2017 中国自动化大会
（CAC 2017）
论文集

山东·济南

2017 年 10 月 20-22 日

A new IFNTSM controller design for the BWB aircraft with parameter uncertainties

Yan Jianing, Zhang Jing, Liu Pin, Yang Lingyu
School of Automation Science and Electrical Engineering
Beihang University
Beijing, China
yanglingyu@buaa.edu.cn

***Abstract*—Aiming at attitude control of the advanced Blend-wing-body (BWB) layout aircraft with aerodynamic, center of gravity and inertial parameters' uncertainties, a new improved fast non-singular terminal sliding mode control (IFNTSM) method is proposed in this paper. Compared to the conventional terminal sliding mode (TSM) control method, IFNTSM guarantees to reach the equilibrium in an improved finite time for any initial state and also effectively deals with system uncertainties. Moreover, the proposed method addresses the singularity problem in the conventional TSM controller design. Simulation results verify the ability and effectiveness of the IFNTSM controller to improve the convergence rate and cope with strong uncertainties.**

Keywords—terminal sliding mode control; nonsingular; parameter uncertainty; attitude control; aircraft

I. INTRODUCTION

For the advanced aircrafts, different kinds of uncertainties are existent during the actual flight, including the fuel consumption, weapon launch, wing icing, and aircraft structural damage. These factors may result in unknown drastic changes in aerodynamic, center of gravity (CG) and inertial parameters of aircrafts, thus having an adverse impact on flight control performance and safety. To deal with this problem, many scientific research institutions have conducted in-depth studies on advanced flight control of the aircraft with uncertain parameters. The control methods that are often used include robust control, adaptive control, sliding mode control and integrated control which uses various control methods.

The applications of adaptive control to cope with uncertainties usually set the uncertain parameters as adaptive parameters and keep adjusting the control law to achieve the desired tracking performance [1-4]. Reference [2] proposed a multi-variable spline adaptive control combined with dynamic inverse control for the general aerodynamic uncertainties of the high-performance aircraft. Reference [4] used the adaptive backstepping control to cope with the aerodynamic parameter uncertainties and external disturbances, and then indeed achieved good control performance.

Robust control has also been used to solve the system uncertainties [5-8]. Reference [5] investigated a robust control method using the disturbance estimator as well as constant compensation to deal with the uncertainties caused by complex aerodynamic effects in the close formation flight. References [7] and [8] provided improved approaches for robust conservatism existing in the H_∞, μ-synthesis control methods. Reference [7] presented an optimization method to effectively find the maximum robust stability index of the system and reduce the conservatism of robust control design, then three corresponding algorithms were given in reference [8].

Sliding-mode control has been studied extensively and used in many applications due to its simplicity and robustness, especially the invariance to system disturbance and parameter variations. Selecting the longitudinal LPV model of Boeing 747 as object, reference [9] designed the control law based on sliding mode control and disturbance observer, which could cope with the uncertainties existing in the aerodynamic derivatives of aircraft. Based on the longitudinal model of the hypersonic vehicle, reference [10] proposed a recursive sliding-mode controller with the nonlinear disturbance observer to reduce the influences of aerodynamic parameter perturbation and unknown dead-zone nonlinearity.

Compared to the limitations of other control methods, such as robust conservatism, calculation amount and convergence rate of adaptive control, the sliding-mode control shows more excellent control performance to the uncertain parameters system. As such, sliding-mode control is selected as the main scheme in this article. For the specific BWB aircraft with parameter uncertainties, a new improved fast nonsingular terminal sliding mode control (IFNTSM) method is proposed, which can guarantee to reach the equilibrium in an improved finite time for any initial state, and also address the singularity problem in the conventional TSM controller. Besides, this method possesses the invariance property of sliding-mode control to external disturbances and parameter variations.

The first section of this article introduces the research background. In the second section, the six degree-of-freedom model of the BWB aircraft with aerodynamic, CG and inertial parameters' uncertainties is built. Then the detailed design process and stability analysis of the proposed IFNTSM method are presented in the third section. Then it is followed by the simulation analysis of control laws, including the convergence rate and robustness. Conclusions are stated in the final section.

II. MODELING OF THE BWB AIRCRAFT WITH PARAMETER UNCERTAINTY

The six degree-of-freedom model of aircraft with CG

This work was supported by the National Natural Science Foundation of China under Grant No. 61273099 and No. 61304030.

variations can be expressed as [11]:

$$\begin{bmatrix} \dfrac{d(\boldsymbol{V}_{\text{o}})_b}{dt} \\ \dfrac{d(\boldsymbol{\omega})_b}{dt} \end{bmatrix} = \begin{bmatrix} m\mathbf{I}_{3\times3} & -m\mathbf{D} \\ m\mathbf{D} & \mathbf{I} \end{bmatrix}^{-1} \cdot \left\{ \begin{bmatrix} (\boldsymbol{F})_b \\ (\boldsymbol{M})_b \end{bmatrix} - \begin{bmatrix} m\mathbf{\Omega} & -m\mathbf{\Omega}\mathbf{D} \\ m\mathbf{\Omega}\mathbf{D} & \mathbf{\Omega}\mathbf{I} - m\mathbf{V}\mathbf{D} \end{bmatrix} \begin{bmatrix} (\boldsymbol{V}_{\text{o}})_b \\ (\boldsymbol{\omega})_b \end{bmatrix} \right\}$$

where

$$\mathbf{\Omega} = \begin{bmatrix} 0 & -r & q \\ r & 0 & -p \\ -q & p & 0 \end{bmatrix}, \mathbf{V} = \begin{bmatrix} 0 & -w & v \\ w & 0 & -u \\ -v & u & 0 \end{bmatrix}$$

$$\mathbf{D} = \begin{bmatrix} 0 & -\Delta z_{cg} & \Delta y_{cg} \\ \Delta z_{cg} & 0 & -\Delta x_{cg} \\ -\Delta y_{cg} & \Delta x_{cg} & 0 \end{bmatrix}, \mathbf{I} = \begin{bmatrix} I_{xx} & -I_{xy} & -I_{xz} \\ -I_{xy} & I_{yy} & -I_{yz} \\ -I_{xz} & -I_{yz} & I_{zz} \end{bmatrix}$$

m is the mass of the aircraft, $\mathbf{I}_{3\times3}$ represents the 3×3 identity matrix, $\boldsymbol{\omega} = [p, q, r]^{\mathrm{T}}$ is the aircraft angular velocities, $\boldsymbol{V}_{\text{o}} = [u, v, w]^{\mathrm{T}}$ is the linear velocities, $(\boldsymbol{F})_b$, $(\boldsymbol{M})_b$ are the total external forces and moments in the body-fixed frame, Δx_{cg}, Δy_{cg}, Δz_{cg} are the CG variations in body axis, I_{xx}, I_{yy}, I_{zz} are the aircraft moments of inertia, I_{xy}, I_{xz}, I_{yz} are the aircraft products of inertia.

Denote $(\boldsymbol{M})_b$ as $\boldsymbol{U}$, then the aircraft angular motion can be expressed as

$$\dot{\boldsymbol{\omega}} = -\mathbf{I}^{-1}\mathbf{\Omega}\mathbf{I}\boldsymbol{\omega} + \mathbf{I}^{-1}\boldsymbol{U} - \mathbf{I}^{-1}(m\mathbf{\Omega}\mathbf{D})^T\boldsymbol{V} - \mathbf{I}^{-1}m\mathbf{D}\dot{\boldsymbol{V}} \quad (1)$$

Based on (1), the uncertainties of aerodynamic and inertial parameters are further introduced. These parameters' uncertainties can be expressed as

$$\Delta\mathbf{I} = k_I \cdot \mathbf{I}, \ \Delta C_i = k_C \cdot C_i \quad (2)$$

and the influence of aerodynamic parameters' uncertainties on the aircraft motion (1) can be described as

$$\Delta\boldsymbol{U} = \Delta C_i \cdot QSb \quad (3)$$

where ΔC_i, $\Delta\mathbf{I}$ represent the aerodynamic and inertial parameters' uncertainties, Q is the dynamic pressure, S is the wing area, b is the wingspan.

Based on (2) and (3), the aircraft angular motion with aerodynamic, CG and inertial parameters' uncertainties can be obtained as

$$\begin{aligned} \dot{\boldsymbol{\omega}} = &-\mathbf{I}^{-1}\Omega\mathbf{I}\boldsymbol{\omega} + \mathbf{I}^{-1}\boldsymbol{U} + \mathbf{I}^{-1}[\Delta\boldsymbol{U} - \frac{k_I}{1+k_I}(\boldsymbol{U} + \Delta\boldsymbol{U}) \\ &- \frac{1}{1+k_I}(m\mathbf{\Omega}\mathbf{D})^T\boldsymbol{V} - \frac{1}{1+k_I}m\mathbf{D}\dot{\boldsymbol{V}}] \end{aligned} \quad (4)$$

Considering the Euler angle $\boldsymbol{X}$ and the angular velocity $\boldsymbol{\omega}$ satisfy

$$\boldsymbol{\omega} = \mathbf{P}_{\mathbf{X}} \cdot \dot{\boldsymbol{X}}$$

where

$$\mathbf{P}_{\mathbf{X}} = \begin{bmatrix} 1 & 0 & -\sin\theta \\ 0 & \cos\phi & \cos\theta\sin\phi \\ 0 & -\sin\phi & \cos\theta\cos\phi \end{bmatrix} \quad \boldsymbol{X} = \begin{bmatrix} \phi \\ \theta \\ \psi \end{bmatrix}$$

ϕ, θ, ψ denote the roll, pitch and yaw angles respectively.

Take the derivative of the equation above, we have

$$\dot{\boldsymbol{\omega}} = \mathbf{P}_{\dot{\mathbf{X}}} \cdot \dot{\boldsymbol{X}} + \mathbf{P}_{\mathbf{X}} \cdot \ddot{\boldsymbol{X}}$$

Then (4) can be rewritten as

$$\begin{aligned} \ddot{\boldsymbol{X}} = &-(\mathbf{I}\mathbf{P}_{\mathbf{X}})^{-1}(\mathbf{\Omega}\mathbf{I}\mathbf{P}_{\mathbf{X}} + \mathbf{I}\mathbf{P}_{\dot{\mathbf{X}}})\dot{\boldsymbol{X}} + (\mathbf{I}\mathbf{P}_{\mathbf{X}})^{-1}\boldsymbol{U} + (\mathbf{I}\mathbf{P}_{\mathbf{X}})^{-1}[\Delta\boldsymbol{U} \\ &- \frac{k_I}{1+k_I}(\boldsymbol{U} + \Delta\boldsymbol{U}) - \frac{1}{1+k_I}(m\mathbf{\Omega}\mathbf{D})^T\boldsymbol{V} - \frac{1}{1+k_I}m\mathbf{D}\dot{\boldsymbol{V}}] \end{aligned} \quad (5)$$

Conveniently, the aircraft attitude equations can be described as

$$\ddot{\boldsymbol{X}} = \mathbf{F} \cdot \dot{\boldsymbol{X}} + \mathbf{B} \cdot \boldsymbol{U} + \boldsymbol{d}(\boldsymbol{t}) \quad (6)$$

where

$$\mathbf{F} = -(\mathbf{I}\mathbf{P}_{\mathbf{X}})^{-1}(\mathbf{\Omega}\mathbf{I}\mathbf{P}_X + \mathbf{I}\mathbf{P}_{\dot{\mathbf{X}}})$$

$$\mathbf{B} = (\mathbf{I}\mathbf{P}_{\mathbf{X}})^{-1}$$

$$\boldsymbol{d}(\boldsymbol{t}) = (\mathbf{I}\mathbf{P}_{\mathbf{X}})^{-1}[\Delta\boldsymbol{U} - \frac{k_I}{1+k_I}(\boldsymbol{U} + \Delta\boldsymbol{U}) - \frac{1}{1+k_I}(m\mathbf{\Omega}\mathbf{D})^T\boldsymbol{V} - \frac{1}{1+k_I}m\mathbf{D}\dot{\boldsymbol{V}}]$$

$\boldsymbol{d}(\boldsymbol{t})$ represents all the disturbances including uncertainties of the aerodynamic, CG and inertial parameters.

III. IFNTSM Attitude Control Law Design

Aiming at the system uncertainties, a new improved fast non-singular terminal sliding mode control (IFNTSM) method is proposed in this article. Compared with the conventional terminal sliding mode control (TSM), the proposed method not only effectively copes with the parameter uncertainties, but also accelerates the convergence rate in the sliding mode phase. Besides, the new method can address the singularity problem in the TSM controller.

The sliding surface is chosen as:

$$s = e + \gamma e^{mh/ng} + \alpha e^{h/g} + \frac{1}{\beta}\dot{e}^{h/g} \quad (7)$$

where α, β, γ are all integers and satisfy $\alpha, \beta, \gamma > 0$, g, h, m, n are all positive odds and satisfy $h > g, m > n$.

When the system state enters into the sliding mode phase $s = 0$, (7) can be further expressed as

$$\dot{e}^{h/g} = -\beta\gamma e^{mh/ng} - \beta\alpha e^{h/g} - \beta e \tag{8}$$

As can be seen, similar to the traditional TSM controller, IFNTSM has fast convergence rate close to the equilibrium point ($|e|<1$), and furthermore, the item $-\beta\gamma e^{mh/ng}$ speeds up the convergence rate of the state far from equilibrium ($|e|>1$).

The non-singular TSM concept can be described as

$$s = e + \frac{1}{\beta}\dot{e}^{h/g}$$

and the dynamics will reach $e=0$ in a finite time determined by

$$T_N = \int_0^{e(0)} \frac{1}{(\beta e)^{g/h}} de = \frac{|e(0)|^{1-g/h}}{\beta^{g/h}(1-g/h)}$$

For the proposed IFNTSM method, the exact time to reach zero is solved as

$$T_{IFN} = \int_0^{e(0)} \frac{1}{(\beta e + \alpha\beta e^{h/g} + \beta\gamma e^{mh/ng})^{g/h}} de$$

Compared T_{IFN} with T_N, we have $T_{IFN} < T_N$, which means IFNTSM improves the convergence toward equilibrium and guarantees to reach the equilibrium in a bounded finite time.

Based on the aircraft model, IFNTSM attitude control law is designed. First, the model (6) is rewritten as

$$\begin{cases} \dot{\boldsymbol{x}}_1 = \boldsymbol{x}_2 \\ \dot{\boldsymbol{x}}_2 = \mathbf{F}\boldsymbol{x}_2 + \mathbf{B}\boldsymbol{u} + \boldsymbol{d}(\boldsymbol{t}) \end{cases} \tag{9}$$

And the system error is defined as

$$\begin{cases} \boldsymbol{e} = \boldsymbol{x}_1 - \boldsymbol{r} \\ \dot{\boldsymbol{e}} = \dot{\boldsymbol{x}}_1 - \dot{\boldsymbol{r}} = \boldsymbol{x}_2 - \dot{\boldsymbol{r}} \\ \ddot{\boldsymbol{e}} = \dot{\boldsymbol{x}}_2 - \ddot{\boldsymbol{r}} \end{cases} \tag{10}$$

where $\boldsymbol{x}_1$, $\boldsymbol{r}$ represent the attitude angle and the command, respectively.

Take the derivative of the equation (7) and select the reaching law as

$$\dot{\boldsymbol{s}} = -\frac{h}{\beta g}\mathbf{Q_1} \cdot \boldsymbol{\varepsilon}\,\mathrm{sgn}(\boldsymbol{s}) \tag{11}$$

where $\mathbf{Q_1} = diag(\dot{e}_i^{h/g-1})$, $\boldsymbol{\varepsilon} = diag(\varepsilon_i)$, $i=1,2,3$.

Thus the control law can be obtained as

$$\boldsymbol{u} = -\mathbf{B}^{-1}[\mathbf{F}\boldsymbol{x}_2 - \ddot{\boldsymbol{r}} + \frac{\beta g}{h}\dot{\boldsymbol{e}}^{2-h/g} + \frac{\beta\gamma m}{n}\mathbf{Q}_3\dot{\boldsymbol{e}}^{2-h/g} + \alpha\beta\mathbf{Q}_2\dot{\boldsymbol{e}}^{2-h/g} + \boldsymbol{\varepsilon}\,\mathrm{sgn}(\boldsymbol{s})] \tag{12}$$

where $\mathbf{Q}_2 = diag(e_i^{h/g-1})$, $\mathbf{Q}_3 = diag(e_i^{mh/ng-1})$, $i=1,2,3$.

As can be seen in (12), the negative power terms will not appear if the conditions $2>h>g>1, m>n$ are satisfied. Then the singularity problem is avoided.

Theorem 1: For the system (9), the improved fast non-singular terminal sliding mode control law (12) is stable if the conditions $\varepsilon_i > |d_i(t)|, i=1,2,3$ are satisfied, i.e., the system error $\boldsymbol{e}$ can converge to zero in a finite time.

Proof: Select the Lyapunov function as

$$\mathbf{V} = \frac{1}{2}\boldsymbol{s}^T\boldsymbol{s}$$

Take the derivative of $\mathbf{V}$, then we have

$$\dot{\mathbf{V}} = \boldsymbol{s}^T \cdot (\dot{\boldsymbol{e}} + \frac{\gamma mh}{ng}\mathbf{Q_3}\dot{\boldsymbol{e}} + \frac{\alpha h}{g}\mathbf{Q_2}\dot{\boldsymbol{e}} + \frac{h}{\beta g}\mathbf{Q_1}\ddot{\boldsymbol{e}})$$

Substitute (10) and (12) into this equation and obtain

$$\begin{aligned} \dot{\mathbf{V}} &= \boldsymbol{s}^T \cdot \{\dot{\boldsymbol{e}} + \frac{\gamma mh}{ng}\mathbf{Q_3}\dot{\boldsymbol{e}} + \frac{\alpha h}{g}\mathbf{Q_2}\dot{\boldsymbol{e}} + \frac{h}{\beta g}\mathbf{Q_1} \\ &\quad \cdot[\mathbf{F}\boldsymbol{x}_2 - \mathbf{F}\boldsymbol{x}_2 + \ddot{\boldsymbol{r}} - \frac{\beta g}{h}\dot{\boldsymbol{e}}^{2-h/g} - \frac{\beta\gamma m}{n}\mathbf{Q_3}\dot{\boldsymbol{e}}^{2-h/g} \\ &\quad -\alpha\beta\mathbf{Q_2}\dot{\boldsymbol{e}}^{2-h/g} - \boldsymbol{\varepsilon}\,\mathrm{sgn}(\boldsymbol{s}) + \boldsymbol{d}(\boldsymbol{t}) - \ddot{\boldsymbol{r}}]\} \\ &= -\boldsymbol{s}^T \cdot \frac{h}{\beta g}\mathbf{Q_1}[\boldsymbol{\varepsilon}\,\mathrm{sgn}(\boldsymbol{s}) - \boldsymbol{d}(\boldsymbol{t})] \\ &= -|s_i|\frac{h}{\beta g}\dot{e}_i^{h/g-1}[\varepsilon_i - d_i(t)] \quad i=1,2.3 \end{aligned}$$

Since g, h are positive odds and satisfy $1<h/g<2$, $\dot{e}_i^{h/g-1}>0$ is always true when $\dot{e}_i \neq 0$. If $\varepsilon_i > |d_i(t)|$, then $\dot{\mathbf{V}} = \boldsymbol{s}^T \cdot \dot{\boldsymbol{s}} \le -\eta\|\boldsymbol{s}\|$ holds.

When $\dot{e}_i = 0$, Substitute the control law (12) into (9) then we obtain

$$\ddot{\boldsymbol{e}} = \boldsymbol{d}(\boldsymbol{t}) - \boldsymbol{\varepsilon}\,\mathrm{sgn}(\boldsymbol{s})$$

Similarly if $\varepsilon_i > |d_i(t)|$, thus

$$\ddot{e}_i = d_i(t) - \varepsilon_i\,\mathrm{sgn}(s_i) \Rightarrow \begin{cases} \ddot{e}_i \le -\eta & s_i > 0 \\ \ddot{e}_i \ge \eta & s_i < 0 \end{cases}$$

Then according to Ref. [12], the system states from any initial position can converge to the sliding surface.

As discussed above, $\dot{\mathbf{V}} = \boldsymbol{s}^T \cdot \dot{\boldsymbol{s}} \le -\eta\|\boldsymbol{s}\|$ always holds and the reachability of the sliding surface is satisfied, i.e., the system states can converge to $\boldsymbol{s}=0$. When the states reach the sliding surface, for any initial error $\boldsymbol{e}(0)$, the dynamics will reach $\boldsymbol{e}=0$ in a finite time determined by T_{IFN}.

In brief, if the system uncertainties satisfy $|d_i(t)| < \varepsilon_i$, the proposed IFNTSM control law (12) guarantees that the error

$\boldsymbol{e}$ can converge to zero in a finite time.

IV. SIMULATION RESULTS

To verity the improvement ability of IFNTSM in convergence rate and coping with parameter uncertainties, numerical simulations are carried out to compare IFNTSM with the conventional non-singular terminal sliding mode control (NTSM) method.

To solve the chattering phenomena of sliding mode control, the following continuous function is used to instead of the $\mathrm{sgn}(\cdot)$ function in (12), and the concrete form of continuous function is shown as below.

$$\dot{s}_i = \varepsilon_i \cdot \frac{s_i}{|s_i| + \sigma}$$

Besides, the saturation and rate limits of control surface actuators are summarized in Table 1.

TABLE 1. SATURATION AND RATE LIMITS OF ACTUATORS

	Inner elevon	Outer elevon	Split drag rudder
Deflection limit, deg	[-20,20]	[-20,20]	[-20,20]
Rate limit, deg/s	[-100,100]	[-100,100]	[-100,100]

First, the proposed IFNTSM method is compared with the conventional NTSM method and fast non-singular terminal sliding mode control (FNTSM) to verify the improvement on the convergence rate. The model of the BWB aircraft is trimmed at height 12 km and 0.6 Mach. Rectangular shape commands of 5 and 4 degrees are given to the roll and pitch axes. Each rectangular signal ascends at 2s and descends at 8s. Fig.1, 2 and 3 show the comparison of attitude response among different controllers of IFNTSM, NTSM and FNTSM. And the corresponding parameters of the three controller are listed in Table 2.

As can be seen in Fig.1-3, the attitude responses of IFNTSM can quickly and precisely track the given commands. Compared with the conventional NTSM, FNTSM methods, the proposed IFNTSM method significantly improves the tracking speed and reduces the settling time. The tracking speeds of pitch and roll channels are increased by 40.0%, 48.8% respectively.

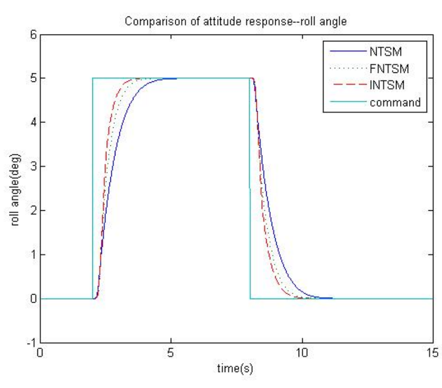

Fig. 1. Roll angle responses

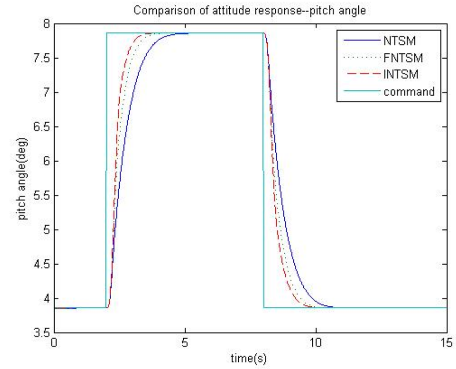

Fig. 2. Pitch angle responses

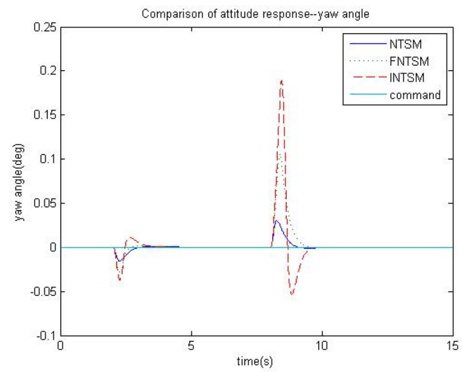

Fig. 3. Yaw angle responses

TABLE 2. CONTROL PARAMETERS OF THE DIFFERENT THREE METHODS

	Sliding surface							Reaching law	
	α_i	β_i	γ_i	h	g	m	n	ε_i	δ_i
NTSM	1			13	11			20	0.8
FNTSM	1	1		13	11	11	9	20	0.8
IFNTSM	1	1	1	13	11	11	9	20	0.8

Then the parameter uncertainties are introduced to verify the robustness performance. The uncertainties are set as:

a. CG uncertainty:

$$\begin{bmatrix} \Delta x_{cg} & \Delta y_{cg} & \Delta z_{cg} \end{bmatrix} = \begin{bmatrix} 10\% c_A & 2\% c_A & 2\% c_A \end{bmatrix}$$

b. aerodynamic parameter uncertainty: $k_C = -30\% \sim 30\%$

c. inertial parameter uncertainty: $k_I = -30\% \sim 30\%$

The aerodynamic and inertial parameters' uncertainties are usually time-varying and unknown, and the representative form used in the simulation is shown in Fig. 4.

Fig.5, 6 and 7 show the attitude tracking performance of IFNTSM in roll, pitch and yaw channels respectively while the uncertainties are continuously in existence.

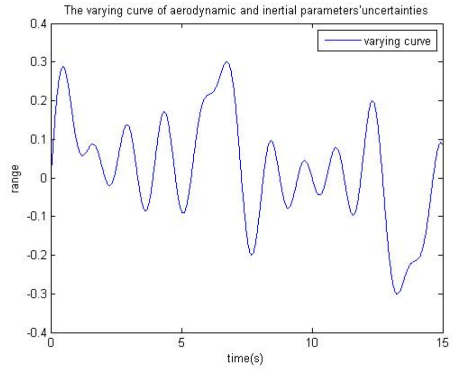

Fig. 4. Aerodynamic and inertial parameters' uncertainties

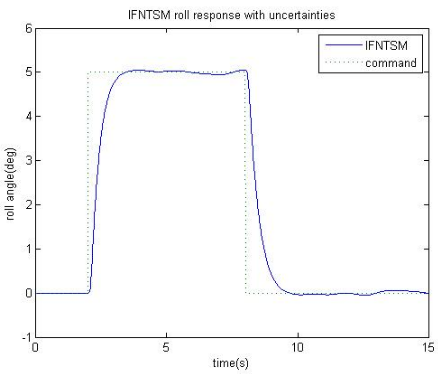

Fig. 5. IFNTSM roll response with uncertainties

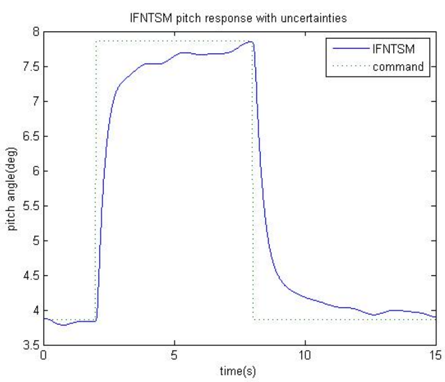

Fig. 6. IFNTSM pitch response with uncertainties

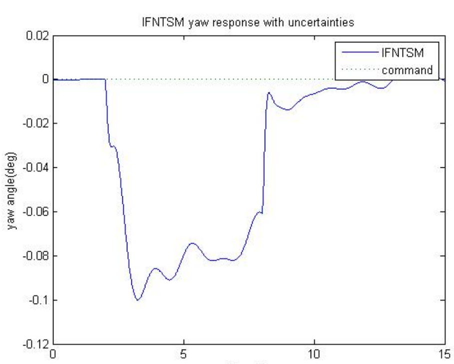

Fig. 7. IFNTSM yaw response with uncertainties

As can be seen in Fig.5-7, the aerodynamic, CG and inertial parameters' uncertainties indeed generate some small effects on the control performance of IFNTSM. However, the attitude responses in roll, pitch and yaw channels can also track the commands well, which verifies the robustness and ability of IFNTSM in coping with parameter uncertainties.

CONCLUSION

In this article, a new improved fast non-singular terminal sliding mode control method is proposed for the BWB aircraft with aerodynamic, CG and inertial parameter uncertainties. First, a six degree-of-freedom model of aircraft with parameter uncertainties is built. Then the attitude control law is designed based on the proposed method, and the stability is proved by using the Lyapunov theory. The proposed IFNTSM method can guarantee the robustness to system parameter variations and reach the equilibrium in an improved finite time for any initial state. Finally, the effectiveness of IFNTSM is verified by simulations.

REFERENCES

[1] R. Kamalapurkar, B. Bialy, L. Andrews, et al, "Adaptive RISE feedback control strategies for systems with structured and unstructured uncertainties," AIAA Guidance, Navigation, and Control (GNC) Conference, Boston, MA, pp.1-11, 2013.

[2] H.J. Tol, C.C. DeVisser, L.G. Sun, et al, "Multivariate spline-based adaptive control of high-performance aircraft with aerodynamic uncertainties," Journal of Guidance, Control, and Dynamics, vol. 39, pp. 781-800, 2016.

[3] P. van Gils, E. Van Kampen, C.C. DeVisser, et al, "Adaptive incremental backstepping flight control for a high-performance aircraft with uncertainties," AIAA Guidance, Navigation, and Control (GNC) Conference, San Diego, California, pp.1-26, 2016.

[4] X. Zhang, S.Z. Chen, and H. Zhou, " Research on the adaptive inverse control for hypersonic vehicle with uncertainties," Missiles and Space Vehicles, vol. 343, pp.73-76, 2016.

[5] Q. Zhang, and H.H. Liu, "Robust design of close formation flight control via uncertainty and disturbance estimator," AIAA Guidance, Navigation, and Control (GNC) Conference, San Diego, California, pp.1-15, 2016.

[6] Y. Dai, Z. Wu, C. Yang, et al, "Robust aeroservoelastic design with structural and trim parameter uncertainties," AIAA Guidance, Navigation, and Control (GNC) Conference, Minneapolis, Minnesota, pp.1-14, 2012.

[7] R.K. Yedavalli, N. Devarakonda, R. Chandavarkar, et al, "ECO-inspired robust control design algorithms for linear systems with real parameter uncertainty," AIAA Guidance, Navigation, and Control (GNC) Conference, National Harbor, Maryland, pp.1-16, 2014.

[8] N. Devarakonda, P. Sar, and R.K. Yedavalli, "Robust flight control design for linear systems with real parameter uncertainty," Journal of Guidance, Control, and Dynamics, vol.38, pp. 2195-2200, 2015.

[9] Z. Chao, A. Tsourdos, and M. Lone, "Disturbance observer based sliding mode controller design for large civil aircraft," AIAA Guidance, Navigation, and Control (GNC) Conference, Boston, MA, pp.1-10, 2013.

[10] C. Chen, G.F. Ma, Y.C. Sun, et al, "Recursive sliding mode control for hypersonic vehicle based on nonlinear disturbance observer," ACTA ARMAMENTARII, vol. 37, pp. 840-850, 2016.

[11] J. Zhang, L. Yang, and G. Shen, "Modeling and attitude control of aircraft with center of gravity variations," 2009 IEEE Aerospace conference, Big Sky, MT, pp.1-11, 2009.

[12] Y. Feng, X. Yu, and Z. Man, "Non-singular terminal sliding mode control of rigid manipulators," Automatica, vol.38, pp.2159-2167, 2002.