10th ANKARA INTERNATIONAL AEROSPACE CONFERENCE **AIAC-2019-030**
18-20 September 2019 - METU, Ankara TURKEY

ADAPTIVE TUNING OF DEFENDING MISSILE UNDER ENVIRONMENTAL CONSTRAINTS

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ABSTRACT

In this research, an adaptive controller is tuned and applied to proportional navigation system under environmental constraints. The purpose of this controller is to keep the stability of the missile during flight time, while the missile is affected by environmental effects such as gravity and drag force caused by the air. The proposed control system is tuned using particle swarm optimization (PSO) method, and the fitness function of the PSO technique is optimized to reduce the miss distance value of the missile at impact time, as well as to minimize the deviation error of the missile from the right track between the missile and the target The simulation results proved the stability of the missile in terms of trajectory stability during flight, miss distance, incidence angle, normal acceleration and radar's gimbal angle stability and tracking performance.

INTRODUCTION

Proportional Navigation Guidance (PNG) is the most used system for guiding defending missiles, however, when dealing with missiles that changes its altitude during flight times, the gains of the proportional navigation guidance system should also be changed to accommodate to the environmental effects such as air drag force and gravity effects. It has been found that these effects could be obtained by knowing some states of the missile such as incidence angle, and attitude of the missile. In this research, an adaptive controller is applied to the proportional navigation system and tuned with particle swarm optimization in order to stabilize the missile during flight time. A two-dimensional look-up table is constructed for different values of incidence angles and missile speed. Then, a particle swarm optimization method is used to obtain the required gains for the three-loop proportional navigation guidance system. Research in [Shah, Samar, & Bhatti, 2010] aimed to design an adaptive control system for controlling the roll behavior of a short-range missile. The main problem investigated by this paper is to design a robust control system that could maintain the roll angle to a value near to zero. The adaptive controlling method is used to control the missile under varying launch angle as well as dynamic pressure. Also, the robustness of the control system is tested under aerodynamic effects and other disturbances. The simulation results proved the ability of the controller in rejecting aerodynamic effects and the other disturbances while keeping the roll angle to a value near to

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zero. In [Xiangrong Tong, Hongchao Zhao, & Guohu Feng, 2006] an overload control system is proposed to control an anti-warship missile. The purpose of this controller is to remove the non-minimum phase behavior of missile overload output. By this control technique, an adaptive global sliding mode method is proposed in order to control the missiles' modified output system, and it's used to evaluate the upper bound value of the lumped uncertainty. Also, a simulation example is performed to demonstrate the validity of the proposed system. In [Yang, Li, & Shi, 2009] a self-adaptive fuzzy proportional integral derivative control system is used to control a longitudinal channel for an autopilot system. This autopilot system is integrated within unmanned aerial vehicles (UAV) and combined with the traditional PID control system along with a fuzzy controlling method. The simulation results proved the simplicity of integrating the self-adaptive fuzzy PID. Also the proposed controller is compared with the traditional PID controller in terms of the dynamic performance of the controlled object, and the proposed controller also could be applied to a time-varying object and nonlinear systems such as UAV. In [Yue Zhang, 2011], the three-loop adaptive autopilot system is researched in order to guarantee the precision of the initial launching of the trajectory. By this research, the autopilot system is simplified and a three-loop adaptive control method and stability is designed. The simulation results showed that the proposed system was effective in controlling the disturbance of the initial launching and adapt to the change in dynamic characteristics as well as keeping the control stability. In [Shi & Zhao, 2017], the uncertain aerodynamics of the spinning projectile autopilot is investigated. A robust feedback adaptive control system is designed and applied for a double-channel spinning projectile. In order to accomplish that, the first step is to develop a dynamic model for the projectile, and an adaptive controller is applied to the autopilot system. Then a robust compensation system is designed and integrated to the adaptive controller to accomplish its robustness. The closed loop adaptive system performance is tested through simulations. In addition, the simulation results proved that the autopilot system eliminated the high frequency oscillations. Apart from these researches, our proposed intelligent tuning technique with its simple implementation method, proved to have a very good stability for the missile during flight time through the second norm minimization as well as excellent hitting accuracy by integrating the miss distance optimization technique into the fitness function of the PSO tuning method. The rest of the paper is organized as follows: Section 2 introduces the principles of the proportional navigation guidance system and the integrated adaptive controller. Section 3 includes the system design and tuning method. Section 4 contains the simulation results of the proposed control system.

METHOD

Proportional Navigation Control System

The main idea of the proportional navigation control system is to find lateral acceleration value in order to rotate the missile and correct the deviation of the missile velocity from the line between the missile and the target. By this concept, the proportional navigation control system will work on maintaining the rotation rate of the missile and line of sight equals to each other. The engagement geometry of a missile distant from a target could be shown in Figure 1.

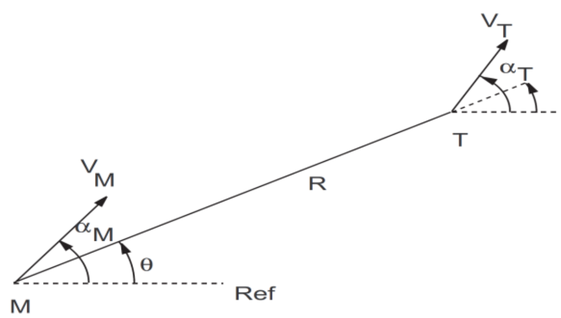

Figure 1: Engagement geometry of missile distant from the target.

The normal acceleration that should be exerted by the missile in order to rotate the missile and keep its velocity direction aligned with the line of sight could be expressed using the following equations:

$$V_r = \dot{R} = -V_M cos(\alpha_M - \theta) + V_T cos(\alpha_T - \theta)$$

$$V_\theta = \dot{\theta R} = -V_M sin(\alpha_M - \theta) + V_T sin(\alpha_T - \theta)$$

$$\dot{\alpha_M} = \frac{a_M}{V_M}$$

$$a_M = V_M N \dot{\theta}$$

where R is the range between the target and the missile, a_M is the lateral acceleration exerted to rotate the missile, α_M is the angle between the direction of the missile and the axis of reference, α_T is the angle between the direction of the target and the axis of reference, N is referred to as the navigation constant and θ is the angle formulated by the line between the missile and target, and the reference axis.

Missile Dynamics

The dynamics of the missile are formulated by obtaining the pitch moment as well as the forces applied on each axis of the missile body. The forces on each axis and the moment could be obtained using the following equations:

$$F_x = 0.5 C_x \rho V^2 S_{ref}$$

$$F_z = 0.5 C_z \rho V^2 S_{ref}$$

$$M = 0.5(C_m + q)(\rho V^2 S_{ref} * D_{ref})$$

where Fx is the force exerted on the x-axis of the missiles' body, Fz is the force exerted on the z-axis of the missiles' body, M is the missiles' pitch moment, Sref is the cross-sectional area for the reference of the missile, Dref is the diameter of the missiles' body, V is the velocity exerted by the missile, Cx and Cz are multiplication coefficients that are related to the incidence angle and velocity of the missile. These coefficients are needed to change during flight time depending on the velocity and incidence angle of the missile and they are necessary to stabilize the missile during flight. The pitch moment and forces exerted on each axis are then employed to obtain the acceleration exerted by the missile with respect to each axis. Forces and pitch moment values are then used to evaluate the acceleration of the missile at each axis, and the rate of the missiles' incidence angle, by that, the motion of the missile is formulated as shown in the following equations:

$$A_x = -q v_x - g sin(\theta) + \frac{F_x}{m}$$

$$A_z = q v_z + g cos(\theta) + \frac{F_z}{m}$$

$$\dot{q} = \frac{M}{I}$$

$$\dot{\theta} = q$$

where θ is the pitch angle, Ax is the acceleration exerted by the missile at the x-axis, Az is the acceleration exerted by the missile at the z-axis, m is the mass of the missile, q is the rotation rate exerted by the missile, g is the gravity force, vx is the velocity of the missile with respect to the x-axis, vz is the velocity of the missile with respect to the z-axis, and I is the inertia exerted by the missile.

Autopilot Systems

The two-loop and three-loop autopilot systems have been used for guiding missiles in recent years [Fan & Xiao-qian, 2015]. The two-loop autopilot control system employs two loops to measure the motion signal of the missile and feed it back to the forward path of the autopilot control system. The first loop is concerned with body rate signal, which is measured by a rate gyro and then fed it back. The second loop is concerned with the missile acceleration, measured by an accelerometer, and this loop is the main feedback loop. The three-loop guidance system has been designed especially to be integrated for the radar-guided missile in order to remove the coupling effect introduced by a radome and parasitic loop. The conventional two-loop autopilot guidance system contains a rate-damping loop that acts as a damper and accelerometer loop that provides the lateral acceleration used to rotate the missile. However, we can convert this two-loop autopilot system to three-loop autopilot just by adding a synthetic stability loop [Abd-elatif, Qian, & Bo, 2016]. The flight dynamic of the missile is highly dependent on the aerodynamics of the missile which is affected significantly with altitude and velocity of the missile [Kantue, 2017]. In this research, the tracking performance of the missile is investigated and optimized using an adaptive control technique. The problem is to design an autopilot system that could generate a normal acceleration by deflecting the fins of the missile, this normal acceleration is responsible for putting the missile on the right track in order to hit the target efficiently without oscillation. Figure 2 shows a standard three-loop autopilot system.

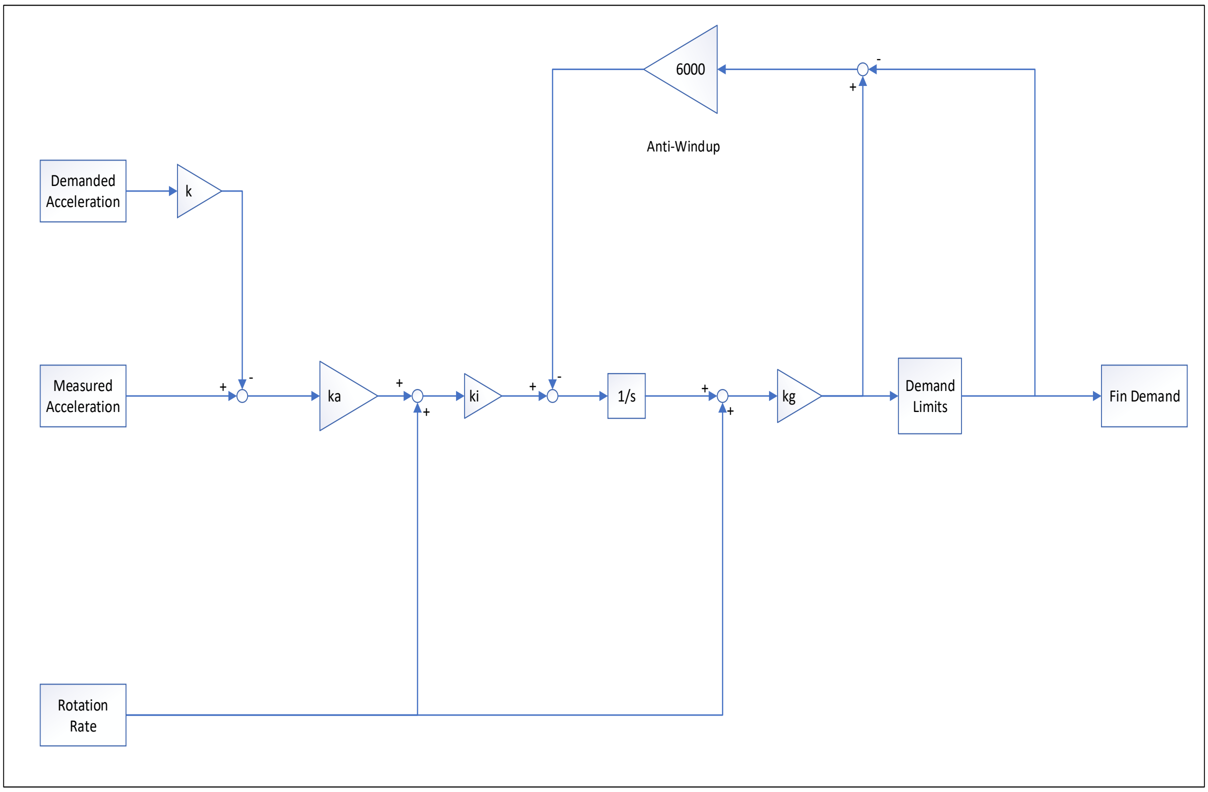

Figure 2: The standard three-loop autopilot system.

Tuning the Proposed Adaptive Controller

The gains of the proposed controller have been tuned using particle swarm optimization technique along with H_2 Norm values. Firstly, the deviation of the missile velocity from the line between the missile and target at each time step is evaluated and saved in a vector (Error_Vector) as seen in Figure 3.

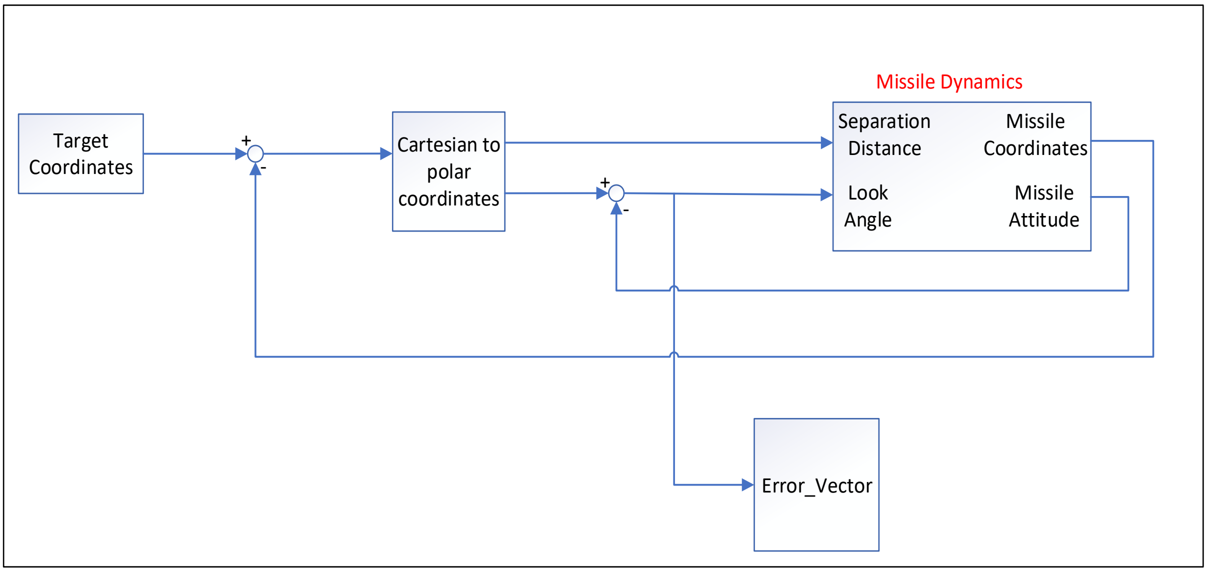

Figure 3: The feedback control system of the missile.

Then, particle swarm optimization method is used to fill a lookup table for each gain in the three-loop autopilot system as seen in Figure 4.

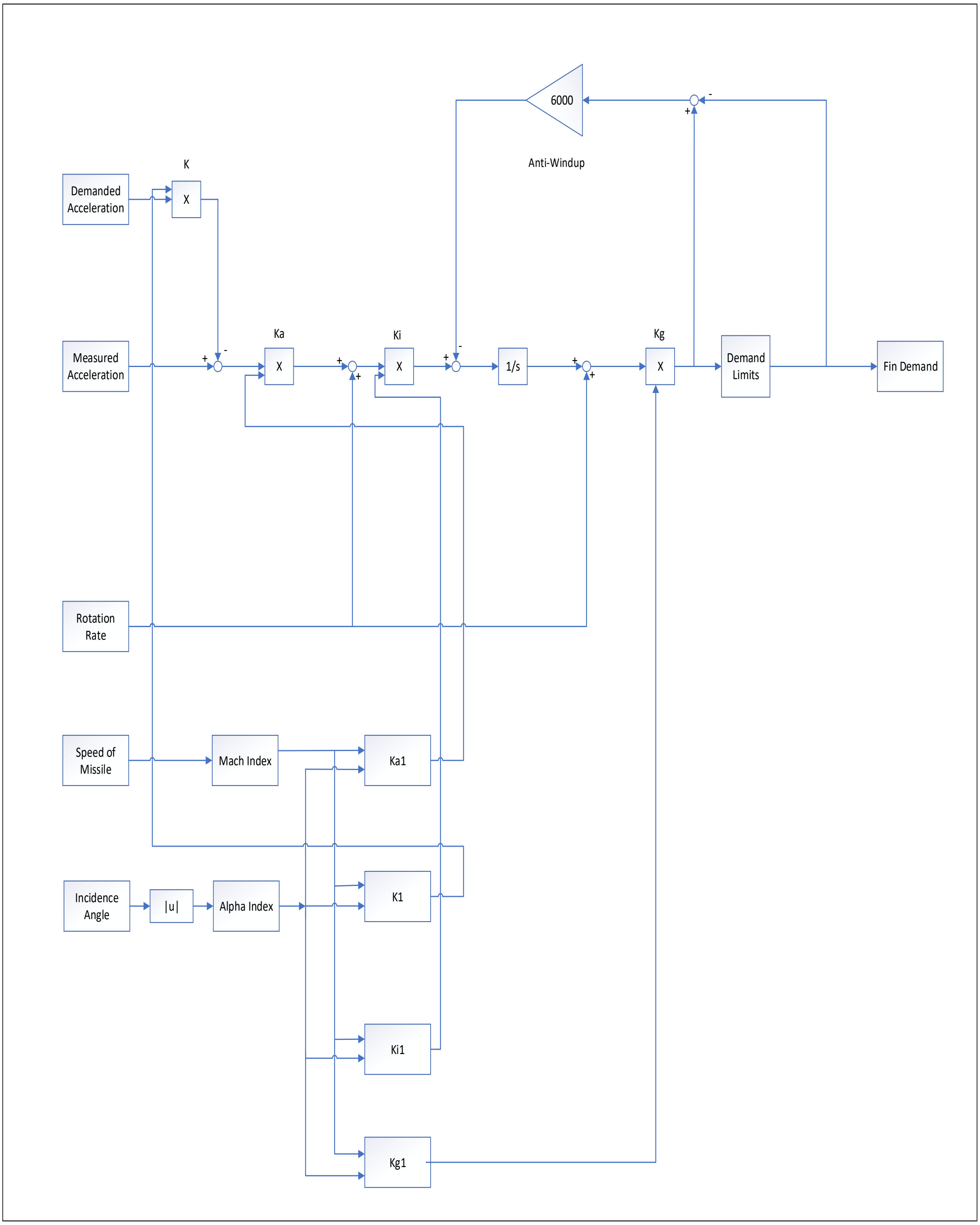

Figure 4: The proposed adaptive autopilot system.

After that, the second norm (H_2) is applied to Error_Vector values, and the miss distance value is also measured during the impact time. The fitness function of the particle swarm optimization as shown in Algorithm 1 will work to minimize the values of miss distance and the second norm (H_2) by searching the space for the best values of the lookup table for each gain of the three-loop autopilot system.

Algorithm 1: The intelligent tuning algorithm of the adaptive controller

Input: Random values for the parameters of the lookup tables.

Output: Optimal values for the parameters of the lookup tables.

Start:

Step 1: Set the initial values for the parameters of the lookup tables.

Step 2: Perform a test on the closed-loop system in order to evaluate the miss distance and the second norm values based on the previous controller parameters.

Step 3: Minimize the second norm and miss distance values using PSO.

Step 4: Obtain the final values of the lookup tables parameters which yield the smallest possible values of the second norm and miss distance of the missile during flight test.

End.

The following matrices express the sliding mode gains applied to the autopilot control system as functions of velocity and incidence angle.

$$\mathrm{Ka}(\mathrm{V},\alpha) = \begin{bmatrix} \mathrm{Ka}_{1,1} & \mathrm{Ka}_{1,2} & \cdots & \mathrm{Ka}_{1,5} & \mathrm{Ka}_{1,6} \\ \mathrm{Ka}_{2,1} & \mathrm{Ka}_{2,2} & & \mathrm{Ka}_{2,5} & \mathrm{Ka}_{2,6} \\ \vdots & & \ddots & & \vdots \\ \mathrm{Ka}_{8,1} & \mathrm{Ka}_{8,2} & \cdots & \mathrm{Ka}_{8,5} & \mathrm{Ka}_{8,6} \\ \mathrm{Ka}_{9,1} & \mathrm{Ka}_{9,2} & & \mathrm{Ka}_{9,5} & \mathrm{Ka}_{9,6} \end{bmatrix} \tag{20}$$

$$\mathrm{K}(\mathrm{V},\alpha) = \begin{bmatrix} \mathrm{K}_{1,1} & \mathrm{K}_{1,2} & \cdots & \mathrm{K}_{1,5} & \mathrm{K}_{1,6} \\ \mathrm{K}_{2,1} & \mathrm{K}_{2,2} & & \mathrm{K}_{2,5} & \mathrm{K}_{2,6} \\ \vdots & & \ddots & & \vdots \\ \mathrm{K}_{8,1} & \mathrm{K}_{8,2} & \cdots & \mathrm{K}_{8,5} & \mathrm{K}_{8,6} \\ \mathrm{K}_{9,1} & \mathrm{K}_{9,2} & & \mathrm{K}_{9,5} & \mathrm{K}_{9,6} \end{bmatrix} \tag{21}$$

$$\mathrm{Ki}(\mathrm{V},\alpha) = \begin{bmatrix} \mathrm{Ki}_{1,1} & \mathrm{Ki}_{1,2} & \cdots & \mathrm{Ki}_{1,5} & \mathrm{Ki}_{1,6} \\ \mathrm{Ki}_{2,1} & \mathrm{Ki}_{2,2} & & \mathrm{Ki}_{2,5} & \mathrm{Ki}_{2,6} \\ \vdots & & \ddots & & \vdots \\ \mathrm{Ki}_{8,1} & \mathrm{Ki}_{8,2} & \cdots & \mathrm{Ki}_{8,5} & \mathrm{Ki}_{8,6} \\ \mathrm{Ki}_{9,1} & \mathrm{Ki}_{9,2} & & \mathrm{Ki}_{9,5} & \mathrm{Ki}_{9,6} \end{bmatrix} \tag{23}$$

$$\mathrm{Kg}(\mathrm{V},\alpha) = \begin{bmatrix} \mathrm{Kg}_{1,1} & \mathrm{Kg}_{1,2} & \cdots & \mathrm{Kg}_{1,5} & \mathrm{Kg}_{1,6} \\ \mathrm{Kg}_{2,1} & \mathrm{Kg}_{2,2} & & \mathrm{Kg}_{2,5} & \mathrm{Kg}_{2,6} \\ \vdots & & \ddots & & \vdots \\ \mathrm{Kg}_{8,1} & \mathrm{Kg}_{8,2} & \cdots & \mathrm{Kg}_{8,5} & \mathrm{Kg}_{8,6} \\ \mathrm{Kg}_{9,1} & \mathrm{Kg}_{9,2} & & \mathrm{Kg}_{9,5} & \mathrm{Kg}_{9,6} \end{bmatrix} \tag{23}$$

SIMULATION RESULTS

The proposed sliding mode control has been applied to the three-loop autopilot system of a missile and then tested with the following performance metrics: stability of missile trajectory when following the target, normal acceleration performance in tracking demanded normal acceleration, Incidence angle behavior and fin demands. Figure 5 shows 0the motion of the missile when tracking the target. As shown in the figure, the missile shows smooth motion during flight time.

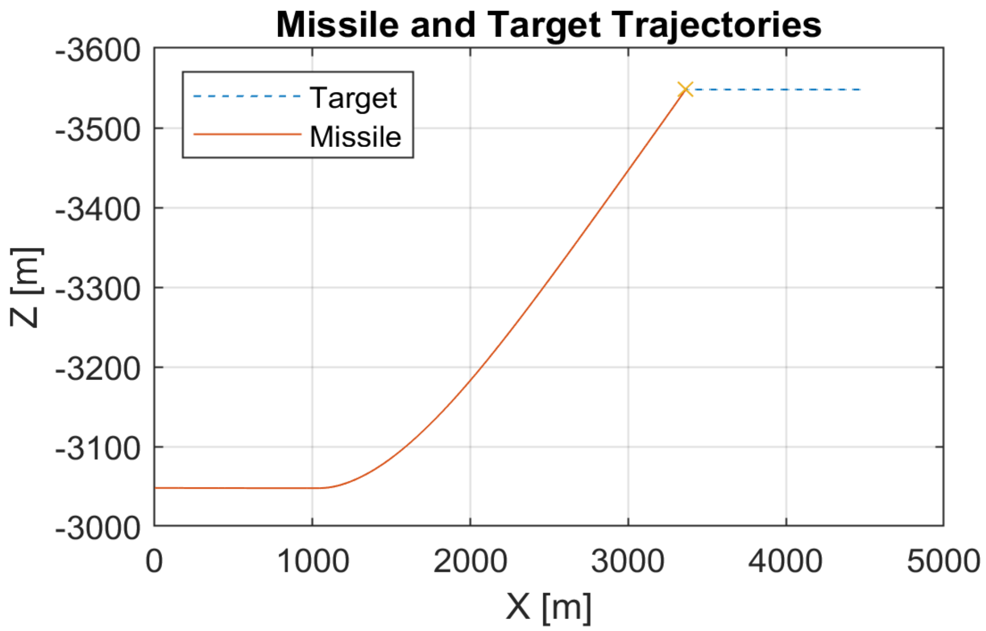

Figure 5: Missile and target trajectories.

Figure 6 shows the demanded normal acceleration generated to direct the missile towards the target. As shown in the figure, when the target is detected by the radar system of the missile at about 0.9 seconds, the missile started to change the normal acceleration demands to rotate the missile towards the target. Also, the missile was able to track the generated acceleration demands efficiently. The miss distance value that is measured during flight time was $8.7908*10^{-4}$ and the second norm value was 4.6281.

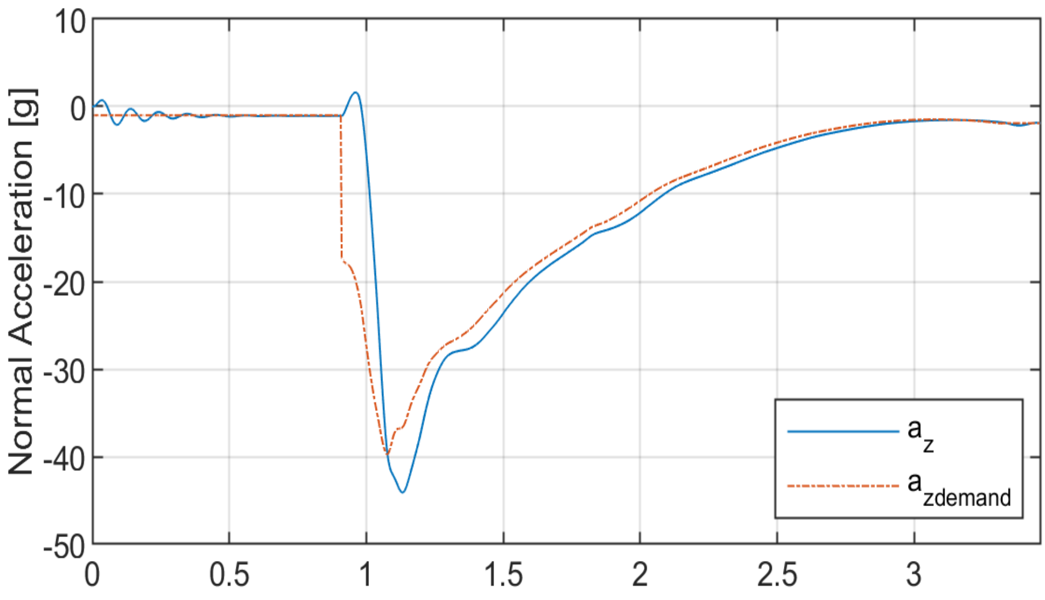

Figure 6: Normal acceleration vs demanded normal acceleration.

Figure 7 shows the true look angle vs. gimbal, as seen in the figure, the gimbal angle of the radar system is rotating between 30 to -30 degrees to search for the target, when the target seconds at about 0.9 seconds, the radar was able to keep track of the true missile location efficiently without oscillations during flight.

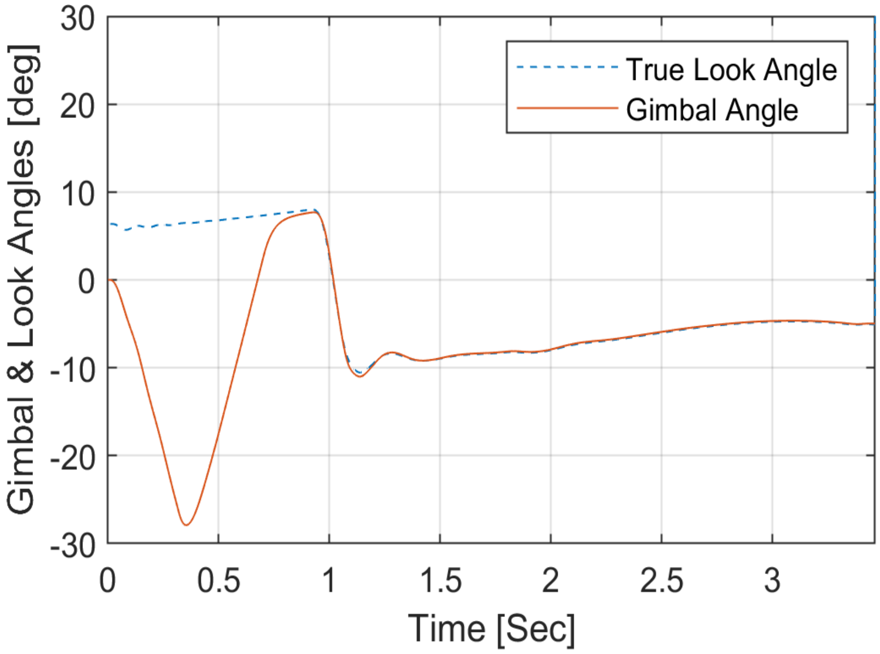

Figure 7: True look angle vs gimbal angle.

Figure 8 shows the incidence angle of the missile during flight. As seen in the figure, the incidence angle changed suddenly at about 0.9 seconds when the target is found by the missile in order to direct the missile toward the target. Then it started to decrease smoothly until the missile impacted with the target.

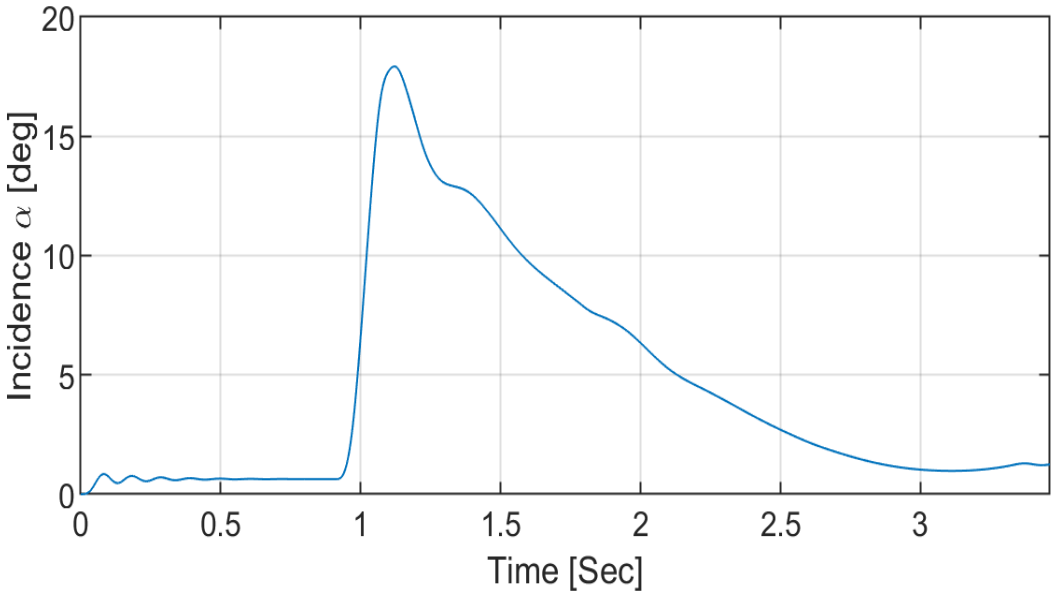

Figure 8: Incidence angle of the missile during flight.

CONCLUSIONS AND FUTURE WORK

By this research, an adaptive sliding mode controller is designed for a three-loop autopilot system using particle swarm optimization method. The proposed controller is applied for a defending missile and the simulation results proved the stability of the controller during flight time in terms of incidence angle stability, tracking ability of the radar system, normal acceleration stability and its performance in tracking the demanded normal acceleration and the second norm value. The accuracy of the missile in hitting the target is also measured and proved in term of miss distance value. As future work, the controller could be designed to work under noise and external disturbances.

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