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Integral Modified Linear Quadratic Regulator Method for Controlling Lateral Movement of Flying Wing in Rotational Roll Mode

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Abstract: The studies on fix wing Unmanned Aerial Vehicle (UAV), especially, flying wing type has progressed very rapidly. Therefore, fix-wing UAV can be used to support disaster mitigation and monitoring. But in order to make UAV completes the mission, it needs an autonomus flight control system. There were two types of control method in order to control the movement of flying wing, i.e., longitudinal and lateral. Control of lateral motion is considered more important because it serves to prevent air turbulence that can cause the air to move is not in accordance with the mission that has been set. This research aims to design control lateral movement that is focused on using the roll rotational motion mode control method Linear Quadratic Regulator (LQR) with the addition of integral in dealing with steady state error that occurred. Experiments conducted by simulating the model airframe using DATCOM and control simulation using MATLAB. The results showed that by using modifeied LQR control with integral components added, the system is able to stabilize the rotational motion roll in a short time and minimal overshoot. In addition, the experiment also showed that steady-state error in the system can be eliminated with a rise time = 0.189 sec, overshoot = 0.688%, settling time = 0.301 sec and steady state error = 0%.

Key words: Autonomous flight control, integral modified LQR, stability, steady state error, experiments, UAV

INTRODUCTION

Currently research on flying robot which is also known as Unmanned Aerial Vehicle (UAV) has been progressing very rapidly. This can be seen from various UAV utilization which is programmed to perform a variety of missions as missions that are monitoring the volcano area, disaster mitigation, plantations mapping and other missions in the vast area (Mardiatno *et al*., 2015).

Not all types of UAVs can be used in these missions. Only UAV with the capability to fly long distance and cruising in long endurance can finish ithe mission. One type of UAV that meets the specifications is flying wings. This type of UAV movements are controlled by the wing elevators which is a combination between the wings elevon to the upward or downward and the wing aileron to the right or to the left movement.

A robust flight control system is required in order to demonstrate the capability of flying wings, being able to fly within a large area, long distance and long durability (Priyambodo *et al*., 2016). Without such a control system, the flying wing is not responsive to the characteristic change. Also, it makes the flying wing is not stable. In the worse conditions without a robust control system, the flying wing can be stalled.

Lateral axis control is very important in the success of the use of flying wing. It is related to the roll axis movement of the airframe. A such control will protect the airframe from turbulence when cruising in a straight direction or in a turn maneuver.

In order to control the airframe lateral stability, a Linear Quadratic Regulator (LQR) method can be implemented. LQR control is an optimal control system which has a good accuracy due to the liniearization. LQR control method is an optimal control system with linearization process approaches that have high accuracy. This control system consists of feedback amplifier component for each controlled state and supported by the ability to regulate the system robustly (Athans and Falb, 2016). However, this kind of control system still has limitation in order to handle the steady state error. Due to this limitation, the airframe be able to have a multi-overshoot response to the steady state error. As a result the airframe can perform rolling movement suddently, until upside-down position. This accident happens when the steady state error is beyond the stabilizing capability of the control system.

To overcome that limitation, it needs to add an extended algorithm to LQR control method in order to handle response-time, overshoot and steady state problem. By using this modified algorithm the airframe will be more stable. Consequnetly, the flying wing type UAV can be used to conduct missions with the best performance.

MATERIALS AND METHODS

Unmaned Aerial Vehicle (UAV) and flying wing: Unmanned Aerial Vehicle (UAV) is an unmanned airframe system that has ability to do various types of remote sensing mission-based on video and photos, either for civilian or military purposes. The missions that can be done by UAV include surveillance, reconnaisance, monitoring, air patrols, high-resolution aerial photography, goods transportation, etc. (Nonami *et al*., 2010). UAV is well-known as unmanned airframe and also known as UAS (Unmanned Aircraft System) in United States. UAV is defined as a plane without a pilot which uses aerodynamic forces to fly, either automatically with the help of autopilot or remotely operated with the help of a remote control and can carry a payload of weapons or not.

One of UAV type (Unmanned Aerial Vehicle) is a flying wing. Flying wing is a form of a fixed wing airframe which has elevator (combined Ailerons and Elevon) on both wings without any ruder wings in the rear of the plane. The airframe has the capability of glading in the air with a long distance flying. This airframe uses 1 piece of rotor (propeller) and 2 servo motors in regulating the movement of flying (Priyambodo *et al*., 2016).

Dynamics of flying wing airframe: The coordinate systems of flying wing airframe include force style, torque, translational linear speed and angular velocity for each axis orientation. The component parameters fuselage axis coordinate system illustrated clearly in Fig. 1.

X, Y and Z is the external force of airframe, L, M and N is external torque airframe, u, v and w are linear velocity, p, q and r is velocity of angle for each angle orientation (roll, pitch and yaw). In addition, there are two additional important components in controlling fly such as angle α (angle of attack) and angle β (slideslip angle). The angle α represents the angle formed between x-axis and the wind on the fuselage from e x-z direction. Meanwhile, the angle β represents the angle that is formed between the x-axis and the wind that on the fuselage from x-y. Both of those angle is illustrated in Fig. 2.

There are six equations that describe equation of dynamic object and three kinematic equations with the addition of inertia I_{xx} equation, I_{yy}, I_{zz} and orientation I_{xz} show the movement of airframe in the form, written in Eq. 1 and in the form of non-linear systems (Markin, 2010):

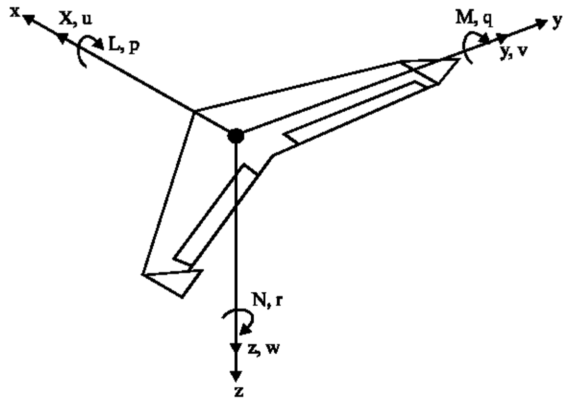

Fig. 1: Fuselage axis coordinate system (Markin, 2010)

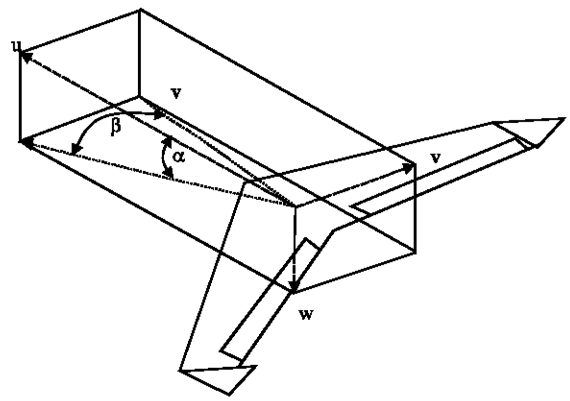

Fig. 2: Angle of attack and sideslip angle (Markin, 2010)

$$\begin{bmatrix} \dot{u} \\ \dot{v} \\ \dot{w} \\ \dot{p} \\ \dot{q} \\ \dot{r} \\ \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} = \begin{bmatrix} -g\sin\theta + rv - qw + \frac{X}{m} \\ g\cos\theta\sin\phi - ru + pw + \frac{Y}{m} \\ g\cos\theta\sin\phi + qu - pv + \frac{Z}{m} \\ \frac{1}{I_{xx}}\left[L + I_{xz}(\dot{r} + pq) + I_{yy} - I_{zz})qr\right] \\ \frac{1}{I_{yy}}\left[M + I_{xz}(r^2 + p^2) + I_{zz} - I_{xx})rp\right] \\ \frac{1}{I_{zz}}\left[N + I_{xz}(\dot{p} + qr) + I_{xx} - I_{yy})pq\right] \\ p + (q\sin\phi + r\cos\phi)\tan\theta \\ q\cos\phi - r\sin\phi \\ (q\sin\phi + r\cos\phi)\sec\theta \end{bmatrix} \quad (1)$$

In describing the dynamic characteristics approach in more detail, then there are two theories that are

Table 1: Variables in the longitudinal and lateral motion

Longitudinal variables	Lateral variables
u, w, q, θ	v, p, r, ϕ, ψ

involved. Those two theories are Small Disturbance Theory (SDT) and Small Angle Theory (SAT) as equation involving a state of change disorder airframe system on Eq. 1. The new equation for each state is shown by Eq. 2 (Markin, 2010) which is aimed to simplify the equation and then it is assumed that there is no discruption components at all by providing a zero value:

$$\begin{aligned}
\Delta\dot{u} &= \frac{\Delta X}{m} - g\Delta\theta\cos\theta_0 \\
\dot{v} &= \frac{\Delta Y}{m} + g\phi\cos\theta_0 - u_0 r \\
\dot{w} &= \frac{\Delta Z}{m} + g\Delta\theta\sin\theta_0 - u_0 q \\
\dot{p} &= \left(I_{xx}\ I_{zz} + I^2{}_{xz}\right)^{-1}\left(I_{zz}\Delta L - I_{xz}\Delta N\right) \\
\dot{q} &= \frac{\Delta M}{I_y} \\
\dot{r} &= \left(I_{xx}\ I_{zz} + I^2{}_{xz}\right)^{-1}\left(I_{xz}\Delta L - I_{xx}\Delta N\right) \\
\Delta\dot{\theta} &= q \\
\dot{\phi} &= p + r\tan\theta_0 \\
\dot{\psi} &= r\sec\theta_0
\end{aligned} \tag{2}$$

Based on the Eq. 2, the equation is divided into two groups such as the equation of motion which describe longitudinal and lateral. Variable states of both movements are shown in Table 1.

Linear quadratic regulator: Linear quadratic regulator is called linear because its model and controller form is in linear form. It is also called as quadratic because of its cost function is quadratic. While, it is called as regulator because of its reference is not in the form of a function of time (Whidborne, 2007). Method of LQR control system is illustrated in a block diagram as shown in Fig. 3.

Based on Fig. 3, an equation of linear control system on LQR controlling method with constant value D is equal to zero because in designing the control system provided with the value of the feedback (feedback), the parameter D will be ignored and considered does not influence the outcome, so the system can be written through Eq. 3 and 4:

$$\dot{x} = Ax + Bu \tag{3}$$

$$y = Cx \tag{4}$$

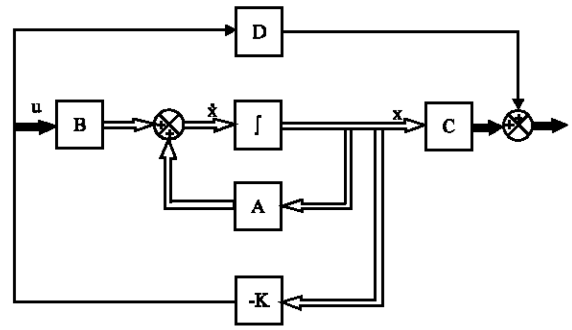

Fig. 3: Diagram blok of LQR control system

Where:
x_{n*1} = State system
u_{mxn} = Input process
y = Output process
A = Matrix system A_{n*n}
B = Matrix system B_{m*n}
C = Matrix system C_{1*n}
K = Gain feedback

By entering system u and K can be obtained through Eq. 5 and 6:

$$u = -Kx \tag{5}$$

$$K = R^{-1}B^{T}P \tag{6}$$

In designing the control system to be optimal, then it performs a minimizing of energy (cost function/quadratic function) which is defined through index of performnce in the interval [t_0, 8] in Eq. 7 (Athans and Falb, 2016) in which this function involves matrix Q and R to create a non negative matrix system:

$$J = \frac{1}{2}\int_{t^0}^{\infty}\left(x^{T}Q_x + u^{T}R_u\right)dt \tag{7}$$

Where:
t^0 = First section
∞ = Last section
Q = Semi definite positive matrix
R = Definite positive matrix

Problems of regulator in this method can be solved by using algebratic Riccati equation which can be searched through the Eq. 8:

$$A^{T}P + PA + Q - PBR^{-1}B^{T}P = 0 \tag{8}$$

Ballast parameters of Q and R in the equation is the matrix that affects a large size of matrix P which is

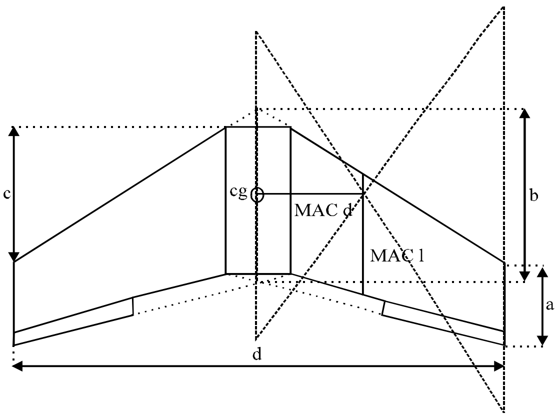

Fig. 4: Design of mechanical flying wing

Table 2: Desription and value of variables

Variables	Description	Values
a	Tip chord	2.25×10^{-1} m
b	Root chord	4.0×10^{-1} m
c	Sweep	3×10^{-1} m
d	Wing span	1.9 m

conducted an experiment to get the most optimal K where Q is semi-definite positive matrix and R is positive definite. Weighting Q and R is based on the following two things (Domingues, 2009):

- The larger of Q value, getting larger value K strength, therefore it accelerates the system to reach steady-state (intermediate state cost function)
- The bigger the value of R, it will reduce the value of K and decrease the value of K strenght and decelerate steady state (energy drive)

Mechanic desain of flying wing airframe: Flying wing airframe is designed depending on the needs of research that is called as monitoring mission. The airframe was built by using polyfoam material in medium sizewhich is suitable for flying at an altitude of 100 m above sea level and in medium speed of wind. Plan design and the results design are shown in Fig. 4.

Variables on the mechanical design are a size of the components of airframe flying wing. The sizes of these variables are presented in detail in Table 2.

Furthermore, these measurements are used to calculate the position of the center of gravity as well as other parameters that will be used as a determinant of co-eficient value on motion characteristics to fly the airframe. The parameters are calculated with the help of CG Software calculator. The result of these calculations is shown in Table 3.

Based on the draft plan of the airframe mechanics, airframe can be built manually. The result of the draft implementation plan is shown in Fig. 5 with a payload that has been adapted to the CG positions of airframe.

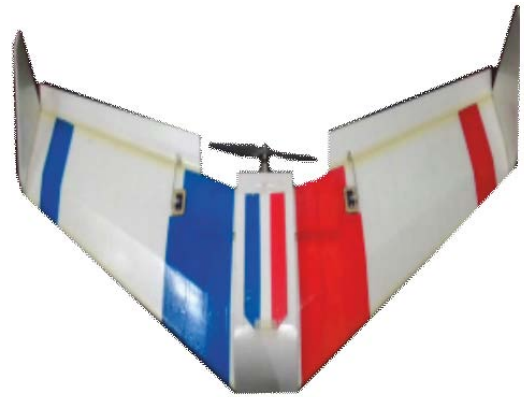

Fig. 5: Mechanic design implementation

Table 3: Airframe parameters

Parameters	Values
Wing area	3.57×10^{-1} m^2
MAC distance (MAC d)	2.44×10^{-1} m
MAC length (MAC l)	3.38×10^{-1} m
Center of gravity (cg d)	2.30×10^{-1} m

Flying wing identification: Movement of the flying wing is divided into two groups, namely longitudinal movement associated with pitching motion and the translational motion in the direction of the x-axis and lateral air associated with rolling motion and the translational motion in the direction of the y-axis plane. The focus of this study is on controlling air lateral movement. Based on Table 1, the variables affecting the lateral movement of the airframe is v, p, r, ϕ, ψ where the variable can be obtained through the Eq. 2 with a non-linear approach. These equations must be converted into the form of a linear function that is in the form of state space as the basic form linear quadratic optimal control. Equation 2 contains variables of lateral motion which is converted into a linear function and involving components force/style and torque disturbance on the system. Application of these functions is designed through the method of Taylor expansion.

On the lateral motion, variable longitudinal does not affect the style of lateral interference, so the style of these disorder and interference can be described by the Eq. 9:

$$\Delta Y = f(v,p,r,\phi,\psi,\delta) \tag{9}$$

where, δ represents the application of the force control with components that occur on the y-axis. Throughout Taylor expansion through, then disorder force on the y-axis will have the Eq. 10:

$$\Delta Y = \frac{\partial Y}{\partial v}v(t)+\frac{\partial Y}{\partial p}p(t)+\frac{\partial Y}{\partial r}r(t)+\Delta Y_C(t) = Y_v(t)+Y_p(t)+Y_r(t)+\Delta Y_C(t) \quad (10)$$

where, the parameter Y_v, Y_p and Y_r is the stability of derivative and c indicates the application of the controlling force through the actuator. On the lateral air controllers, actuators are used in the form of a servo motor to adjust the wing aileron and rudder which parameter aileron and rudder can be shown by the equation system function in Eq. 11:

$$\Delta Y_C = T_{\delta a}\Delta\delta_a + Y_{\delta t}\Delta\delta_t \quad (11)$$

After subtituting each force and torque to the lateral component of the Eq. 2, it will get the state space linear form that is shown in Eq. 12:

$$\begin{bmatrix}\Delta\dot{B}\\ \Delta\dot{p}\\ \Delta\dot{r}\\ \Delta\dot{\phi}\end{bmatrix} = A_{long}\begin{bmatrix}\Delta B\\ \Delta p\\ \Delta r\\ \Delta\phi\end{bmatrix} + B_{long}\begin{bmatrix}\Delta\delta_a\\ \Delta\delta_r\end{bmatrix} \quad (12)$$

In lateral controlling, component ψ is not included in the state vector and the inertia of the x-axis and z-axiz produce yield control of sideslip angle changes which are included in the state vector.

Lateral dynamic of flying wing airframe consists of three modes namely dutch roll, roll mode and a spiral mode. Research that will be done is focused more on stage of roll mode. Thus, the components of vector used is roll (ϕ) and angular velocity roll (p) with the changes in the stability of derivatives rudder which is equal to zero because flying wing does not have any rudder wings, so that Eq. 12 is reduced to the Eq. 13:

$$\begin{bmatrix}\Delta\dot{p}\\ \Delta\dot{\phi}\end{bmatrix} = A_{lat}\begin{bmatrix}\Delta p\\ \Delta\phi\end{bmatrix} + B_{lat}[\Delta\delta_a]$$

$$\begin{bmatrix}\Delta\dot{p}\\ \Delta\dot{\phi}\end{bmatrix} = \begin{bmatrix}\left(\frac{L_p}{I'_x} + I'_{zx}N_p\right) & 0\\ 1 & 0\end{bmatrix}\begin{bmatrix}\Delta p\\ \Delta\phi\end{bmatrix} + \begin{bmatrix}\left(\frac{L_{\delta a}}{I'_x} + I'_{zx}N_{\delta a}\right)\\ 0\end{bmatrix}[\Delta\delta_a] \quad (13)$$

The output is written in Eq. 14:

$$y = \begin{bmatrix}0 & 0\\ 0 & 1\end{bmatrix}\begin{bmatrix}\Delta p\\ \Delta\phi\end{bmatrix} \quad (14)$$

where, the parameters of inertia I'_x and I'_{zx} can be obtained by calculating the Eq. 15 and 16:

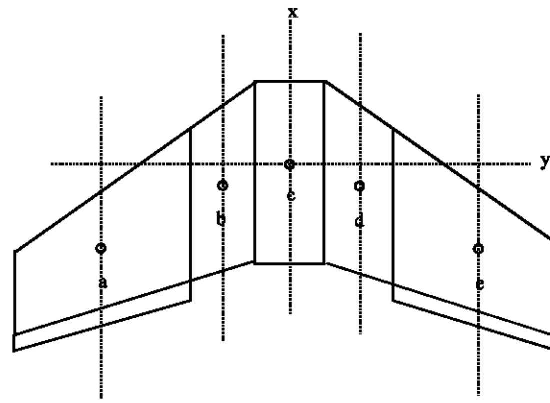

Fig. 6: Points of inertia in the airframe

$$I'x = \left(I_xI_z - I_{xz}^2\right)/I_z \quad (15)$$

$$I'zx = I_{xz}/\left(I_xI_z - I_{xz}^2\right) \quad (16)$$

Moment of inertia I_x and I_z also product of inertia I_{zx} is calculated through the calculation of the airframe by using mechanism of dividing 5 and three point of inertia section. The division is indicated in Fig. 6.

The points "a" through "e" is a point of "cg" for each field that affects "cg" of central airframe in determining great value moment of inertia and inertia product. Point "a" and point "b" and "e" are point where inertia inertia $I_{GKS2'}$ occurs. Point "b" and "d" result in inertia occurrence of $I_{GKS1'}$. Great value for the moment of inertia at the point "a" and "e" and also "b" and "e" are equal, so that, the calculations can be done once and then it is multiplied two. According to the basic of the inertia moment of I_x and I_z and also inertia product of I_{xz} can be obtained through the Eq. 17-19:

$$I_x = 2I_{Gx_{KS2}} + 2I_{Gx_{KS1}} + 1_{Gx_{KP}} \quad (17)$$

$$I_z = 2I_{Gz_{KS2}} + 2I_{Gz_{KS1}} + 1_{Gz_{KP}} \quad (18)$$

$$I_{xz} = 2I_{Gzx_{KS2}} + 2I_{Gzx_{KS1}} + 1_{Gzx_{KP}} \quad (19)$$

Meanwhile, the parameter of L_p, N_p, $L_{\delta a}$ and $N_{\delta a}$ is included in derivative latera as depicted in Eq. 20-23 (Bagheri, 2014):

$$L_p = \frac{QSb^2}{2u_0I_x}C_{lp} \quad (20)$$

$$N_p = \frac{QSb^2}{2u_0I_x}C_{Np} \quad (21)$$

Tabel 4: Lateral parameter values

Parameters	Values/Unit
Q	1.21 kg/m^3
S	3.57×10^{-1} m^2
b	1.09 m
C_{l_p}	-2.51×10^{-3}
C_{N_p}	1.13×10^{-4}
$C_{l_{\delta a}}$	2.54×10^{-2}
$C_{N_{\delta a}}$	1.82×10^{-3}
I_x	4.19×10^{-2} kg/m^2
I_z	5.06×10^{-2} kg/m^2
I_{zx}	7.67×10^{-4} kg/m^2
u_0	1.46×10^{1} m/sec^2

$$L_{\delta a} = \frac{QSb}{I_x} C_{l_{\delta a}} \quad (22)$$

$$N_{\delta a} = \frac{QSb}{I_z} C_{N_{\delta a}} \quad (23)$$

Where:

Q	=	The density of air
S	=	Wing area
b	=	Wing span
u_0	=	The velocity of airframe
$C_{l_p}, C_{N_p}, C_{l_{\delta a}}$ and $C_{N_{\delta a}}$	=	The co-efficient of rolling and aileron

The coefficients are obtained from the simulation of airframe which was designed by using software datcom. In the simulation to find the characteristics of airframe in form of algorithm given on datcom, longitudinal motion is assumed stable with output 0, then the speed of airframe for the environment is set at 14.61 m/sec at an altitude of 100 m. The result of the simulation is in the form of coefficient roll motion as well as three-dimensional shape of airframe.

The result of coefficient values of parameters is presented in Table 4, along with other parameters that create the value of state vector on function component of linear state space. From the parameters which consists identified values shown lateral state space (Eq. 13):

$$\begin{bmatrix}\Delta \dot{p}\\ \Delta \dot{\phi}\end{bmatrix} = \begin{bmatrix}-0.02498 & 0\\ 1 & 0\end{bmatrix}\begin{bmatrix}\Delta p\\ \Delta \phi\end{bmatrix} + \begin{bmatrix}6.7732\\ 0\end{bmatrix}[\Delta\delta_a]$$

Intergal added LQR control design: LQR control system still has flaws in eliminating steady state error that occurs in the system, so that it can be solved with the addition of the control components in the form of integral control. Integrator capabilities in eliminating steady state error will also be able to eliminate interference on inpit system when integrator is placed before the parameters of the control system disorders. The structure of this integral component additions make the block diagram control of LQR become equal as in Fig. 7, that is LQR control with the addition of the concept of tracking with any reference. where r is the reference value or the expected value, e is an error and Ki is constants integral control.

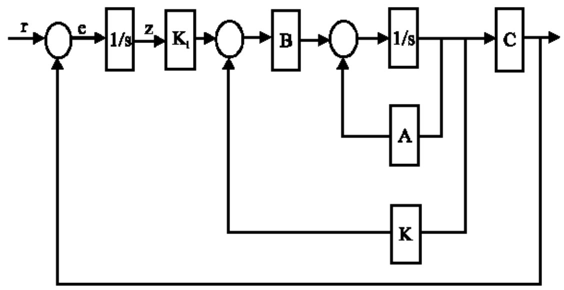

Fig. 7: Integral added LQR control

Giving an extra integrator can be done through the augmentation of state linear equation system by adding a new state as shown in Fig. 7. The state is in form of state z which is the result of integration on resulting error value (e). State space model of Eq. 3 is modified by adding integrator and it will be in the same form of Eq. 24:

$$\begin{bmatrix}\dot{x}\\ \dot{z}\end{bmatrix} = \begin{bmatrix}A & 0\\ -C & 0\end{bmatrix}\begin{bmatrix}x\\ z\end{bmatrix} + \begin{bmatrix}B\\ 0\end{bmatrix}u + \begin{bmatrix}0\\ I\end{bmatrix}r \quad (24)$$

From Eq. 24, state space model of lateral system in rollflying wing mode in Eq. 13 become Eq. 25:

$$\begin{bmatrix}\Delta \dot{p}\\ \Delta \dot{\phi}\\ \Delta z\end{bmatrix} = \left[\begin{pmatrix}\frac{L_p}{I'x} + I'_{zx} N_p & 0\\ 1 & 0\end{pmatrix}\right]\begin{bmatrix}\Delta p\\ \Delta \phi\\ \Delta z\end{bmatrix} + \begin{bmatrix}\left(\frac{L_{\delta a}}{I'x} + I'_{zx} N_{\delta a}\right)\\ 0\\ 0\end{bmatrix}[\Delta\delta_a] + \begin{bmatrix}0\\ 0\\ 1\end{bmatrix} r \quad (25)$$

e which is represented by z state on the state model can be obtained from the difference between the output system to the expected reference value (e = state z = y-r) where output-system controlled is roll angle ($y = \phi$). So, the state equation of output variables in Eq. 14 will be equal as Eq. 26:

$$y = \begin{bmatrix}0 & 0 & 0\\ 0 & 1 & 0\\ 0 & 0 & 0\end{bmatrix}\begin{bmatrix}\Delta p\\ \Delta \phi\\ z\end{bmatrix} \quad (26)$$

Minimizing energy (cost function/quadratic function) which is defined through performance index in the interval of $[t^0, \infty]$ with the addition of an integral state taht is shown by Eq. 27:

$$J = \frac{1}{2}\int_{t^0}^{\infty}\left(x^T R_{zz} x + u^T R_{uu} u\right)dt \quad (27)$$

Therefore, the input of new system can be obtained from the Eq. 28:

$$u = -K(x - x_d) - K_i z + u_d \tag{28}$$

where by x_d a reference which is usually minimally assumed as possible that is can be ignored and u_d is an input of variable disorder that is often assumed to produce fault system which is equal to zero. In order to facilitate the search for the value of feedback gain u_d is determined at zero.

RESULTS AND DISCUSSION

Simulation of lateral stability control of UAV motion rotation in flying wing using LQR method: Model of state space variables designed to build LQR control on the lateral control of the rotational motion roll has the following parameters:

$$A_{lat_{roll}} = \begin{bmatrix} -0.02498 & 0 \\ 1 & 0 \end{bmatrix}$$

$$B_{lat_{roll}} = \begin{bmatrix} 6.7732 \\ 0 \end{bmatrix}$$

$$C_{lat_{roll}} = [0.1]$$

$$D_{lat_{roll}} = \begin{bmatrix} 0 \\ 0 \end{bmatrix}$$

Characteristics of LQR control on the lateral control rotation geral roll for equation model of state space airframe when flying over an altitude of 100 m above the ground and at speeds of m/sec^2, it is simulated by using MATLAB Software assistance in which the characteristic control with the best results can be obtained by weighting Q and R.

Weighting given is done by varying the value of Q to state roll angle and roll angular velocity, R is set constant at a value of 1. The first test, variety of weighting is conducted on Q for state of roll angle with weight Q-state on roll angular velocity and R is equal to 1 that tested at 1 rad reference value. The result of simulation is completed with the characteristic of roll angle output control. Figure 8 shows the results.

Based on the characteristic of result control by tuning variation of weighting Q state roll angle, it can be cocluded that the greater value of the weighting in the state controlled, the faster accelerating into the system to

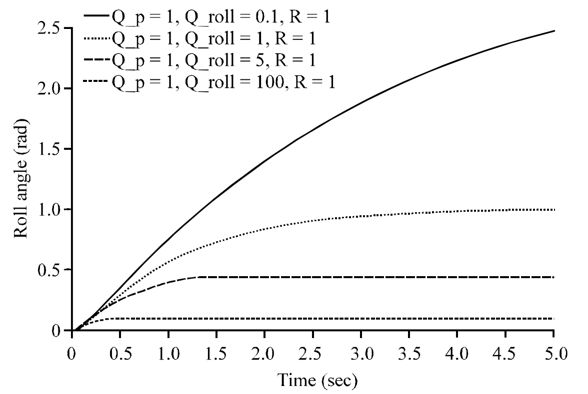

Fig. 8: Step response for variation of roll angle with weighted Q

achieve steady state. This can be seen in the rapid rise time value for each increase in the weighting value Q. This condition occurs because the larger the value of Q will produce a greater feedback gain (K) and it will improve responsiveness system. However, the increase greater value of the weighting Q then steady state error that occurs will be even greater, where the steady state error is the difference in the value system in steady state. In addition, the smaller weighting Q state that is controlled by roll angle, the system will run into oscillation.

The system will have a good stability if it meets the criteria overshoot that occurs a maximum of 5% of steady state, the speed of the system in achieving steady state that is represented by rise time of <1 sec and settling time that is produced <3 sec as well as errors that occur or maximum steady state error is 2% from the largest deviation of the system where the maximum rotational motion of the flying wing is 45^0, so that, the maximum error that occurs is 0.90 (Ogata, 2010).

Although, the value of research time has not gained the criteria of the system which has good control but the variations of the weighting of Q-state roll angle that can meet the stability criteria are weighting with the value $Q_p = 1$, $Q_\phi = 1$ and $R = 1$. These characteristics still can be improved by tuning the weighting Q for state roll of angular velocity. Therefore, the second test is done by tuning the weighting of $Q_\phi = 1$ dan $R = 1$. The tuning weighting is simulated by varying the value of Q_p. The simulation gives result in output characteristic of stability system that is completed with K values as shown in Fig. 9.

Figure 9 indicates that the greater the value of Q_p, the rise time value will increase more in the GCC and it will result in system to be less responsive and the overshoot that occurs will be more pressured and even to be eliminated. In other words, the variable state of angular

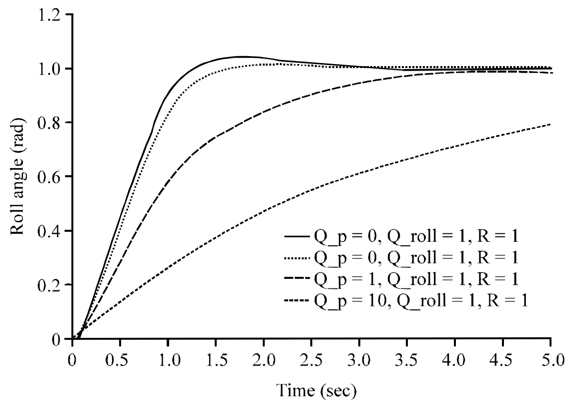

Fig. 9: Step response of Q variation of roll angular velocity

velocity is a state that serves as a buffer in a controlled system. This case happens because the state of angular velocity is first derivative of the roll angle. Therefore, when the state is multiplied by the value of strengthening feedback gain that will occur is avoid the surge value and cause a lack of responsiveness of the system under control.

From the results of the weighting, the characteristic bof best control syste is found in the weighting $Q_p = 0, 1$, $Q_\phi = 1$ and R = 1, although, there are still any overshoot and steady state error.

Simulation of lateral stability control in UAV flying wing of rotational motion by using integral added LQR method: Statespace that is formed on the lateral control of the roll rotational motion with the addition of the integral component which has parameters model as:

$$A_{lat-iroll} = \begin{bmatrix} -0.02498 & 0 & 0 \\ 1 & 0 & 0 \\ 0 & 1 & 0 \end{bmatrix}$$

$$B_{lat-iroll} = \begin{bmatrix} 6.7732 \\ 0 \\ 0 \end{bmatrix}$$

$$C_{lat-iroll} = \begin{bmatrix} 0 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 1 & 0 \end{bmatrix}$$

$$D_{lat-iroll} = \begin{bmatrix} 0 \\ 0 \\ 0 \end{bmatrix}$$

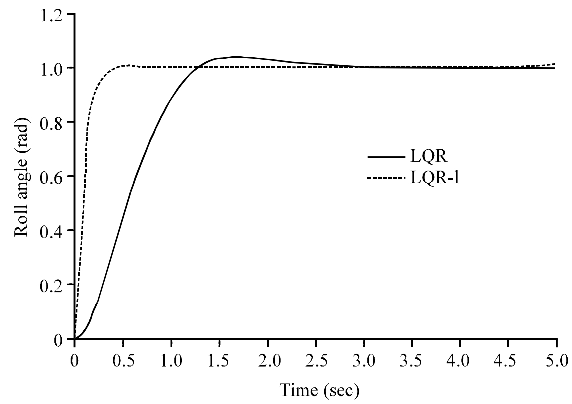

Fig. 10: Step response stability lateral movement stability rotation roll using LQR and integral added LQR

Statespace model with the addition of this integral is simulated by using LQR control method command in MATLAB Software with adding several commands of variable component inside. In order to obtain the output corresponding to the concept, the syntax is written in MATLAB:

Hi = ss(Ai-Bi*Ki, Bi*Ki*xd - [0; 0; Ci*xd], Ci, 0)

where, xd is a variable of placeholder matrix as a reference input system. The expereiment that has been conducted shows taht the tuning of weighting Q and R,lateral control of roll rotational motion is done to get the best integra K and K and it obtained $Q_p = 5.235$, $Q_\phi = 1000$, $Q_z = 1.234$. Weighting Q for state angular velocity is increased to a value figure of more than 1 and it is because their integral components make the system to be more responsive than before, so, if K is at state of angular velocity is regulated small and it is not too significant to dampen the system and to eliminate the overshoot that occurs.

The result of the simulation of LQR control with the addition of integral is compared with the results of LQR simulation by tuning the best of Q and R to know the different characteristics of the control. These results are presented in graphical form in Fig. 10.

Figure 10 shows that implementing LQR method to the roll rotational motion of flying wing, obtains a rise time = 0.906 sec, overshoot = 1.145%, settling time = 1.011 sec and steady state error = 0.01%. Meanwhile, implementing Integal Added LQR method obtains rise time = 0.189 sec, overshoot = 0.688%, settling time = 0.301 sec and steady state error = 0%.

The result shows that with the addition of integral to the LQR, improve the response significantly in stabilizing the roll rotational motion of airframe based on the reference that is given by <1 sec and in a very minimum overshoot. In addition, there is no error that occurs when the system reaches a steady state or there is no occurence of steady-state error. Conceptually, this thing can be reached in the system because the integral is able to cope with the steady-state error. The lateral movement in the roll rotation mode has a steady state type 0 that is 0 type with entry input system in the form of a step in which the Steady-state Error (SSE) of this type can be calculated by an Eq. 29 (Ogata, 2010):

$$SSE = \frac{1}{1+K} \quad (29)$$

Therefore, the solving problem in eliminating the interference and system errors in a stable state is integrating once of the system error that is then multiplied in the form of constant reinforcement. The component is added in a closed loop system of roll rotational motion of lateral system in the type of flying wing UAV airframe.

CONCLUSION

Controlling lateral roll rotational motion to linear approach that is solved by using LQR control method with the addition of integral, it generates significant characteristics in the system. The addition of integral makes the stability system can be achieved, it is not only able to stabilize in a short time and overshoot that occurs at a minimum but also the steady state error in the system can be eliminated.

RECOMMENDATIONS

Future research will design a controller with a non-linear approach that still remain in optimum control as backstepping method, so that, it can be collaborated with LQR method that can make the system to be more optimal in dealing with disturbances in different conditions.

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