MISSILE AUTOPILOT CONTROLLED BY AERODYNAMIC LIFT AND DIVERT THRUSTERS VIA SECOND-ORDER SLIDING MODE

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Abstract: A new approach to the design of a kinetic-kill longitudinal autopilot steering the missile trajectory by the combination of aerodynamic lift and divert-thrusters and with its attitude oriented by attitude-thrusters is presented. The pitch plane autopilot design is based on high order sliding mode control. A robust high accuracy tracking of the missile normal acceleration guidance command is achieved in presence of considerable model uncertainties created by the interactions between the airflow and the thruster's jets. Results of the computer simulation demonstrate excellent robust high accuracy tracking performance of the proposed design.

Keywords: Missile, Sliding Mode Control

1. INTRODUCTION

Controlling missile interceptions of maneuvering targets in the medium endo-atmospheric (20000 to 40000 m) domain is a challenging problem due model uncertainties created by the interactions between the airflow and the thruster's jets as shown by Kennedy et al, 1998. Avoiding intercepting in this altitude range in order to reduce the mentioned unwanted phenomena might not be always possible, especially in the case of short-range ballistic threats. The need that interceptors outmaneuver the target's increasingly larger maneuver amplitudes requires kill-vehicle larger and larger divert control capabilities. While it always possible to achieve lateral divert, using only divert thruster, this approach to increasing missile lateral divert comes at price, by leading to larger and thus more expensive interceptors. An efficient solution to this problem is the combined use of aerodynamic-divert and thruster-divert, which may increase the total divert acceleration capability by up to 100%. Typical dual-thrust control missile equipped with divert and attitude control systems (DACS) is shown in Fig. 1.

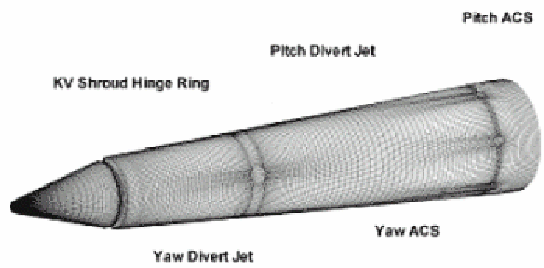

Fig. 1 Dual-thrust controlled endo-atmospheric missile-interceptor

The combination of direct forces with traditional lift created by the angle-of-attack has also been envisaged in aviation, using spoilers as a direct lift device concurrently with the lift created by the angle-of-attack, as a mean of achieving gusts alleviation was proposed by Sun (1996). Shama and Cloutier (1993) studied a two-loop missile autopilot where the inner-loop pitch angle command, calculated by an inversion is the pseudo-control of the outer-loop. The decoupling of the two loops requires the inner-loop to be faster than the outer-loop. Another problem is posed by the accuracy of the inversion due to model uncertainties.

This approach generally requires some modal decomposition using linear techniques, which are adversely affected by model uncertainties, and as the missile flight conditions vary rapidly requires an effective gain scheduling. Possible solutions to the inversion are the usage of disturbance observers as proposed by Tournes (1998a) and Chen (2000).

Previous design by Tournes (2001) integrated the guidance and the automatic pilot using Subspace Stabilization Theory introduced by Johnson (1973) to simultaneously guide the missile and control its attitude while achieving the invariance of the subspaces which represent desired divert and attitude error-responses in error. In spite of the extensive development of robust control techniques, *sliding mode control* (SMC), (see Edwards and Spurgeon, 1998 and Utkin et al. 1999) stays the main choice in handling bounded uncertainties/disturbances and unmodeled dynamics. The idea is in steering the system trajectory in finite-time to custom-chosen sliding manifold and keeping it thereafter by means of high-frequency switching control and making that manifold insensitive to matched disturbances. Tournes (1996a, b), Shtessel et al. (2000b) applied sliding mode control technique to the design of aircraft autopilots, Tournes (1998b), Shtessel et al. (2000a) applied it to a reusable launch vehicle flight control. While stabilization of the sliding variable in SISO systems by means of the traditional SMC requires the system relative degree to be equal to one with respect to the sliding variable, and high frequency control switching leads to chattering that is difficult to avoid or attenuate, high order sliding mode control (HOSM) mitigating these difficulties was used by Levant 1998, 2001 and by Shtessel and Shkolnikov 2003b. The applications of and high order sliding mode control to a variety of supersonic missiles integrated guidance and control designs as proposed by Shtessel et al. (2002, Shkolnikov et al. 2001, Shtessel and Shkolnikov (2003) and Levant et al. (2000) demonstrated its effectiveness for aircraft flight control and aerospace vehicle control designs. In this paper the second order sliding mode (SOSM)–based autopilot is designed for controlling missile interceptions of maneuvering targets in the pitch plane in the medium endo-atmospheric domain using a combination of aerodynamic lift and divert-thrusters control. Model uncertainties created by the interactions between the airflow and the thruster's jets are taken into account. This paper is structured as follows. Section 2 presents the dynamic model of the vehicle. The control architecture is discussed in Section 3. Section 4 presents the design of the autopilot. Computer simulations are discussed in Section 5. Finally, Section 6 presents the conclusions.

2. MISSILE PICH PLANE MODEL

Missile airframe dynamics in a pitch plane is given by

$$\begin{aligned}\dot{\alpha} = q - \frac{\rho(z)VS}{2m} C_{L_\alpha}[1+d_\alpha]\alpha - \frac{Tmax_\Delta}{mV}[1+d_\Delta]\cos(\alpha)F_\Delta \\ - \frac{Tmax_\delta}{mV}[1+d_\delta]\cos(\alpha)F_\delta + \frac{g\cos(\gamma)}{V}\end{aligned} \tag{1}$$

$$\begin{aligned}\dot{q} = \frac{\rho(z)V^2Sa_c}{2I_{yy}} C_{L_\alpha}[1+d_\alpha]\alpha + \frac{\rho(z)VS\overline{C}^2}{2I_{yy}} C_{M_q}[1+d_\alpha]q \\ + \frac{Tmax_\Delta}{I_{yy}}[1+d_\Delta]a_\Delta\cos(\alpha)F_\Delta + \frac{Tmax_\delta}{I_{yy}}[1+d_\delta]a_\delta\cos(\alpha)F_\delta\end{aligned} \tag{2}$$

$$\dot{\gamma} = q - \dot{\alpha} \tag{3}$$

Model disturbances are introduced in the form of multiplying factors $d_\alpha, d_\delta, d_\Delta$ applied respectively to aerodynamic lift, attitude and divert thrusts. They are caused by the interactions between attitude and thruster jets and the system of shockwaves. At a given time the three disturbance factors are supposed to be de-coupled, and given the rapid firing rate of the thrusters, they are also decoupled from time sample to time sample. It is worth noting that same random samples are applied in α-equation and in the q-equation. Random disturbance samples $d_\alpha, d_\delta, d_\Delta$ are uniformly distributed in intervals $\pm D_\alpha, \pm D_\delta, \pm D_\Delta$ respectively. For simplicity sake, longitudinal velocity is assumed to be constant
We are going to design a pitch-autopilot using Eqs. (1-3) based on a SOSM-based design that achieves asymptotic tracking of given in current time command to missile acceleration normal to velocity vector, $\Gamma_{com}(t)$ by means of F_i, $i = \delta, \Delta$ in presence of modeling uncertainties $d_\alpha, d_\delta, d_\Delta$.

3. CONTROL ARCHITECTURE

The normal acceleration command Γ_{com} is filtered in a block identified in Figure 2 as *Command Profiles.*

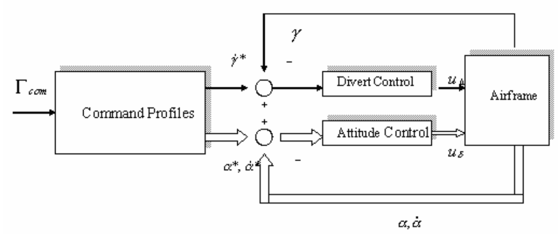

Fig. 2 Missile Flight Control (Autopilot) Architecture

The purpose of the block is to filter the acceleration command, to generate corresponding flight path angle and flight path rate and flight path acceleration $\gamma^*, \dot{\gamma}^* \ddot{\gamma}^*$ respectively and to calculate associated aerodynamic angle and aerodynamic angles $\alpha^*, \dot{\alpha}^*$ that would be required to achieve filtered flight path angle profiles, assuming perfect model knowledge. The filter also restricts the magnitude of commanded angle-of-attack. The block identified as *Attitude Control* calculates the attitude thruster control F_δ the tracking of the angle-of-attack and of the angle-of-attack rate profiles require. The tracking of commanded angle-of-attack actually constitutes an open loop control of missile acceleration. It is clear that even assuming a perfect model, and a perfect tracking of commanded angle-of-attack, this open loop control could not achieve by itself satisfactory tracking of filtered flight path angle rate. The block identified as *Divert Control* achieves the tracking of the commanded flight path angle rate. It calculates the control of divert thruster F_Δ, in the presence of the normal acceleration created by the aerodynamic lift. This acceleration constitutes a "cooperating" matched disturbance with respect to divert control. In traditional designs divert control must track exactly the commanded flight path angle rate minus the estimated or modeled flight path angle rate achieved by the aerodynamic maneuver by itself. In the proposed Sliding Mode based design, this is not required due to the property that sliding mode controllers compensate for the effects of matched disturbances. The advantage of such approach is that as the missile altitude and velocity vary, so does the contribution of the aerodynamic lift to the total divert acceleration, until, that contribution ceases to be significant at which point attitude control simply regulates the angle-of-attack to zero. The simplest thruster mode of operation is on-off, where effective thrust is obtained by adjusting the duration of the pulse-width. When linear control designs are used, the pulse-width mode of operation requires complex re-designs of the control to accommodate this mode of operations, which most of the time degrade the linear control performance. We are going to show that the sliding mode control design achieves the same level of performance using pulse-width modulation than with continuous thrusters.

4. CONTROL DESIGN

4.1 Command Profiles

Smoothed command acceleration and its first time derivative, $\Gamma^*, \dot{\Gamma}^*$ are calculated using the second order filter given by

$$\Gamma^* = \frac{\omega^2 \Gamma_{com}}{s^2 + 1.4\omega s + \omega^2} \tag{4}$$

where $\Gamma_{com}(t)$ is a command coming from the missile guidance system. The angle-of-attack maneuver aimed at alleviating the lateral divert effort required by divert thruster is calculated as follows. The reference flight path angle profile $\dot{\gamma}^*$ is calculated as function of normal command acceleration Γ^* and the corresponding gravity component as

$$\dot{\gamma}^* = \frac{\Gamma^*}{V} + \frac{g}{V}\cos(\gamma^*)\ . \tag{5}$$

Using the time derivative of command acceleration generated by the filter, Eq. (4) one can calculate $\ddot{\gamma}^*$ the second time derivative of the flight path angle profile as

$$\ddot{\gamma}^* = \frac{\dot{\Gamma}^*}{V} - \frac{g}{V}\dot{\gamma}^*\sin(\gamma^*)\ . \tag{6}$$

Likewise, integrating $\dot{\gamma}^*$ yields the reference flight path angle profile as

$$\gamma^* = \int \dot{\gamma}^* d\tau\ . \tag{7}$$

Assuming perfect knowledge of missile aerodynamics the angle of attack profile α^* sought and its $\dot{\alpha}^*$ derivative are calculated using Eqs. (1)-(3) as follows:

$$\alpha^* = \frac{\dot{\gamma}^* + g\cos(\gamma^*)/V}{Z_\alpha} \tag{8}$$

$$\dot{\alpha}^* = \frac{\ddot{\gamma}^* - g\sin(\gamma^*)\dot{\gamma}^*/V}{Z_\alpha} \tag{9}$$

Given the disturbances discussed before, the value of Z_α is not very accurate, which yet is not a problem, because the purpose of the attitude control is just to create a "cooperative" disturbance to the guidance command, which will alleviate the control effort left to the divert thrusters, and for that matter it needs not to be very accurate. This is also the case when the commanded angle-of-attack is saturated to its maximum acceptable value, commanded acceleration cannot be matched by the effects of the commanded angle-of-attack.

4.2 Design of the Attitude Control

Introducing angle of attack tracking error $e_\alpha = \alpha^* - \alpha$, differentiating (1) and combining it with (3) we obtain

$$\ddot{e}_\alpha = \ddot{\alpha}^* - f_\alpha - M_\delta F_\delta \tag{10}$$

where $M_\delta = \dfrac{Tmax_\delta a_\delta}{I_{yy}}$ and disturbance f_α, given by

$$\begin{aligned} f_\alpha = {} & \frac{\rho(z)V^2 S a_c}{2I_{yy}} C_{L_\alpha}[1+d_\alpha]\alpha + \frac{\rho(z)VS\bar{C}^2}{2I_{yy}} C_{M_q}[1+d_\alpha](\dot{\gamma}+\dot{\alpha}) \\ & + \frac{Tmax_\Delta}{I_{yy}}[1+d_\Delta]a_\Delta\cos(\alpha)F_\Delta + \frac{Tmax_\delta}{I_{yy}}[d_\delta]a_\delta\cos(\alpha)F_\delta \\ & - \frac{\rho(z)VS}{2m} C_{L_\alpha}[1+d_\alpha]\dot{\alpha} - \frac{Tmax_\Delta}{mV}[1+d_\Delta]\cos(\alpha)\dot{F}_\Delta \\ & - \frac{Tmax_\delta}{mV}[1+d_\delta]\cos(\alpha)\dot{F}_\delta + \frac{g\sin(\gamma)}{V} + \frac{Tmax_\Delta}{mV}[1+d_\Delta]\sin(\alpha)\dot{\alpha}F_\Delta \\ & + \frac{Tmax_\delta}{mV}[1+d_\delta]\sin(\alpha)\dot{\alpha}F_\delta \end{aligned} \tag{11}$$

accounts for the effects of all the other terms, including disturbance d_δ. The sliding variable σ_α is chosen as

$$\sigma_\alpha = \varpi^2 \int e_\alpha d\tau + 1.4\varpi e_\alpha + \dot{e}_\alpha; \quad \varpi = 10\, rad/s\ . \tag{12}$$

Equation (12) shows that, by achieving $\sigma_\alpha = 0$, the tracking error e_α converges to zero asymptotically according to the eigenvalues of Eq. (12), and hence the problem becomes now to stabilize $\sigma_\alpha = 0$, the dynamics of which are described by

$$\dot{\sigma}_\alpha = \ddot{\alpha}^* - f_\alpha + \varpi^2 e_\alpha + 1.4\varpi\dot{e}_\alpha - M_\delta F_\delta \tag{13}$$

Equation (13) can be robustly stabilized at zero using traditional sliding mode control as proposed by Edwards and Spurgeon (1998) and Utkin et al. (1999), which exhibits a high frequency switching. The problem is that the thrusters need to switch on and off with a sufficient frequency, usually as large as *500 Hz*, possible out of the realm of the technology. In this paper we suggest to use second order sliding mode control as proposed by Levant 1998 and 2001, in particular super-twisting algorithm, that generates a continuous control function and, which is in the same time completely robust to the effects of the disturbances. This continuous control function is transformed to a set of pulses with a given duty cycle by means of Pulse Width Modulation (PWM) or sample-quantization techniques. Also it is assumed that the system (1)-(3) is a minimum phase, otherwise some specially developed sliding mode control techniques, i.e. based on dynamic sliding manifold or system center technique could be employed as proposed by Levant (2000, 2001), Shkolnikov and Shtessel (2001). Indeed, as shown by Edwards and Spurgeon (1998) and Levant (2001) a solution $z(t) \in R^1$ and its derivative $\dot{z}(t)$ of differential equation

$$\dot{z} + \alpha |z|^{1/2} sign(z) + \beta \int sign(z) d\tau = \xi(t) \quad (14)$$

converge to zero in a finite time if $\alpha > 0.5\sqrt{C}$ and $\beta \geq 4C$ with $|\xi(t)| \leq C$.

Based on (14) the following control, called a super-twisting one, can provide finite time convergence of σ_α dynamics (13) to the second order sliding mode, i.e. $\sigma_\alpha = \dot{\sigma}_\alpha = 0$

$$F_\delta = \alpha_M |\sigma_\alpha|^{1/2} sign\sigma_\alpha + \beta_M \int sign\sigma_\alpha d\tau \quad (15)$$

Remark. Taking into account an attitude thruster actuator dynamics $\dot{F}_\delta = \frac{1}{\tau}(-F_\delta + u_\delta)$ it is easy to show that the $\sigma_\alpha -$ dynamics is of a relative degree 2, and the control law (15) should be redesigned in terms of u_δ using twisting, prescribed convergence law algorithms (see Levant, 1998) or nonlinear dynamic sliding manifold-based second order sliding mode control (see Shkolnikov and Shtessel, 2003). However, in this work the designed dual thruster autopilot is simulated using attitude thruster control law (15) that is fed to the input of the actuator.

We will discuss at the end of this section, two implementation of the control law, the first based on a sampled-quantized approach, the second on a pulse-width modulated approach with a dead band.

4.3 Design of the Divert Control

The equation governing the flight path angle is given by

$$\dot{\gamma} = \frac{\rho(z)VS}{2m} C_{L_\alpha}[1 + d_\alpha]\alpha + \frac{Tmax_\Delta}{mV}[1 + d_\Delta]\cos(\alpha)F_\Delta + \frac{Tmax_\delta}{mV}[1 + d_\delta]\cos(\alpha)F_\delta - \frac{g\cos(\gamma)}{V} \quad (16)$$

where the applicable control is F_Δ. Introducing flight path angle tracking error $e_\gamma = \gamma^* - \gamma$, the sliding surface representing the desired error-dynamics is given by

$$\sigma_\gamma = \psi \int e_\gamma d\tau + e_\gamma, \quad \psi = 20\, rad/s \quad (17)$$

Differentiating (17) we obtain the flight path angle sliding variable σ_γ dynamics

$$\dot{\sigma}_\gamma = \dot{\gamma}^* - f_\gamma - Z_\alpha \alpha^* + \psi e_\gamma - Z_\Delta F_\Delta,$$

$$Z_\Delta = \frac{T_{\max\Delta}}{mV}, \; Z_\alpha = \frac{\rho SV}{2m} C_{L\alpha} \quad (18)$$

where

$$f_\gamma = \frac{\rho(z)VS}{2m} C_{L_\alpha}[d_\alpha]\alpha + \frac{Tmax_\Delta}{mV}[d_\Delta]\cos(\alpha)F_\Delta + \frac{Tmax_\delta}{mV}[1 + d_\delta]\cos(\alpha)F_\delta - \frac{g\cos(\gamma)}{V} \quad (19)$$

is considered as a disturbance accounting for the effects of all the other terms, including d_Δ.

Taking into account a divert thruster actuator dynamics given by

$$\dot{F}_\Delta = \frac{1}{\tau}(-F_\Delta + u_\Delta) \quad (20)$$

it is easy to show that the sliding variable σ_γ dynamics is of relative degree 2, and control u_Δ is designed in a twisting second order sliding mode control format (see Levant, 2001), that stabilizes $\sigma_\gamma = \dot{\sigma}_\gamma = 0$ in a finite time

$$u_\Delta = \alpha_\Delta sign(\dot{\sigma}_\gamma) + \beta_\Delta sign(\sigma_\gamma), \quad (21)$$

where $\dot{\sigma}_\gamma$ is estimated using second order sliding mode differentiator (see Levant 1998).

4.4 Control Implementations

The control is achieved by selectively opening the valves opposite to desired thrust. The simplest implementation is on-off implementation where at least one valve must be opened. Two control implementations were tested, the first one with a sample and hold, the second with a pulse-width modulation.

The sample and hold strategy is based on the combination of a sampler and a quantizer. The attitude control was sampled at 500 Hz and the divert control was sampled at 200 Hz. It is clear that the 500 Hz sampling rate is significantly too high, and that rates in the 200 Hz would be more conservative. Unfortunately, for such sampling rate, the attitude control exhibits small oscillations. Both controls were quantized over 1000 levels. The pulse width modulated control implementation relating the pulse width control to the continuous control F_δ and u_Δ is based on a generic equation for PWM of control u

$$u_{pulse} = deaedband(0.4, (1 + \sin(2\pi f_{pulse} t)u) \quad (22)$$

5. SIMULATION

5.1 Simulation set up

The parameters of generic dual-thrust controlled endo-atmospheric missile are given in the Table 1

Table1 Missile Characteristics

Variable name	Description	Value
m	Missile mass (kg)	100
I_{yy}	Pitch moment of inertia (kg m^2)	4
V	Missile longitudinal velocity (m/s)	3500
C_{L_α}	Lift gradient coefficient	2
a_α	Distance of the aerodynamic center from the Cg along -x body axis (m)	0
a_δ	Distance of the application point of attitude thrust from the Cg along –x body axis (m)	-0.3
a_Δ	Distance of the application point of divert thrust from the Cg along x-body axis (m)	-0.01
$Tmax_\delta$	Nominal magnitude of attitude thrust (N)	3000
$Tmax_\Delta$	Nominal magnitude of divert thrust (N)	10000

D_α	Maximum magnitude of aerodynamic lift disturbance	0.2
D_δ	Maximum magnitude of attitude thrust disturbance	0.3
D_Δ	Maximum magnitude of divert thrust disturbance	0.3
S	Reference surface (m^2)	0.0707
$\bar{C}$	Reference length (m)	0.3
τ	Attitude and divert thruster time constant (s)	0.005

Flight altitude, and kill vehicle velocity are 25000 m and 3500 m/s respectively. Commanded normal acceleration is presented in Figure 3, with peak commanded acceleration three times larger than the divert acceleration achievable with Cg thruster alone. The reference acceleration profile is generated by a second order filter (4) with a characteristic frequency $\omega = 30$ rad /s. Initial flight path angle, altitude and velocity are given by $\gamma(0) = 0.5\, rad; z(0) = 25000\, m$

$V(0) = 3500\, m/s$. For simplicity sake, longitudinal velocity is assumed to be constant. Normalized thruster responses are represented by first order transfer functions as in (20).

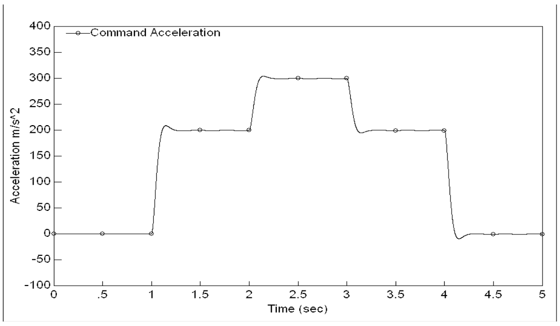

Fig. 3 Commanded acceleration

The thruster dynamic (20) are included in the nonlinear simulation model. Using this simulation scenario we are going to show that it is possible to track a commanded acceleration three times as large as achievable by divert thruster alone and still achieve this challenge combining the aerodynamic maneuver and the divert thrusters, thereby showing that that combination brings together the best of both words, that is the additional acceleration achievable with the aerodynamic maneuver and the time constant permitted by divert thrusters.

5.2 Simulation Results

The scenario was run for the two control implementations. In the first set of results associated to the sample-and hold option no angle-of-attack saturation limits are imposed. The second set of results, associated to the pulse-width modulation implementation was obtained with an angle-of-attack saturation $\alpha_{\max} = 0.5; \alpha_{\min} = -0.5$. Corresponding results are interesting because saturation conditions are often associated to control losses. As performances obtained are identical only the former set of results is presented. The most significant result is evidently the quasi-perfect tracking of prescribed flight path angle, represented in Fig. 4. The angle-of-attack tracking performance that is shown Figure 5, exhibits an almost perfect tracking with the exception of short intervals, where control authority is not sufficient to maintain the sliding mode. Examination of the plot of the α-sliding surface, Figure 6, shows that at times 1, 2, 3 and 4 sec, σ_α exhibits a large growth due to lack of control authority. The same analysis is conducted on σ_γ the γ-surface is shown Fig. 7. Based on Fig. 8 one can note that the peaks of the equivalent value of attitude thrust is larger than the value of the divert thruster, which explains why the attitude thruster is more prone to saturation.

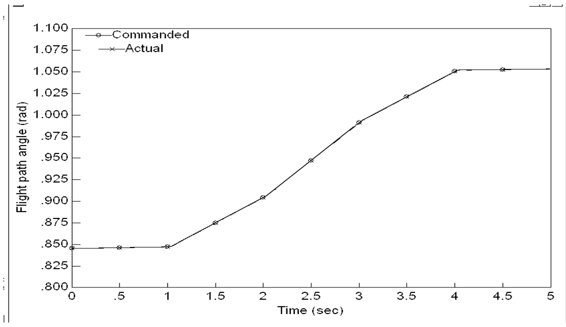

Fig. 4 Flight path angle tracking with sample-and-hold implementation

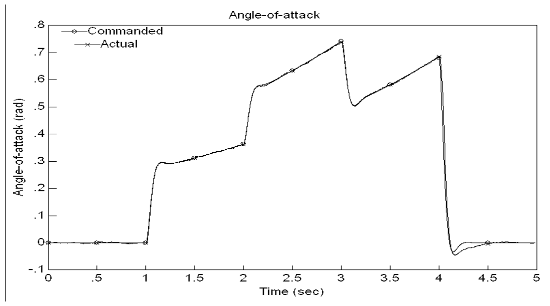

Fig. 5 Angle-of-attack with sample-and-hold implementation

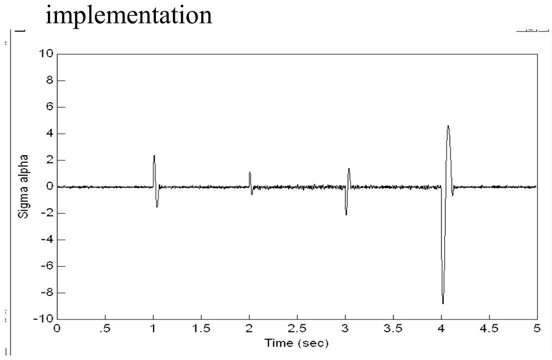

Fig. 6 Attitude σ_α with sample-and-hold implementation

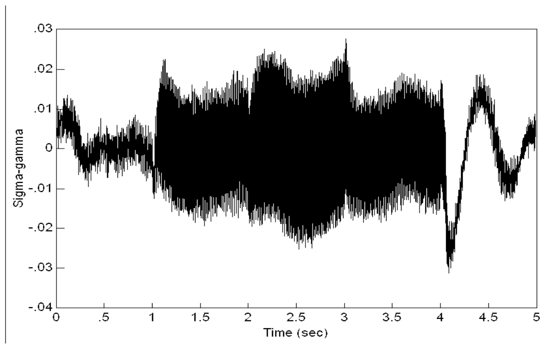

Fig. 7 Divert σ_γ with sample-and-hold implementation

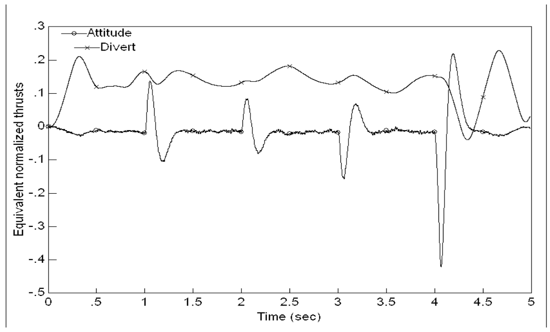

Fig. 8 Equivalent attitude and divert normalized control amplitudes with sample-and hold-implementation

6. CONCLUSIONS

A new approach to the design of a kinetic-kill longitudinal autopilot steering the missile trajectory by the combination of aerodynamic lift and divert-thrusters and with its attitude oriented by attitude-thrusters is presented. The pitch plane autopilot design is based on high order sliding mode control. The combination of thruster and aerodynamic maneuvers presents the control designer facing with up to 30% relative model uncertainties caused by the strong interaction between the thruster jets and the shock wave system around the body, compounded by the 200 Hz firing rate of which makes the real-time estimation of the uncertainties by the disturbance observers very difficult. The paper proposes a control architecture where the missile angle-of-attack is steered to create a cooperative disturbance to the lateral divert control. The controls of the angle-of-attack and of the flight path angle are both based on second order sliding mode control. Performance obtained with preferred pulse-width modulation control is similar to the performances that would be obtained using more complex continuous thrusters. The design shows a remarkable robustness to very rapidly changing model uncertainties created by thruster firings. The control design finally is simple and requires very limited knowledge of the plant model.

REFERENCES

Chen, W.H. (2002). Observer Enhanced Dynamic Inversion Control of Missiles. *AIAA Journal of Guidance Control and Dynamics* **25**-1, 161-166.

Edwards, C. and Spurgeon, S (1998) *Sliding Mode Control: Theory and Applications,* Taylor & Francis, Bristol.

Johnson, C.D. (1973). Stabilization of Linear Dynamical Systems with Respect to Arbitrary Linear Subspaces *Journal of Mathematical Analysis and Applications* **44**-1, 173-185.

Kennedy, Walker, B. Mikkelsen, C (1998). AIT Real Gas Divert Jet Interactions; Summary of Technology. AIAA Paper 98-5188, 225-234.

Levant, A. (1998). Robust exact differentiation via sliding mode technique. *Automatica,* 34-3, 379-384.

Levant, et al. (2000), Aircraft Pitch Control via Second Order Technique. *AIAA Journal of Guidance, Control, and Dynamics,* **23**-4, 586-594.

Levant, A. (2001). Universal single-input-single-output (SISO) sliding-mode controllers with finite-time convergence. *IEEE Transactions on Automatic Control*, **46**-9, 1447–1451.

Shama, J. and Cloutier, J. (1993). Gain Scheduled Missile Autopilot Design Using Linear Parameter Varying Transformations. *AIAA Journal of Guidance Control and Dynamics* **16**-2.

Shtessel, Y. Hall, C. and Jackson, M. (2000). Reusable Launch Vehicle Control in Multiple Time Scale Sliding Modes *AIAA Journal on Guidance, Control, and Dynamics,* **23**-6, 1013-1020.

Shtessel, Y.B., Buffington, J. and Banda, S. (2002). Tailless Aircraft Flight Control Using Multiple Time Scale Re-configurable Sliding Modes. *IEEE Transactions on Control Systems Technology*, **10**-2,, 288-296.

Shtessel, Y.B. and Shkolnikov, I A. (2003). Aeronautical and Space Vehicle Control in Dynamic Sliding Manifolds. *International Journal of Control,* **76**-9/10, 1000-1017.

Shtessel, Y.B. and Shkolnikov, I.A. (2003). Integrated Guidance and Control of Advanced Interceptors Using Second Order Sliding Modes. *Proceedings of the Conference on Decision and Control,* 4587-4592.

Shkolnikov. I.A. and Shtessel, Y. B. (2001). Aircraft Nonminimum Phase Control in Dynamic Sliding Mode Manifolds. *AIAA Journal of Guidance, Control, and Dynamics,* **24**-3.

Shkolnikov, I.A. Shtessel, Y.B. and Lianos, D. (2001). Integrated Guidance-Control System of a Homing Interceptor: Sliding Mode Approach. *Proceedings of the Conference on Guidance, Navigation, and Control,* AIAA Paper 2001-4218.

Sun, X.D. Woodgate, K.G. and Allwright, J.C. (1996) Nonlinear Inverse Dynamics Control of Aircraft using Spoilers. *AIAA Journal of Guidance Control and Dynamics* **19**-2, 475-482.

Tournes, C. and Shtessel, Y.B. (1996) Sliding Mode Control Applied to Aircraft Control. *Proceedings of the AIAA Guidance Navigation and Control Conference,* AIAA Paper 96-3692.

Tournes, C. and Shtessel, Y.B. (1996) Non Minimum Phase Output Tracking Using Dynamic Sliding Mode Manifolds. *Proceedings of the Conference on Decision and Control,* 2071-2076.

Tournes, C. and Johnson, C.D. (1998). Aircraft Guidance and Control Using Subspace Stabilization Control Techniques. *Proceedings of the AIAA Guidance Navigation and Control Conference,* AIAA Paper 98-4123.

Tournes, C. Landrum, B., Shtessel, Y. B., and Hawk, C. (1998) Ramjet Powered Reusable Launch Vehicle Control by Sliding Mode. *AIAA Journal of Guidance Control and Dynamics* **21**-3.

Tournes, C., Paschal N. and Wilkerson, P.W. (2001) Integrated Terminal Guidance and Automatic Pilot Using Subspace-Stabilization. *Proceedings of the AIAA Guidance Navigation and Control Conference.*

Utkin, V. Guldner, J. and Shi, J. (1999) *Sliding Modes in Electromechanical Systems*, Taylor and Francis, London.