11th ANKARA INTERNATIONAL AEROSPACE CONFERENCE **AIAC-2021-054**
8-10 September 2021 - METU, Ankara TURKEY

DEVELOPMENT OF A ROLL COMMAND CONTROLLER FOR HEADING AUTOPILOT OF AUTONOMOUS FIXED WING AIRCRAFT

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ABSTRACT

In the last decade, guidance technologies have been integrated with unmanned aerial vehicles to decrease human factors. Guidance methods are developed depending on aircraft kinematics and/or control methods in order to reach given coordinates autonomously with minimum time, fuel consumption, and cross-error value. In this paper, a new high-simplicity lateral guidance method is developed taking into account heading angle, rolling angle, and lateral and longitudinal coordinates. A new approach is used to calculate commanded roll angle to guide fixed-wing aircraft towards waypoints with the least possible maneuvers. The paper also includes simulation results that prove the efficiency and applicability of this method.

Keywords: *Autonomous Flight, Waypoint Guidance Method, Fixed Wing Aircraft, Autopilot Design, Roll Command, Outer loop control.*

INTRODUCTION

With the development of unmanned aircraft systems, fixed-wing autonomous flight control became an important subject in both academic and military fields. Near future technologies such as urban air vehicle transportation and delivery drones rise the necessity of autonomous flight control. While some autonomous flight methods use geometric algorithms to guide aircraft toward desired coordinates, so-called waypoints, other methods apply optimal control to find the best path to reach waypoints. [Medagoda and Gibbens, 2010] developed a method where aircraft chases a synthetic waypoint that runs along a path toward the desired waypoint. [Park et. al, 2004] invented a new method for tracking non-linear trajectories. [Sujit et. al, 2013] discussed different methods of classical path following like e.g. Line-of-Sight, Vector Field-based algorithms, Carrot Chasing, and LQR Path Following algorithm. [Beard and McLain, 2012] explained in detail Vector Field-based algorithm application is 3D space. Other researchers developed controllers to follow the optimum path toward waypoints [Ailon and Zohar, 2010] [Breivik and Fossen, 2005] [Ratnoo et. al, 2011]. In [He et. Al, 2020], a method of optimal control and guidance law combination is developed for better energy performance and minimum waypoint following effort.

The main purpose of this paper is to develop a new waypoint guidance method that uses simple sensors and data and avoids the complexity of guidance methods and limitations of sensors. For that reason, the newly developed method depends only on heading angle, coordinates, and roll angle. This guidance method contains 2 factors that need to be optimized in order to head waypoints directly without oscillating around the path. For this paper, the trial and error method is used for optimization and to show the effect of each factor over waypoint

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approach performance. After that, the method is tested in simulation environment for 10 waypoints.

METHOD

Fixed Wing Aircraft Dynamics

Engineers use the 6DoF model in order to understand fixed-wing aircraft motion. 3 DoF represent linear motion and other 3DoF are angular motion. As represented in Figure 1, aircraft linear velocities are u, v, and w and angular velocities are q (pitching), p (rolling), and r (yawing). In general, two types of axes are used for motion analysis. While the first axes type is relative to the wind vector, the second type is relative to aircraft body axes. 6DoF equation of motion of fixed-wing is represented in [Roskam 1998] as follows:

$$\dot{u} = rv - qw - g\sin\theta + q_d\frac{S}{m}\left[C_X(\alpha) + C_{X_q}(\alpha)\frac{cq}{2V} + C_{X_{\delta e}}(\alpha)\right]$$

$$\dot{v} = pw - ru + g\cos\theta\sin\phi + q_d\frac{S}{m}\left[C_{Y_0} + C_{Y_\beta} + C_{Y_p}\frac{bp}{2V} + C_{Y_r}\frac{br}{2V} + C_{Y_{\delta a}}\delta a\right]$$

$$\dot{w} = qu - pv + g\cos\theta\cos\phi + q_d\frac{S}{m}\left[C_z(\alpha) + C_{Z_q}(\alpha) + C_{Y_p}\frac{cp}{2V} + C_{Z_{\delta e}}(\alpha)\delta e\right]$$

$$\begin{aligned}\dot{p} = {} & \frac{I_{xz}(I_x - I_y + I_z)}{I_xI_z - I_{xz}^2}pq - \frac{I_z(I_z - I_y) + I_{xz}^2}{I_xI_z - I_{xz}^2}qr + \frac{I_z}{I_xI_z - I_{xz}^2}q_d bSC_{l_0} + C_{l_\beta}\beta \\ & + \left[C_{l_p}\frac{bp}{2V} + C_{l_r}\frac{br}{2V} + C_{l_{\delta a}}\delta a\right] + \frac{I_{xz}}{I_xI_z - I_{xz}^2}q_d bS[C_{n_0} + C_{n_\beta}\beta + C_{n_p}\frac{bp}{2V} + C_{n_r}\frac{br}{2V} \\ & + C_{n_{\delta a}}\delta a]\end{aligned} \quad (1)$$

$$\dot{q} = \frac{(I_z - I_x)}{I_y}pr - \frac{I_{xz}}{I_y}(p^2 - r^2) + \frac{1}{I_y}q_d Sc[C_{m_0} + C_{m_\alpha}\alpha + \frac{C_{m_q}c}{2V}q + C_{m_{\delta e}}\delta e]$$

$$\begin{aligned}\dot{r} = {} & \frac{I_x(I_x - I_y) + I_{xz}^2}{I_xI_z - I_{xz}^2}pq - \frac{I_{xz}(I_x - I_y + I_z)}{I_xI_z - I_{xz}^2}qr + \frac{I_{xz}}{I_xI_z - I_{xz}^2}q_d bS * [C_{l_0} + C_{l_\beta}\beta + C_{l_p}\frac{bp}{2V} + C_{l_r}\frac{br}{2V} \\ & + C_{l_{\delta a}}\delta a] + \frac{I_x}{I_xI_z - I_{xz}^2}q_d bS[C_{n_0} + C_{n_\beta}\beta + C_{n_p}\frac{bp}{2V} + C_{n_r}\frac{br}{2V} + C_{n_{\delta a}}\delta a]\end{aligned}$$

where *V* is true speed, *g* is gravity acceleration, q_d is dynamic pressure, *α* is angle of attack, *θ* is pitching angle, *β* is side-slip angle, *Φ* is rolling angle, *δe*, *δa* and *δr* are control surfaces, *q, p* and *r* are angular velocities, *u, v* and *w* are linear velocities, *I* is moment of inertia, *m* is mass, *c* is mean chord, *S* is wing area, *b* is wing span and *C's* are aerodynamic coefficients.

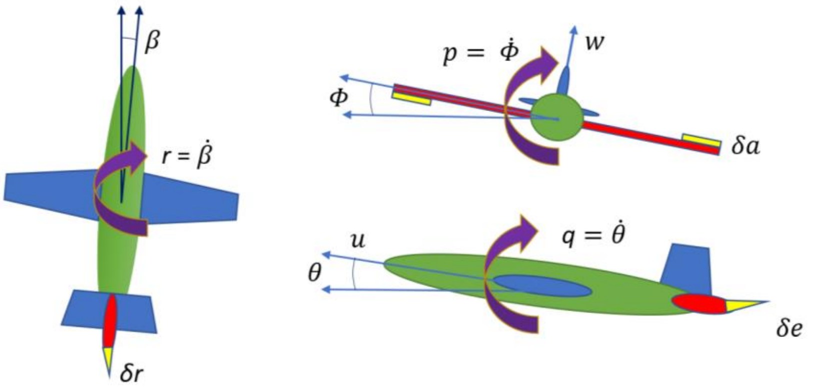

Figure 1. Fixed Wing Aircraft Dynamics

Linearization

Each type of aircraft motion depends on many variables, which means that aircraft motion is a non-linear motion. Analyzing and simulating nonlinear motion is difficult. Linearizing the system around cruise condition's equilibrium point is more practical and acceptable for some systems, fixed-wing aircraft is one of them. Linearization is done using the small perturbations method. In this method, Linear and angular accelerations are accepted to be zero. Free stream velocity is constant and equal to u velocity, which means that v and w velocities are zero. These assumptions are ideal for analyzing aircraft dynamics in cruise conditions.

Transfer functions and state-space matrices are used for linearized systems representation. While transfer functions are preferred in Single Input – Single Output systems, state-space is more efficient in Multi Input – Multi Output systems. For that reason, using state space for a linearized aircraft system is more logical.

State space is represented as follows:

$$\begin{aligned} \dot{x} &= Ax + Bu \\ y &= Cx + Du \end{aligned} \tag{2}$$

where *A* is the system dynamics matrix, *B* is control surfaces dynamics matrix, *C* is output matrix, *D* is feedforward matrix, *x* is state vector, *u* is control surfaces vector and *y* is output vector. For this study, the assumption of *y* = *x*` will be done. Moreover, *C* will be accepted as an identity matrix and *D* will be a zero matrix.

Linearized longitudinal, lateral and directional dynamics of fixed wing is represented using state space as follows:

$$\begin{gathered} \begin{bmatrix} \Delta\dot{u} \\ \Delta\dot{w} \\ \Delta\dot{q} \\ \Delta\dot{\theta} \end{bmatrix} = \begin{bmatrix} X_u & X_w & 0 & x_{\dot{u}} \\ Z_u & Z_w & u_0 & 0 \\ M_u + M_{\dot{w}} Z_w & M_w + M_{\dot{w}} Z_w & M_q + M_{\dot{w}} u_0 & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} \Delta u \\ \Delta w \\ \Delta q \\ \Delta\theta \end{bmatrix} + \begin{bmatrix} X_{\delta e} \\ Z_{\delta e} \\ M_{\delta e} + M_{\dot{w}} Z_{\delta e} \\ 0 \end{bmatrix} [\Delta\delta_e] \\ \begin{bmatrix} \Delta\dot{\beta} \\ \Delta\dot{p} \\ \Delta\dot{r} \\ \Delta\dot{\Phi} \end{bmatrix} = \begin{bmatrix} Y_\beta/u_0 & Y_p/u_0 & -(1 - Y_r/u_0) & g/u_0 \\ L_\beta & L_p & L_r & 0 \\ N_\beta & N_p & N_r & 0 \\ 0 & 1 & 0 & 0 \end{bmatrix} \begin{bmatrix} \Delta\beta \\ \Delta p \\ \Delta r \\ \Delta\Phi \end{bmatrix} + \begin{bmatrix} 0 & Y_{\delta r}/u_0 \\ L_{\delta a} & L_{\delta r} \\ N_{\delta a} & N_{\delta r} \\ 0 & 0 \end{bmatrix} \begin{bmatrix} \Delta\delta_a \\ \Delta\delta_r \end{bmatrix} \end{gathered} \tag{3}$$

where *L, M* and *N* are moment stability derivatives and *X, Y* and *Z* are force stability derivatives. For this linearization, wind relative axes and body relative axes are assumed to be the same [Napolitano, 2012].

Autonomous Fixed Wing Autopilot Design

In this paper, Ryan Navion's aircraft mathematical model is implemented using aerodynamic coefficients from [Nelson, 1997]. PID controller is used. The autopilot system is represented in Figure 2. PID coefficients are tuned to reach an appropriate rising time, settling time, and overshoot ratio for aircraft dynamics and maneuverability. Limitations are applied to controller outputs as aircraft control surfaces have upper and lower deflection limits. Desired inputs are calculated by the Waypoint Guidance Algorithm which calculates desired rolling angle depending on the current coordinates of the aircraft and its heading angle and waypoint location. Waypoint Guidance Algorithm is explained in detail in the next subheading.

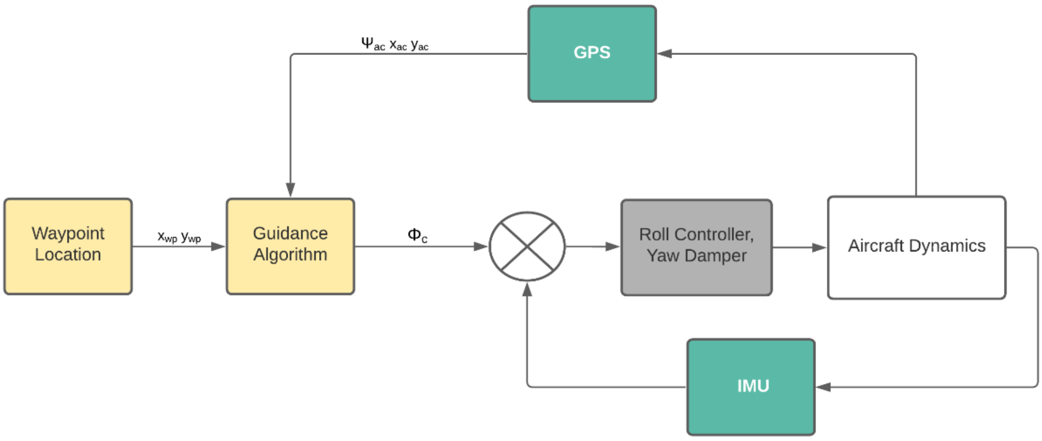

Figure 2. Autopilot Design

Waypoint Guidance

Many approaches types are investigated in literature for waypoint guidance. [Beard and McLain, 2012] used Lyapunov Vector Field to approach waypoints with a required heading angle. Approaching maneuver aggression varies with respect to factors in the vector field equation. The synthetic waypoint guidance method is applied by [Medagoda and Gibbens, 2010] to guide aircraft toward waypoints by following the desired path. This method is widely used in small UAVs flight controllers. In other methods, aircraft is guided using modern control techniques which find the optimum path to reach the waypoint with the possible minimum time or fuel consumption [Ailon and Zohar, 2010] [Breivik and Fossen, 2005] [Ratnoo et. Al, 2011].

In this study, a new method is developed to guide with the possible fewer aileron deflections, in other words, fewer maneuvers. Moreover, the method is designed to be able to guide fixed-wing aircrafts outdoor using GPS (for coordinates and heading angle) and IMU (for rolling angle) data only. The main aim of the new waypoint guidance method is to make the distance between aircraft location and waypoint coordinates goes to zero. Good to mention that longitudinal guidance is not taken into account and aircraft is assumed to fly at constant altitude. Altitude controller is discussed in detail by [Nelson, 1997].

In this paper, heading angle, coordinates, and rolling angle are assumed to be calculated perfectly in the simulation environment. To estimate the relative location of waypoints with respect to aircraft, the following equations are applied:

$$Y_{wp/ac} = -(X_{wp} - X_{ac})\sin\Psi + (Y_{wp} - Y_{ac})\cos\Psi \tag{4}$$

$$X_{wp/ac} = (X_{wp} - X_{ac})\cos\Psi + (Y_{wp} - Y_{ac})\sin\Psi \tag{5}$$

where X_{ac} and Y_{ac} are aircraft location which are estimated using flight path equations from [Napolitano, 2012], Ψ is heading angle and X_{wp} and Y_{wp} are waypoint coordinates, all with respect to earth axes frame.

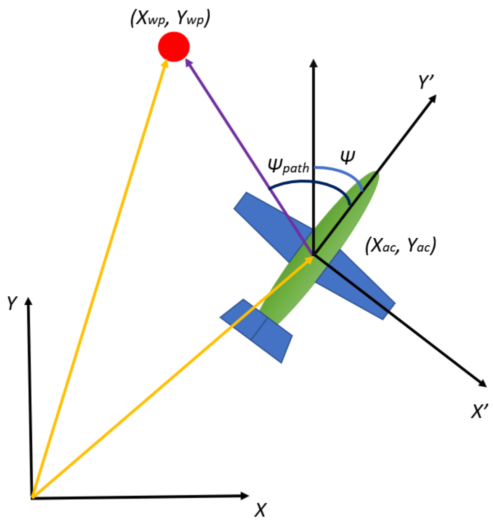

Figure 3. Waypoint Guidance

As it is clear from Figure 3, heading angle is the variable to be controlled in order to reach the waypoint. A relationship between Ψ_{path} and command rolling angle Φ_c must be built where $-\Phi_{max} < \Phi_c < \Phi_{max}$.

$$\Phi_{req} = \left(\frac{1}{\Phi_{max}}\right)^{n-1} \times \Psi_{path}^{n} \times k \tag{6}$$

where Φ_{max} is the maximum banking angle of the aircraft, Ψ is the heading angle of the aircraft and n is the factor of maneuver aggression. Ψ_{path} is the desired heading angle. Ψ_{max} is a value setted by the user. This value makes the aircraft to roll with banking angle of Φ_{max} in the case Ψ_{path} is bigger than Ψ_{max}. *k* represents the relationship between Ψ_{max} and Φ_{max}. *k* will be discussed later in the next section.

It is observable from Figure 4 that the rise in *'n'* leads to lower Φ_{req} angle as Ψ goes to zero in compare with Φ_{req} with higher '*n*' values. This will cause less aggressive maneuvers during waypoint tracking. However, very high *'n'* values may make the aircraft approaches waypoints with higher cross-error value. This shows the importance of optimizing *'n'* factor.

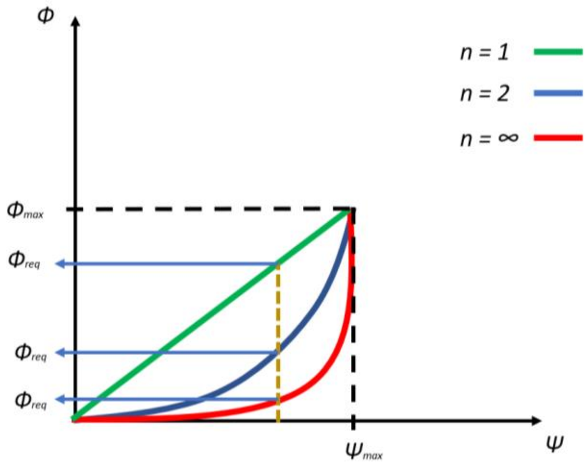

Figure 4. 'n' effect on Ψ ~ Φ relationship

Figure 5 represents the effect of *‘k’* factor. *‘k’* factor equation is:

$$k = \left(\frac{\Phi_{max}}{\Psi_{max}}\right)^{n} \tag{7}$$

Ψ_{max} is responsible of changing *‘k’* as Φ_{max} is totally depending on aircraft dynamics and maneuverability. Higher values of Ψ_{max} leads to smoother trajectory during autonomous flight.

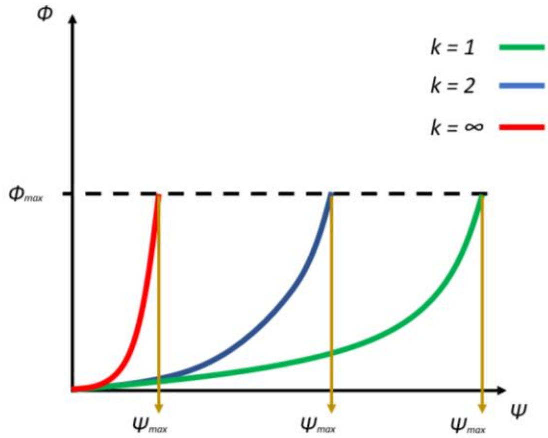

Figure 5. 'k' effect on Ψ ~ Φ relationship

SIMULATION RESULTS

Factors discussed in the previous section need to be optimized in order to reach the better guiding performance. Performance criteria will be less oscillation around the path and less time consuming -or shorter distance-. For this paper, trial and error method will be sued to choose the optimum values. This method will be applied by doing several flight simulations with different values for each factor.

‘n’ factor is responsible of maneuver aggression. In linear relationship between Ψ and Φ, where *n=1*, aircraft reach waypoints with almost zero cross-error distance. Even so, this costs more aggressive maneuvers to reach the right heading angle and leads to oscillation around the path during the approach. In the other side, higher values of *n* guide the aircraft with lower maneuvers. However, high values of *n*, such as 20, are not efficient in choosing the short path and it leads to more time consumed. In fact, simulation results show that *n=2* gives better approaching maneuver in comparison with others.

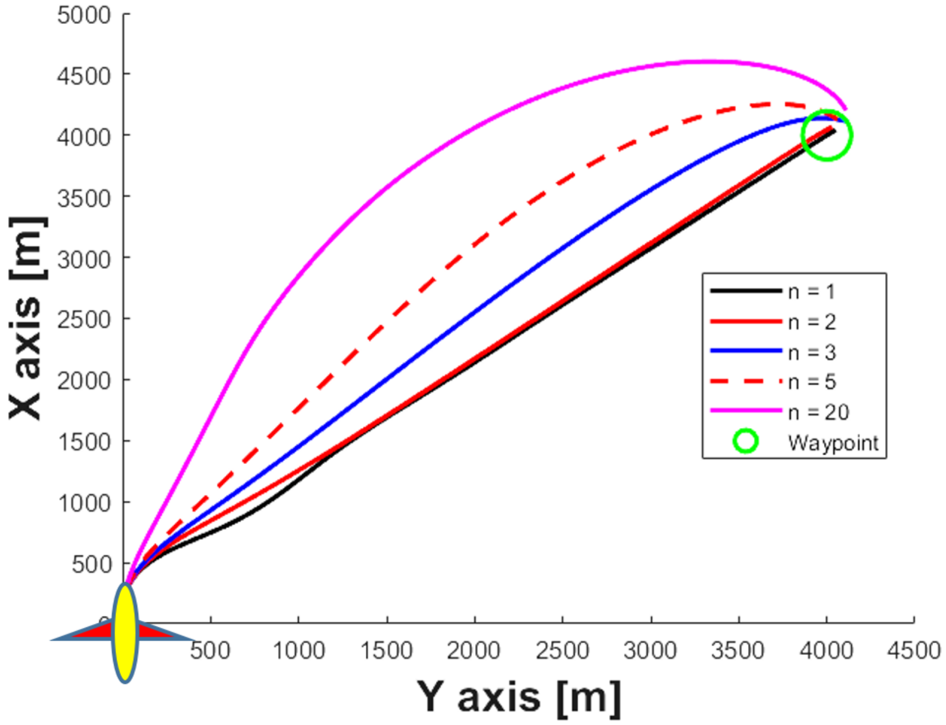

Figure 6. Effect of 'n' factor [Φ_{max}=45˚, Ψ_{max}=45˚]

Ψ_{max} with values less than 15˚ causes higher oscillation around path. This will cause low efficient approach in tracking close waypoints. Yet, higher values such as 90˚ cost uncertainty in reaching the waypoint, which means very high cross error. Simulation results show that 45˚ is the best value for this situation.

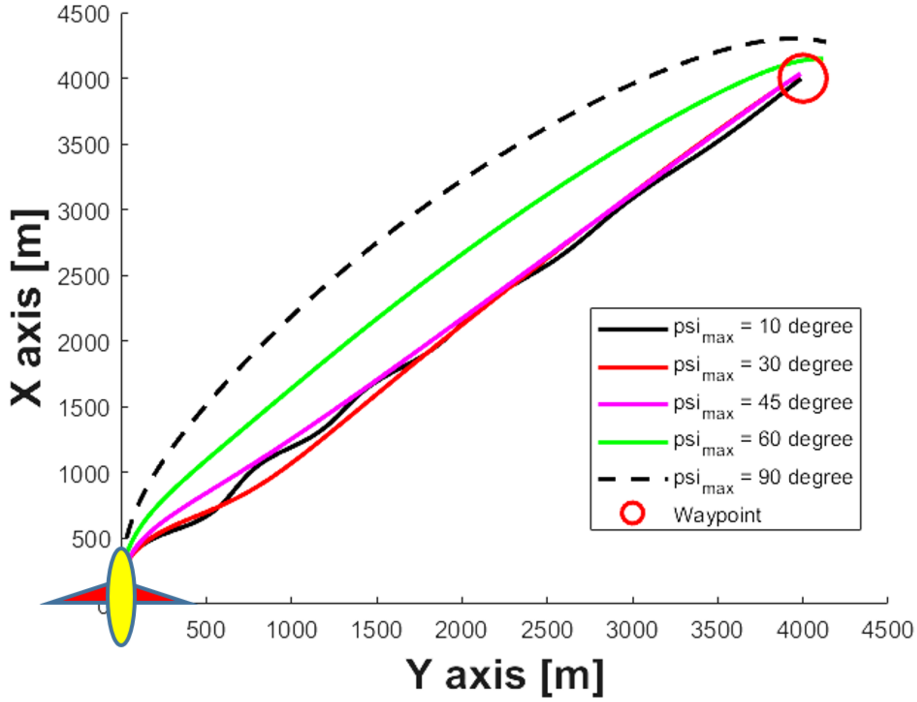

Figure 7. Effect of 'Ψ_{max}' [Φ_{max}=45˚, n=2]

Maximum roll angle varies from aircraft to another. It depends on aircraft dynamics, structure and maximum aileron deflection angle. In Figure 8, a flight simulation is done to show that values chosen are optimum and give efficient results only for aircraft with Φ_{max}=45˚ and n=2. Results also show that optimization of *'k'* and *'n'* is very important and the optimum value varies depending on Φ_{max} of the aircraft.

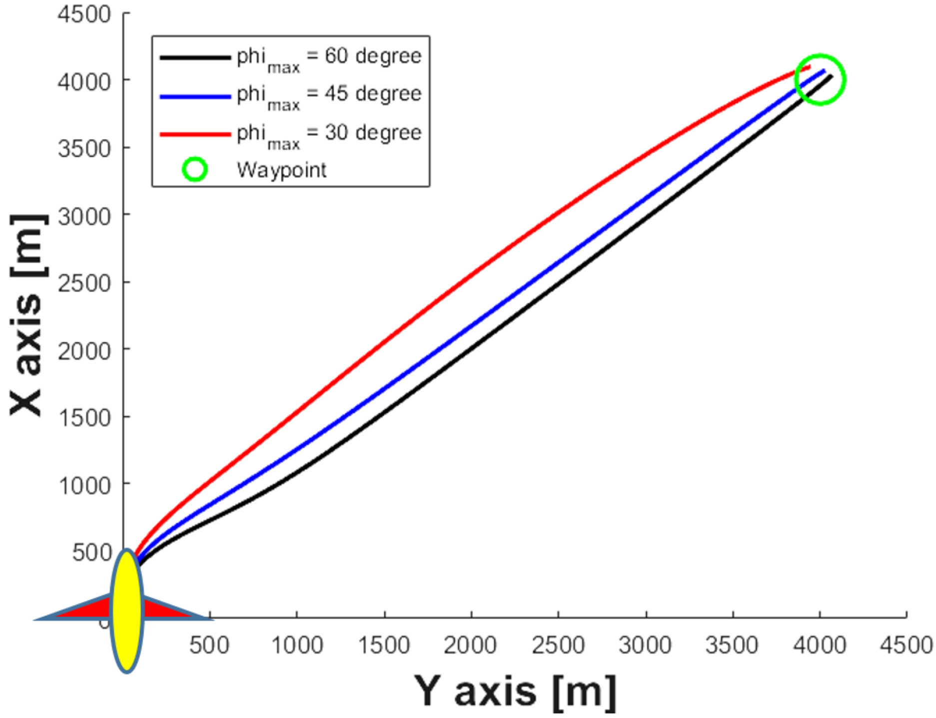

Figure 8. Effect of 'Φ_{max}' [Ψ_{max}=45˚, n=2]

Figure 9 shows that the new developed method is successful in guiding aircraft autonomously between waypoints.

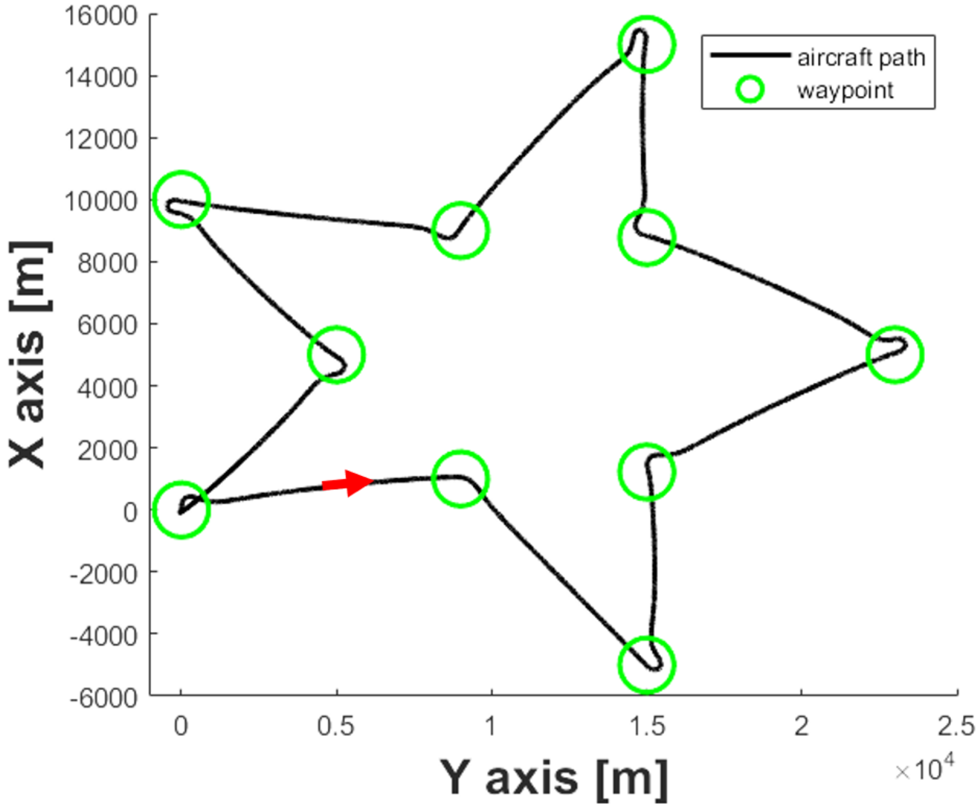

Figure 9. Autonomous Flight of 10 Waypoints

Coordinates of waypoints are inputted in order inside algorithms. In order to switch between waypoints, a minimum cross error value is chosen, for example *d*. Whenever the distance between aircraft and waypoint became less that *d*, aircraft switches to the next waypoint and starts to track it.

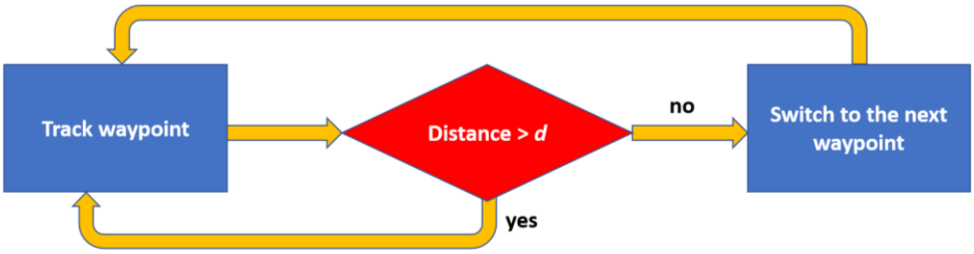

Figure 10. Waypoint switch algorithm

Conclusion and future works

UAVs can operate autonomously using GPS and IMU data only. To demonstrate that, a longitudinal-lateral-directional autopilot is designed for a fixed wing aircraft mathematical model using State Space model and PID controller. After that, a new waypoint lateral guidance method is developed and successfully tested in simulation environment. Method uses only aircraft and waypoint coordinates, heading angle and banking angle. All these data are assumed to be calculated perfectly in simulation environment. These 4 parameters are chosen to be new method parameters as they are able to be simply calculated by IMU, GPS and simple microcontroller. The simplicity of method and data used will help authors in doing an experimental flight to test the efficiency of the method in practice and compare it with methods used in other flight controllers. Method also has 2 factors to be optimized according to flight dynamics. The importance of optimizing method's factors is represented in detail with respect to criteria such as flight duration and less aileron deflections.

Briefing

This project is done with the financial support of The Scientific and Technological Research Council of Turkey (TÜBİTAK).

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