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Backstepping design of missile guidance and control based on adaptive fuzzy sliding mode control

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Abstract This paper presents an integrated missile guidance and control law based on adaptive fuzzy sliding mode control. The integrated model is formulated as a block-strict-feedback nonlinear system, in which modeling errors, unmodeled nonlinearities, target maneuvers, etc. are viewed as unknown uncertainties. The adaptive nonlinear control law is designed based on backstepping and sliding mode control techniques. An adaptive fuzzy system is adopted to approximate the coupling nonlinear functions of the system, and for the uncertainties, we utilize an online-adaptive control law to estimate the unknown parameters. The stability analysis of the closed-loop system is also conducted. Simulation results show that, with the application of the adaptive fuzzy sliding mode control, small miss distances and smooth missile trajectories are achieved, and the system is robust against system uncertainties and external disturbances.

1. Introduction

Missile guidance and control systems are usually designed separately due to the assumption that there is a spectral separation between the guidance loop and the control loop. Based on this paradigm, a number of past missile systems which guarantee outstanding performance have been designed. However, it can be argued that this design paradigm cannot fully exploit synergistic relationships between the two subsystems or strictly maintain the stability of the overall system.[1] On the other hand, the spectral separation assumption may be invalid, especially at the end-game phase of the interception.[2] Integrated guidance and control (IGC) design was first put forward in Ref.[3], and has received much attention in recent years.[4–8] It was shown that IGC designs have the potential to enhance missile performance by viewing the two subsystems as an integrated system and accounting for the coupling between guidance and control dynamics.

Various control methods have been adopted in IGC designs. A small-gain theorem based IGC law was designed in Ref.[1] for missiles steered by both canard and tail controls, and the stability of the overall system could be guaranteed without the assumption that the angle between line-of-sight (LOS) and missile velocity was almost invariable. An IGC law based on adaptive output feedback and backstepping techniques was designed in Ref.[7] for formation flight, which was translated into better transient and steady-state range tracking performance. An IGC law based on the state-dependent

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Riccati equation approach for a moving-mass-actuated missile was designed in Ref.[8], and miss distances which were much less than the diameter of the missile were achieved. The nonlinear optimal control technique, the θ–D method, was employed in Ref.[9] to design an IGC law, and the controller did not require online computation of the state-dependent Riccati equation.

Sliding mode control (SMC) is another typical method in IGC designs. SMC is known to be an efficient control technique applicable to a wide class of nonlinear systems, due to its insensitivity to model uncertainties and external disturbances after reaching the sliding phase. SMC has been addressed in some previous studies for IGC designs.[2,10–14] Koren et al.[2] chose the zero-effort miss distance as the sliding variable. A robust SMC controller was then designed to deal with both system uncertainties and the difference between nonlinear and linear design methods. Shima et al.[10] defined the same sliding surface as that in Ref.[2]. Based on their approach, small distances could be achieved even in stringent interception scenarios. Hou and Duan[11] proposed an IGC scheme for homing missiles against ground fixed targets, and an SMC-based adaptive nonlinear control law was designed to guarantee a missile hit a target accurately with a desired impact attitude angle. Based on the assumption that each of the three channels of an IGC model can be independently designed, Yamasaki et al.[12] introduced an IGC design approach for a path-following uninhabited aerial vehicle. Dong et al.[13] developed a robust higher-order sliding mode (HOSM) based IGC law, in which the IGC design problem was considered to be equal to the stabilization of a third integral chain system. Zhao et al.[14] proposed a SMC-based nonlinear IGC strategy which took the higher-order dynamics of the system into account.

Although SMC has been widely applied to IGC designs, some problems still exist. Nearly all existing approaches are based on the assumption that the nonlinear functions in an IGC model could be accurately obtained. In practice, such an assumption may not be always guaranteed. In this paper, an IGC law based on adaptive fuzzy sliding mode control is firstly presented. The developed approach, when compared with the existing results, is novel in that the IGC law can guarantee high performance without the assumption that the coupling nonlinear functions in the integrated model can be accurately obtained.

2. Model derivation

2.1. Engagement kinematics

The planar engagement geometry is depicted in Fig. 1, where OXY is a Cartesian inertial reference frame, and M and T represent the missile and the target, respectively. The corresponding equations of motion between the missile and the target are as follows:[1]

$$\dot{R} = V_{\mathrm{T}}\cos(q-\theta_{\mathrm{T}}) - V_{\mathrm{M}}\cos(q-\theta_{\mathrm{M}}) \tag{1a}$$

$$R\dot{q} = -V_{\mathrm{T}}\sin(q-\theta_{\mathrm{T}}) + V_{\mathrm{M}}\sin(q-\theta_{\mathrm{M}}) \tag{1b}$$

where R is the relative range, q is the LOS angle, θ_{M} and θ_{T} are the missile and target flight path angles, respectively, and V_{M} and V_{T} are the missile and target velocities, respectively. Differentiating Eq. (1b) followed by the substitution of Eq. (1a), we get

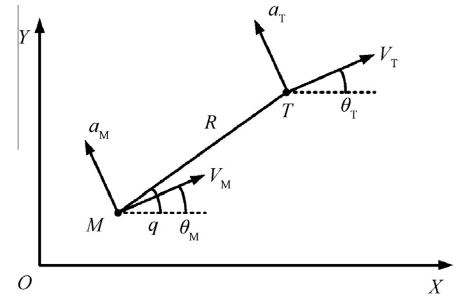

Fig. 1 Planar engagement geometry.

$$R\ddot{q} + 2\dot{R}\dot{q} = -\dot{V}_{\mathrm{T}}\sin(q-\theta_{\mathrm{T}}) + \dot{V}_{\mathrm{M}}\sin(q-\theta_{\mathrm{M}}) + V_{\mathrm{T}}\dot{\theta}_{\mathrm{T}} \times \cos(q-\theta_{\mathrm{T}}) - V_{\mathrm{M}}\dot{\theta}_{\mathrm{M}}\cos(q-\theta_{\mathrm{M}}) \tag{2}$$

Assume that $\dot{V}_{\mathrm{M}} = \dot{V}_{\mathrm{T}} = 0$, and define $V_q = R\dot{q}, a_{\mathrm{T}} = V_{\mathrm{T}}\dot{\theta}_{\mathrm{T}}$, and $a_{\mathrm{M}} = V_{\mathrm{M}}\dot{\theta}_{\mathrm{M}}$. Eq. (2) can be rewritten as

$$\dot{V}_q = -\frac{\dot{R}}{R}V_q + a_{\mathrm{T}}\cos(q-\theta_{\mathrm{T}}) - a_{\mathrm{M}}\cos(q-\theta_{\mathrm{M}}) \tag{3}$$

where a_{M} and a_{T} are the missile and target accelerations, respectively.

2.2. Missile dynamics

The planar missile dynamics are given by[15]

$$\dot{\alpha} = \frac{1}{mV_{\mathrm{M}}}(-T_{\mathrm{M}}\sin\alpha - L + mg\cos\theta_{\mathrm{M}}) + \omega_z \tag{4}$$

$$J_z\dot{\omega}_z = M_0 + M_{\delta_z}\delta_z \tag{5}$$

$$\dot{\vartheta} = \omega_z \tag{6}$$

$$\alpha = \vartheta - \theta_{\mathrm{M}} \tag{7}$$

where α is the angle of attack, m is the missile mass, T_{M} is the thrust of the missile, L is the lift force, ω_z is the pitch rate, J_z is the moment of inertia about z-axis, δ_z is the deflection angle for pitch control, ϑ is the pitch angle, M_{δ_z} is the control contribution to the angular acceleration, and $M_0 = M_0(\alpha, Ma, h, V_{\mathrm{M}}, \omega_z)$ represents the angular acceleration contributions from all other sources such as the angle of attack α, the Mach number Ma, the height h, and so on. M_0 is often approximated as follows:[16]

$$M_0 = M_\alpha\alpha + M_{\omega_z}\omega_z \tag{8}$$

where M_α and M_{ω_z} are the angular acceleration contributions from the angle of attack and pitch rate, respectively.

The lift force (L) and relative parameters ($M_\alpha, M_{\omega_z}, M_{\delta_z}$) are as follows:

$$\begin{cases} L = 57.3Qs\left(c_y^\alpha\alpha + c_y^{\delta_z}\delta_z\right) \\ M_\alpha = 57.3Qslm_z^\alpha\alpha \\ M_{\omega_z} = \dfrac{Qsl^2m_z^{\omega_z}}{V_{\mathrm{M}}} \\ M_{\delta_z} = 57.3Qslm_z^{\delta_z} \end{cases} \tag{9}$$

where Q is the dynamic pressure, s is the aerodynamic reference area, l is the reference length, c_y^α and $c_y^{\delta_z}$ are the lift force derivatives with respect to α and δ_z, respectively, and $m_z^\alpha, m_z^{\omega_z}$, and $m_z^{\delta_z}$ are the pitch moment derivatives with respect to α, ω_z, and δ_z, respectively.