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Theoretical and Simulation Based Approach for Controlling Aircraft Longitudinal and Lateral Yaw Damper Movement Using PID Controller

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Abstract- In this manuscript we consider two different parameters of DC-8 aircraft and extend the work as original research for controlling the longitudinal and lateral yaw damper movement. Here we consider both the theoretical and numerical aspect of aircraft dynamics by modeling the control surfaces i.e., elevators and lateral yaw damper. For controlling these control surfaces we design an intelligent PID controller and examine the overall performance of the system primarily based on time response specification. The simulation results generated are plotted and evaluated between controller response v/s deflection of control surfaces i.e., horizontal stabilizer and vertical stabilizer/rudder**.** The controller is designed based on dynamical model of aircraft for which equations are derived governing input to elevator, and rudder, which are used to control aircraft longitudinal and directional stability of aircraft. A quantitative analysis of PID controller has been carried out in MATLAB 2014a Simulink© environment for all the two movements of aircraft based on time response specification.

Keyword- Pitch, Yaw, Elevators, Rudder, PID

I. INTRODUCTION

The two basic control movements of aircraft which are taken into consideration are longitudinal and lateral yaw damper movement of DC-8 aircraft. In the present paper we have considered it as valuable approach to work on these parameters i.e., longitudinal, and Yaw movement control and implemented it with PID controller. These are important parameters and need precise evaluation at different stages in flight during which aircraft changes its transition from one state to another and performs complex maneuver. The pitch movement of aircraft is categorized under longitudinal stability whereas roll and yaw are categorized under lateral stability.

A set of control surfaces known as elevators, ailerons and rudder known as primary control surfaces are used for controlling aircraft longitudinal, roll and Yaw movement respectively. Here in the present work we are considering two control movements of aircraft i.e., longitudinal and yaw movement. ***Elevators*** are movable control surfaces located at the back of fixed wing aircraft which causes the aircraft to climb and descend and also to obtain sufficient lift from the wings to keep the aircraft in level flight at various speeds. ***Rudder*** also known as vertical stabilizer is used to control yaw movement of aircraft. The rudder generally provides for the control of yaw (nose right or nose left). Some aircraft's are provided with dual rudders, each of which is split into two separately actuated sections. The rudder control system also incorporates, most often, a yaw damper which receives inputs from a yaw rate gyro and provides additional signals to the rudder power control unit so as to move the aircraft in the direction opposing the yaw motion and in proportion to the yaw rate [1]. A device known as actuator is used to implement the longitudinal and yaw movement of aircraft. The purpose of actuator is to avoid stress to pilot's command to move the control surfaces, so that they can move with ease.

A lot of work has already been initiated in this particular field. Some of the recent work carried out is discussed here: The first kind of control technique applied for longitudinal control of aircraft is based on L1 adaptive control. The adaptive law and control law of control augmentation systems are designed so that tracking error rapidly converges and keeping robust the stochastic sliding mode control to stabilize the decoupled longitudinal dynamics by using linear matrix inequalities (LMIs) [2], the second technique is based on stochastic sliding mode control method with Linear matrix inequalities(LMIs) [3], the third technique is base on combination of Fuzzy-PID controller with nonlinearities taken into consideration [4] and the fourth technique is based on Fuzzy logic control of longitudinal motion of an aircraft based on Takagi–Sugeno modeling [5], whereas for aircraft lateral yaw damper the model is implemented for the first time with PID controller and no literature is available for this particular movement of aircraft.

The following figure1 shows basic control surfaces of aircraft.

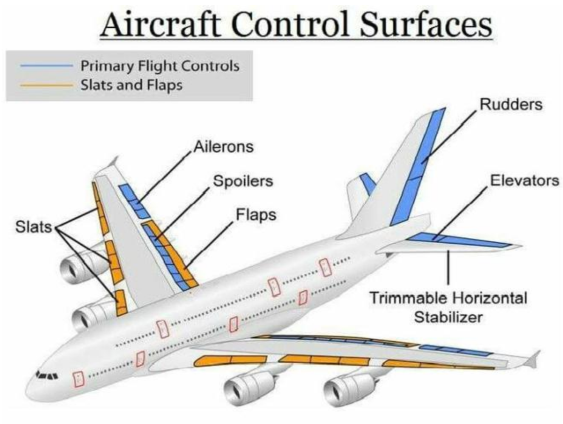

Figure 1 Basic control surfaces of Fixed wing Aircraft

The following paper is organized in 5 sub-sections. In section 2 mathematical modeling of longitudinal and lateral yaw damper of DC-8 aircraft is repotted, in section 3 methodology giving designing of PID controller is given which is implemented with the following two parameters of aircraft, section 4 gives detail comparison of the results between the original data without controller and present work with PID controller based on time response specification and also values of omega, zeta and tau are obtained for designing of actuator. The last section of the paper, 5 gives a detailed conclusion of the paper presented here. Especially for lateral yaw damper movement the work repotted is for the first time as there is no literature available for this.

II. MATHEMATICAL MODELING OF AIRCRAFT FOR LONGITUDINA AND YAW CONTROL MOVEMENT

In this section of the paper, a brief description for modeling of aircraft longitudinal and lateral yaw damper of DC-8 aircraft control movement is discussed, showing different equations utilized for movement of elevators, and rudder.

Modeling for Aircraft Longitudinal Movement i.e., ELEVATORS- The equation of motion for an aircraft is derived using a moving coordinate system fixed to the aircraft. The $x-y-z$ axes are referred to as body axes. The x-axis is aligned with the longitudinal axis of the airplane. The equations are based on Newton's laws of motion for a rigid body in translation and rotation. The result is a system of six coupled nonlinear differential equations. Three of the six equations expressed accelerations $\dot{u}, \dot{v}, \dot{w}$ in terms of body axis velocities u, v, w, angular velocities p, q, r and external, aerodynamics, and gravitational forces acting on the plane. The remaining three equations relate the angular accelerations $\dot{p}, \dot{q}, \dot{r}\ to\ p, q, r$ and moments produced by the external and aerodynamics forces about the plane's centre of mass.

The plane's attitude is fixed by three rotations of $x-y-z$ axes starting from an orientation initially aligned with the $x'-y'-z'$ axes of the inertial coordinate system. The angular rotation θ, ψ, ϕ are called Euler angles and denote the roll, pitch and yaw of the plane, respectively. Solution to the flight dynamics equation yields u, v, w in the $x-y-z$ body axis coordinate system. The velocity vector v is converted from body axis components u, v, w to inertial components $\dot{x}', \dot{y}', \dot{z}'$ by transformation matrix C_e^b.

$$\begin{bmatrix} \dot{x}' \\ \dot{y}' \\ \dot{z}' \end{bmatrix} = C_e^b \begin{bmatrix} u \\ v \\ w \end{bmatrix} \quad (2.1)$$

$$C_e^b = \begin{bmatrix} \cos\theta\cos\psi & \sin\phi\sin\theta\cos\psi - \cos\phi\sin\psi & \cos\phi\sin\theta\cos\psi + \sin\phi\sin\psi \\ \cos\theta\sin\psi & \sin\phi\sin\theta\sin\psi + \cos\phi\cos\psi & \cos\phi\sin\theta\sin\psi - \sin\phi\cos\psi \\ -\sin\theta & \sin\phi\cos\theta & \cos\phi\cos\theta \end{bmatrix} \quad (2.2)$$

The longitudinal dynamics respond to changes in elevator deflection and thrust. Elevator deflection and thrust result from changes to the yoke and throttle by pilot. Here we consider specifically pitch and attitude response of aircraft to changes in elevator deflection when the plane is flying at a constant cruising speed in horizontal flight under steady-state condition.

Sine our interest is solely in the longitudinal dynamics, specifically pitch and attitude response of aircraft to change in elevator deflection when the plane is flying at a constant cruising speed in horizontal flight under steady-state conditions regarding this particular section, therefore we consider some equations relating to elevator control surface, as given below. From the figure for the plane to be in level flight the velocity vector v must be horizontal, the flight angle$y = 0$, and the pitch is equal to the angle of attack. The plane is pitched slightly in order for the wings to develop sufficient lift to overcome gravity. The steady state conditions are shown in figure with v_o (horizontal cruising speed), $\bar{u}$ (longitudinal speed), $\bar{w}$ (speed in z-direction), $\bar{\theta}$ (pitch), and $\bar{\alpha}$ (angle of attack). The elevator input and engine thrust necessary to maintain these conditions are $\overline{\delta_e}$ and$\overline{\delta_r}$, respectively.

The derivation in u, α, θ, w and q from their steady-state operating level are

$$\Delta u = u - \bar{u},\ \Delta w = w - \bar{w} = w,\ \Delta\alpha = \alpha - \bar{\alpha},$$

$$\Delta\theta = \theta - \bar{\theta}, \Delta q = q - \bar{q} = q \quad (2.3)$$

Since we considering only changes in elevator deflection,

$$\Delta\delta_e = \delta_e - \overline{\delta_e}, \Delta\delta_T = \delta_T - \overline{\delta_T} = 0 \quad (2.4)$$

The state vector Δx in a linearized model of longitudinal dynamics can be chosen as either $[\Delta u\ \Delta w\ \Delta q\ \Delta\theta]^T$ or$[\Delta u\ \Delta\alpha\ \Delta q\ \theta]^T$. The

Relationship between u, w and α is

$$\tan\alpha = \frac{w}{u} \qquad (2.5)$$

For small angle of attack,$\tan\alpha = \sin\alpha / \cos\alpha \approx \alpha$. Replacing $\tan\alpha$ in equation (2.5) with α and solving for w give

$$w = u\alpha \qquad (2.6)$$

Solving for $u, \alpha\ and\ w$ in equation (2.3) and substituting the results into equation (2.6)

$$\bar{w} + \Delta w = (\bar{u} + \Delta u)(\bar{\alpha} + \Delta\alpha) = \overline{u\alpha} + \bar{u}\Delta\alpha + \bar{\alpha}\Delta u + \Delta u\Delta w \qquad (2.7)$$

Suppose a linearized model of aircraft cruising in level flight under steady-state conditions with v_o=500 ft/s and $\bar{\alpha} = \bar{\theta}$= 0025 rad (2.86°) is

$$\frac{d}{dt}\begin{bmatrix}\Delta u\\ \Delta\alpha\\ \Delta p\\ \Delta\theta\end{bmatrix} = \begin{bmatrix}-0.04 & 11.59 & 0 & -32.2\\ -0.00073 & -0.65 & 1 & 0\\ 0.000048 & -0.49 & -0.58 & 0\\ 0 & 0 & 1 & 0\end{bmatrix}\begin{bmatrix}\Delta u\\ \Delta\alpha\\ \Delta p\\ \Delta\theta\end{bmatrix} + \begin{bmatrix}0 & 0.1\\ 0 & 0\\ -0.014 & 0\\ 0 & 0\end{bmatrix}\begin{bmatrix}\Delta\delta_e\\ \Delta_T\end{bmatrix} \qquad (2.8)$$

Where

Choosing the output $\Delta y = \Delta\underline{x} = [\Delta u\ \Delta\alpha\ \Delta q\ \Delta\theta]^T$ leads to the system of state equations $\Delta\dot{\underline{x}} = A\Delta\underline{x} + B\Delta\underline{u}, \Delta y = C\Delta\underline{x} + D\Delta\underline{u}$ with A and B the matrices in equation (6), C equals to the 4*4 identity matrix and D is a 4*2 matrix of zeros.

The linearized equations in state variable form can be converted to a transfer function matrix relating the four outputs $\Delta u(s), \Delta\alpha(s), q(s) and\ \Delta\theta(s)$ to the two inputs$\Delta\delta_e(s) and\ \Delta\delta_T(s)$. The transfer function relating elevator input to aircraft pitch is therefore [6].

$$G_{\Delta\delta e}^{\Delta\theta}(s) = \frac{\Delta\theta(s)}{\Delta\delta e(s)} = \frac{-0.0141s^2 - 0.0097s - 0.0005}{s^4 + 1.2700s^3 + 0.9247s^2 + 0.0406s + 0.0125}$$

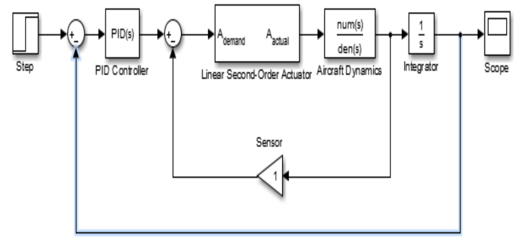

Figure 2 Block diagram for longitudinal control of aircraft.

Following values of time constant (T), natural frequency (ω_n)and zeta (ζ) are obtained by solving equations for these parameters as mentioned in table 1 for linear second-order actuator.

Table 1

parameters	Min	Max
T	0.7524	153.67
$\boldsymbol{\omega_n}$	0.0065	1.3290
ζ	0.10621	1

Design of Lateral Yaw Damper

Here we consider a case study of Douglas DC-8 aircraft for controlling Lateral Yaw Damper movement. When an aircraft has a low speed at a high altitude, the Dutch-roll properties of the aircraft deteriorate. To prevent this, a yaw damper is used. In this example, the design of a yaw damper is illustrated. The aircraft lateral dynamics is specified in state-space form. The design of a pure proportional controller is done using yaw rate feedback to improve the closed-loop damping. The yaw rate response to a rudder command generally includes contributions from all lateral natural modes. Although the Dutch roll is most significant, the spiral and roll subsidence also contribute to this response. Thus, all modes must be adequately stabilized. Moreover, one should avoid the continued and sustained use of the rudder and this can be avoided by employing a suitable washout filter. The rudder servo actuator and washout filter transfer functions are assumed to

$$\frac{\xi}{\xi_c} = \frac{6}{(s+6)}\ \ and\ W(s) = \frac{s}{(s+0.3)}$$

$$\frac{d}{dt}\begin{bmatrix}\Delta v\\ \Delta p\\ \Delta r\\ \Delta\emptyset\end{bmatrix} = \begin{bmatrix}-0.1008 & 0 & -468.2 & 32.2\\ -0.00579 & -1.232 & 0.397 & 0\\ 0.00278 & -0.0346 & -0.257 & 0\\ 0 & 1 & 0 & 0\end{bmatrix}\begin{bmatrix}\Delta v\\ \Delta p\\ \Delta r\\ \Delta\emptyset\end{bmatrix} + \begin{bmatrix}13.48416\\ 0.392\\ -0.864\\ 0\end{bmatrix}\xi\ .$$

The transfer function relating the aileron input to roll angle is

$$\frac{\Delta r}{\xi} = -\frac{\{0.864s^3 + 1.127s^2 + 0.0598s + 0.126\}}{\{s^4 + 1.5898s^3 + 1.7820s^2 + 1.9171 + 0.0124\}}$$

The yaw rate root locus plot with no compensation is shown in Figure 8.50. The proportional gain is chosen as Kr = 3. The bank angle root locus plot with the washout filter is shown in Figure 8.51. The proportional gain is increased to Kr = 7.5. The transfer function of the washout filter is [7]

$$W(s) = \frac{s}{(s + 0.3)}$$

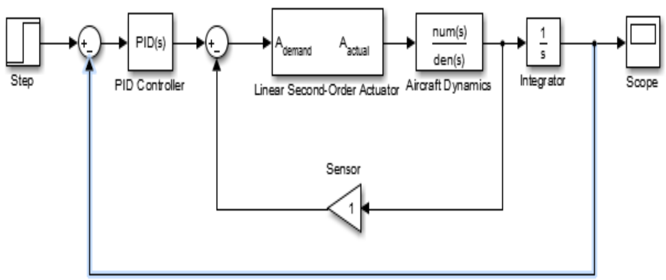

Figure 3 Block diagram for aircraft lateral yaw damper

Following values of T, ω_n and ζ are obtained by solving equations for these parameters as mentioned in table 2 for linear second-order actuator.

Table 2

parameters	Min	Max
T	1.6078	76.7144
ω_n	0.1193	0.9370
Z	0.1093	0.6638

III. METHADOLOGY

PID CONTROLLER-The proportional-integral-derivative (PID) controller is the most common and reliable intelligent controller used in variety of applications, such as in control loop feedback mechanism in industrial control systems, aerospace, etc. The parameters to be tuned is a hurdle in controller design, which has a great effect on the performance of the industrial control systems, especially for those controlled plants with high order and time delays.

We consider here a PID controller in a closed-loop system using the schematic shown in Fig. 4 and expressed as in Equation (1). The input r (t) is the desired process value or "set point", and the output y (t) is the actual output measured by detection equipment. The variable $e (t) = r (t) - y (t)$ represents the tracking error, which will be sent to the PID controller, and the controller computes the proportion, derivative and the integral of this error signal.

$$u(t) = k_p e(t) + k_i \int_0^t e_t \ d(t) + k_d \frac{d}{dt} e(t)$$

The control signal u(t) sent to the plant, is equal to the proportional gain (kp) times the magnitude of the error plus the integral gain (ki) times the integral of the error plus the derivative gain (kd) times the derivative of the error. It is generally known that the dynamic performance of a control system is often measured by four major characteristics of the closed-loop step response, i.e., Rise Time (tr), Overshoot (σ%), Settling Time (ts) and Steady-state Error (ess).

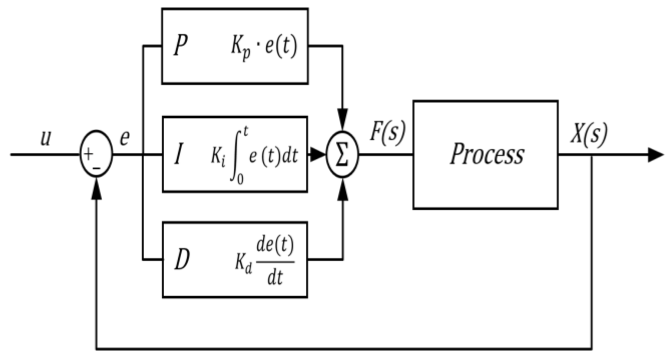

Fig.4 Block diagram of PID controller in closed loop

More specifically, ess of the system under the step response is the difference between the input u(t) and the output y(t) when t → ∞. tr is the time it takes for the output signal y(t) to go from 10% to 90% of its steady-state value. ts is time that y(t) enters and stays in the interval [y(∞)−Δy, y(∞)+Δy], where the Δy is usually defined as either 2% or 5% of the steady-state value y(∞). The overshoot σ is defined using the following ratio: σ = yM − y (∞) y (∞), (2) where yM is the peak value. When we design a controller, it is expected to have a short starting time, high response speed, small overshoot and tracking error, and good robustness [8].

IV. SIMULATION RESULTS AND DISCUSSION

In this section of paper simulation results obtained for aircraft longitudinal, and Lateral yaw damper movement are compared with the standard model. Following responses are obtained in MATLAB showing two different models of Aircraft representing longitudinal and lateral yaw damper movement of aircraft. Using MATLAB Simulink model we can depict the behavior of the system as it would do in real conditions.

A. Aircraft Longitudinal movement

Figure 4.1 and 4.2 shows the step input response of aircraft elevator for longitudinal control movement for open loop system and for closed loop system respectively, showing elevator deflection v/s time

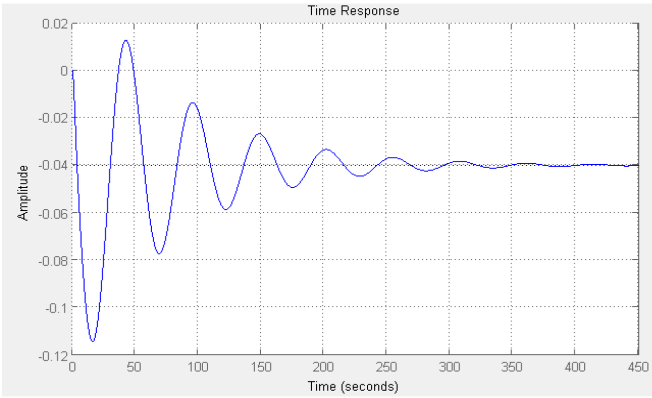

Figure 4.1 Step input response of elevator without controller.

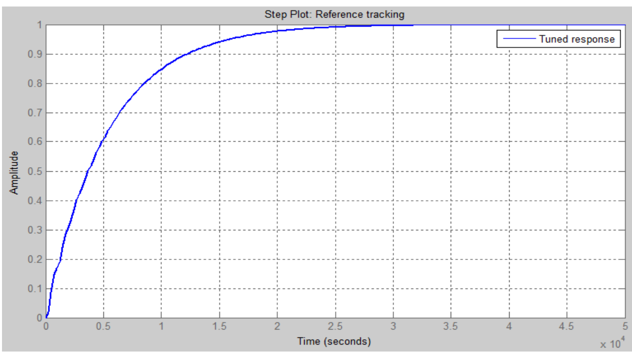

Figure 4.2 Step response of elevator with PID controller

B. Aircraft Lateral Yaw damper movement control

Figure 4.3 & 4.4 shows the root locus plot for rudder , with and without wash filter, and figure 4.5 and 4.6 shows step input response of rudder for open loop and closed loop i.e., without and with controller.

Table 3 shows detail comparison of results with original work for longitudinal and lateral yaw damper movement of aircraft implemented with pid controller. Considering detail time response specification of the system it can be seen from the transient response that pid controller stabilizes the system immediately performing task with high level of accuracy and providing great robustness to the system

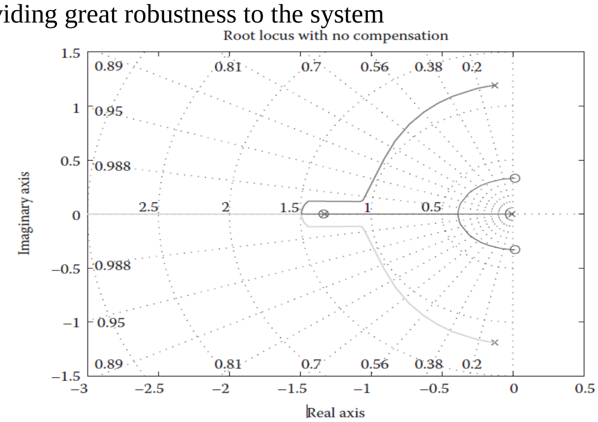

Figure 4.3 Rudder to Yaw rate root locus

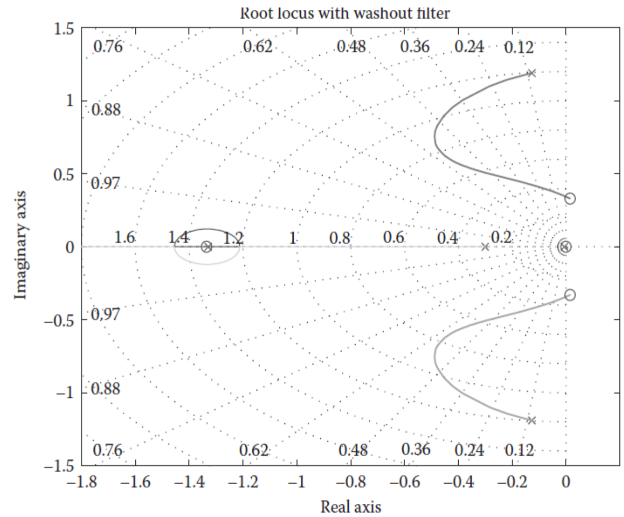

Figure 4.4 Yaw rate root locus plot without wash filter

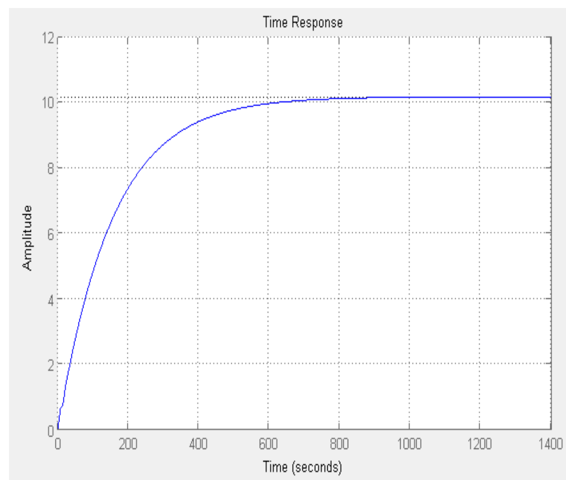

Figure 4.5 Aircraft rudder yaw rate step response without controller

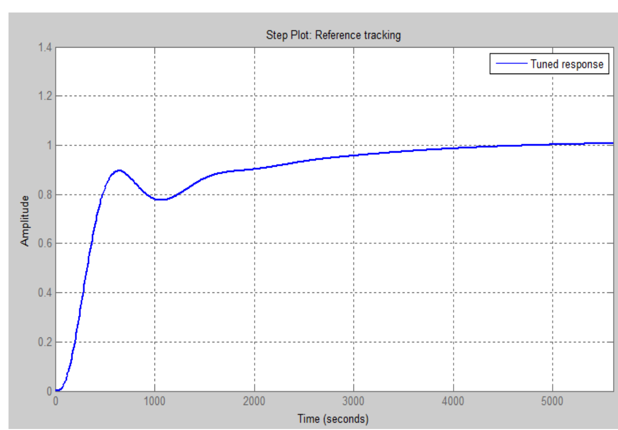

Figure 4.6 Aircraft rudder yaw rate response with PID controller

Table 3

Control action	**Without controller**				**With PID controller**			
Parameters	t_r	t_s	*overshoot*	*peak*	t_r	t_s	overshoot	peak
Longitudinal	*2.6 sec*	*311 sec*	*186%*	*-0.114*	*0.00862 Sec*	*0.000133 sec*	0.929%	1.01
Yaw	*337 sec*	*602 sec*	*0%*	*10.2*	*27 sec*	*0.00107 sec*	1.87%	1.02

V. CONCLUSION

In this present paper two case studies were repotted i.e., aircraft longitudinal movement, and lateral yaw damper movement, and the same work is further processed and implemented with pid controller and the results are evaluated by making comparison with the original mentioned work. The results discussed in previous section shows the behaviors of system with and without controller in terms of time response specification. Further this work can be evaluated by implementing it with other intelligent and adaptive controllers such as Neuro-Fuzzy, Adaptive Fuzzy, ANN based controllers. In this paper all the observation are made without taking into account the effect of disturbances which occur in the environment acting on a body of Aircraft in the air, such as Hydrodynamic forces, radiation force, Excitation force and Drag Force.

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