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Design And Implementation of Robust Control Methodologies in the Control of Lateral Axis of Aircraft

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ABSTRACT

The automatic flight control system of an aircraft is very complicated in nature with lot of logics and operational constraints. The study of the detailed automatic flight control and thrust control involves complicated logics for the control functions with closed loop control systems which can be realized in the software. The work is focussed on the redesign of automatic flight control system using robust control methods in MATLAB/Simulink environment. This paper aims at a complete study, analysis, design and performance monitoring of new generation autopilot system using Linear Quadratic Regulator and Sliding Mode Control methods. Comparison of the results of simulation of the above gives an optimal performance of the autopilot system.

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INTRODUCTION

An aircraft can be capable of many very intensive tasks: helping the pilot focus on the overall status of the aircraft and flight. Autopilots can automate tasks such as maintaining the altitude, climbing or descending to an assigned altitude, turning to and maintaining an assigned heading, intercepting a course, guiding aircraft between the way points that make up a route programmed into an FMS and flying a precision or non-precision approach. The fast growth of aircraft designs from less capable airplane to the present day high performance military, commercial and general aviation aircraft required the development of many technologies. The automatic control system plays an important role in monitoring and controlling of many of the aircraft's subsystems for which an autopilot is designed that controls the principal axes of the aircraft leading to the safe landing of the aircraft during adverse weather conditions (Nelson, R.C., 1998). Generally the aircraft is free to rotate around the three axes which are perpendicular to each other. In this paper, optimal and robust controllers, LQR and Sliding mode controllers are developed for controlling the yaw axis of the aircraft. The performances of these controllers are investigated and compared by simulating in MATLAB environment.

2. Modelingof Yaw control system:

An aircraft is free to rotate around the three axes which are perpendicular to each other and intersect at the plane's centre of gravity. There are two dynamical equations of motion present for an aircraft, where lateral dynamic equations of motion represents the dynamics of aircraft with respect to the lateral axis and longitudinal dynamic equations of motion represents the dynamics of aircraft with respect to the longitudinal axis (Donald Mclean, 1990; www.nasa.gov(01.03.2011).Figure.1 shows the forces , moments and velocity components in the body –fixed co-ordinate where X_B, Y_B, Z_B denoted the aerodynamic force components.φandδr represents the orientation of aircraft in the earth axis system and rudder deflection angle respectively. L,M,N denotes the aerodynamic moment components. p,q,r represents the angular rates of roll, pitch and yaw axis and u,v,w represents the velocity components of roll, pitch and yaw axis.Figure.2 shows the angular orientation and velocities of gravity vector. During the modeling of yaw control systems, few assumptions are considered.

1. The aircraft is steady state cruise at constant altitude and velocity, thus the thrust and drag cancel out each other.
2. The changes in yaw angle do not change the speed of an aircraft under any circumstances.

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Yaw control problem is a lateral problem and the work is developed to control the yaw angle of an aircraft in order to stabilize the system when an aircraft performs yawing motion.

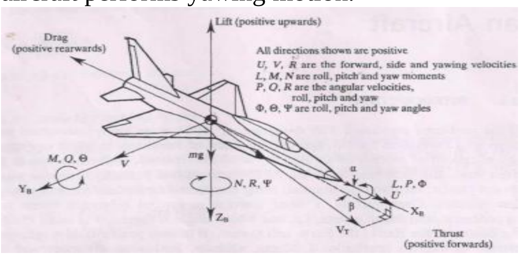

Fig. 1: Aerodynamic forces, moments and velocity components.

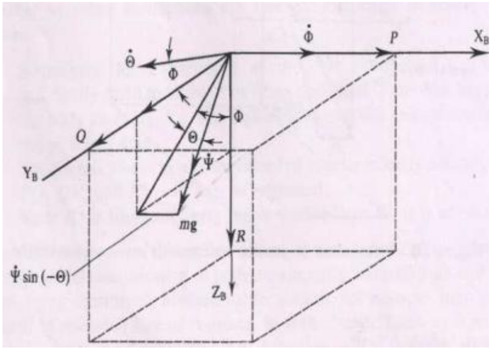

Fig. 2: Angular orientation and velocities of gravity vector.

With respect to figure.1 &2, the force and moment equations are given in equation (1) to (3).

$Y + mg\cos\theta\sin\varphi = m[\dot{v} + ru - pw]$ (1)

$L = I_{xx}\dot{p} - I_{xz}\dot{r} + qr(I_{zz} - I_{yy})$ (2)

$N = -I_{xz}\dot{p} - I_{zz}\dot{r} + pq(I_{yy} - I_{xx}) + I_{xz}qr$(3)

The above equations are nonlinear and simplified by considering the aircraft to comprise two components: a mean motion that represents the equilibrium or trim conditions and a dynamic motion which accounts for the perturbations about the mean motion (Seckel, E. and J.J. Moris, 1971). Thus every motion variable is considered to have two components.

$U \triangleq U_o + \Delta u,\ \triangleq Q_o + \Delta q\ ,\ R \triangleq R_o + \Delta r\ ,\ M \triangleq M_o + \Delta m\ ,\ Y \triangleq Y_o + \Delta y\ ,\ P \triangleq P_o + \Delta p\ ,\ L \triangleq L_o + \Delta l\ ,\ V \triangleq V_o + \Delta v\ ,\ \delta \triangleq \delta_o + \Delta\delta$ (4)

The reference flight condition is assumed to be symmetric and the propulsive forces are assumed to beconstant.

$v_o=q_o=u_o=r_o=\varphi_o=\Psi_o=0$ (5)

The complete linearized equations of motion are obtained as below where sideslip angle is used.

$\left(\frac{d}{dt} - Y_v\right)\Delta v - Y_p\Delta p + (u_0 - Y_r)\Delta r - (g\cos\theta_0)\Delta\varphi = Y\delta r.\Delta\delta r$ (6)

$-L_v\Delta v + \left(\frac{d}{dt} - L_p\right)\Delta p - L_r\Delta r = L\delta r.\Delta\delta r + L\delta a.\Delta\delta$ (7)

$-N_v\Delta v + \left(\frac{d}{dt} - N_r\right)\Delta r - N_p\Delta p = N\delta r.\Delta\delta r + N\delta a.\Delta\delta$(8) (8)

Substituting, $\Delta v=\Delta\beta$, $Y_v=Y_\beta$, $L_v=L_\beta$, $N_v=N_\beta$and $\Delta_\beta=\Delta_v/u_o$

$\left(\frac{Y_\beta}{u_o}\right)\Delta\beta + \frac{Y_p}{u_o}\Delta p - \left(1 - \frac{Y_r}{u_0}\right)\Delta r + \frac{(g\cos\theta_0)}{u_0}\Delta\varphi = \frac{Y\delta r}{u_0}.\Delta\delta r$ (9)

$-L_\beta\Delta\beta + (L_p)\Delta p + L_r\Delta r = L\delta r.\Delta\delta r + L\delta a.\Delta\delta a$ (10)

$-N_\beta\Delta\beta + (N_p)\Delta p + N_r\Delta r = N\delta r.\Delta\delta r + N\delta a.\Delta\delta a$ (11)

Using the equations (9),(10) ,(11) the state space model for the roll and yaw control problem can be formulated.In this work, the modelling is based on the data from the general aviation aircraft NAVION.The yaw control problem has the input as rudder deflection angle and output as change in the yaw angle of aircraft. The lateral directional stability derivative parameters for Navion (Ryan, C., Struett, 2012) are taken from the table.1, which are the standard datas of the general aviation aircraft Navion.

$$\begin{bmatrix}\dot{\Delta}_\beta \\ \dot{\Delta}_p \\ \dot{\Delta}_r \\ \dot{\Delta}_\varphi\end{bmatrix} = \begin{bmatrix}\frac{Y_\beta}{u_0} & \frac{Y_p}{u_0} & -\left[1 - \frac{Y_r}{u_0}\right] & \frac{g\cos\theta_0}{u_0} \\ L_\beta & L_p & L_r & 0 \\ N_\beta & N_p & N_r & 0 \\ 0 & 1 & 0 & 0\end{bmatrix}\begin{bmatrix}\Delta\beta \\ \Delta p \\ \Delta r \\ \Delta\varphi\end{bmatrix} + \begin{bmatrix}0 & \frac{Y\delta r}{u_o} \\ L\delta a L\delta r \\ N\delta a N\delta r \\ 0 & 0\end{bmatrix}\begin{bmatrix}\Delta\delta a \\ \Delta\delta r\end{bmatrix} \quad (12)$$

Table 1: Lateral Stability Derivatives.

Lateral Derivatives	Components		
	X-Force Derivatives	Rolling moment Derivatives	Yawing moment Derivatives
Pitching velocities	$Y_v = 0.254$	$L_v = -0.091$	$N_v = 0.025$
Sideslip angle	$Y_\beta = -44.6$	$L_\beta = -15.84$	$N_\beta = 4.3$
Rolling rate	$Y_p= 0$	$L_p= -8.349$	$N_p= -0.342$
Yawing rate	$Y_r = 0$	$L_r = 2.086$	$N_r = -0.76$
Rudder Deflection	$Y_{\delta r} =12.43$	$L_{\delta r} = -2.67$	$N_{\delta r} = -4.79$
Aileron Deflection	$Y_{\delta a} = 0$	$L_{\delta a}= -28.68$	$N_{\delta a} = -0.216$

Australian Journal of Basic and Applied Sciences, 9(16) Special 2015, Pages: 77-81

The data required for the state space representation are substituted from the table.1, to get the state space representation for the yaw control system as in equation (13).

$$\begin{bmatrix}\dot{\Delta}_\beta\\ \dot{\Delta}_p\\ \dot{\Delta}_r\\ \dot{\Delta}_\varphi\end{bmatrix}=\begin{bmatrix}-0.254 & 0 & -1 & 0.184\\ -15.84 & -8.349 & 2.19 & 0\\ 4.3 & -0.342 & -0.76 & 0\\ 0 & 1 & 0 & 0\end{bmatrix}\begin{bmatrix}\Delta\beta\\ \Delta p\\ \Delta r\\ \Delta\varphi\end{bmatrix}+\begin{bmatrix}0.07\\ -2.67\\ -4.79\\ 0\end{bmatrix}[\Delta\delta r] \quad (13)$$

3. Design of Control Methodologies:

The work is emphasized to develop a yaw control scheme for the yaw angle of aircraft. For this two control methodologies (i.e.) Linear Quadratic Regulator (LQR) and Sliding Mode Control methodologies were proposed. The performance of both the control strategies were investigated and compared.

3.1 Design of Linear Quadratic Regulator (LQR):

This method is based on the manipulation of the equations of motion in state space and the system can be stabilized using full state feedback system.
Consider the state and output equations describing thelongitudinal equations of motion.

$$\dot{X}(t) = Ax(t) + Bu(t)$$
$$y(t) = Cx(t) + Du(t) \quad (14)$$

In the LQR design[8], the lqr function in Matlab can be used to determine the value of the vector'K' which is used to find the feedback control law.
$u(t) = -kx(t) + \Delta\delta e.N$(15)
This control law has to minimize the performance index,$J = \int_0^\infty (X^T QX + u^T Ru)dt$where, Q- state cost matrix, R- performance index matrix .Here R=1 and $Q = C^T C$.
Fig.3 shows the full state feedback controller with reference input. For the present study, the value of

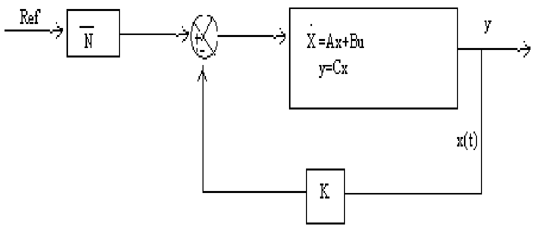

Fig 3. Full state feedback controller

'K' is to be determined. The controller is tuned by varying the elements in Q matrix which is done in m-file code .With the below values ,
R=1& Q= [0 0 0 0;0 0 0 0;0 0 0 0;0 0 0 1500], the following value of K= [5.29 -3.106 -0.999 -38.682] with$\bar{N} = -38.73$ is obtained.

3.2 Design of Sliding Mode Control(SMC):

Sliding mode control is a non-linear method(Edwards, C.,) which alters the dynamics of the non-linear system by the application of a high frequency switching control. This is a variable structure control. The aim of this is to drive the system state from an initial condition x(0) to the state space origin as t→∞. The j[th] component U_j(j=1,2,….m) of the state feedback control vector U(x) has a discontinuity on the j[th]switching surface which is a hyperplane 'M_j' passing through the state origin.
Defining the hyper plane by
$M_j = \{x: C_j x = 0\}, j = 1,2, \dots\dots m$ (16)
Consider the system
$\dot{x}(t) = Ax(t) + Bu(t)$ (17)
From equation.(14), the sliding mode satisfies the condition
$S = Cx(t) = 0 \ , t \geq t_s$ (18)
C= mxnmatrix ,t_s is the time when sliding subspace is reached.Differentiating eqn.(16) with respect to time and substituting for eqn.(15), we get
$C\dot{x}(t) = CAx(t) + CBu(t) = 0, t \geq t_s$ (19)
$CBu(t) = -CAx(t)$ (20)
C is the hyperplane matrix so that $|CB| \neq 0$

The switching surface design is predicted based on the knowledge of the system behaviour in sliding mode and this behaviour depends on the parameters of the switching surface. So achieving the switching surface design requires analytically specifying the motion of the state trajectory in a sliding mode. Therefore the method of equivalent control is essential. Equivalent control constitutes a control input when exciting the system, produces the motion of the system on the sliding surface whenever the initial state is on the surface. So Eqn.(18) can be rearranged to get equivalent control as below

$$U_{eqt} = -(CB)^{-1}CAx(t)$$
$U_{eqt} = -Kx(t)$ (21)
$\dot{x}(t) = [A - Bk]x(t)$ (22)

Eqn.(20) gives the system equation for the closed loop system dynamics during sliding. The choice of 'C' determines the matrix 'K'. The purpose of the control "Ueqt" is to drive the state in to the sliding subspace and maintain thereafter in it. The (n-m) dimensional switching surface imposes 'm' constraints on the plant dynamics in a sliding mode. So the plant dynamics should be converted to a regular form. The nominal linear system can then be expressed as

$$\dot{y_1}(t) = A_{11}y_1 + A_{12}y_2$$
$$\dot{y_2}(t) = A_{21}y_1 + A_{22}y_2 + B_2 u(t) \quad (23)$$

This plays an important role in bringing the solution of the reachability problem and the sliding condition is equivalent to

$$C_1 y_1(t) + C_2 y_2(t) = 0$$
$$y_2(t) = -\frac{C_1}{C_2} y_1(t)$$

Australian Journal of Basic and Applied Sciences, 9(16) Special 2015, Pages: 77-81

$y_2(t) = -Fy_1(t)$ (24)

$\dot{y}_1(t) = [A_{11} - FA_{12}]\, y_1(t)$ (25)

Eqn.(23) is the reduced order equivalent system.

Design of Slidinghyperplane(Edwards, C.; John, Y., 1993):

To design the hyperplane , the quadratic performance $J = \frac{1}{2}\int_{t_s}^{\infty} x^T Q x dt$ can be minimised, where Q> 0 is positive definite symmetric matrix and t_s is the time of attaining the sliding mode. The matrix 'F' is determined once the matrix Riccati equation is solved.

a) Design of feedback control law:

Once the sliding surface has been selected, then the reachability problem can be solved. This involves the selection of a state feedback control function which will drive the state 'x' in to the sliding subspace and thereafter maintain in it. This control law has two components:

1. A linear control law (U_l) to stabilize the nominal linear system.
2. Discontinuous component (U_n)

And the control law,

$U(t) = U_l(t) + U_n(t)$ (26)

Where, $U_l(t) = U_{eq}(t) = -(CB)^{-1}CAx(t) = -(CB)^{-1}[CA - \varphi C]x(t)$ (27)

The non-linear component is defined to be

$U_n(t) = \rho\frac{S}{\|S\|} = \rho\frac{Cx(t)}{\|Cx(t)\|}, \rho > 0$ (28)

'C' is the symmetric positive definite matrix satisfying the Lyapunov equation,

$C\varphi + \varphi^T C = -I$,φ is any stable design matrix.

The control law is

$U(t) = -(CB)^{-1}[CA - \varphi C]x(t) + \rho\frac{Cx(t)}{\|Cx(t)\|}$(29)

4. Simulation and Results:

A Control system for the yaw axis is simulated using LQR and SMC and the results of simulation are analysed and presented for comparison. To investigate the performance of the control strategy, the time domain specifications are analysed. The values of K for the yaw control problems in LQR is obtained asK= [5.29 -3.106 -0.999 -38.682] and $\bar{N} = -38.73$

The control law in SMC has the value,U(t)=[-10.93 -5.41 2.1815]

The figure.4 and figure.5 shows the response of the yaw control system using LQR and SMC respectively.

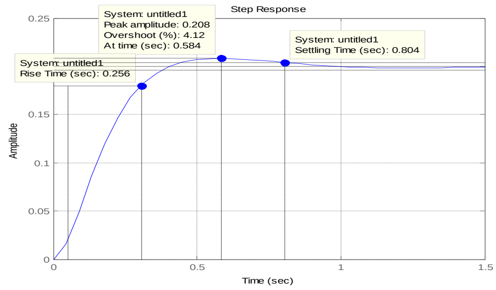

Fig. 4: Response of system for LQR.

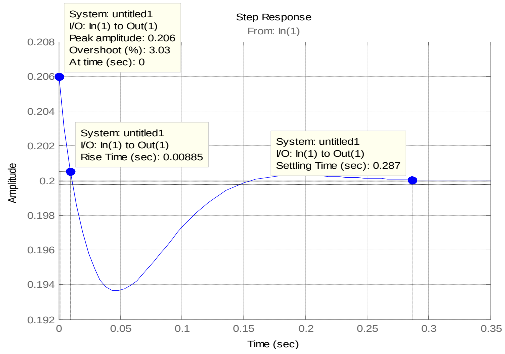

Fig. 5: Response of system for SMC.

Performance Characteristics	LQR	SMC
Rise time (tr)	0.256 sec	0.00885 sec
Settling time (ts)	0.804 sec	0.287 sec
% Overshoot (%Mp)	4.12%	3.03%

Table2: Comparison of performance characteristics.

Table.2 shows the performance characteristics of both LQR and SMC controllers .By comparing the performance characteristics in table.2, it is clear that the sliding mode controller has the fastest response and gave an optimal performance than the LQR controller in controlling the yaw angle of the aircraft.

5. Conclusion:

The work emphasizes the design of an autopilot for controlling the yaw axis of aircraft which was done using LQR and SMC on the MATLAB environment. The controllers were designed and the responses were analysed and verified in time domain. From the result of simulation it is observed that the sliding mode controller gives an optimal performance in controlling the yaw axis of the aircraft efficiently than the LQR by handling the effect of disturbances in the system.

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