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INTEGRATED HOMING GUIDANCE-CONTROL SYSTEM FOR ENDO-ATMOSPHERIC INTERCEPT IN SLIDING MODES [1]

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Abstract: The enabling methods for advanced interceptors to achieve the hit-to-kill accuracy against targets performing evasive maneuvers including spiraling motion are developed on the basis of one unified theory for the purpose of design and analysis, viz. a contemporary nonlinear robust control theory such as the sliding mode control. The integration of guidance and flight control systems is achieved in a two-loop guidance and flight control system designed in the combined state space of engagement kinematics and vehicle dynamics. The designed guidance-control system performance is verified via computer simulations using a generic endo-atmospheric interceptor model.

Keywords: sliding mode control, guidance and control, robust control.

1. INTRODUCTION

The miss distance as an ultimate performance criterion of a homing interceptor is crucially dependent on all its subsystems (guidance, navigation and control (Garnell and East, 1977; Zarchan, 1998) working together in a closed loop, such as the homing loop. This situation calls for the integrated design of all interceptor modules: sensor information processing and necessary data estimation, a homing guidance law, and flight control (autopilot).

The primary goal of this work is to develop the enabling technology for advanced interceptors to achieve the hit-to-kill accuracy against targets performing evasive maneuvers including spiraling motion, considering all subsystems on the basis of one unified theory for the purpose of design and analysis, viz. a contemporary nonlinear robust control theory such as the sliding mode control (SMC). The scope of this paper is restricted to the perfect information scenario (noise is not involved, no sensors, filters, estimators), such that the integration of guidance computer and flight control system is addressed only. The main concern here is to use minimum possible information in order to achieve the goal (target intercept) in presence of uncertainties and disturbances acting in the homing loop. Processing noisy navigation data with the aid of SMC-based fliters/observers is the issue to be explored in a different work.

In this work, an integrated two-loop guidance and flight control system is designed to incorporate variety of guidance strategies and robustly enforce them regardless of target maneuvers, atmospheric disturbances, and dynamic uncertainty of airframe-actuator. The ideas of backstepping approach and the relative degree approach are used in the two-loop sliding mode control system, much similar as has been done for the aircraft control in (Shtessel *et al*., 1999). Modern achievements of the emerging control technique, higher order sliding modes, (Fridman and Levant, 1996) are employed in order to upgrade algorithms in guidance and flight control systems of an advanced interceptor (kinetic energy kill vehicle). Different strategies to the homing missile guidance problem are studied and particular benefits from the standpoint of an integrated guidance-control system

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that robust control theory can potentially contribute to its solution are identified.

2. INTERCEPT STRATEGY AND SLIDING MODE GUIDANCE

2.1 Introduction to Sliding Mode Guidance

Missile guidance law synthesis and performance analysis, especially homing missile guidance, is a very attractive problem for application of a broad band of control and system theory methods. One approach is taken by the group of methods employing geometric ideas in the feedback control design, where the intercept strategy is identified first and then the intercept problem is transformed into the output regulation problem. Among these are feedback linearization control applications (Bezick *et al.*, 1995) and sliding mode control (SMC) applications (Brierly and Longchamp, 1990; Babu *et al.*, 1994; Zhou *et al.*, 1999; Moon and Kim, 2000). Recent application of SMC theory to the homing missile guidance resulted in a series of very effective algorithms in terms of smaller acceleration advantage required for intercept of weaving targets as compared to ProNav and augmented ProNav guidance.

In this work, SMC-based guidance algorithms are to be used as a basis for building an integrated guidance-flight-control system. The main idea is to enforce given closed-loop dynamics for LOS rate in presence of unmodeled before missile dynamics (aerodynamics + airframe + actuator), using the input voltage signal to the actuator as control. In this case engagement kinematics and missile dynamics are integrated into one state space, and the intercept problem is transformed into the output regulation problem (a given constraint keeping). The given closed-loop dynamics for LOS rate is selected to be such as it would be with SMC-guidance applied to a zero-lag guidance system.

2.2 Intercept Strategy: Geometric Approach

Consider planar engagement kinematics without account for gravity. In polar coordinate system the relative position is presented by $\mathbf{R} = (r, \lambda)$, where r = range along Line-Of-Site (LOS), and λ = LOS angle. The state model of homing-missile engagement process is obtained

$$\begin{cases} \dot{r} = V_r, \\ \dot{V}_r = r\omega_\lambda^2 + A_{T\,r} - \sin(\lambda - \gamma_M)n_L, \\ \dot{\lambda} = \omega_\lambda, \\ \dot{\omega}_\lambda = \dfrac{1}{r}\left(-2V_r\omega_\lambda + A_{T\,\lambda} - \cos(\lambda - \gamma_M)n_L\right), \end{cases} \tag{1}$$

where we consider ω_λ as a commanded output, missile normal acceleration as a control input, and projections of target acceleration along and orthogonal to LOS, $A_{T\,r}, A_{T\,\lambda}$, are considered as unknown bounded disturbances. The system (1) can be written also as

$$\begin{cases} \dot{r} = V_r, \\ \dot{V}_r = r\omega_\lambda^2 + A_{T\,r} - \sin(\lambda - \gamma_M)n_L, \\ \dot{\lambda} = \dfrac{V_\lambda}{r}, \\ \dot{V}_\lambda = -\dfrac{V_r V_\lambda}{r} + A_{T\,\lambda} - \cos(\lambda - \gamma_M)n_L, \end{cases} \tag{2}$$

where $V_\lambda = r\omega_\lambda$ is a transversal component of relative velocity in the reference frame rotating with LOS. For a direct hit, it's necessary to keep $V_r < 0$. The most critical component in V_r-dynamics, centrifugal acceleration, $r\omega_\lambda^2 = \dfrac{V_\lambda^2}{r}$, is rapidly growing as LOS keep rotating. It provides almost instant reverse in V_r direction, as r becomes small. The most radical decision is to eliminate this term, i.e. to keep $\omega_\lambda = 0$. In other words, the goal to keep $\dot{\mathbf{R}} = \mathbf{V_R}$ vector to be anti-collinear to $\mathbf{R}$ vector is one possible strategy. In this case the reverse in $\dot{r}$ will happen only after r crossing zero (direct hit the target). Zeroing the LOS rate was the primary goal for many developed guidance laws (Zarchan, 1998). The control task in this case is to counteract target acceleration driving ω_λ to zero. However, to hit the target, it's sufficient to nullify the transversal component of relative velocity, V_λ, only at the final moment; i.e. if $V_\lambda \to 0$ as $r \to 0$, then $\mathbf{V_R}$ will lie along LOS, and heading error will be zero. Foe example, one of the firsts and one of the most practical strategies in guidance, proportional navigation (PN), nullifies heading error gradually during the flight and eliminates it completely at the end (ideally) (Zarchan, 1998). Moreover, even if ω_λ is allowed to grow, though not faster than $\dfrac{1}{r^\alpha}, \alpha < 1$ as $r \to 0$, we obtain $V_\lambda = r \cdot \dfrac{1}{r^\alpha} = r^{1-\alpha} \to 0$ as $r \to 0$. On the other hand, centrifugal acceleration term in V_r-dynamics should stay limited, as $r \to 0$, otherwise the reverse in V_r is inevitable. Thus, at least, one has to provide $r\omega_\lambda^2 \le M$, some M, i.e. $\omega_\lambda \propto \dfrac{1}{r^{\alpha'}}, \alpha' \le \frac{1}{2}$. So, the least suitable ω_λ behavior is

$$\omega_\lambda = \frac{C_o}{\sqrt{r}}, \text{or } V_\lambda = C_o\sqrt{r}. \tag{3}$$

Now, the following task can be formulated:
Stabilize the system (1) or (2) on the manifold

$$\sigma_1 = \omega_\lambda = 0, \text{ or } \sigma_1 = V_\lambda = 0 \tag{4}$$

or

$$\sigma_2 = \omega_\lambda - \frac{C_o}{\sqrt{r}} = 0, \text{ or } \sigma_2 = V_\lambda - C_o\sqrt{r} = 0 \tag{5}$$

where the quantity $\sigma_i, i = 1,2$ determines the system (1) or (2) output to stabilize to zero.

2.3 Sliding Mode Guidance Design

The chosen intercept strategy transforms the intercept problem to an output regulation problem, which can be considered as a fundamental problem in geometric methods of control synthesis. The control goal of the output regulation problem is to stabilize $\sigma_i, i = 1,2$ and to satisfy the condition $\forall t \geq t_o,\ V_r < 0$ provided given limits on control input, $n_L \leq n_{L\ \max}$, and known bounds of uncertainties variations.

In this paper, we employ continuos SMC control design developed by Brown *et al.*, 2000. There, the SMC retains property of finite time convergence, which is the essence of a sliding mode, in absence of uncertainties, as opposed to asymptotic convergence in a linear control law. From (5) the σ-dynamics is identified as (we omit the subscript)

$$\dot{\sigma} = -\frac{V_r V_\lambda}{r} + A_{T,\lambda} - \frac{c_0 V_r}{2\sqrt{r}} - \cos(\lambda - \gamma_M) n_L , \quad (6)$$

so the commanded acceleration, n_c, for the missile normal acceleration n_L is selected to be

$$n_c = \frac{1}{\cos(\lambda - \gamma_M)} \left(\frac{N' V_r V_\lambda}{r} - \frac{c_0 V_r}{2\sqrt{r}} + \rho \frac{\sigma}{|\sigma|^{0.5}} \right) . \quad (7)$$

From (7), the closed loop σ-dynamics is derived as

$$\dot{\sigma} = \frac{(N'-1) V_r \sigma}{r} + \frac{c_0 (N'-1) V_r}{\sqrt{r}} + A_{T,\lambda} - \rho \frac{\sigma}{|\sigma|^{0.5}} . \quad (8)$$

When *r* is large, the last term in (8) dominates over the other providing for the convergence of σ to the small domain $|\sigma| \leq L_1, |\dot{\sigma}| \leq L_2$, where its size can be computed using the analysis in (Brown *et al.*, 2000). When *r* is approaching to zero and still $V_r < 0$, the first term provides for the finite-time collapse of the σ-dynamics (8), such that $\sigma \to 0$ as $r \to 0, V_r < 0$.

3. INTEGRATION OF GUIDANCE AND FLIGHT CONTROL SYSTEMS: 2-LOOP SMC APPROACH

3.1 Vehicle Dynamics

We consider the following planar vehicle model of a homing interceptor (kinetic energy kill vehicle KEKV) in pitch plane without account for gravity as in (Shkolnikov *et al.*, 2000)

$$\begin{bmatrix} \dot{\alpha} \\ \dot{V} \\ \dot{\theta} \\ \dot{q} \end{bmatrix} = \begin{bmatrix} -\frac{X}{mV}\sin\alpha + \frac{Z}{mV}\cos\alpha + q \\ \frac{X}{m}\cos\alpha + \frac{Z}{m}\sin\alpha \\ q \\ \frac{\bar{q}Sl}{I_{yy}}\left(C_{m_o}(\cdot) + \Delta C_m(t)\right) \end{bmatrix} + \begin{bmatrix} 0 \\ 0 \\ 0 \\ \frac{\bar{q}Sl}{I_{yy}} C_m^{\delta_e}(\cdot) \end{bmatrix} \cdot \delta_e , \quad (9)$$

where it's assumed that the general nonlinear model is affine in controls, i.e. in case of only aerodynamic control (δ_e is a virtual deflection, which is allocated by the fin's mixing logic) we have

$$C_m(\alpha, \dot{\alpha}, Mach, q, \delta_e, ...) = C_{m_o}(\cdot) + C_m^{\delta_e}(\cdot) \cdot \delta_e ;$$

$$X = \bar{q}S\left(C_X(\cdot) + \Delta C_X(t)\right),$$

$$Z = \bar{q}S\left(C_Z(\cdot) + \Delta C_Z(t)\right),\ \bar{q} = \tfrac{1}{2}\rho V^2 ,$$

where $\Delta C_X(t), \Delta C_Z(t), \Delta C_m(t)$ are the external disturbances.

The normal acceleration, n_L is the following system output

$$n_L = \frac{\bar{q}S}{m} C_L(\alpha, ...) = -A_x \sin\alpha + A_z \cos\alpha = \dot{\gamma}_M \cdot V , \quad (10)$$

where $\gamma_M = \theta - \alpha$ is the flight path angle, $A_x = \frac{X}{m}, A_z = \frac{Z}{m}$.

It's also assumed that the model (9) is of minimum phase. In case of aerodynamic control, it means that $\frac{\partial C_L}{\partial \delta_e} = 0,\ \frac{\partial C_D}{\partial \delta_e} = 0$, which is true, if the extra small fins are employed. The actuators have the highly nonlinear dynamics of unknown order, the relative degree is supposed to be equal to one, such that the input/output dynamics for each individual fin channel can be presented as

$$\dot{\delta}_i = -a(\delta_i, t)(\delta_i - u_i),\ i = 1,2,..., \quad (11)$$

(*i* is fin number), where the unmodeled internal dynamics will be considered as a time varying bounded disturbance, and be accounted as the part of the nonlinear term $a(\delta_i, t)$, which is allowed to be non-smooth but bounded (backlash, rate saturation etc.), since it's matched by the control u_i.

3.2 Model Behavior of the Homing Loop

Now, the problem is to design the control input to the actuator, $u(t)$, in order to enforce the given closed loop performance of a homing loop robustly to uncertainty of vehicle dynamics. Under the SMC guidance of a generic format

$$n_c = -\frac{4 V_r V_\lambda}{r} + f(V_\lambda, r) , \quad (12)$$

the closed-loop V_λ-dynamics are

$$\dot{V}_\lambda = \frac{3 V_r V_\lambda}{r} - f(V_\lambda, r) + A_{T,\lambda} . \quad (13)$$

It was shown in simulations that even in presence of $A_{T,\lambda}$ the closed-loop V_λ-dynamics provides for target intercept. So, the ideal model to follow will be selected as

$$\dot{V}_\lambda = \frac{3 V_r V_\lambda}{r} - f(V_\lambda, r) , \quad (14)$$

although we know that bounded forcing disturbance to this system is tolerable. The model behavior (14) should be robustly enforced by control *u* in presence of vehicle dynamics uncertainties and disturbances and target maneuvers. This problem will be solved using higher order SMC approach (Fridman and Levant, 1996) and backstepping ideas in a two-loop controller structure (Shtessel *et al.*, 1999).

The first loop (outer loop) will be designed to enforce (14) using missile body pitch rate, *q*, as a virtual control. We assume that pitch rate is measurable and, therefore, the correspondent pitch rate tracking error signal can be used to create the second (inner loop) control loop.

3.3 Outer Loop SMC Design

Now we consider the composite state space of the systems (1) and (9). If we try to hold the following constraint

$$\sigma_o = \dot{V}_\lambda - \frac{3V_r V_\lambda}{r} + f(V_\lambda, r) = 0\,, \qquad (15)$$

then even under small residual perturbation , such that $\sigma_o \neq 0$ but $|\sigma_o(t)| \leq M$, some $M > 0$, we achieve good closed-loop performance, meaning that our ideal model (14) will behave similar to the system (13), which is proven to be satisfactory.

Following the SMC approach, we call the constraint to be kept (15) as the sliding manifold in the outer loop, where σ_o is the sliding quantity of the manifold (15). To stabilize σ_o to zero, its dynamics is identified

$$\dot{\sigma}_o = -4\left[\frac{\left(\dot{V}_r V_\lambda + V_r \dot{V}_\lambda\right) r - V_r^2 V_\lambda}{r^2}\right] - \dot{A}_{T,\lambda} -$$

$$\left(\sin(\lambda - \gamma_M)\left(\frac{V_\lambda}{r} - \frac{n_L}{V_M}\right) n_L + \cos(\lambda - \gamma_M)\dot{n}_L\right) + \frac{\partial f}{\partial V_\lambda}\dot{V}_\lambda + \frac{\partial f}{\partial r} V_r \qquad (16)$$

One can rearrange the terms in (16) and write it in the short notation as

$$\dot{\sigma}_o = \varphi(\cdot) - \cos(\lambda - \gamma_M)\dot{n}_L\,, \qquad (17)$$

where all the uncertainty is lumped into the term $\varphi(\cdot)$. Designing a continuous SMC control in terms of $\dot{n}_L$, which is to be enforced as $\dot{n}_{Lc}$, we obtain similar to the procedure for the guidance law on missile normal jerk $\dot{n}_{Lc}$

$$\dot{n}_{Lc} = \frac{1}{\cos(\lambda - \gamma_M)} \rho_o \frac{\sigma_o}{|\sigma_o|^{0.5}}\,. \qquad (18)$$

So, the inner loop design can be build with respect to the normal jerk tracking, however having pitch rate as an available measurement, one can recalculate (18) into command for pitch rate. From the model (9) we identify

$$q = \frac{1}{V}\left(T_\alpha \dot{n}_L + \int \dot{n}_L d\tau\right), \qquad (19)$$

where V is missile speed, and T_α is known as the turning rate coefficient

$$T_\alpha = \frac{\alpha}{\dot{\gamma}_M} = V\left(\frac{m}{\bar{q}S}\right)\left(\frac{\partial C_L}{\partial \alpha}(\cdot)\right)^{-1}. \qquad (20)$$

Considering T_α as a known slowly varying quantity, we finally obtain the following profile for the missile pitch rate to follow

$$q_c(t) = \frac{1}{V\cos(\lambda - \gamma_M)}\left(T_\alpha \rho \frac{\sigma_o}{|\sigma_o|^{0.5}} + \int \rho \frac{\sigma_o}{|\sigma_o|^{0.5}} d\tau\right) \qquad (21)$$

Thus, a command on missile maneuver is obtained in terms of pitch rate command. It can be considered as the guidance command. However, to obtain this command we had to consider the composite state space of engagement kinematics and vehicle dynamics. The second (inner loop) will be designed next to robustly enforce pitch rate command (21) in presence of uncertainties and disturbances to the airframe-actuator dynamics.

3.4 Inner Loop SMC Design

The regulated output in the inner loop is the pitch tracking error

$$e = q_c(t) - q\,. \qquad (22)$$

From (9),(11) we determine that the relative degree of the input-output dynamics for e

$$\ddot{e} = \varphi_1(\cdot) - \frac{T_\alpha}{V}(T_\alpha)^{-1}\omega_n^2(\cdot)a(\cdot)u\,, \qquad (23)$$

is equal to two. Thus, one can apply the second order SMC approach (Krupp *et al*., 2000) to provide for finite time convergence of tracking error to zero robustly to time-varying additive uncertainty $\varphi_1(\cdot)$ and multiplicative uncertainties of airframe dynamics $\omega_n^2(\cdot)$ (equivalent undamped natural frequency of airframe $\omega_n^2(\cdot) = \frac{\bar{q}Sl}{I_{yy}} C_m^{\delta_e}(\cdot)$) and actuator dynamics $a(\cdot)$ (actuator bandwidth). The only requirement is to know the limit of $\varphi_1(\cdot)$ variations, and the nominal value and sign of the term $\frac{1}{V}\omega_n^2(\cdot)a(\cdot)$.

Given that the control law for u, based on nonlinear dynamic sliding manifold (NDSM) design (Krupp et al., 2000) is obtained as

$$u = \frac{V}{\omega_n^2 a}\rho\,\mathrm{sgn}(J),\; J = e + \chi,$$
$$\dot{\chi} = 5\frac{e}{|e|^{1/2}} - 40\frac{J}{|J|^{2/3}}\,, \qquad (24)$$

where the nominal values for uncertainties $\omega_n^2(\cdot)$ and $a(\cdot)$ are used, and the coefficient ρ is selected according to the upper absolute limit for additive uncertainty $\varphi_1(\cdot)$.

Thus, using only output feedback $e = q_c(t) - q$, the control voltage to the actuator (11) provides for the output e convergence to zero in a finite time.

4. SIMULATION EXAMPLE

We simulate the KEKV airframe with the following equivalent characteristics (Shkolnikov et al., 2000) $T_\alpha = 3.75\sec$, $\omega_n = 2\pi \cdot 1.8\, rad/s$, trimmed uncompensated damping 0.01 (untrimmed airframe is unstable), actuator bandwidth 20*Hz*, missile speed in projection to the plane of engagement is 473 m/s. The rest of simulation data and conditions is the same as in (Shkolnikov et al., 2000).

The integrated guidance-control system is selected as follows.

First, the closed-loop engagement kinematics is selected as being under SMC guidance but in absence of target maneuver, i.e.

$$\sigma_o = \dot{V}_\lambda - \frac{3V_r V_\lambda}{r} + 5SAT_4(V_\lambda)\,.$$

Second, from (21) the "virtual control" for pitch rate is obtained as

$$q_c(t)=\frac{1}{V\cos(\lambda-\gamma_M)}\left(T_\alpha\rho\frac{\sigma_o}{|\sigma_o|^{0.5}}+\int\rho\frac{\sigma_o}{|\sigma_o|^{0.5}}d\tau\right),$$

the outer loop control ("guidance law"),
Finally, the control input to the actuator is selected to be

$$u=\frac{V}{\omega_n^2}100\,\mathrm{sgn}(J),\ J=e+\chi,\ \dot{\chi}=5\frac{e}{|e|^{0.5}}-40\frac{J}{|J|^{2/3}},$$

the inner loop control.
Simulation are shown in Figs.23-30.
The work of outer loop Result: Miss distance is zero.

Discussion of results: The discontinuous control signal is passing through the actuator producing smooth control surfaces deflections ("equivalent" normalized deflection is shown in Fig.5, in reality discontinuous voltage is passing through fins mixing stage logic, and then each command is actuated via individual fin actuator (11)). The sliding mode $J=0$ on the auxiliary dynamic sliding manifold J (Fig.4) provides for exact pitch rate tracking in a finite time (Fig.3). Pitch rate as the virtual control in the outer loop holds given constraint $\sigma_o=0$ with good quality, which is enough to perform intercept. In Fig. 1 one can see that the ratio of missile/target acceleration in projection to LOS is close to one, when transient is over and missile repeats target maneuver as that appears in projection to LOS. Here, the stability of the homing loop is supported up to the point, when the "direct hit" condition is inevitable.

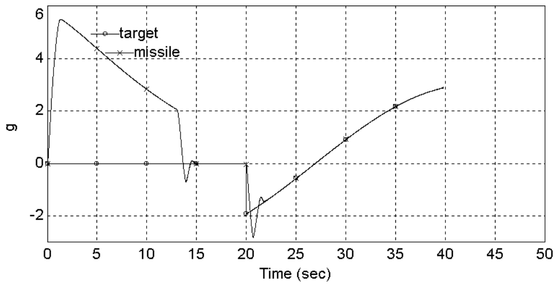

Fig.1 Target and missile accelerations in projection to LOS

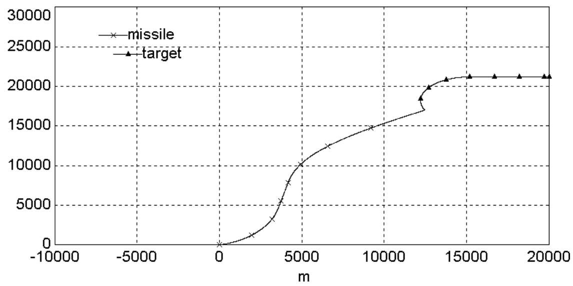

Fig.2 Trajectory of missile and target

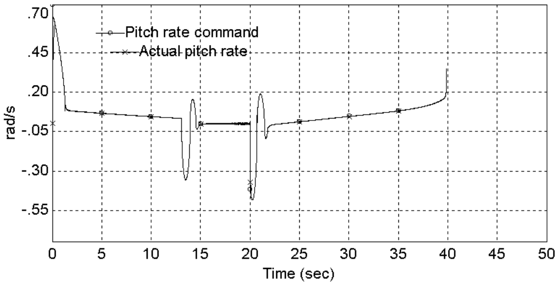

Fig.3 Pitch rate tracking

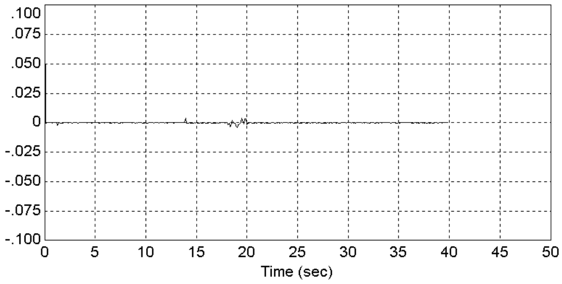

Fig.4 The sliding quantity *J* of the NDSM

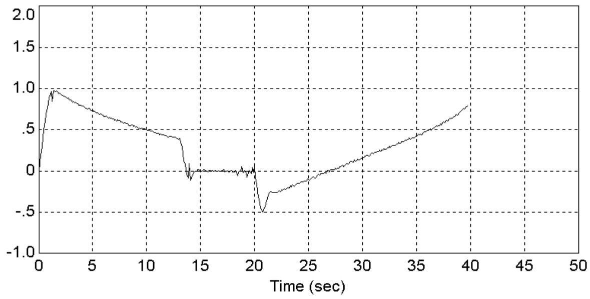

Fig.5 Equivalent normalized control surface deflection

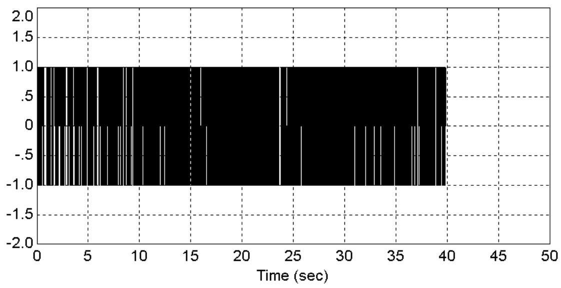

Fig.6 Normalized voltage to the actuator

5. CONCLUSIONS

In this work, an integrated two-loop guidance and flight control system is designed to incorporate variety of guidance strategies and robustly enforce them regardless of target maneuvers, atmospheric disturbances, and dynamic uncertainty of airframe-actuator.
The ideas of backstepping approach, the relative degree approach and the second order sliding mode control are used in the two-loop sliding mode control system. Different strategies based on SMC geometric approach to the homing missile guidance problem are studied. Particular benefits of the presented integrated guidance-control system include

- robustness to agile target motion and phase lag in the homing loop (the outer "guidance" loop system),
- robustness to atmospheric disturbances, and dynamic uncertainty of airframe-actuator of the inner loop flight control system,
- the reduced acceleration ratio requirements due to prolonged stability of the homing loop,
- zero time lag in the inner loop due to finite time convergence of the pitch rate tracking dynamics (collapse of airframe-actuator dynamics in the terminal nonlinear dynamic sliding manifold).

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