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Enhancing the stabilization of aircraft pitch motion control via intelligent and classical method

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Abstract. The pitching movement of an aircraft is very important to ensure passengers are intrinsically safe and the aircraft achieve its maximum stability. The equations governing the motion of an aircraft are a complex set of six nonlinear coupled differential equations. Under certain assumptions, it can be decoupled and linearized into longitudinal and lateral equations. Pitch control is a longitudinal problem and thus, only the longitudinal dynamics equations are involved in this system. It is a third order nonlinear system, which is linearized about the operating point. The system is also inherently unstable due to the presence of a free integrator. Because of this, a feedback controller is added in order to solve this problem and enhance the system performance. This study uses two approaches in designing controller: a conventional controller and an intelligent controller. The pitch control scheme consists of proportional, integral and derivatives (PID) for conventional controller and fuzzy logic control (FLC) for intelligent controller. Throughout the paper, the performance of the presented controllers are investigated and compared based on the common criteria of step response. Simulation results have been obtained and analysed by using Matlab and Simulink software. The study shows that FLC controller has higher ability to control and stabilize the aircraft's pitch angle as compared to PID controller.

1. Introduction

Aircraft technology has transformed human life by enabling regional and international transportation of people and goods. Aviation has essentially reduced the travelling time and cost, and revolutionized travelling experience with introduction of in-flight luxuries that are not commonly available in other modes of transportation like ground vehicles, trains or boat. However, an aircraft has to be controlled with a different control system to guarantee flight passengers are safe and the aircraft is in fly-worthy conditions. There are many critical parts in an aircraft that need to be perfectly controlled and one of them is the elevator. The elevator influences aircraft motion by affecting the pitch movement [1, 2]. There are many intelligence methods that are widely used for aviation control such as proportional, integral and derivatives (PID), linear quadratic control and fuzzy logic control (FLC) [3-5]. The FLC system has become one of today's successful technology to control sophisticated systems. It addresses system complexity by resembling the human decision making and is able to generate precise solutions given certain approximate input information. It closes the gaps in engineering control design that has been limited by rigid mathematical techniques such as linear manipulate design.



The development of automated control system has played a crucial role in the growth of civil and military aviation. Aircraft and missile structures are being assembled with control structures that offer stability, disturbance attenuation and reference signal tracking, even as their aerodynamic coefficients vary over a huge dynamic range due to large Mach-altitude fluctuations and uncertainties resulting from inaccurate wind tunnel measurements [6, 7]. The aggregate of nonlinear dynamics, modelling uncertainties and parameter variant in characterizing an aircraft and its operating environment are the major problems in flight control system [8]. To reduce the complexity of analysis, the aircraft is often assumed as an inflexible body and the aircraft's motion includes a small deviation from equilibrium flight condition [7, 9]. One of the objectives of a pitch control machine is to manipulate or assist the pilot to steer the plane by guiding the pitch steadily. With a pitch controller, the aircraft will go back to the desired attitude within a reasonable duration of time after a pitch disturbance. Moreover, the pitch controller can quickly make a pitch change after a given command. There are many problems faced in developing a pitch controller for an aircraft longitudinal motion. Researchers have varying opinion on which controller to be applied to ensure adequate suitability and minimum error when simulations are implemented [10]. By using a simulation tool, it can be observed whether the output is following the desired input or not. In order to get the best results, the best controller needs to be selected to minimize the errors in the simulation.

A simulation on the behaviours of the pitch motion of an aircraft is implemented in this study. In addition, a control system is developed to enhance the performance of previous control system, using intelligence method for the pitch angle control. Throughout this paper, the aircraft dynamic system is modelled with mathematical equations. Mathematical modelling involves the process of describing the dynamics of the system in a set of differential equations. By applying the physical law governing a particular system, which is from Newton's Laws, the system dynamics for many mechanical systems can be obtained. The motions of an airplane consist of longitudinal and lateral motions. However, to narrow down the scope of study, only equation of longitudinal motion is considered. The modelling derivation states that the summation of all external forces acting on the aircraft's body is identical to the time fee of exchange of the momentum of aircraft's frame. Furthermore, the summation of the outside moments acting on the aircraft's frame is identical to the time fee of exchange of the moment of angular flight momentum. In this paper, two proposed controllers are designed, namely PID and FLC. The proposed controller is then enhanced by combining the controllers which are proportional (P), proportional and derivatives (PD), PID and FLC to improve the aircraft response. Analyses are implemented on each type of controllers, whether it has fulfilled the criteria needed for the system. Several mathematical models are used to perform the simulations.

2. Pitch Motion of an Aircraft System

The pitch of aircraft is controlled through an elevator that is commonly positioned at the rear of the plane, parallel to the wing that houses the ailerons [11]. Pitch rate manipulation typically responds very fast to the adjustments in control input compared to pitch angle. Pitch control is a longitudinal challenge and the design of the autopilot system that controls the pitch of an aircraft is complex. By moving the elevator backwards, the pilot controls the elevator up (a function of bad camber) resulting in the downwards pressure on the horizontal tail to expand. Angle of attack on the wings increases, so the aircraft nose is pitched up and lift is usually produced. In micro-lighting fixtures and grasp gliders, the pitch movement is reversed and the pitch control device is much less complicated. Hence, whilst the pilot manipulate backwards the elevator, it produces a nostril-down pitch and the angle of attack at the wing is reduced. The pitch angle of an aircraft is manipulated via adjusting the angle and therefore boosting the force of the rear elevator. Figure 1 shows the pitch angle and the direction of an aircraft.

A few assumptions need to be made before the modelling process is done. First, the aircraft is at a steady state cruising at constant altitude and velocity, hence the thrust and drag cancel out each other while the lift and weight balance out each other. Second, the change in pitch angle does not change the speed of an aircraft under any circumstance. Moreover, the atmosphere in which the plane is flying is assumed to be undisturbed, thus forces and moment due to atmospheric disturbance are taken as zero.

Besides that, the pitch angle is assumed not to be more than 90 degree or 1.6 radians. If the pitch angle goes beyond that limit, it will damage the whole operation of the aircraft. The angle of deflection also has its limitation that need to be considered. The input for the system is from zero to the limited value in the system whereas the output for the system is also limited at a certain range of angles.

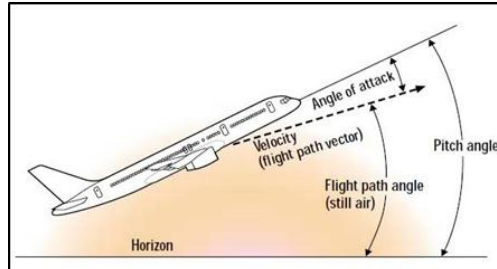


Figure 1: The pitch angle with the direction of an aircraft

The operational framework of this study includes the equation from free body of an aircraft taking the lift at certain pitch angle, which are derived to form three equations [12, 13]. Those equations are in first order system before being converted into a transfer function. The following dynamic equations in Equation 1 to Equation 3 describe the longitudinal dynamics involving the pitch motion of a typical aircraft, where u , v and w represents forward, side and lateral velocities, respectively, while p , q and r are roll, pitch and yaw rates, respectively.

$$F_x = m(\dot{u} + qw - rv) \quad (1)$$

$$F_z = m(\dot{w} + pv - qu) \quad (2)$$

$$M_y = I_y \dot{q} + rp(I_x - I_z) + I_{xz}(p^2 - r^2) \quad (3)$$

It is necessary to solve completely the aircraft problem with the following assumptions for the aircraft motion, as in Equation 4 to Equation 9.

$$\text{Rolling rate: } p = \dot{\phi} - \psi \sin\theta \quad (4)$$

$$\text{Yawing rate: } q = \dot{\theta} \cos\phi + \dot{\psi} \cos\theta \sin\phi \quad (5)$$

$$\text{Pitching rate: } r = \dot{\psi} \cos\theta \cos\phi - \dot{\theta} \sin\phi \quad (6)$$

$$\text{Pitch angle: } \dot{\theta} = q \cos\phi - r \sin\phi \quad (7)$$

$$\text{Roll angle: } \dot{\phi} = p + q \sin\phi \tan\theta + r \cos\phi \tan\theta \quad (8)$$

$$\text{Yaw angle: } \dot{\psi} = (q \sin\phi + r \cos\phi) \sec\theta \quad (9)$$

The equations are linearized by adding perturbation equation and small disturbance theory to replace the nonlinearities in the aircraft general dynamic equations [14]. The gravitational force components along the x, y and z-axis can be easily shown to be as in Equation 10 to Equation 12, respectively.

$$F_{xg} = -mg \sin\theta \quad (10)$$

$$F_{yg} = mg \cos\theta \sin\phi \quad (11)$$

$$F_{zg} = mg \cos\theta \cos\phi \quad (12)$$

Then, Equation 1 to Equation 3 are deduced to Equation 13 to Equation 15, respectively.

$$F_x = m(\dot{u} + qv - rv) + mg \sin\theta \quad (13)$$

$$F_z = m(\dot{w} + pv - qu) - mg \cos\theta \cos\phi \quad (14)$$

$$M_y = I_y \dot{q} + rp(I_x - I_z) + I_{xz}(p^2 - r^2) \quad (15)$$

The equations above can be transformed into the state-space form as stated by Equation 16 to Equation 18. The model in state-space form is used for controller design. Moreover, the parameters used for the domestic aircraft are tabulated in Table 1.

$$\begin{bmatrix} \dot{\Delta\alpha} \\ \dot{\Delta q} \\ \dot{\Delta\theta} \end{bmatrix} = \begin{bmatrix} \frac{Z_\alpha}{u_0} & 1 & 0 \\ M_\alpha + \left[M_\alpha \cdot \frac{Z_\alpha}{u_0} \right] & M_q + M_{\dot{\alpha}} & 0 \\ 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} \Delta\alpha \\ \Delta q \\ \Delta\theta \end{bmatrix} + \begin{bmatrix} -\left[\frac{Z_\alpha}{u_0} \right] \\ -\left[M_\alpha + \left[M_\alpha \cdot \frac{Z_\alpha}{u_0} \right] \right] \\ 0 \end{bmatrix} \Delta\delta \quad (16)$$

$$\begin{bmatrix} \dot{\Delta\alpha} \\ \dot{\Delta q} \\ \dot{\Delta\theta} \end{bmatrix} = \begin{bmatrix} -2.02 & 1 & 0 \\ -6.9868 & -2.9476 & 0 \\ 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} \Delta\alpha \\ \Delta q \\ \Delta\theta \end{bmatrix} + \begin{bmatrix} 0.232 \\ 0.0203 \\ 0 \end{bmatrix} \Delta\delta e \quad (17)$$

$$Y = \begin{bmatrix} 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} \Delta\alpha \\ \Delta q \\ \Delta\theta \end{bmatrix} + \begin{bmatrix} 0 \end{bmatrix} \quad (18)$$

Table 1: Parameters of domestic aircraft [2]

Longitudinal Derivatives	Components		
	Dynamics Pressure and Dimensional Derivative $Q = 36.8 \text{ lb/ft}^2$, $QS = 6771 \text{ lb}$, $QSC = 38596 \text{ ft.lb}$, $(\bar{c}/2u_0) = 0.016s$		
	X-Force (S^{-1})	Z-Force (Z^{-1})	Pitching Moment (FT^{-1})
Rolling velocities	$X_u = -0.045$	$Z_u = -0.369$	$M_u = 0$
Yawing velocities	$X_w = 0.036$, $X_{\dot{w}} = 0$	$Z_w = -2.02$, $Z_{\dot{w}} = 0$	$M_w = -0.05$, $M_{\dot{w}} = -0.051$
Angle of attack	$X_\alpha = 0$, $X_{\dot{\alpha}} = 0$	$Z_\alpha = -355.42$, $X_{\dot{\alpha}} = 0$	$M_\alpha = -8.8$, $X_{\dot{\alpha}} = -0.8976$
Pitching rate	$X_q = 0$	$Z_q = 0$	$M_q = -2.05$
Elevator deflection	$X_{\delta_e} = 0$	$Z_{\delta_e} = 0$	$M_{\delta_e} = 0$

3. Controller Design

Two types of controller are implemented in this study: PID and FLC controllers. The PID acts as the benchmark while the FLC controller is to enhance the performance of PID controller. Software used for the simulation are Matlab and Simulink version 2016b.

The most common industrial controller is based totally on the PID set of regulations. PID is known as control loop feedback mechanism that is applied on top of structured things. PID controller attempts to compensate the error between a measured variable and its desired reference value by mathematical calculations. The corrective values from PID will be the input into the controller, producing stabilized controlled movements. The PID controller calculation involves three separate parameters that consist of proportional, integral and derivatives values, as indicated by Equation 19 and Figure 2. Proportional value decides the reaction to the current error whereas the integral value is based on the sum of recent errors. On the other hand, derivatives component determines the reaction to the cost at which the error has since changed.

$$u(t) = K_p e(t) + K_i \int_0^t e(\tau) d\tau + K_d \frac{de(t)}{dt} \quad (19)$$

The gain is known as the critical advantage whereas the frequency of oscillation is known as crucial frequency. There are other conventional tuning methods such as the Cohen-Coon method that is used

when the main design criterion is related to the disturbance rejection but it may only be used for first order models consisting of massive manner delays [15]. Such relay auto-tuning technique eliminates the possibility of flying the aircraft near the steadiness limit.

In the meantime, FLC is a superior approach for sorting and when dealing with statistics. It has also been proven to be the step forward for most control system applications since it mimics the human manipulative common sense. It can be embedded into anything from small hand-held products to the massive computerized manipulative systems. It makes use of imprecise but very descriptive language to address input statistics similar to a human operator. It is very robust and forgiving of operator and facts input, and usually works when first applied with little or no tuning. A novel technique for FLC controller is employed for stabilization of the pitch control in this study [7, 11]. Figure 3 shows the overall closed-loop system for FLC control systems for the pitch control of an aircraft. The input to the fuzzy controller is the error, which measures the system performance, and the rate at which the error changes. The output is the change to be applied to the control signal. From Figure 3, the error is computed by comparing the desired point with the aircraft output. The change of error is generated by deriving the error. The error and change of error are fed to the fuzzy controller through a multiplexer.

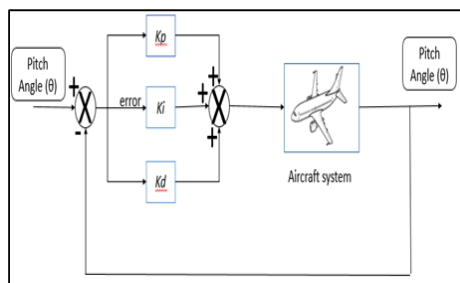


Figure 2: PID controller system

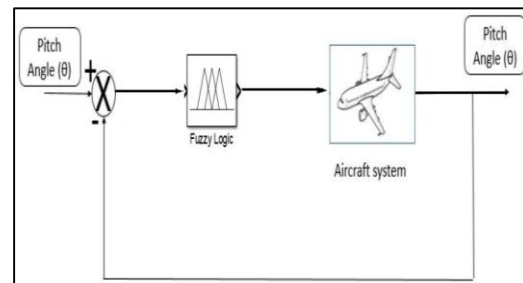


Figure 3: FLC controller system

FLC has two inputs (i.e. error change and error) and one output (i.e. control signal) as shown in Figure 4, Figure 5 and Figure 6, respectively. The variables that implies inputs and output have been classified as N(Negative), Z(Zero), P(Positive). Since there are three fuzzy variables and each fuzzy variable has three membership functions, the fuzzy controller for pitch control of an aircraft has a total on nine membership functions. Each membership function is constrained to be triangular such that each has three parameters. Table 2 shows the fuzzy rule employed in this study that is based on 3 x 3.

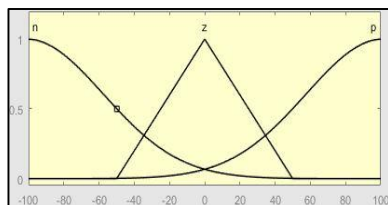


Figure 4: Fuzzy set for the input “error”

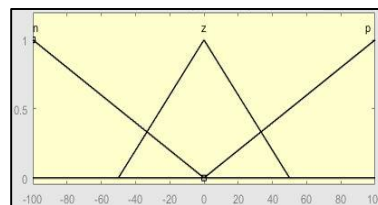


Figure 5: Fuzzy set for the input “Δerror”

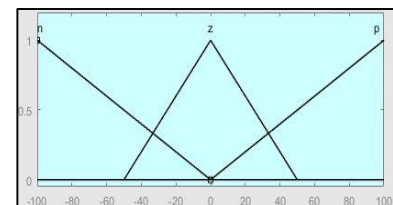


Figure 6: Fuzzy set for the output “output”

Table 2: The fuzzy rule base

$\Delta_{error} (de)$	error (e)		
	N	Z	P
Negative (N)	N	N	P
Zero (Z)	N	Z	P
Positive (P)	N	P	P

4. Result and

Discussion

The simulations are analysed based on the mathematical model for the aircraft pitch motion system. Firstly, the performance of the controllers are observed on the pitch angle of the aircraft without any feedback. Then, the results obtained are analysed using the classical approach and intelligent method, which is PID and FLC, respectively. From Figure 7, it can be seen that the pitch control system is not stable without a controller. The curve of the pitch angle is approaching infinity as the time increases. Therefore, a feedback controller needs to be designed in order to stabilize the system. In this study, a unit step command is required in order for the pitch angle to follow a reference value of 0.2 radian. The controller output is made proportional to the error where the proportionality constant called the proportional gain, K_P is introduced. The K_P will effectively reduce the rise time but does not eliminate the steady-state error. This controller is capable of maintaining the output steady-state value at desired value as shown in Figure 8. From Figure 8, K_P is set to be 1.5. Increasing the value K_P will decrease the rise time but at the same time increase the overshoot of the system. When the K_P is increased, it will produce large oscillations before the system reaches the steady-state value. The output response gives fast response with a rise time of 0.388 seconds, settling time of 1.871 seconds, percentage of overshoot of 4.8% and percentage of steady-state error of 0.05%.

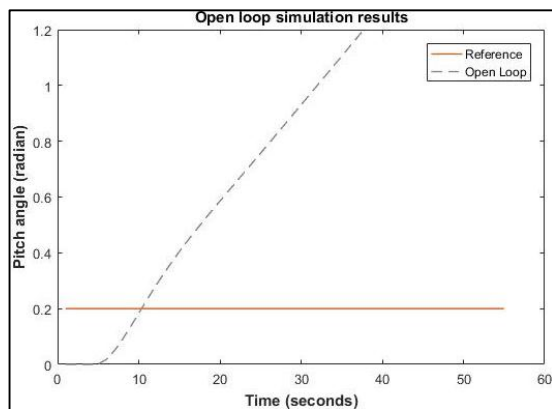


Figure 7: Without controller

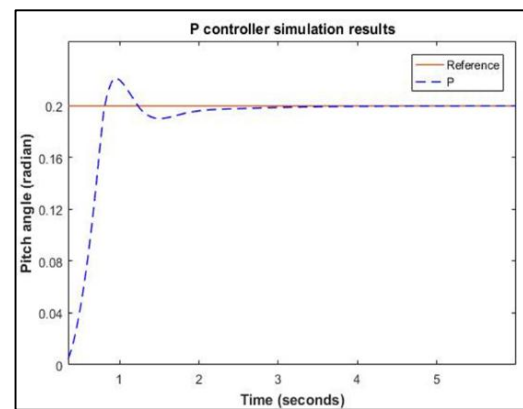


Figure 8: With P controller

Figure 9 illustrate the response of PD controller for pitch angle to the unit step reference input. The derivatives component is used to create damping in dynamic system and thus stabilizing its behaviour. The derivatives action is proportional to the rate of change of error measurement, which compensates the errors by changing the measurement and improving the system transient response. This increases the stability of the system. The PD controller provides a very fast response with settling time of 1.098 seconds and rise time of 0.23 seconds, but the response has a sharp peak. The transient response has an overshoot around 2.3%. In addition, it also gives a percentage of steady-state about 0.05%. Increasing K_D will make the response become a little bit slower, hence a decrease of the overshoot.

The value of K_P , K_I and K_D are dependent on each other. Changing one of them can affect the other two. Effects of each controller gain K_P , K_I and K_D on a closed-loop system will make a system more stable. K_P will have the effect of reducing rise time but it will never eliminate the steady-state error. K_I will have the effect of eliminating the steady-state error but it will make the transient response worse. K_D will have the effect in increasing the stability of the system by reducing overshoot and improving the transient response. Using Ziegler-Nicols method and after several trials and errors, the parameters value for the PID controller are determined as $K_P= 12.45$, $K_I= 1.75$ and $K_D= 0.12$. The simulation results for PID controller are shown in Figure 10. It can be observed that the response of PD controller is a little bit slower than PID controller in term of rise time and settling time, which both values for the latter controller are 0.16 seconds and 1.89 seconds, respectively.

Meanwhile, Figure 11 shows the output response of pitch control by using the FLC approach where the two inputs and one output have been applied to the system. The FLC provides good performance in term of settling time, percentage of overshoot, percentage of steady-state error and rise time. From the

results obtained, it shows that the pitch angle follows the reference value. This controller is able to provide a better response than the PID controller and does not produce any overshoot. The response is comparatively faster where the settling time is about 0.96 seconds and the rise time is 0.16 seconds.

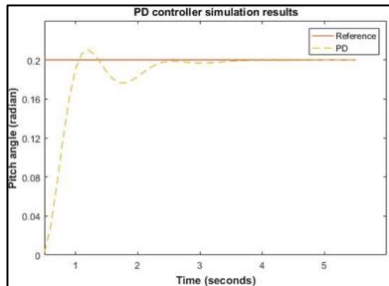


Figure 9: PD controller

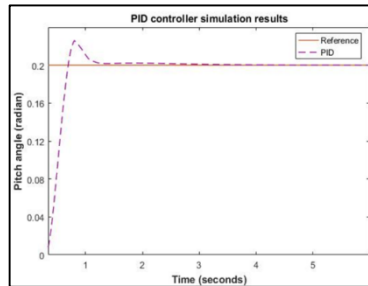


Figure 10: PID controller

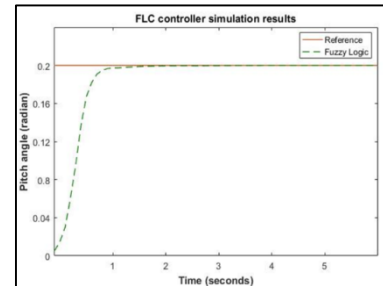


Figure 11: FLC controller

Moreover, it can be observed that the intelligent based approach gives the best output performance in term of settling time, overshoot percentage, steady state error and rise time. From the results shown in Figure 8 to Figure 11, the PD controller takes the longest time to settle down with 3.5 seconds. The highest percentage of overshoot goes to the P controller whereas the highest percentage of steady-state error is obtained by P and PD controller for 0.05. Significantly, the FLC controller has 0% overshoot compared to the other controllers. Based on these results, it can thus be concluded that FLC is the best controller to enhance the pitch motion of the aircraft compared to the PID.

Furthermore, in order to know whether the PID controller is robust or not, a new input pitch angle is set and a disturbance is applied. The disturbance is reflected on the cross wind effect that will hit the plane while on air in pitch motion scenario. The PID controller system is tested with new input pitch angle of 0.1 radians. Meanwhile, the 0.3 step input that acts as disturbance is added to the system to examine the robustness of the controller. The results obtained are shown in Figure 12 and Figure 13, which show that the PID controller still give a good performance by achieving the criteria needed. The PID controller settles down and follows the desired input of 0.1 radians. It is concluded that the PID controller is robust due to its performance from different input response and with applied disturbance.

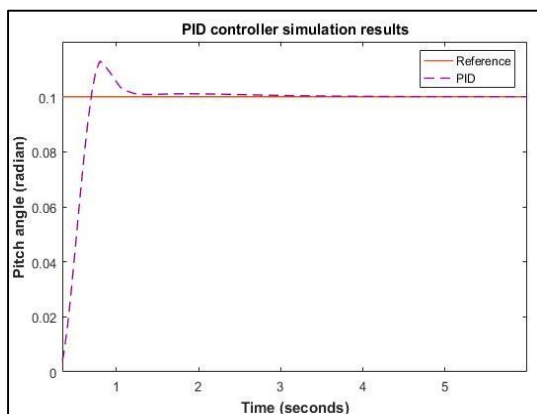


Figure 12: PID controller for input 0.1 radians and 0.3 step input disturbance effect

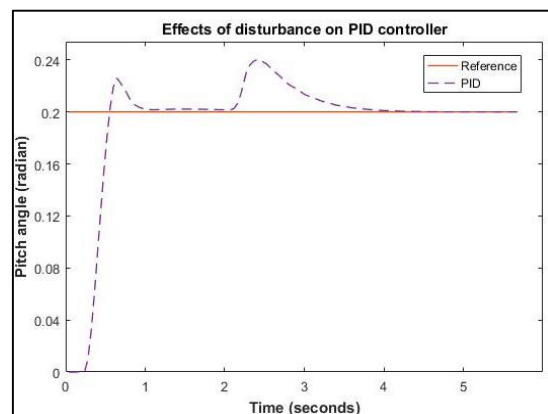


Figure 13: PID controller for input 0.2 radians and 0.3 step input disturbance effect

The FLC controller is also tested with similar disturbance criteria as the PID to see its robustness. Similar to PID controller, the chosen value for new input pitch angle is 0.1 radians and 0.3 step input is applied as disturbance on both chosen values of pitch angle. From Figure 14, it shows that the FLC controller still give a good performance by achieving the criteria required. The FLC controller settles down and follows the desired input 0.1 radians. When there is a disturbance, the steady-state error is

observed because it does not reach the settling time for 0.2 radians and 0.1 radians. Some tunings need to be done to improve the performance of the FLC when applied with a 0.3 step input disturbance for both input.

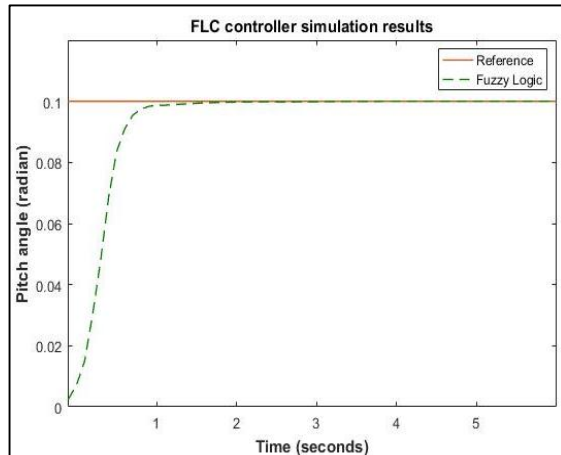


Figure 14: FLC controller for input 0.1 radians and 0.3 step input disturbance effect

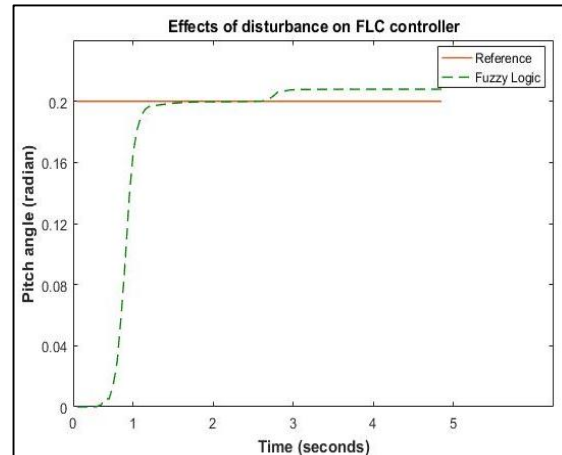


Figure 15: FLC controller for input 0.2 radians and 0.3 step input disturbance effect

5. Conclusion

The validated model of pitch control of an aircraft is very helpful in developing the control strategy for an actual system. Pitch control of an aircraft requires a pitch controller to maintain the pitch angle at the desired value. This can be achieved by reducing the error signal, which is the difference between the output angle and the desired angle. In this study, two methods in designing a controller have been explored: PID and FLC controllers. It is difficult to design a third order system, thus for PID controller method, the rate feedback is added to improve the system performance. From the results, it can be concluded that the controllers designed are all capable to control the pitch angle of an aircraft using a linearized system. Simulation and analysis of the results have shown that FLC controller gives the best performance in relative to the PID controller. However, further improvement needs to be done on the FLC controller in order to improve its performance such that it becomes more robust and much better response can be achieved. The limitation in the FLC controller design should be eliminated by adding more membership functions to the controller.

Acknowledgement

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