0.1067	2.6355
NNESO-BSM	x	51.42	0.4892	122.5393
	y	26.80	-100	1.7522
	z	38.50	0.0895	2.5482
NNESO-IBSM	x	50.76	0.2121	61.3793
	y	23.34	-100	1.7277
	z	34.90	0.0895	2.2690

From Figs 18 and 19, it can be seen that the output tracking responses begin oscillating after the actuator faults at 50s, 80s, and 120s. The responses of NNEO-INBSM have quicker convergence in attitude motions than the responses of the NNEO-BS control, this demonstrates the NNEO-INBSM design has stronger FTC capability. The RMSE for the NNEO-IBSM and NNEO-BSM designs are smaller than those of NNEO-BS method, which shows poor robust performance of the backstepping control. Compared with the NNEO-IBSM, the tracking responses of the NNEO-INBSM control have smaller overshoots, lower settling time and tracking errors, see Table 3, which show that the NNEO-INBSM design has better robust capability to cope with actuator faults, model uncertainties and gust disturbances. The actuator faults, parameter uncertainties and gust disturbances have been processed by the NNEO-INBSM due to the good approximation capability of the RBF neural network and robustness of the IBSM control. The associated responses of V_T , α , and β and angular rate responses are shown in Figs 20 and 21.

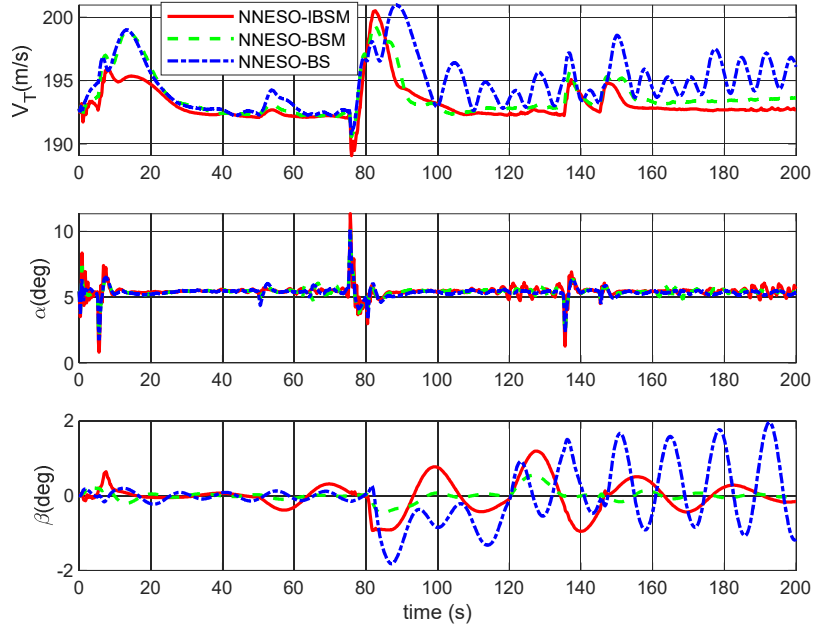


Fig. 20 Output responses of V_T , α , and β under Scenario III.

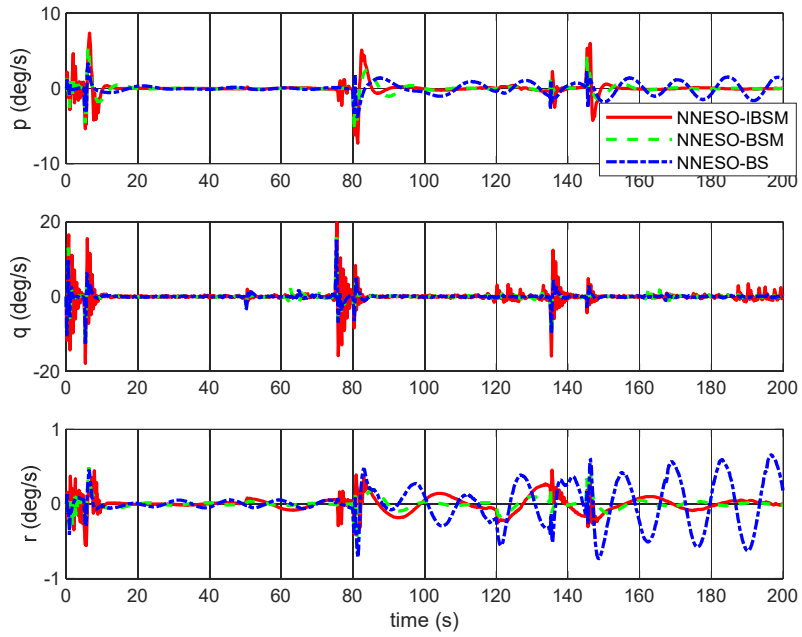


Fig. 21 Angular rate tracking responses under Scenario III.

From Fig.20 and Fig.21, it can be seen that the responses by using NNESO-BS have more oscillation in the presence of actuator faults and model uncertainties. Compared with NNESO-IBSM, the responses of angular rates by using the NNESO-BSM and the NNESO-BS have more fluctuation than by using the NNESO-IBSM, see Fig.21. The corresponding control inputs are shown in Fig.22.

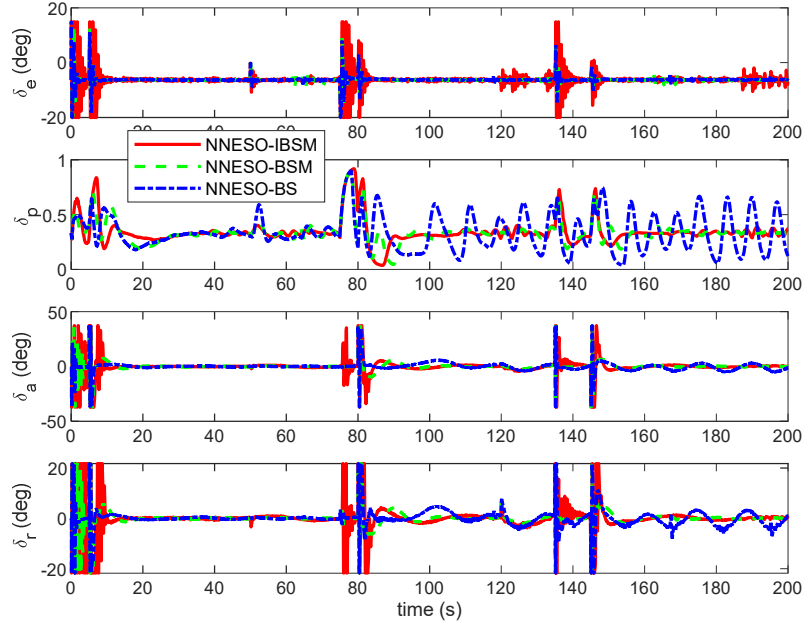


Fig. 22 Control inputs under Scenario III.

The thrust and rudder inputs from the NNESO-BS have large fluctuation, which show the NNESO-BS needs more input and has a poorer robust performance. Compared with the NNESO-BSM, the NNESO-IBSM control is smooth and quickly convergent, which shows better performance for the NNESO-IBSM. The disturbances have been observed via the NNESO observers as seen in Figs 23~ 24.

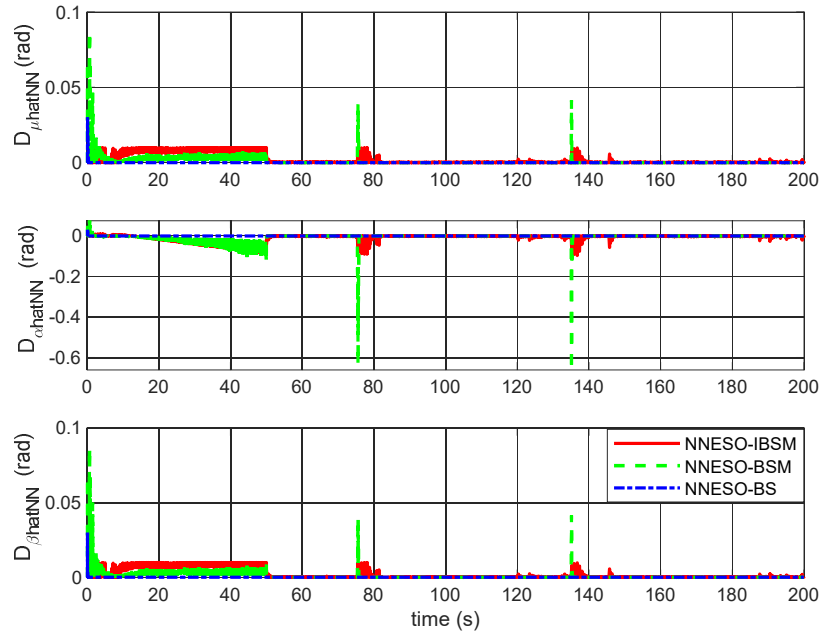


Fig. 23 Attitude disturbance, model uncertainty and fault estimation by NNs under Scenario III.

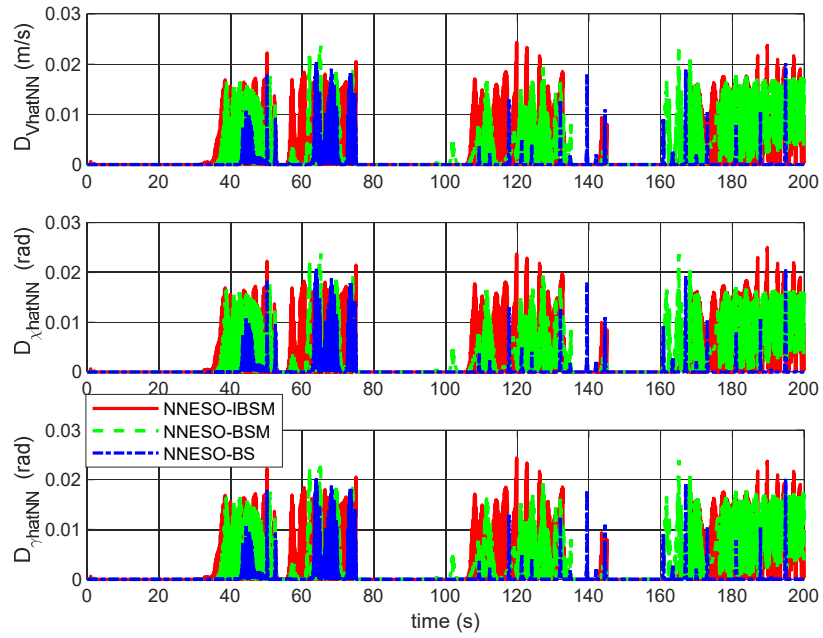


Fig. 24 Flight path disturbance, model uncertainty and fault estimation by NNs under Scenario III.

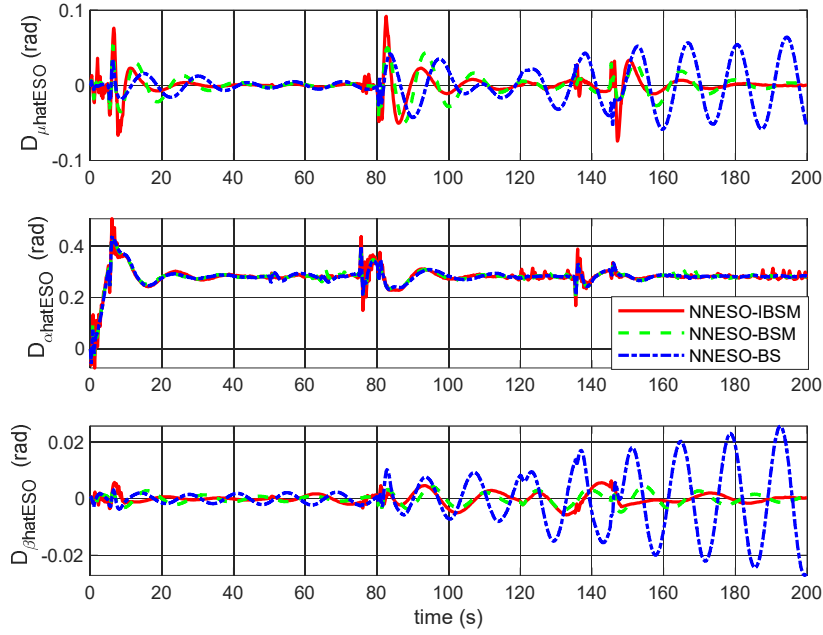


Fig. 25 Attitude disturbance, model uncertainty and fault estimation by the ESO under Scenario III.

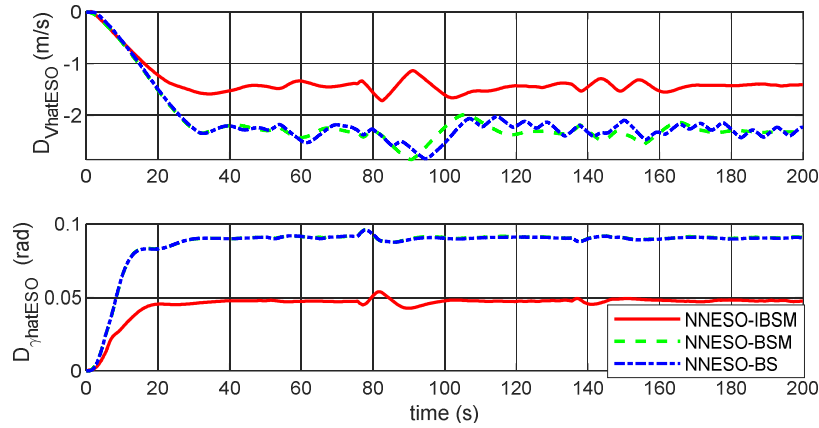


Fig. 26 Flight path disturbance, model uncertainty and fault estimation by the ESO under Scenario III.

Figs 23 - 26 show that gust disturbance, fault estimation and model uncertainties are estimated by the NNESO, which are consistent with Figs.8-11. Fig.25 shows the μ and β disturbance estimations by the ESO-BS has big oscillation and become divergent, which illustrates the NNESO-BS has worst performances under Scenario III. Additionally, by using the NNESO, the burden of the IBISM control input are greatly reduced, see Fig.5 and Fig.22.

Consider some special case, for example, the sideslip angle is not equal zero, set the input $\beta = 5\text{deg}$, and then Eq.(12) is replaced by Eq.(11). By using the proposed control, the simulation results are shown as follows

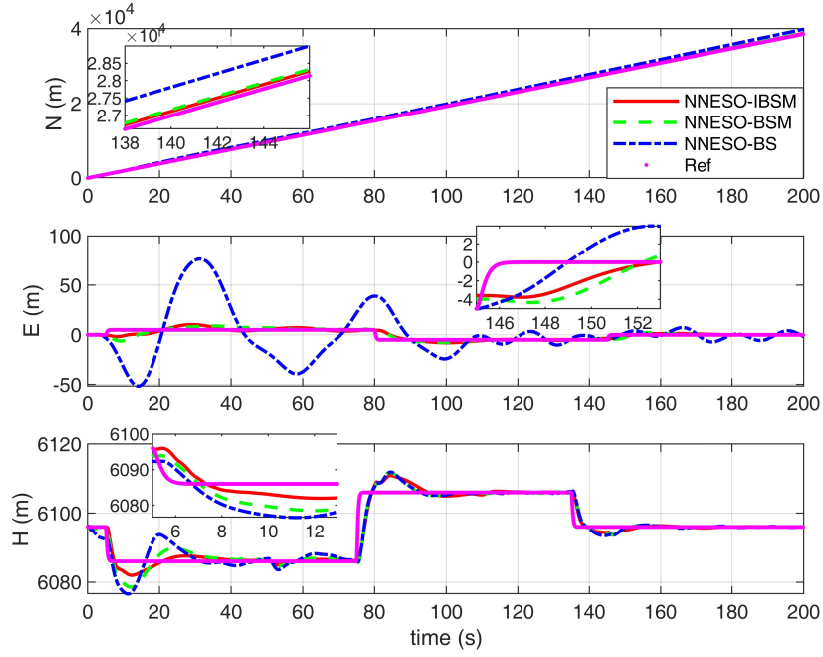


Fig. 27 Position tracking responses under $\beta = 5\text{deg}$.

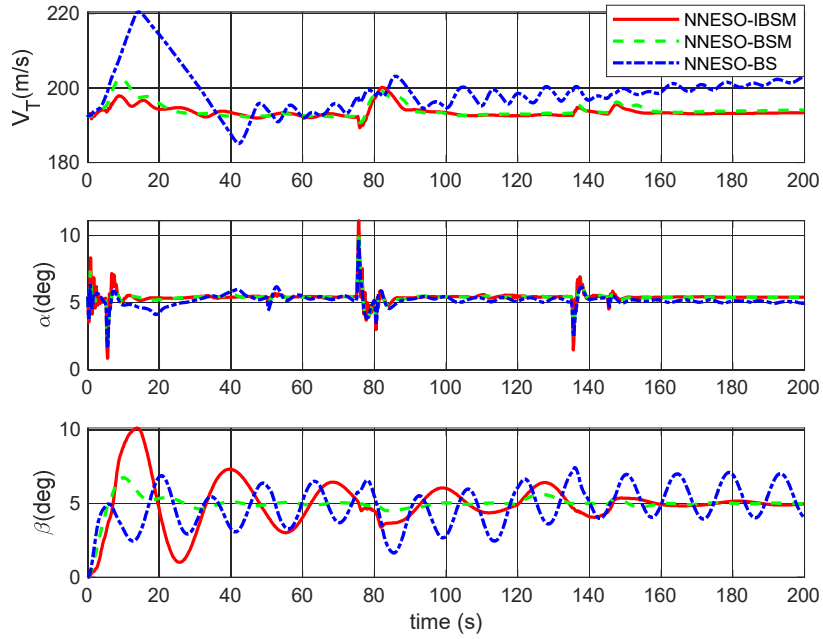


Fig. 28 Output responses of V_T , α , and β under $\beta = 5\text{deg}$

Figs 27 - 28 show that IBSM has the best tracking accuracy for position tracking control, and the responses of sideslip angle converge to the desired value. The responses of the NNESO-BS control have big overshoots with long

time oscillation, this demonstrates the NNESO-BS design has weak capability when the sideslip angle input is nonzero.

For safe pilot, the pitch angle are confined in $[-\pi/6, \pi/6]$ and the roll angle are confined in $[-\pi/3, \pi/3]$, and the associated flight path angle responses are shown as follows

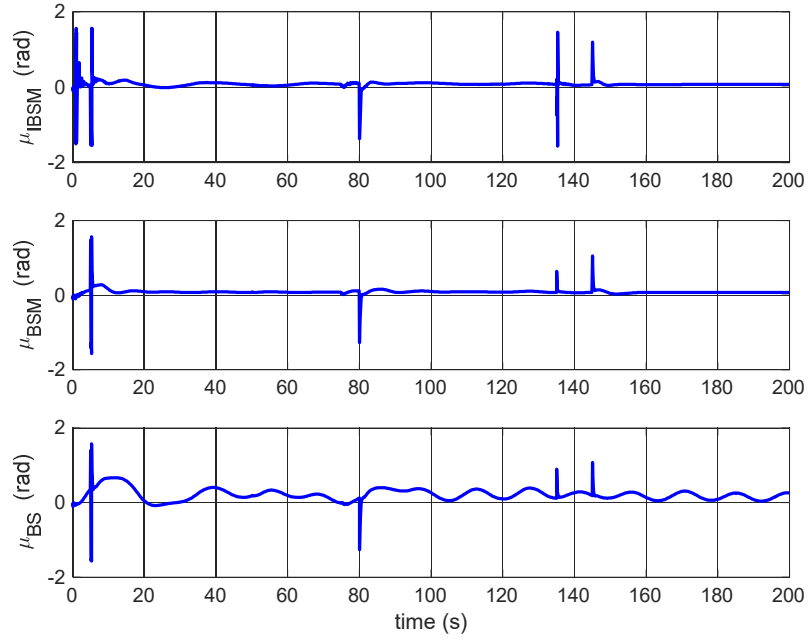


Fig. 29 Output responses of flight path angle μ under $\beta = 5\text{deg}$

Fig.29 shows the flight path angle μ is free without constraints.

V. Conclusion

A NNESO-INBSM approach is proposed for the control of the FW aircraft. The precise trajectory tracking controller is designed by three cascaded loops of attitude, flight path and position control. The developed controller not only stabilizes the attitude and angular rate in presence of gusty winds and model parameter uncertainties, but also accurately tracks the desired trajectory for the FW aircraft with actuator faults. Furthermore, a Neural Network based Extended State Observer (NNESO) is designed to estimate external disturbances, model uncertainties and unknown faults. Stability analysis shows that the closed-loop system is stable by Lyapunov theory. Three scenarios of unknown gust wind disturbances, model uncertainties and actuator faults have been simulated and show the proposed NNESO-IBSM control robustness. Compared with BS and BSM controllers with NNESO, the NNESO-

IBSM control achieves better robust tracking performance despite gust environment, parametric uncertainties, and actuator faults. Therefore, the effectiveness and capability of the NNESO-IBSM design are demonstrated.

Appendix: Detailed Design of the Attitude Controller

The attitude controller is derived as follows.

Step 1 Construct a candidate Lyapunov function V_1 as

$$V_1 = \frac{1}{2} \mathbf{z}_1^T \mathbf{z}_1. \quad (52)$$

By Eq.(24), we get

$$\dot{\mathbf{z}}_1 = \dot{\mathbf{x}}_3 - \dot{\mathbf{x}}_{3d} = \mathbf{f}_3 + \mathbf{G}_3 \mathbf{x}_4 - \dot{\mathbf{x}}_{3d} \quad (53)$$

To make the derivative function $\partial V_1 / \partial \mathbf{z}_1$ negative definite, a reference for $\mathbf{x}_4$ can be designed as

$$\mathbf{x}_{4,ref} = \mathbf{G}_3^{-1} (-\mathbf{c}_3 \mathbf{z}_1 + \dot{\mathbf{x}}_{3d} - \mathbf{f}_3) \quad (54)$$

and then

$$\dot{V}_1 = \mathbf{z}_1^T (\dot{\mathbf{x}}_3 - \dot{\mathbf{x}}_{3d}) = \mathbf{z}_1^T (\mathbf{f}_3(\mathbf{x}_3) - \mathbf{c}_3 \mathbf{z}_1 + \dot{\mathbf{x}}_{3d} - \mathbf{f}_3(\mathbf{x}_3) - \dot{\mathbf{x}}_{3d}) = -\mathbf{z}_1^T \mathbf{c}_3 \mathbf{z}_1 < 0 \quad (55)$$

where $\mathbf{c}_3$ is a diagonal matrix to be determined and meeting $c_3 > 0$, i.e., $\mathbf{c}_3 = \text{diag}(k_{31}, k_{32}, k_{33})$, and k_{3i} ($i = 1, 2, 3$) are positive constant.

Furthermore, define the angular rate tracking error as $\mathbf{z}_2 = \mathbf{x}_4 - \mathbf{x}_{4ct}$ ($\mathbf{z}_2 = \mathbf{x}_4 - \mathbf{x}_{4d}$), and then by Eq.(26) the resulting dynamics are

$$\dot{\mathbf{z}}_2 = \dot{\mathbf{x}}_4 - \dot{\mathbf{x}}_{4d} = \mathbf{f}_4(\mathbf{x}_3, \mathbf{x}_4) + \mathbf{G}_4(\mathbf{x}_3, \mathbf{x}_4) \mathbf{u} + \mathbf{D}_\omega - \dot{\mathbf{x}}_{4d}. \quad (56)$$

where $\mathbf{x}_{4ct}$ is angular rate output of the ESO. The incremental dynamics of the angular rate motion are derived by using a first-order Taylor series expansion around the condition at $t - \Delta t$ (denoted by the subscript 0) as

$$\dot{\mathbf{x}}_4 = \dot{\mathbf{x}}_{4,0} + \left. \frac{\partial \mathbf{f}_4(\mathbf{x}_3, \mathbf{x}_4)}{\partial \mathbf{x}} \right|_0 \Delta \mathbf{x} + \left. \frac{\partial \mathbf{G}_4(\mathbf{x}_3, \mathbf{x}_4)}{\partial \mathbf{u}} \right|_0 \Delta \mathbf{u} + \Delta \mathbf{D}_\omega + \mathcal{O}(\mathbf{x}, \mathbf{u}) \quad (57)$$

where $\mathcal{O}(\mathbf{x}, \mathbf{u})$ are the high-order terms of the expansion remainder.

Step 2

To improve system adaptiveness an SMC control is adopted and its sliding surface s is described as

$$s = \lambda_1 \mathbf{z}_1 + \mathbf{z}_2, \quad (58)$$

where λ_1 is the control gains to be determined. Consider attraction and invariance of SMC, it meets the condition for the state to reach the sliding mode surface s in finite time t_r and to remain on it, i.e.,

$$s\dot{s} < 0, \quad t < t_r \quad (59)$$

$$s = \dot{s} = 0, \quad t \geq t_r \quad (60)$$

And the reachability condition is designed by the sliding mode surface (58) as

$$\dot{s} = -\hbar s - \varsigma \operatorname{sgn}(s), \quad (61)$$

so we get

$$s\dot{s} = -\hbar s^2 s - \varsigma |s| \quad (62)$$

Next suppose that the disturbance effect can be approximated by the neural network. Construct the second candidate Lyapunov function as

$$V_2 = V_1 + \frac{1}{2} s^T s + \frac{1}{2} (1/\gamma_D) \tilde{D}_\omega^T \tilde{D}_\omega + \frac{1}{2} (1/\gamma_W) \tilde{W}^T \tilde{W}, \quad (63)$$

where $\tilde{W}$ is estimation error of the weight function W meeting $\tilde{W} = W^* - \hat{W}$, $\tilde{D}_\omega$ is estimation error of the slow attitude disturbance D_ω meeting $\tilde{D}_\omega = D_\omega - \hat{D}_\omega$ and $\dot{\tilde{D}_\omega} \approx 0$, γ_W and γ_D are positive constants that determines the convergence speed of the NN approximation and D_ω estimation. By using (53), we get

$$\dot{z}_1 = f_3 + G_3 x_{4d} - \dot{x}_{3d} + G_3 z_2 = -c_3 z_1 + G_3 z_2 \quad (64)$$

Then differentiating (63) yields

$$\begin{aligned} \dot{V}_2 &= z_1^T \dot{z}_1 + s^T \dot{s} + (1/\gamma_D) \tilde{D}_\omega^T \dot{\tilde{D}_\omega} + (1/\gamma_W) \tilde{W}^T \dot{\tilde{W}} \\ &= -z_1^T c_3 z_1 + z_1^T G_3 z_2 + s^T (\lambda_1 \dot{z}_1 + \dot{z}_2) + (1/\gamma_D) \tilde{D}_\omega^T \dot{\tilde{D}_\omega} + (1/\gamma_W) \tilde{W}^T \dot{\tilde{W}}. \end{aligned} \quad (65)$$

Substituting (23), (35), (55), and (56) into (65) yields

$$\begin{aligned} \dot{V}_2 &= -z_1^T c_3 z_1 + z_1^T G_3 z_2 + s^T (\lambda_1 (-c_3 z_1 + G_3 z_2) + (f_4 + G_4 u + D_\omega - \dot{x}_{4d})) + (1/\gamma_D) \tilde{D}_\omega^T (\dot{D}_\omega - \dot{\hat{D}_\omega}) - (1/\gamma_W) \tilde{W}^T \dot{\tilde{W}} \\ &= -z_1^T c_3 z_1 + z_1^T G_3 z_2 + s^T (\lambda_1 (-c_3 z_1 + G_3 z_2) + f_4 + G_4 u + D_\omega + W^{*T} h + \varepsilon - \dot{x}_{4d}) - (1/\gamma_D) \tilde{D}_\omega^T \dot{\tilde{D}_\omega} - (1/\gamma_W) \tilde{W}^T \dot{\tilde{W}} \end{aligned} \quad (66)$$

By using the sliding surface (58) and reaching law (61), a desired control input (in non-incremental form) is designed as follows to make (66) semi-negative definite,

$$u_{BSM} = G_4^{-1} \left(-\lambda_1 (-c_3 z_1 + G_3 z_2) - f_4 + \dot{x}_{4d} - (\hat{D}_\omega + \hat{W}^T h) - G_3^T z_1 - (\hbar s + \varsigma \operatorname{sgn}(s)) \right), \quad (67)$$

Since

$$\mathbf{s}^T \tilde{D}_\omega = \tilde{D}_\omega^T \mathbf{s}, \quad (68)$$

Substituting (67) and (68) into (66) yields

$$\begin{aligned} \dot{V}_2 &= -\mathbf{z}_1^T c_3 \mathbf{z}_1 + \mathbf{z}_1^T G_3 \mathbf{z}_2 - \mathbf{s}^T G_3^T \mathbf{z}_1 - \mathbf{s}^T (\hbar \mathbf{s} + \varsigma \operatorname{sgn}(\mathbf{s})) + \mathbf{s}^T (\tilde{W}^T \mathbf{h} + \varepsilon) + \mathbf{s}^T \tilde{D}_\omega - (1/\gamma_D) \tilde{D}_\omega^T \dot{\tilde{D}}_\omega - (1/\gamma_W) \tilde{W}^T \dot{\tilde{W}} \\ &= -\mathbf{z}_1^T c_3 \mathbf{z}_1 + \mathbf{z}_1^T G_3 \mathbf{z}_2 - (\boldsymbol{\lambda}_1 \mathbf{z}_1 + \mathbf{z}_2)^T G_3^T \mathbf{z}_1 - \hbar \mathbf{s}^T \mathbf{s} - \varsigma \|\mathbf{s}\| + \mathbf{s}^T (\tilde{W}^T \mathbf{h} + \varepsilon) + \tilde{D}_\omega^T \left(\mathbf{s} - (1/\gamma_D) \dot{\tilde{D}}_\omega \right) - (1/\gamma_W) \tilde{W}^T \dot{\tilde{W}} \\ &= -\mathbf{z}_1^T (c_3 + \boldsymbol{\lambda}_1 G_3^T) \mathbf{z}_1 - \hbar \mathbf{s}^T \mathbf{s} - \varsigma \|\mathbf{s}\| + \mathbf{s}^T (\tilde{W}^T \mathbf{h} + \varepsilon) + \tilde{D}_\omega^T \left(\mathbf{s} - (1/\gamma_D) \dot{\tilde{D}}_\omega \right) - (1/\gamma_W) \tilde{W}^T \dot{\tilde{W}} \end{aligned} \quad (69)$$

Choose the update law of the NN weight function as

$$\dot{\tilde{W}} = \gamma_W \mathbf{s}^T \mathbf{h}(x) \quad (70)$$

By (30), if the ESO makes disturbance estimation approximate $\gamma_D \mathbf{s}$, that is,

$$\dot{\tilde{D}}_\omega = \dot{\tilde{D}}_\omega = -\beta_{03} \operatorname{fal}(\varepsilon_3, 0.25, \delta_w) \rightarrow \gamma_D \mathbf{s}, \text{ or } \left\| -\beta_{03} \operatorname{fal}(\varepsilon_3, 0.25, \delta_w) - \gamma_D \mathbf{s} \right\| \leq \varepsilon_D \|\tilde{D}_\omega\| \quad (71)$$

where ε_D is a positive constant, then substituting (70) and (71) into (69) yields

$$\dot{V}_2 \leq -\mathbf{z}_1^T (c_3 + \boldsymbol{\lambda}_1 G_3^T) \mathbf{z}_1 - \hbar \mathbf{s}^T \mathbf{s} - \varsigma \|\mathbf{s}\| + \mathbf{s}^T \varepsilon + \bar{c} \quad (72)$$

where $\bar{c} = \varepsilon_D \tilde{D}_\omega^T \tilde{D}_\omega$. Define Q as a positive definite symmetric matrix

$$Q = \begin{bmatrix} c_3 + \boldsymbol{\lambda}_1 G_3^T + \hbar \boldsymbol{\lambda}_1^2 & \boldsymbol{\lambda}_1 \hbar \\ \boldsymbol{\lambda}_1 \hbar & \hbar \mathbf{I}_n \end{bmatrix} \quad (73)$$

where $\mathbf{I}_n$ denotes the $n \times n$ unit matrix. Let

$$\mathbf{Z}_{12} = \begin{bmatrix} \mathbf{z}_1^T & \mathbf{z}_2^T \end{bmatrix}^T, \quad (74)$$

this then yields

$$\begin{aligned} \mathbf{Z}_{12}^T Q \mathbf{Z}_{12} &= \boldsymbol{\lambda}_1 \mathbf{z}_1^T G_3^T \mathbf{z}_1 + \hbar \boldsymbol{\lambda}_1^2 \mathbf{z}_1^T \mathbf{z}_1 + 2 \boldsymbol{\lambda}_1 \hbar \mathbf{z}_1^T \mathbf{z}_2 + \hbar \mathbf{z}_2^T \mathbf{z}_2 \\ &= \boldsymbol{\lambda}_1 \mathbf{z}_1^T G_3^T \mathbf{z}_1 + \hbar (\boldsymbol{\lambda}_1 \mathbf{z}_1 + \mathbf{z}_2)^T (\boldsymbol{\lambda}_1 \mathbf{z}_1 + \mathbf{z}_2) \\ &= \mathbf{z}_1^T (c_3 + \boldsymbol{\lambda}_1 G_3^T) \mathbf{z}_1 + \hbar \mathbf{s}^T \mathbf{s}. \end{aligned} \quad (75)$$

Substituting (75) into (72) yields

$$\dot{V}_2 \leq -\mathbf{Z}_{12}^T Q \mathbf{Z}_{12} - \varsigma \|\mathbf{s}\| + \mathbf{s}^T \varepsilon + \bar{c}. \quad (76)$$

Let $c_k = \min\{2|Q|, 2\varepsilon_D\}$, and if $\varsigma > \bar{\varepsilon} > 0$, then

$$\dot{V}_2 \leq -c_k V_2 + \bar{c} \quad (77)$$

According to LaSalle-Yoshizawa Lemma [36], the closed-loop system tracking error will exponentially converge, and

$$V_2 \leq (V_2(0) - \bar{c}/c_k) e^{-c_k t} + \bar{c}/c_k \quad (78)$$

Hence

$$\frac{1}{2} Z_{12}^T Z_{12} \leq V_2, \|Z_{12}\| \leq \sqrt{\frac{2\bar{c}}{c_k}}, \quad (79)$$

that is, the desired tracking error will exponentially converge to the set

$$\|z_1\| + \|z_2\| \leq \sqrt{\frac{2\bar{c}}{c_k}}, \quad (80)$$

and

$$\|x_3\| = \|z_1 + x_{3d}\| \leq \|\tilde{z}_1\| + \|x_{3d}\|, \|x_4\| = \|z_2 + x_{4d}\| \leq \|z_2\| + \|x_{4d}\|. \quad \blacksquare \quad (81)$$

By using Eq.(57) and Eq.(66), it yields

$$\begin{aligned} \dot{V}_2 = & -z_1^T c_3 z_1 + z_1^T G_3 z_2 + \lambda_1 s^T (-c_3 z_1 + G_3 z_2) - (1/\gamma_D) \tilde{D}_\omega^T \dot{\hat{D}}_\omega - (1/\gamma_W) \tilde{W}^T \dot{\hat{W}} \\ & + s^T \left(\dot{x}_{4,0} + \frac{\partial f_4(x_3, x_4)}{\partial x} \bigg|_0 \Delta x + \frac{\partial G_4(x_3, x_4)}{\partial u} \bigg|_0 \Delta u + \Delta D_\omega + O(x, u) + W^{*T} h + \varepsilon - \dot{x}_{4d} \right) \end{aligned} \quad (82)$$

then the incremental control input is obtained as (35).

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