Autonomous Flight of Unmanned Aerial Vehicle (UAV) by using Linear Quadratic Regulator (LQR)

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Abstract

The linear quadratic regulator based autopilot is premeditated for level flight of unmanned aerial vehicle (UAV) which has been achieved lucratively. Diverse modes are utilized for controlling the rates of UAV. The pitch rate is controlled by utilizing the phugoid mode, the roll rate is controlled by utilizing the roll mode and the spiral mode is utilized for yaw rate. The linear quadratic regulator proposes sturdiness at the rate of accomplishment to subjugation over the turbulences and instabilities in order to conquer the favored performance. The FlightGear simulator is utilized for the authentication of the performance of the autopilot. The level flight has been conquered successfully and has been legitimated by utilizing the FlightGear simulator. The simulated results as well as the FlightGear simulator results or the legitimated results are presented in this paper. Microsoft Visual Studio has been employed as a conduit between the MATLAB and FlightGear Simulator.

Keywords: Linear Quadratic Regulator, Autopilot, Aircraft, Level Flight, Flight Gear Simulator, Unmanned Aerial Vehicle

Nomenclature

\[ M = \text{pitch moment} \]
\[ N = \text{yaw moment} \]
\[ P = \text{roll rate} \]
\[ \phi = \text{roll angle} \]
\[ \theta = \text{pitch angle} \]
\[ u = \text{axial velocity perturbation} \]
\[ v = \text{lateral velocity perturbation} \]
\[ w = \text{normal velocity perturbation} \]
\[ L = \text{roll moment} \]
\[ Q = \text{pitch rate} \]
\[ R = \text{yaw rate} \]
\[ X = \text{axial force} \]
\[ Y = \text{lateral force} \]
\[ Z = \text{normal force} \]
\[ X_u = \text{change in force in x-direction with airspeed} \]
\[ M_u = \text{change in pitching moment with airspeed} \]
\[ Z_u = \text{change in force in z-direction with airspeed} \]
\[ X_w = \text{change in force in x-direction with normal velocity} \]
\[ M_w = \text{change in pitching moment with normal velocity} \]
\[ Z_w = \text{change in force in z-direction with normal velocity} \]
\[ X_q = \text{change in force in x-direction with pitch rate} \]
\[ M_q = \text{change in pitching moment with pitch rate} \]
\[ Z_q = \text{change in force in z-direction with pitch rate} \]
\[ x_{s_k} = \text{change in force in x-direction with pitch rate} \]
\[ Y_{\beta} = \text{change in force in y-direction with sideslip angle} \]
\[ L_{\beta} = \text{change in rolling moment with sideslip angle} \]
\[ N_{\beta} = \text{change in yawing moment with sideslip angle} \]
\[ Y_p = \text{change in force in y-direction with roll rate} \]
\[ L_p = \text{change in rolling moment with roll rate} \]
\[ N_p = \text{change in yawing moment with roll rate} \]
\[ Y_r = \text{change in force in y-direction with yaw rate} \]
\[ L_r = \text{change in rolling moment with yaw rate} \]
\[ N_r = \text{change in yawing moment with yaw rate} \]
\[ L_{\delta_A} = \text{change in rolling moment with aileron deflection} \]
\[ N_{\delta_A} = \text{change in yawing moment with aileron deflection} \]
\[ L_{\delta_R} = \text{change in rolling moment with rudder deflection} \]
\[ N_{\delta_R} = \text{change in yawing moment with rudder deflection} \]

1. Introduction

The intention of this paper is to premeditate the autopilot for the UAV by employing the linear quadratic regulator technique. Unmanned aerial vehicles have consequence and implication in both military and civilian territory. In the military systems, they can be employed for security, communication, target designation, weapon delivery and intelligence intentions. In the civilian systems, they can be employed for forest fire detection, weather forecast and agriculture intentions. The main consequence and implication of scheming autopilot is to achieve the flight without human intervention. The main purpose of this research work is to scheme such directing method which can work like human pilot and can be augmented with robust control system that can provide sturdiness against the atmospheric and external disturbances and instabilities. In past many of the systems of UAVs have been designed such as; Donovan, W.R [1] has designed the remote sensing system in the cryosphere for unmanned aerial vehicle. Underwood, S [2] has determined the performance and emission properties of the unmanned aerial vehicle by using the turbo diesel engine using JET-A fuel. Burns, R [3] has designed the avionics and telemetry software system for the unmanned aerial vehicle. Test and evaluation of the Piccolo II autopilot system on a One-Third scale Yaw-54 has been presented by Jager, R [4]. Time plotting framework for remote display of flight data has presented by Esposito, J [5]. Modeling and simulation for the yaw-54 using a 6-DOF model with flight test validation has been presented by Leong, H.J [6]. Robust integrated lateral guidance and control system has designed by Muhammad Zeeshan Babar [7].
2. Mathematical Modeling Description

2.1 Longitudinal State Space Model

According to the M.V Cook [8] the dynamic equations for longitudinal model of an aircraft can be written as

\[ X_u + X_w + X_{\dot{q}} + X_{\dot{w}} + X_{\delta_e} = -mg\theta + X_T u + X_T \delta_t = U\dot{u} \]  \hspace{0.5cm} (1)

\[ Z_u + Z_w + Z_{\dot{q}} + Z_{\dot{w}} + Z_{\delta_e} = Z_T u + Z_T \delta_t = U(\dot{\theta} - U q) \]  \hspace{0.5cm} (2)

\[ M_u + M_w + M_{\dot{q}} + M_{\dot{w}} + M_{\delta_e} = M_T u + M_T \delta_t = qIy \]  \hspace{0.5cm} (3)

For the level flight conditions \( \theta = q \); by re-writing the equations 1-3 in state space form we get

\[ A_i = \begin{bmatrix} X_u & X_w/U & 0 & -g/U \\ \frac{UZ_q}{(U-Z_{\dot{w}})} & \frac{Z_w}{(U-Z_{\dot{w}})} & (U+Z_q)/(U-Z_{\dot{w}}) & 0 \\ \frac{UM_u+M_{\delta_e}UZ}{(U-Z_{\dot{w}})} & \frac{M_w+M_{\delta_e}Z}{(U-Z_{\dot{w}})} & (U+Z_q)/(U-Z_{\dot{w}}) & 0 \\ 0 & 0 & 0 & 1 \end{bmatrix} \]  \hspace{0.5cm} (4)

\[ \Theta_i = \begin{bmatrix} 0 & 0 & 0 & 0 \\ \frac{X_{\delta_e}}{U} & \frac{Z_{\delta_e}}{(U-Z_{\dot{w}})} & \frac{M_{\delta_e}+M_{\delta_e}Z}{(U-Z_{\dot{w}})} & 0 \end{bmatrix} \]  \hspace{0.5cm} (5)

longitudinal dimensionless stability derivatives of Cessna-182 aircraft are given in the table given below [11]

**Table 1. Dimensionless Longitudinal Stability Derivatives for the Cessna-182**

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Value</th>
<th>Symbol</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>( C_L )</td>
<td>0.307</td>
<td>( C_D )</td>
<td>0.032</td>
</tr>
<tr>
<td>( C_{L\alpha} )</td>
<td>4.41</td>
<td>( C_{D\alpha} )</td>
<td>0.121</td>
</tr>
<tr>
<td>( C_{n\alpha} )</td>
<td>-0.613</td>
<td>( C_{Lq} )</td>
<td>1.7</td>
</tr>
<tr>
<td>( C_Lq )</td>
<td>3.9</td>
<td>( C_{nq} )</td>
<td>-12.4</td>
</tr>
<tr>
<td>( C_{L\delta E} )</td>
<td>0.43</td>
<td>( C_{n\delta E} )</td>
<td>-1.122</td>
</tr>
<tr>
<td>( C_{D\delta E} )</td>
<td>0.0</td>
<td>( C_{Dq} )</td>
<td>0.0</td>
</tr>
<tr>
<td>( C_{n\delta_o} )</td>
<td>0.0</td>
<td>( C_{D_o} )</td>
<td>0.0</td>
</tr>
<tr>
<td>( C_{n\delta_u} )</td>
<td>0.0</td>
<td>( C_{L_o} )</td>
<td>0.0</td>
</tr>
<tr>
<td>( C_{D_u} )</td>
<td>0.0</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
The dimensional stability derivatives are figured by utilizing the dimensionless stability derivatives listed in Table 1. The explanation of these computations can be found in reference [9].

Table 2. Dimensional Longitudinal Stability Derivatives for the Cessna-182

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Value</th>
<th>Unit</th>
<th>Symbol</th>
<th>Value</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\alpha$</td>
<td>-0.0011</td>
<td>$\frac{\mu}{\mu}$</td>
<td>$\delta$</td>
<td>-0.0280</td>
<td>$\frac{\mu}{\mu}$</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>1.1371</td>
<td>$\frac{\mu}{\mu}$</td>
<td>$\delta$</td>
<td>0.0</td>
<td>$\frac{\mu}{\mu}$</td>
</tr>
<tr>
<td>$\beta$</td>
<td>0.0</td>
<td>$\frac{\mu}{\mu}$</td>
<td>$\delta$</td>
<td>-0.7010</td>
<td>$\frac{\mu}{\mu}$</td>
</tr>
<tr>
<td>$\gamma$</td>
<td>-0.0104</td>
<td>$\frac{\mu}{\mu}$</td>
<td>$\delta$</td>
<td>0.0</td>
<td>$\frac{\mu}{\mu}$</td>
</tr>
<tr>
<td>$\delta$</td>
<td>-0.1272</td>
<td>$\frac{\mu}{\mu}$</td>
<td>$\delta$</td>
<td>-0.1577</td>
<td>$\frac{\mu}{\mu}$</td>
</tr>
<tr>
<td>$\epsilon$</td>
<td>-0.0012</td>
<td>$\frac{\mu}{\mu}$</td>
<td>$\delta$</td>
<td>-2.5662</td>
<td>$\frac{\mu}{\mu}$</td>
</tr>
<tr>
<td>$\zeta$</td>
<td>-0.0028</td>
<td>$\frac{\mu}{\mu}$</td>
<td>$\delta$</td>
<td>-2.5662</td>
<td>$\frac{\mu}{\mu}$</td>
</tr>
</tbody>
</table>

By substituting the values of dimensional derivatives in equations (4) and (5) we get

$$A = \begin{bmatrix} -0.0011 & 0.0052 & 0 & -0.1463 \\ -0.0104 & -0.0006 & 1.0000 & 0 \\ 0 & -0.7010 & -0.1577 & 0 \\ 0 & 0 & 1.0000 & 0 \end{bmatrix}$$ (6)

$$B_l = \begin{bmatrix} 0 \\ -0.0001 \\ -2.5662 \\ 0 \end{bmatrix}$$ (7)

The above equations (6) and (7) demonstrate the longitudinal state space of the Cessna-182 aircraft. The phugoid mode is executed by utilizing the above state space model in order to control the pitch rate of the aircraft.

### 2.2 Lateral State Space Model

According to the M.V Cook [8] the dynamic equations for lateral model of an aircraft can be written as

$$Y \dot{\phi} + Y_p \phi + Y_r \phi + X \delta_a \delta_a + Y_r \delta_r + m g \phi = U (\dot{\phi} + U_0 \phi)$$ (8)

$$L \dot{\psi} + L_p \phi + L_r \phi + L \delta_a \delta_a + L \delta_r \delta_r = \dot{p} I_z - \dot{\psi}$$ (9)

$$N \dot{\phi} + N_p \phi + N_r \phi + N \delta_a \delta_a + N \delta_r \delta_r = \dot{r} I_z - \dot{\psi} \delta$$ (10)

For level flight conditions $\dot{\phi} = \dot{\psi} = r$; by re-writing the equations 8-10 in state space form we get
The lateral dimensionless stability derivatives of Cessna-182 aircraft are given in the table given below [11]

**Table 3. Dimensionless Lateral Stability Derivatives for the Cessna-182**

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Value</th>
<th>Symbol</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_{\beta}$</td>
<td>-0.393</td>
<td>$C_{\beta}$</td>
<td>-0.0923</td>
</tr>
<tr>
<td>$C_{\phi}$</td>
<td>0.0587</td>
<td>$C_{\phi}$</td>
<td>-0.075</td>
</tr>
<tr>
<td>$C_{\rho}$</td>
<td>-0.0278</td>
<td>$C_{\rho}$</td>
<td>-0.484</td>
</tr>
<tr>
<td>$C_{n}$</td>
<td>0.214</td>
<td>$C_{n}$</td>
<td>0.0798</td>
</tr>
<tr>
<td>$C_{\alpha}$</td>
<td>-0.0937</td>
<td>$C_{\alpha}$</td>
<td>0.229</td>
</tr>
<tr>
<td>$C_{\alpha}$</td>
<td>-0.0216</td>
<td>$C_{\alpha}$</td>
<td>0.187</td>
</tr>
<tr>
<td>$C_{\beta}$</td>
<td>0.0147</td>
<td>$C_{\beta}$</td>
<td>-0.0645</td>
</tr>
<tr>
<td>$C_{\alpha}$</td>
<td>0.0</td>
<td>$C_{\alpha}$</td>
<td></td>
</tr>
</tbody>
</table>

The lateral dimensional stability derivatives are computed by utilizing the dimensionless stability derivatives listed in Table 1-3. The description of these calculations can be found in reference [9].

**Table 4. Dimensional Stability Derivatives for the Cessna-182**

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Value</th>
<th>Unit</th>
<th>Symbol</th>
<th>Value</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>$Y_{\beta}$</td>
<td>-0.0256</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
<td>$L_{\alpha}$</td>
<td>2.7622</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
</tr>
<tr>
<td>$Y_{\rho}$</td>
<td>-0.0079e-4</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
<td>$L_{\delta}$</td>
<td>0.1733</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
</tr>
<tr>
<td>$Y_{\phi}$</td>
<td>0.0011</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
<td>$N_{\beta}$</td>
<td>0.3418</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
</tr>
<tr>
<td>$Y_{\delta}$</td>
<td>0.0</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
<td>$N_{\phi}$</td>
<td>-0.0133</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
</tr>
<tr>
<td>$Y_{\rho}$</td>
<td>0.0122</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
<td>$N_{\rho}$</td>
<td>-0.0448</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
</tr>
<tr>
<td>$L_{\beta}$</td>
<td>-0.0308</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
<td>$N_{\delta}$</td>
<td>-0.1258</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
</tr>
<tr>
<td>$L_{\rho}$</td>
<td>-0.4793</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
<td>$N_{\phi}$</td>
<td>-0.3756</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
</tr>
<tr>
<td>$L_{\phi}$</td>
<td>0.0790</td>
<td>$\frac{\text{deg}}{\text{deg}}$</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
By substituting the values of dimensional derivatives in equations (11) and (12) we get

\[
A_t = \begin{bmatrix}
-0.0001 & 0.0000 & 0.1463 & -1.0000 \\
-0.0308 & -0.4793 & 0 & 0.0790 \\
0 & 1.0000 & 0 & 0 \\
0.3418 & -0.0133 & 0 & -0.0448
\end{bmatrix}
\]

(13)

\[
B_t = \begin{bmatrix}
0 & 0.0122 \\
2.7622 & 0.1773 \\
0 & 0 \\
-0.1258 & -0.3756
\end{bmatrix}
\]

(14)

The above equations (13) and (14) show the lateral state space of the Cessna-182 aircraft. The roll and spiral mode have been implemented by using the above state space model in order to control the roll and yaw rates of the aircraft.

3. Linear Quadratic Regulator (LQR) Synthesis

Suppose the controlled system, the mathematical expression is given as

\[
\dot{x} = Ax + Bu
\]

(15)

\[
y = Cx
\]

(16)

Where,

\[x \in R^n\] is the system state

\[u \in R^m\] is the control input

\[y \in R^p\] is the system output to controlled

It is supposed that \( \text{rank}(C) = p \) and \( \text{rank}(B) = k \geq p \). Full-state feedback case is considered and it is supposed that the x is measured.

The objective of the control is to bring the x to zero while minimizing the quadratic performance index; the mathematical expression is given as,

\[
J = \int_0^\infty (x(t)^TQx(t) + u(t)^TRu(t))dt
\]

(17)

Q is a semidefinite matrix and R is a positive definite matrix. Q and R are chosen such that the preferred results are acquired.

The control law is given as,

\[
u = -Lx
\]

(18)

\[L = R^{-1}B^TS
\]

(19)

Where, S is the unique positive semidefinite and symmetric solution to

\[
A^TS + SA + Q - SBR^{-1}B^TS = 0
\]

(20)

The above expression is called the Algebraic Reccati Equation (ARE).

Both the longitudinal and lateral controllers have been implemented. The pitch rate has been controlled by the phugoid mode. The roll and yaw rates are controlled by the roll and spiral modes. Controllers have been implemented by utilizing the weighted matrices procedure.
4. Simulation Results

In this section, the simulation results of diverse modes are illustrated;

4.1 Phugoid Mode Simulation Results

The simulation results for the phugoid mode are exemplified in this section. The pure phugoid mode response is given in the following figures, which is acquired by utilizing the weight selection method; the desired weights have been selected and the responses of the different states \( u \), \( w \), \( q \) and \( \theta \) have been acquired consequently. These responses are according to the initialization of the state space vector. The simulink block diagram of phugoid mode is given in the figure below:

![Simulink Model of Phugoid Mode LQR Simulation](image1)

**Figure1. Simulink Model of Phugoid Mode LQR Simulation**

![Internal Layout of Aircraft Longitudinal Model](image2)

**Figure2. Internal Layout of Aircraft Longitudinal Model**

The weighted matrices for the phugoid mode are given below:

\[
Q = \begin{bmatrix}
1 & 0 & 0 & 0 \\
0 & 0.1 & 0 & 0 \\
0 & 0 & 1 & 0 \\
0 & 0 & 0 & 1
\end{bmatrix}
\]

(21)

\[
R = \begin{bmatrix}
1 & 0 \\
0 & 1
\end{bmatrix}
\]

(22)

**Table 5. Parameters Attained from Phugoid Mode LQR Simulation**

<table>
<thead>
<tr>
<th>State</th>
<th>Settling Time (sec)</th>
<th>Max. Peak Value (Deg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \dot{u} )</td>
<td>5.0</td>
<td>-0.023</td>
</tr>
<tr>
<td>( \dot{w} )</td>
<td>1.0</td>
<td>-0.13</td>
</tr>
<tr>
<td>( \dot{q} )</td>
<td>1.0</td>
<td>-0.52</td>
</tr>
<tr>
<td>( \dot{\theta} )</td>
<td>1.0</td>
<td>0.0</td>
</tr>
</tbody>
</table>
Figure 3. Phugoid Mode LQR Simulation Response

Figure 4. Phugoid Mode LQR Simulation Responses

Figure 5. Phugoid Mode LQR Simulation Responses
4.2 Roll Mode Simulation Results

The simulation results for the roll mode have been discussed in this section. The pure roll mode response is given in the following figure, which is acquired by utilizing the weight selection method; the desired weights have been selected and the responses of the different states $v$, $p$, $r$ and $\phi$ have been obtained consequently. These responses are according to the initialization of the state space vector. The simulink block diagram of roll mode is given in figure

![Simulink Model of Roll Mode LQR Simulation](image1)

Figure 7. Simulink Model of Roll Mode LQR Simulation

![Internal Layout of Aircraft Lateral Model](image2)

Figure 8. Internal Layout of Aircraft Lateral Model
The weighted matrices for the roll mode are given below

\[
Q = \begin{bmatrix}
1000 & 0 & 0 & 0 \\
0 & 10 & 0 & 0 \\
0 & 0 & 10 & 0 \\
0 & 0 & 0 & 1000
\end{bmatrix}
\]  
\hspace{1cm} (23)

\[
R = \begin{bmatrix}
10 & 0 \\
0 & 10
\end{bmatrix}
\]  
\hspace{1cm} (24)

Table 6. Parameters Attained from Roll Mode LQR Simulation

<table>
<thead>
<tr>
<th>State</th>
<th>Settling Time (sec)</th>
<th>Max. Peak Value (Deg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\dot{\psi}$</td>
<td>0.55</td>
<td>0.0053</td>
</tr>
<tr>
<td>$\dot{\rho}$</td>
<td>0.2</td>
<td>0.0</td>
</tr>
<tr>
<td>$\dot{r}$</td>
<td>0.5</td>
<td>0.001</td>
</tr>
<tr>
<td>$\dot{\phi}$</td>
<td>0.2</td>
<td>0.0</td>
</tr>
</tbody>
</table>

Figure 9. $\dot{\psi}$ Roll Mode LQR Simulation Responses

Figure 10. $\dot{\rho}$ Roll Mode LQR Simulation Responses
4.3 Spiral Mode Simulation Results

The simulation results for the spiral mode have been discussed in this section. The pure roll mode response is given in the following figure, which is acquired by utilizing the weight selection method; the desired weights have been selected and the responses of the different states \(v\), \(p\), \(r\) and \(\phi\) have been obtained consequently. These responses are according to the initialization of the state space vector. The Simulink block diagram of spiral mode is given in figure
Figure 13. Simulink Model of Spiral Mode LQR Simulation

Figure 14. Internal Layout of Aircraft Lateral Model

The weighted matrices for the spiral mode are given below

\[
Q = \begin{bmatrix}
1 & 0 & 0 & 0 \\
0 & 1 & 0 & 0 \\
0 & 0 & 1 & 0 \\
0 & 0 & 0 & 1 \\
\end{bmatrix}
\]  \hspace{1cm} (25)

\[
R = \begin{bmatrix}
10 & 0 \\
0 & 10 \\
\end{bmatrix}
\]  \hspace{1cm} (26)

Table 7. Parameters Attained from Spiral Mode LQR Simulation

<table>
<thead>
<tr>
<th>State</th>
<th>Settling Time (sec)</th>
<th>Max. Peak Value (Deg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \dot{v} )</td>
<td>3.0</td>
<td>-0.016</td>
</tr>
<tr>
<td>( \dot{\rho} )</td>
<td>3.0</td>
<td>-0.27</td>
</tr>
<tr>
<td>( \dot{r} )</td>
<td>2.9</td>
<td>0.018</td>
</tr>
<tr>
<td>( \dot{\phi} )</td>
<td>2.1</td>
<td>0.0</td>
</tr>
</tbody>
</table>
Figure 15. \( \dot{v} \) Spiral Mode LQR Simulation Responses

Figure 16. \( \dot{\theta} \) Spiral Mode LQR Simulation Responses

Figure 17. \( \dot{\phi} \) Spiral Mode LQR Simulation Responses
5. Validation by Using FlightGear Simulator

In this section, the validation results of both longitudinal and lateral modes are presented by using the FlightGear simulator; the longitudinal and lateral controllers have been interfaced with the FlightGear simulator for validation and the validation results have been acquired accordingly. Both the MATLAB and FlightGear Simulator have been interfaced through the visual studio. In the case of longitudinal model, the phugoid mode has been used for controlling the pitch rate, in the case of lateral model; the roll mode has been used for controlling the roll rate as well as the spiral mode has used for controlling the yaw rate. The complete Simulink block diagram of autopilot based on interfacing of MATLAB with FlightGear Simulator and Microsoft Visual Studio is given below:
5.1 Phugoid Mode Validation Results

In this section, the validation results of different states of phugoid mode have been presented, which have been acquired from the FlightGear simulator. The pure phugoid mode response is given in the following figures:

**Figure 20.** Phugoid Mode LQR Validation Responses

**Figure 21.** Phugoid Mode LQR Validation Responses

**Figure 22.** Phugoid Mode LQR Validation Responses
Initially the aircraft was not in the controlled form or not in level form or was in disturbed form that’s why in some states the overshoot is large in beginning after interfacing the controller the overshoot has been reduced to a reasonable range, as these results are the real results of the aircraft, due to this reason these results are damped.

5.2 Roll Mode Validation Results

In this section, the Validation results of different states of roll mode have been acquired from the FlightGear simulator. The pure roll mode response is given in the following figures:

Figure 24. $\ddot{\psi}$ Roll Mode LQR Validation Response

Figure 25. $\dot{\Phi}$ Roll Mode LQR Validation Response
By observing the results of the different states of roll mode, it can be seen that initially there is abrupt change in the overshoot; the reason is that initially the aircraft is not in the controlled form.

5.3 Spiral Mode Validation Results

In this section, the Validation results of different states of spiral mode have been acquired from the FlightGear simulator. The pure spiral mode response is given in the following figures:
6. Conclusion

The designed autopilot based on the linear quadratic regulator (LQR) technique shows good performance and results through which the level flight of the aircraft has been acquired lucratively. The simulated as well as the experimental results show sturdiness against the interruption and disturbances. The simulated and validated results of the
The autopilot has been interfaced with the FlightGear simulator and the victorious level flight has been acquired and the results have been validated accordingly, the validated results also show the robustness against the disturbances and interruption. So by interfacing the autopilot with FlightGear simulator, it has been proved that the designed autopilot works well.

References

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