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Improvement of Pitch Motion Control of an Aircraft Systems

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Abstract

The movement of the aircraft pitch is very important to ensure the passengers and crews are in intrinsically safe and the aircraft achieves its maximum stability.The objective of this study is to provide a solution to the control system that features particularly on the pitch angle motion of aircraft systemin order to have a comfort boarding. Three controllers were developed in these projects which wereproportional integral derivative (PID), fuzzy logic controller (FLC), and linear quadratic regulator (LQR) controllers. These controllers will help improving the pitch angle and achievingthe target reference. By improving the pitch motion angle, the flight will be stabilized and in steady cruise (no jerking effect), hence provides all the passengers withthe comfort zone. Simulation results have been done and analyzed using Matlab software. The simulation results demonstrated LQR and FLC were better than PID in the pitch motion system due to the small error performance. In addition, withstrong external disturbances, a single controller is unable to control the system, thus, the combination of PID and LQR managed to stabilize the aircraft.

Keywords: *pitch angle, aircraft, angle of elevator, linear quadratic regulator, fuzzy logic, proportional integral derivative*

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1. Introduction

In modern technology of flight stability system, the development of the automatic control theory has played a very important part in the growth of aviation technology. This development helps and aids many people such as pilot and engineers in order to conduct the aircraft in automatic position that will reduce the work of workers. In this situation, flight workers can lessen their workload and it also can help the aircraft to land when it comes to bad weather [1].

There are a wide assortment of aerial vehicles with various shapes, sizes, designs, and characteristic. Regularly nations have taken the improvement of aerial vehicles especially on unmanned aerial vehicles (UAVs) as priority [2]. UAVs of aircraft have been broadly developed in recent years and have been actualized in both military and civilian applications such as applications for search and rescue victim, traffic monitoring, wild life concealment, information collection and transmission. Researchers have great interest in UAV development due to the minimization of the risk factors of pilots and the drastic reduction of cost operations [3]. The UAV military world uses it in military operations and reconnaissance missions, while UAV's academic research usesit to observe dangerous states such as volcanoes, natural disaster sites, and observations of conditions of elevated buildings of more than hundred year old [4].

The modern UAV ought to have capabilities of wide airspace flight that undergo the automatic control framework in light of intelligent technique, for example, artificial neural network, fuzzy logic theory, particle swan optimization, and genetic algorithm which have provide novel answers for the flight control framework issues [2]. In order to perform at high level, an aircraft needs to be controlled by different control systems to guarantee the aircraft to be in good conditions. A lot of critical parts in aircraft need to be controlled perfectly and one of

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the most critical parts to be controlled is the elevator [5]. The elevator parts will influence the motions of the aircraft and the pitch movement [6].

Pitch is a motion where a vehicle is moving or pointing upwards in the direction of certain axis. In aircraft system application, pitch only occurs when the vehicle is pointing or moving upwards with nose point upward [7]. In the aircraft stability system that involves movement, there are six motions that can be divided into two parts which are longitudinal and lateral motions. The most important motions in aircraft stability system are pitch, yaw and roll where pitch motions occur to the longitudinal direction. Pitch can be controlled via an elevator which is a moveable part of the tailplane (i.e. horizontal stabilizer) [8]. To make pitch motion upwards for an aircraft, control stick need to pull towards pilot (elevator will move up), while for micro light, horizontal control bar attached with the wing has to push away from the pilot [9]. The pitch process is a risky part of the flight of aircraft, as this can be seen by the number of aircraft crashes occurring during the landing and take-off motion. Pitch has difficulties when the pilot has to perform a specific maneuver to the point of accomplishment carefully [10]. Hence, this project focuses on pitch motion that aircraft endured during a steady cruise on air [11].

Flight dynamics involves the investigation on the execution, steadiness, and control of vehicles flying through the air or in space. The concern is how powers utilization/distribution which affected the behavior of the aircraftimpact its speed and demeanor regarding time. The flight dynamic equation is one of the solution method to get an ideal system such that the input that goes in is the same as the output that goes out as required [12]. One of the real issues of flight control system is because of the mix of nonlinear progression, displaying with numerous vulnerabilities and parameter variety in portraying a flying machine and its working conditions [13]. The airplane movement in free flight is very confusing. By and large, airplane flies in three-pivot plane by controlling the aileron, udder and lift [14].

The aircraft needs to have a comfort boarding and safety precaution for passengers and crews. As such the aircraft needs to be controlled by appropriate types of control systems. By considering disturbance such as cross wind effect and bad weather like heavy rain, the aircraft needs to be stabilized and a lot of critical parts in aircraft need to be controlled without having much problems and one of the most critical parts are the elevator for pitch motion movement [15]. The elevator parts will affect the movement of the aircraft and it will directly affect the pitch motion and thus the stability of the aircraft will be improved.

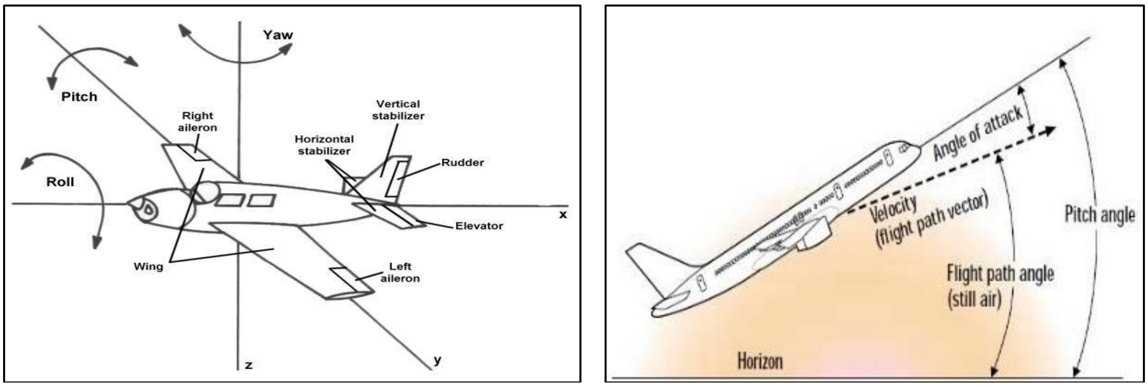
The first contribution of this paper is the developed mathematical model of a pitch controller thatrepresents the actual aircraft system behavior. The mathematical modelling form can describe and present the dynamic system of the aircraft in differential, transfer function and state-space models. By utilizing the Newton's law of motion, the aircraft modelling can be obtained. The modelis for the aircraft that is on air in steady cruise. The model accepts disturbance such as the cross wind effect that may hit the plane while on air in pitch motion.

The second contribution of this paper areon how to reduce and get the exact input as output for the pitching where this involves few parameters and several outputs. The most suitable method that is recommended is the state-space model. State-space modelisa model that uses state variables to describe a system by a set of first-order differential or difference equations and can be controlled and observed for every state of the aircraft movement. This simulation will help the aircraft achieves the requirement.

In this paper, three controller designs are applied: Proportional-Integral-Derivative (PID), Fuzzy Logic Controller (FLC), and Linear Quadratic Regulator (LQR). The performance of each of the controllers will be compared based on the requirement of the step response and observation whether the controller satisfies the requirement of the system. This project will be performed through simulation where several modellings will be used.

2. Aircraft Pitch Model

Mathematical modelling of the dynamic system of an aircraft involves process to describe the dynamics of the system in a set of differential equations. The equation of force is mass with acceleration according to the Newton's law of motion where the acceleration of velocity against time in the direction of *x*, *y* and *z* in vector axis is as shown in Figure 1 [16, 17]. The motions of an airplane consist of longitudinal and lateral motions. However in narrowing down the scope of the project, only the longitudinal motion will be considered in this paper.



(a) (b)

Figure 1. Direction of aircraft; (a) x, y and z vector axis [13], (b) the pitch angle of an aircraft [14]

There are a few assumptions to be made before the modelling process starts. The aircraft should be in constant steady cruise at constant altitude and velocity, then the thrust and drag will be cancelled out and the lift and weight will balance with one another. Then the change of pitch angle will not change the speed of the aircraft. The system will be tested using Matlab configuration to find the stability of the aircraft using bodes plot and root locus technique. The testament of stability is to ensure that the system is needed to be controlled or not. Table 1 displays the vehicle parameters that have been used in the simulation process based on references [11]. The model parameters *u, v*, *w*, *P, Q*, and *R* represent forward velocity, side velocity, lateral velocity, pitch rate, roll rate, and yaw rate respectively.

Table 1. Aircraft Parameters [11]

Longitudinal Derivatives	X-Force (s^{-1})	Z-Force(F^{-1})	Pitching Moment (FT^{-1})
Rolling Velocities	$X_u = -0.0045$	$Z_u = -0.369$	$M_u = 0$
Yawing Velocities	$X_w = 0.036$, $X_{\dot{w}} = 0$	$Z_w = -2.02$, $Z_{\dot{w}} = 0$	$M_w = -0.051$, $M_{\dot{w}} = 0$
Angle of Attack	$X_\alpha = o$, $X_{\dot{\alpha}} = 0$	$Z_\alpha = 0$, $Z_{\dot{\alpha}} = 0$	$M_\alpha = -8.8$, $M_{\dot{\alpha}} = -0.8976$
Pitching Rate	$X_q = 0$,	$Z_q = 0$	$M_q = -2.05$
Elevator Deflection	$X_{\delta e} = 0$	$Z_{\delta e} = -28.15$	$M_{\delta e} = -11.874$

The operational framework of this project basically includes the equation from the free body of an aircraft taking a lift at pitch for certain angle. The three derived equations [14, 16, 18] are in the first order system before they are converted into the transfer function.

In this study, we assume that the complexity of the actual or detailed motion equations of the quadrotor is well explained and available in [14, 16, 18, 19]. After linearization, the following equations were obtained for the longitudinal dynamics of the aircraft keeping in mind that (1), (2) are the force equations and (3) is the momentum equation,

$$\left\{\frac{d}{dx} - x_u\right\}\Delta u - x_w \Delta w + (\text{g}\cos\theta)\Delta\theta = x\delta_e \Delta\delta_e \quad (1)$$

$$-z_u \Delta u + \left\{(1 - z_w)\frac{d}{dt} - z_w\right\}\Delta w - \left\{\left(u_0 + z_q\right)\frac{d}{dt} - g\sin\theta_0\right\}\Delta\theta = z_{\delta e}\Delta\delta_e \quad (2)$$

$$-M_u - \left\{M_w \frac{d}{dt} + M_w\right\}\Delta w + \left\{\frac{d^2}{dt^2} - M_q \frac{d}{dt}\right\}\Delta\theta = M_{\delta e}\Delta\delta_e \quad (3)$$

Dividing with $\Delta\delta_e$ and we obtained

$$\frac{\left(s - \frac{Z_\alpha}{u_0}\right)\Delta\alpha(s)}{\Delta\delta_e(s)} - \frac{\Delta q(s)}{\Delta\delta_e(s)} = \frac{Z_\delta}{u_0} \quad (4)$$

$$-\left(M_\alpha+\frac{M_{\dot\alpha}M_\alpha}{u_0}\right).\frac{\Delta\alpha(s)}{\Delta\delta_e(s)}+\left[s-\left(M_q+M_{\dot\alpha}\right)\right]\frac{\Delta q(s)}{\Delta\delta_e(s)}=M_{\delta e}+M_{\dot\alpha}\frac{Z_\delta}{u_0} \tag{5}$$

By manipulating equation (1-3), and substituting the parameter values of the longitudinal stability derivatives in Table 1, the transfer function for the change in the pitch rate to the change in elevator deflection angle is given as follows:

$$\frac{\Delta q}{\Delta\delta}=\frac{1}{S}.\frac{-\left(M_{\delta e}+\frac{M_\alpha Z_{\delta e}}{u_0}\right)s-\left(\frac{M_\alpha Z_{\delta e}}{u_0}-\frac{M_\alpha Z_\alpha}{u_0}\right)}{s^2-\left(M_q+M_\alpha+\frac{Z_\alpha}{u_0}\right)s+\left(\frac{Z_\alpha M_\alpha}{u_0}-M_\alpha\right)} \tag{6}$$

Hence, the transfer function for the pitch system dynamics of an aircraft is

$$\frac{-(M_{\delta e}+M_\alpha Z_{\delta e}/u_0)s-\left(\frac{M_\alpha Z_{\delta e}}{u_0}-\frac{M_\alpha Z_\alpha}{u_0}\right)}{s^2-\left(M_q+M_\alpha+\frac{Z_\alpha}{u_0}\right)s+(\frac{Z_\alpha M_\alpha}{u_0}-M_\alpha)} \tag{7}$$

The transfer function will be in summation with the cross wind effect disturbance of $\frac{0.2}{s}$,

$$\frac{\Delta q}{\Delta\delta}=\frac{1}{S}.\frac{-(M_{\delta e}+M_\alpha Z_{\delta e}/u_0)s-\left(\frac{M_\alpha Z_{\delta e}}{u_0}-\frac{M_\alpha Z_\alpha}{u_0}\right)}{s^2-\left(M_q+M_\alpha+\frac{Z_\alpha}{u_0}\right)s+(\frac{Z_\alpha M_\alpha}{u_0}-M_\alpha)}+\frac{0.2}{s} \tag{8}$$

The modelling continues into the state-space form as stated. The modelling in state-space form is for the controller design,

$$\begin{bmatrix}\Delta\dot\alpha\\ \Delta\dot q\\ \Delta\dot\theta\end{bmatrix}=\begin{bmatrix}\frac{Z_\alpha}{u_0} & 1 & 0\\ M_\alpha+\left[M_\alpha.\frac{Z_\alpha}{u_0}\right] & M_q+M_{\dot\alpha} & 0\\ 0 & 1 & 0\end{bmatrix}\begin{bmatrix}\Delta\alpha\\ \Delta q\\ \Delta\theta\end{bmatrix}+\begin{bmatrix}-[\frac{Z_\alpha}{u_0}]\\ -[M_\alpha+\left[M_{\dot\alpha}.\frac{Z_\delta}{u_0}\right]]\\ 0\end{bmatrix}\Delta\delta \tag{9}$$

$$\mathrm{Y}=\begin{bmatrix}0 & 0 & 1\end{bmatrix}\begin{bmatrix}\Delta\alpha\\ \Delta q\\ \Delta\theta\end{bmatrix}+[0][\Delta\delta] \tag{10}$$

$$\begin{bmatrix}\Delta\dot\alpha\\ \Delta\dot q\\ \Delta\dot\theta\end{bmatrix}=\begin{bmatrix}-0.313 & 56.7 & 0\\ -0.0139 & -0.4260 & \\ 0 & 56.7 & 0\end{bmatrix}\begin{bmatrix}\Delta\alpha\\ \Delta q\\ \Delta\theta\end{bmatrix}+\begin{bmatrix}0.232\\ 0.0203\\ 0\end{bmatrix}U \tag{11}$$

$$Y=\begin{bmatrix}0 & 0 & 1\end{bmatrix}\begin{bmatrix}\Delta\alpha\\ \Delta q\\ \Delta\theta\end{bmatrix}+[0] \tag{12}$$

3. Controller Design

In this section, the controller is divided into three parts which are the PID, LQR, and FLC controllers. A few assumptions need to be considered before continuing with the modelling process. First, the aircraft is at a steady state cruising at constant altitude and velocity, thus the thrust and drag are cancelled out 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 circumstances. Also, the atmosphere in which the plane flies is assumed undisturbed, thus forces and moment due atmospheric disturbance are considered zero. Besides that, the pitch angle is assumed to be less than 90 deg. or 1.6 rad/sec. If the pitch angle is in that position, it will damage the whole operation of the aircraft. The angle of deflection also has its limitation that needs to be considered. The input for the system will be in range from zero to any limited value in the system. The output for the system will be also limited to certain range of angle.

3.1. PID Controller
The most commonly use controller is PID because it is easy to use and it is only required to tune to three parameters as shown in Figure 2. The three parameters are proportion of error (*P*), integral of error (I) and differential of error (*D*) control. PID is in feedback control mechanism which continuously calculates error as the difference between set point and measured variable [20-22].

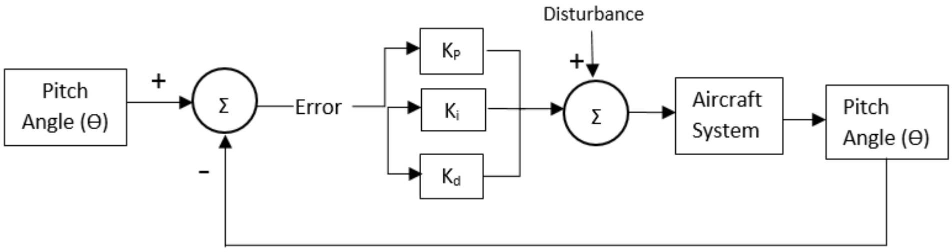


Figure 2. PID block diagram

In PID controller mechanism, *P*stand for now or present values of error wherethe amount of control outputdepends on. Whereas integral, *I*stands for past values of error. If the current output is not meeting the requirement, the integral part will do the fixing to meet the requirement. Lastly, *D*st and s for any errors that may occur in the future which are based on the current rate of change such as

$$U(t) = K_p + K_i \int_0^t e(t)dt + K_d \frac{de(t)}{dt} \quad (13)$$

The effect of PID controller to the system when K_p is increased whilerise time and steady-state error will be decreased.There is a small change in settling time but overshoot will be increased. When $K_i$gainedis increased, rise time will be decreased and steady-state error will be eliminated but overshoot occurs and settling time will be increased because the K_i will check overtime of past error in order to meet the current requirement. If K_d is increased, there will be less overshoot and settling time. The rise time increases and it has no effect on steady state error thus by putting these three gains, it will have a large effect on the system.

3.2. LQR Controller
Thelinear quadratic regulator is a controller that is used in machines or any types of processes such as in aircraft stability which is under mathematical algorithm that minimizes a cost function by weighing two factors supplied by users [23-25]. The cost function is a sum of deviations of key measurements, such as desired altitude or involving any process which is from its desired output. The algorithm in LQR minimizes undesired deviations in order to execute out the error that occurs in the system. The LQR mathematical algorithm also decreases the amount of work done by the control system user for optimizing the controller. Controller construction will be an iterative series of processes for the user to judge as optimal controller which can only be produced through simulation and altering the parameters in order to form a consistent and precise controller.

LQR is also a method of modern control theory technology that is used in state-space approach to analyze a system. Using state space methods is relatively simple to work with multi-output system. LQR problem is a special type of optimal control that deals with linear systems in state and in control and minimization of objective or cost function that are quadratic or the quadratic performance and implement in such algebraic equation. *J* is the parameter index that needs to be minimized by balancing the weight of matrices *Q* and *R*. The cost function needs to be as minimized as possible in order to use the precise *Q* and *R* matrices.

$$J = \int [y^T(t)\ Q\ y(t) + u^T(t) R u(t)] \quad (14)$$

From the (14), *u* is the input which is the multiplication of gain *K* and the state.

$$u = -Kx \tag{15}$$

After getting the value of *Q* and *R* matrices that minimize the cost function to the lowest possible, *Q* and *R* are used in algebraic equation to find the gain *K* for the state-feedback ideal gain placement. The equation *P* needs to be determined by *Riccati*'s algebraic equation.

$$K = (D^T QD + R)^{-1}(B^T P + D^T QC) \tag{16}$$

$$K = (R)^{-1} B^T P \tag{17}$$

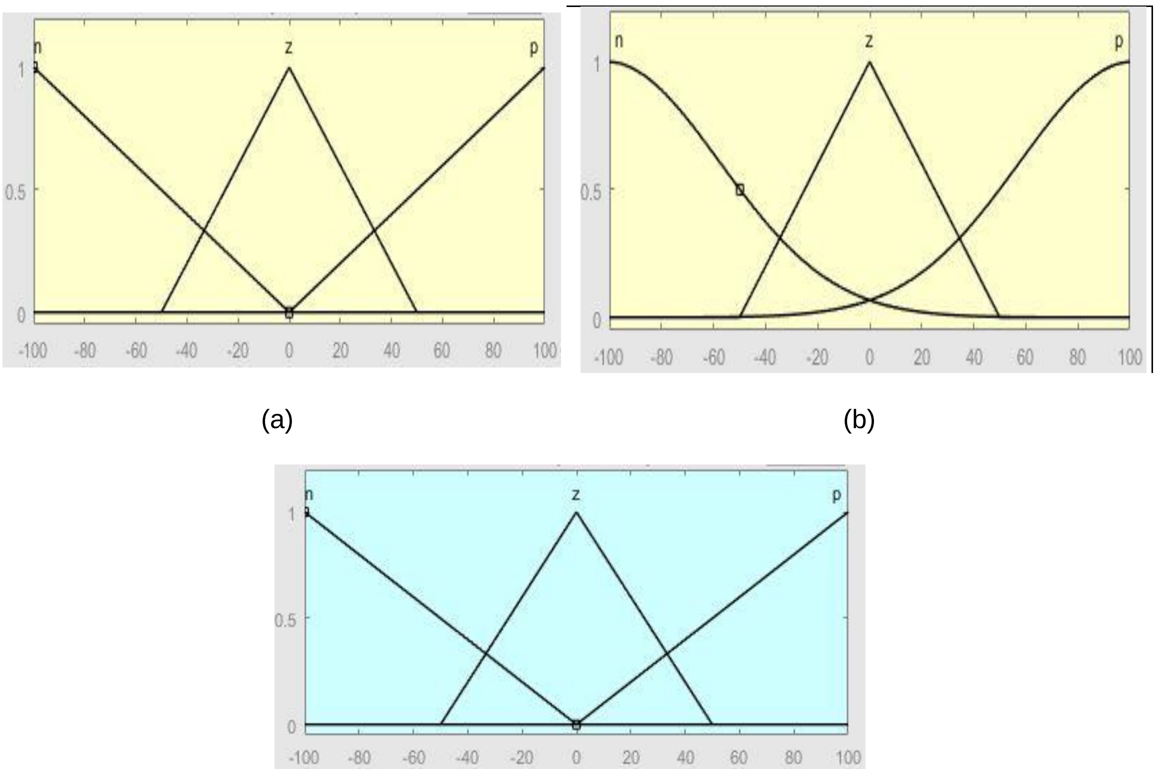
The equation (16) is *Riccati*'s equation. The equation is to determine the value of *P* in order to get the value of gain *K* in equation (16).

$$0 = A^T P + PA + Q - PB(R)^{-1} B^T P \tag{18}$$

Then solve *K* by putting the value of *P* in equation 17.

3.3. Fuzzy Logic Controller

Fuzzy logic control has been used in several applications as mentioned in references [26-29]. Figure 3 shows the membership functions for input and output of an aircraft. The inputs to the fuzzy controller are the error which measures the system performance and the rate of error changes. The output is the change of the control signal. The error is computed by comparing the desired point with the plant output. The change of error is generated by the derivation of the error.



(a) (b)

(c)

Figure 3. Fuzzy set for the (a) input "error", (b) input "Δerror", (c) output "output"

A two input and one output fuzzy pitch control can be designed by defining error as the reference angle minus the measured angle and implementing rules such as the following: "If the error is positive and the change in error is zero, then change throttle position by a positive amount". Another rule might be: "If the error is zero and the change in error is negative, then change throttles position by a negative amount". A rule base is defined intuitively with three membership functions each for the two inputs and one output. For this project, a (3x3) rule base is used as shown in Table 2.

Table 2. Fuzzy Logic Controller Rule Base

Δerror (*de*)	error (*e*)		
	N	Z	P
Negative (N)	N	N	P
Zero (Z)	N	Z	P
Positive (P)	N	P	P

Since there are three fuzzy variables (two inputs and one output) and each fuzzy variable has three membership functions, the fuzzy controller for pitch control of an aircraft has a total of nine membership functions. Each membership function is constrained to be triangular so each membership function has three parameters.

4. Results Analysis and Discussion

The simulation is the comparison between uncontrolled, PID controller, LQR controller, and FLCcontroller in the presence of disturbance such as crosswind effect. All the controller output must be the same as the desired input. The simulation of the aircraft pitch motion is observed in Matlab Simulink version 2015a. The simulation is based on the mathematical model for aircraft pitch motion system as stated in Section 2. The performance of the controllers is observed on the pitch angle of the aircraft. The parameters are aforce in the *X*-direction, force in the *Z*-direction and pitching moments. Furthermore, a disturbance is added known as cross wind effect that occurs at the aircraft pitch motion. The objective here is focused on getting the time response and maintaining the desired pitch angle even with the disturbance effect.

The first simulation involves the PID controller and the response without a controller as shown in Figure 4. From the Figure 4(a), it can be seen that the pitch control system is not stable without a controller. The curve of the pitch's angle is approaching infinity as time increases. Therefore, feedback controller needs to be designed in order to stabilize the system.

In this work, a unit step command is required in order for the pitch angle to follow a reference value at 0.2 radian. The controller output is made proportional to the error and proportionality is called the proportional gain (K_P). The K_P will have the effect of reducing the rise time but never eliminate the steady-state error. This controller is capable of maintaining the output steady-state value at desired value as shown in Figure 4(b). The K_P is set to be 1.5. With the increase of the K_P, the output will decrease the rise time but at the same time it increases the overshoot of the system. By increasing the K_P, it will produce the large oscillation before the system reaches the steady-state value. The output response gives fast response and has the 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%. In this simulation, Proportional and Integral (PI) controller do not provide the desired response, hence the analysis was omitted.

Figure 4(b) shows the corresponding response of PD controller for pitch angle to the unit step reference input. Derivative action can be used to create damping in a dynamic system and thus stabilized its behavior with the derivative action is proportional to the rate of change of the measurement of error. The derivative action can compensate in changing measurement and improve the system transient response and this will affect in increasing the stability of the system. PD controller relatively gives 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 overshoot of about 2.3%. Besides that, it also gives percentage of steady-state of about 0.05%. Increasing K_D will make the response becomes a little bit slower while decrease theovershoot.

The value of K_P, K_I and K_D are dependent on each other. Changing one of these variables can change the effect of the other two. The 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 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 of increasing the stability of the system by reducing the overshoot and improving the transient response. By using Zieglar-Nicols method and with 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 4. However, the response of PID controller is a little bit slower than PD controller in term of rise time and settling time which both values are 0.16 seconds and 1.89 seconds respectively.

As discussed in Section 3.3, two inputs have been applied to FLC controller. Figure 4(c) shows the output response of pitch control. FLC theory is close to human reasoning. These intelligent controllers are derived from some prior information or input-output data of pitch control system. The FLC itself provides good performance in term of settling time, percentage of overshoot, percentage of steady-state error and rise time. As depicted in Figure 4(c), it can be observed that the pitch angle follows the reference value. This controller is able to give a good response and better than PID controller without producing any overshoot. The response is comparatively fast that gives the settling time of about 0.96 seconds and the rise time of 0.16 seconds. The largest value of settling time is due to the PID controller which is 1.89 seconds. The highest percentage of overshoot goes to the P controller and highest percentage of steady-state error is claimed by P and PD controller at 0.05 %. However, only FLC controller has the 0% percentage of overshoot compared to the other controllers.

On the other hand, Figure 4(d) compares the controller performance on the pitch angle without disturbance for uncontrolled, PID and LQR controller. The most obvious finding is the overshoot for PID which is higher than LQR but the same controller gives shorter in rise time. However PID has steady state error of 1.2% while LQR has no steady state error. LQR has slow in rise time but less in overshoot. The settling time for LQR is better and quicker than PID. Thus LQR controller gives less jerking and bouncing of the aircraft while on air.and thus it is is better than PID for the safety and comfort purpose of the passengers.

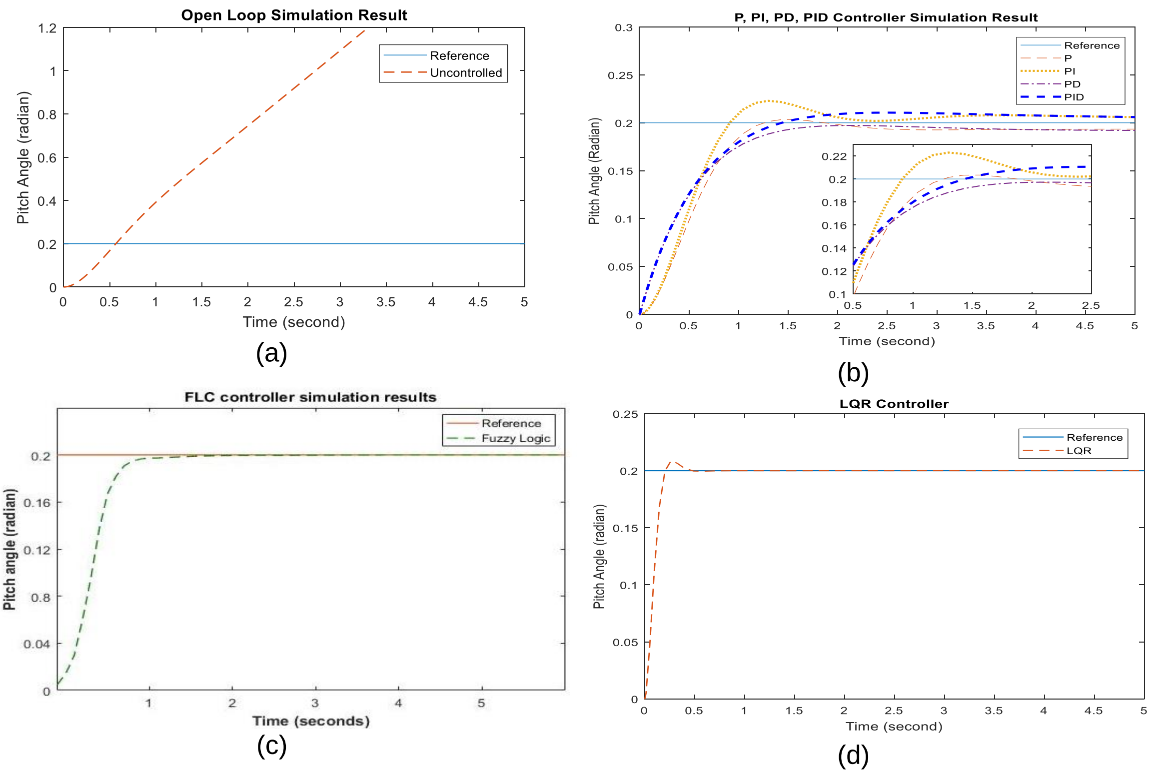


Figure 4. System performance (a) without controller, (b) P, PD, PID controller, (c) FLC controller, (d) LQR controller

Next, in order to know the controllers is robust or not, a new input pitch angle is set and a disturbance is applied. The chosen value for new input pitch angle is 0.1 radians and 0.3 step input is applied as a disturbance on both chosen values of pitch angle. From Figure 5, it shows that the PID controller still gives good performance by achieving the criteria needed. The PID controller still settles down and follows the desired input of 0.1 radians. It can be concluded that the PID controller is robust due to its performance from different input responses and when having a disturbance.

From the previous results, it shows that the FLC controller will give a better performance compare to PID controller by achieving the enhance criteria needed. When the input pitch angle is set to 0.2 radians, it will settle down and achieve the desired input. Figure 5 shows the graph for the input pitch angle of 0.1 radians. As in PID controller, to test the robustness of the FLC controller, a new input pitch angle of 0.2 radians is set and a disturbance of 0.2 step input is applied. From Figure 5, it shows that the FLC controller still gives a good performance by achieving the criteria needed. The FLC controller still settles down and follows the desired input of 0.1 radians. For the disturbance, there is some steady-state error because it does not reach the settling time for 0.2 radians and 0.1 radians. There are some tunings that are needed to be done in the future in order to improve the performance of the FLC when the controller is applied with the 0.3 step input disturbance.

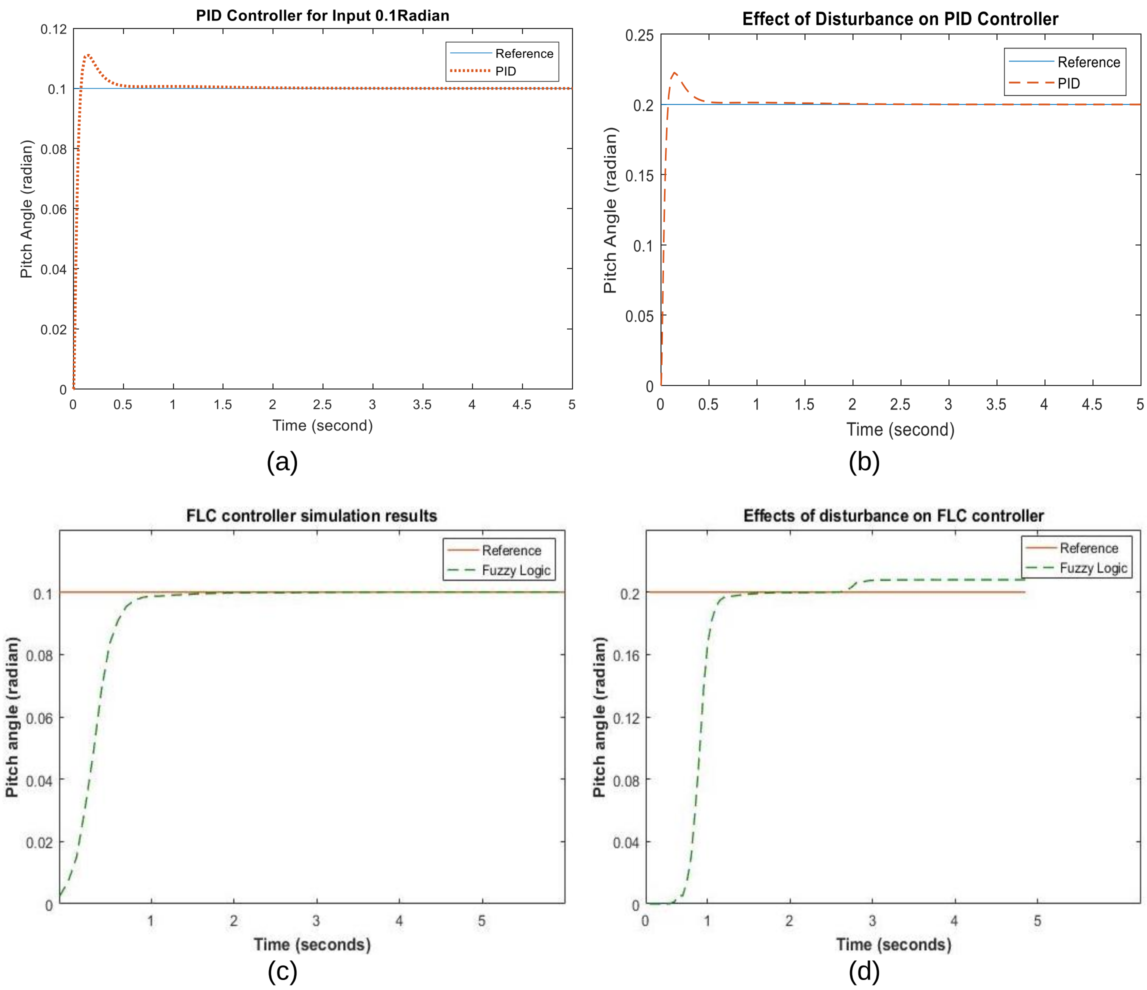
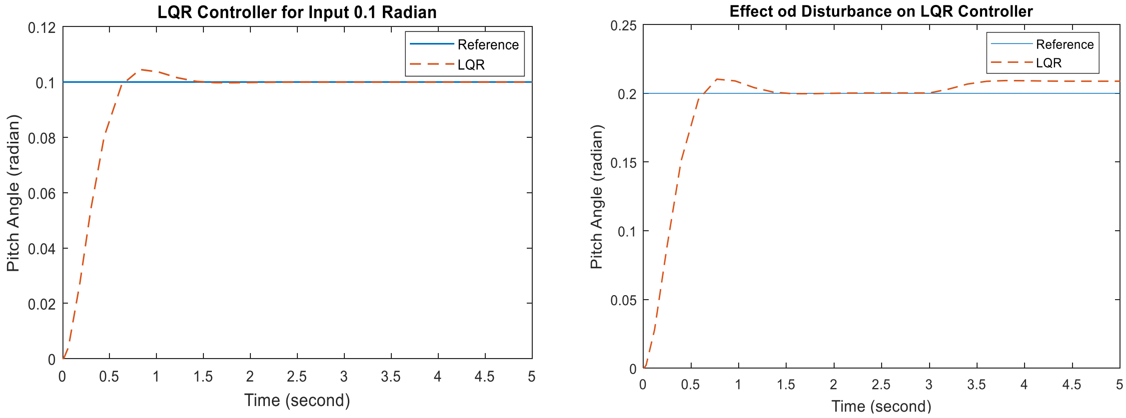


Figure 5. System performance (a) PID: input 0.1 rads, (b) PID: input disturbance 0.2 rads, (c) FLC: input 0.1 rads, (d) FLC: input disturbance 0.2 rads, (e) LQR: input 0.1 rads, (f) LQR: input disturbance 0.2 rads



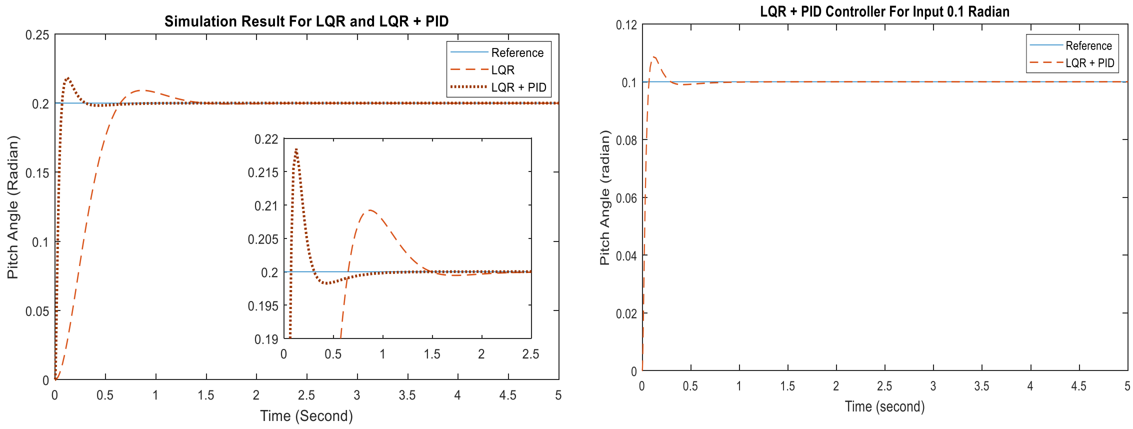
(e) (f)

Figure 5. System performance (a) PID: input 0.1 rads, (b) PID: input disturbance 0.2 rads, (c) FLC: input 0.1 rads, (d) FLC: input disturbance 0.2 rads, (e) LQR: input 0.1 rads, (f) LQR: input disturbance 0.2 rads

Next, Figure 6(a) shows the comparison of uncontrolled, PID, and LQR for pitch angle of aircraft model with cross wind disturbance at three seconds. PID still have the highest overshoot and has steady state error. Both PID and LQR controller is unable to reduce the overshoot at three seconds and thus the aircraft may be facing jerk and collision for three seconds depending on the settling time. Obviously, the pitch angle will not be achieving the desired angle after three seconds.

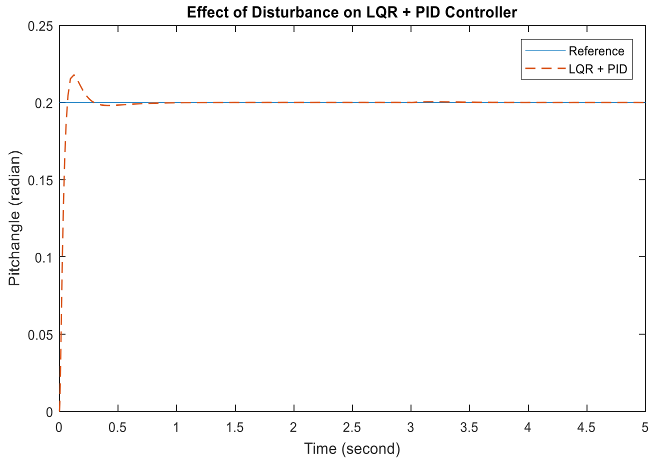
A single controller alone is unable to reduce and eliminate error at 3 seconds as shown in Figure 6(a). Thus, the solution to the elimination of unwanted error is by implementing two controllers in the system. Figure 6(b) illustrates for PID+LQR with disturbance of cross wind effect for an aircraft system which is heavier and having high momentum. From Figure 6(b), the overshoot was managed to be reduced by 0.03% but the steady state error is 1.2%. The disturbance at the three second was managed to be eliminated. So the aircraft will not have the jerking and bounce movement when the cross wind takes place.

Last but not least, the results can be proposed to the aviation companies to improve their performance in serving people across the world. In addition, they can promote a safe and high quality standard of the flight using the control systems that are being used in this research.



(a) (b)

Figure 6. System performance (a) LQR: simulation results for input disturbance 0.2 rads, (b) LQR+PID: input 0.1 rads, (c) LQR+PID: input disturbance 0.2 rads



(c)

Figure 6. System performance (a) LQR: simulation results for input disturbance 0.2 rads, (b) LQR+PID: input 0.1 rads, (c) LQR+PID: input disturbance 0.2 rads

5. Conclusion

The validated model of pitch control of an aircraft is very helpful in developing the control strategy for actual system. Pitch control of an aircraft is a system which requires a pitch controller to maintain the angle at its desired value. This can be achieved by reducing the error signal which is the difference between the output angle and the desired angle. The mathematical model of the dynamic longitudinal equation has been derived successfully and the proposed controllers have been diagnosed and investigated. Three methods for designing a controller are developed; those are PID controller, FLC controller, and LQR controller. It is quite tedious to design the third order system, thus for PID controller method, the rate feedback was added in order to improve the system performance. From the earlier discussion, it can be concluded that all the controller design strategy is capable of controlling the pitch angle of an aircraft from the linearized system. By improving the pitch motion angle, the flight will be stabilized and in steady cruise where the aircraft does not have any jerking effect and all the passengers and the crews will feel comfortable. Further improvement needs to be done for the FLC controller in order to improve more on its performance although it is better than PID controller.

The fuzzy rules or the other parameters such as universe of discourse, defuzzification method has to be tuned to get a better performance. When it comes to some external disturbance such as bad weather or strong wing effect, a single controller alone is unable to solve or control the system that is added with disturbance. Thus, using two controllers consisting of PID and LQR together has managed to stabilize the aircraft in pitch motion situation. Even though the controllers manage to help stabilize the pitch motion system, there are other motions that can be controlled in aircraft such as the yaw and roll motions. In addition, efforts also are devoted in developing more advanced control techniques such as nonlinear control, robust control, adaptive control and others.

The nonlinear control technique should have more advantages in the pitch control system since the system considers the nonlinear parameters.Another controller that we should be focused on is the artificial intelligent controller such neural network, and artificial neural fuzzy interfere system. These controllers are among the best controllers in order to control a complex system such as an aircraft system.

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