

Study of a Longitudinal Autopilot For Different Aircrafts

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Abstract

In this paper the comparative study on the performance of an autopilot is made for four different types of the aircraft i.e. general aviation aircraft, fighter aircraft, jet aircraft and large commercial jet aircraft. This paper describes the performance for all the four types of the aircrafts when same autopilot and same input command is used for all aircrafts in the longitudinal mode. Comparative stability analysis is also done which drives to the conclusion that every aircraft must have its own autopilot due to its characteristics and dynamics which is different from another. Simulation results, root locus and bode plot are presented in each case.

1 Introduction

An autopilot is a mechanical, electrical, or hydraulic system used in an aircraft to relieve the human pilot. The original use of an autopilot was to provide pilot relief during cruise modes. Autopilots perform functions more rapidly and with greater precision than the human pilot. In addition to controlling various types of aircraft and spacecraft, autopilots are used to control ships or sea-based vehicles.

An autopilot is unique pilot. It must provide smooth control and avoid sudden and erratic behavior. The intelligence for control must come from sensors such as gyroscopes, accelerometers, altimeters, airspeed indicators, automatic navigators, and various types of radio-controlled data links. The autopilot supplies the necessary scale factors, dynamics (timing), and power to convert the sensor signals into control surface commands. These commands operate the normal aerodynamic controls of the aircraft.

1.1 Replacement of Human Pilot

In the aircraft control systems, human pilots play a very important role as they have certain advantages. Human pilots are highly adaptable to unplanned situations, means they can react according to the desired conditions. Also they have broad-based intelligence and can communicate well with other humans. But still autopilots have advantages over the human pilot which forced it to replace human pilot (10). These advantages are described below.

- Autopilots have high reaction speed as comparison to human pilot.
- They can communicate well with computers, which is difficult for human.
- They can execute multiple events and tasks at the same time.
- They also eliminate risk to on-board pilot.
- Autopilot relieves human pilot from fatigue.

1.2 Flight Control System

A flight control system is either a primary or secondary system. Primary flight controls provide longitudinal (pitch), directional (yaw), and lateral (roll) control of the aircraft. Secondary flight controls provide additional lift during takeoff and landing, and decrease aircraft speed during flight, as well as assisting primary flight controls in the movement of the aircraft about its axis. Some manufacturers call secondary flight controls auxiliary flight controls. All systems consist of the flight control surfaces, the respective cockpit controls, connecting linkage, and necessary operating mechanisms. Basically there are three type of flight control systems as discussed below.

1.2.1 Mechanical

Mechanical flight control systems are the most basic designs. They are basically unboosted flight control systems. They were used in early aircraft and currently in small aeroplanes where the aerodynamic forces are not excessive. The flight control surfaces (ailerons, elevators, and rudder) are moved manually through a

series of push-pull rods, cables, bell cranks, sectors, and idlers. Since an increase in control surface area in bigger and faster aircraft leads to a large increase in the forces needed to move them, complicated mechanical arrangements are used to extract maximum mechanical advantage in order to make the forces required bearable to the pilots. This arrangement is found on bigger or higher performance propeller aircraft.

Some mechanical flight control systems use servo tabs that provide aerodynamic assistance to reduce complexity. Servo tabs are small surfaces hinged to the control surfaces. The mechanisms move these tabs, aerodynamic forces in turn move the control surfaces reducing the amount of mechanical forces needed. This arrangement was used in early piston-engined transport aircraft and early jet transports. The primary flight control system is illustrated in Figure 1.

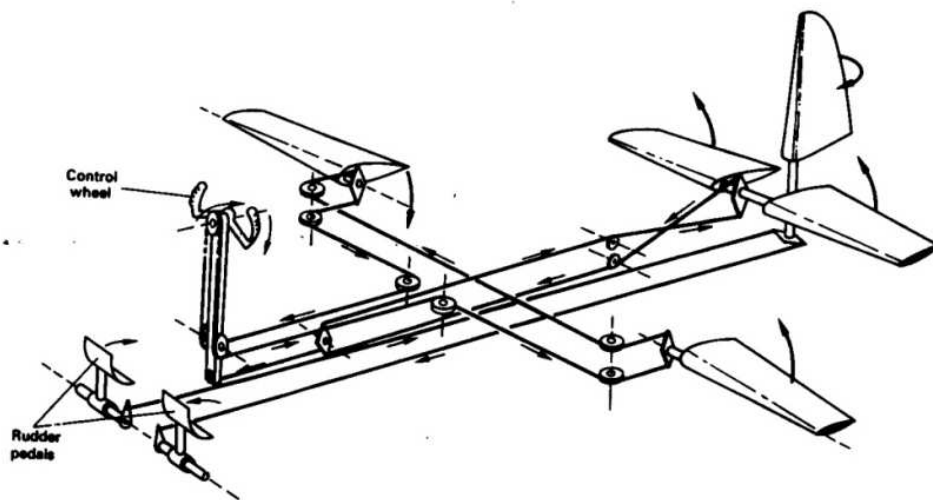


Figure 1: Primary Control System

1.2.2 Hydromechanical

They are power boosted flight control systems. The complexity and weight of a mechanical flight control systems increases considerably with size and performance of the airplane. Hydraulic power overcomes these limitations. With hydraulic flight control systems aircraft size and performance are limited by economics rather than a pilot's strength. In the power-boasted system, a hydraulic actuating cylinder is built into the control linkage to assist the pilot in moving the control surface. A hydraulic flight control systems has 2 parts:

- Mechanical circuit
- Hydraulic circuit

The mechanical circuit links the cockpit controls with the hydraulic circuits. Like the mechanical flight control systems, it is made of rods, cables, pulleys,

and sometimes chains. The hydraulic circuit has hydraulic pumps, pipes, valves and actuators. The actuators are powered by the hydraulic pressure generated by the pumps in the hydraulic circuit. The actuators convert hydraulic pressure into control surface movements. The servo valves control the movement of the actuators. The pilot's movement of a control causes the mechanical circuit to open the matching servo valves in the hydraulic circuit. The hydraulic circuit powers the actuators which then move the control surfaces. These Powered flight controls are employed in high performance aircraft, and are generally of two types (i) power assisted and (ii) power operated, which are shown in the Figures 2 and 3 respectively. Both systems are similar in basic forms but to overcome the aerodynamic loads forces are required, which decides the choice of either of the above system.

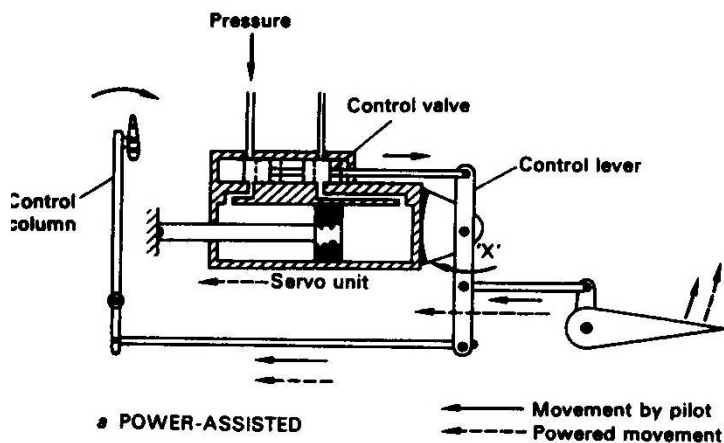


Figure 2: Power Assisted Flight Control System

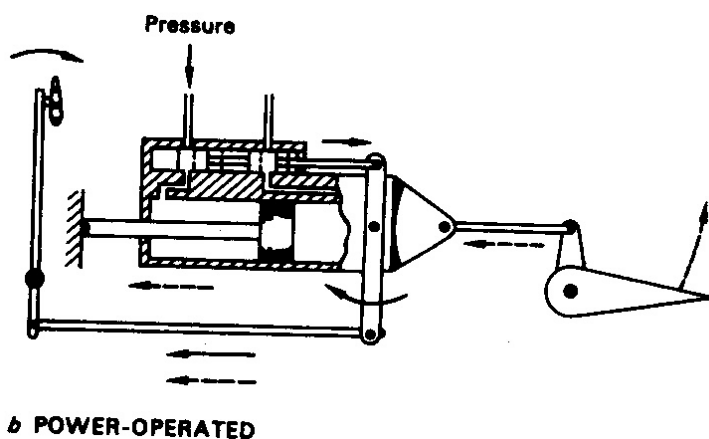


Figure 3: Power Operated Flight Control System

1.2.3 Fly by wire (FBW)

Fly-by-wire is a means of aircraft control that uses electronic circuits to send inputs from the pilot to the motors that move the various flight controls on the aircraft. There are no direct hydraulic or mechanical linkages between the pilot and the flight controls. The total elimination of all the complex mechanical control runs and linkages—all commands and signals are transmitted electrically along wires, hence the name fly-by-wire and also there is interposition of a computer between the pilots commands and the control surface actuators which is incorporated with air data sensors which supply height and airspeed information to the computer. The fly-by-wire system installed in Boeing 767 is shown in Figure 4. Mechanical and hydraulic flight control systems are heavy and re-

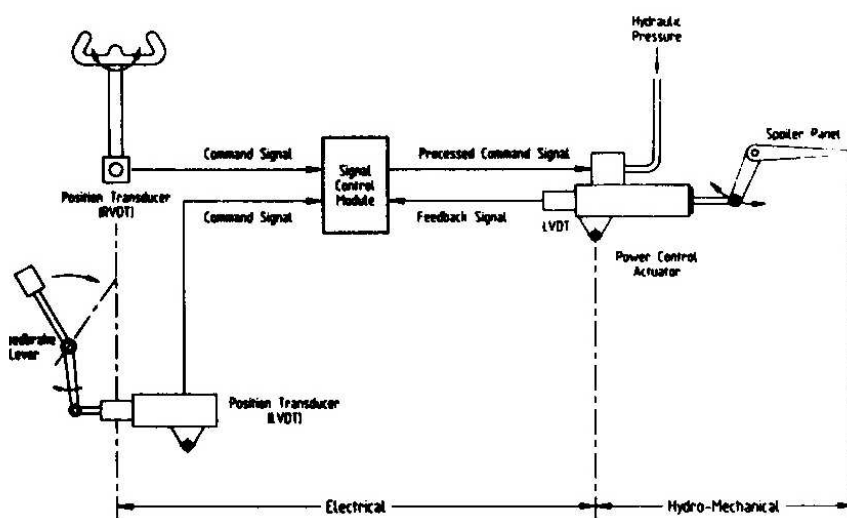


Figure 4: Fly-by-wire System

quire careful routing of flight control cables through the airplane using systems of pulley and cranks. Both systems often require redundant backup, which further increases weight. Another advantages of FBW system are discussed below:

- Provides high-integrity automatic stabilisation of the aircraft to compensate for the loss of natural stability and thus enables a lighter aircraft with a better overall performance.
- Makes the ride much smoother than one controlled by human hands. The capability of FBW systems to maintain constant flight speeds and altitudes over long distances is another way of increasing fuel efficiency. The system acts much like cruise controls on automobiles: less fuel is needed if throttles are untouched over long distances.
- More reliable than a mechanical system because of fewer parts to break or malfunction. FBW is also easier to install, which reduces assembly time and costs. FBW maintenance costs are lower because they are easier to maintain and troubleshoot, and need fewer replacement parts.
- Electrical wires for a flight control system takes up less space inside fuselages, wings, and tail components. This gives designers several options. Wings and

tail components can be designed thinner to help increase speed and make them aerodynamically cleaner, and also to reduce weight. Space once used by mechanical linkages can also be used to increase fuel capacities to give the aircraft greater range or payload.

1.3 History

In the early days of transport aircraft, aircraft required the continuous attention of a pilot in order to fly in a safe manner. This created very high demands on crew attention and high fatigue. The autopilot is designed to perform some of the tasks of the pilot.

The first aircraft autopilot was developed by Sperry Corporation in 1912. The autopilot using gyros to sense the deviation of the aircraft from the desired attitude and servo motors to activate the elevators and ailerons was built under the direction of Dr. E. A. Sperry. The apparatus, called Sperry Aeroplane Stabilizer, installed in the Curtis Flying Boat, won prominence on the 18th June, 1914. Elmer Sperry demonstrated it two years later in 1914, and proved his credibility of the invention, by flying the plane with his hands up (2).

The autopilot connected a gyroscopic attitude indicator and magnetic compass to hydraulically operated rudder, elevator, and ailerons. It permitted the aircraft to fly straight and level on a compass course without a pilot's attention, thus covering more than 80 percent of the pilot's total workload on a typical flight. This straight-and-level autopilot is still the most common, least expensive and most trusted type of autopilot. It also has the lowest pilot error, because it has the simplest controls.

In the early 1920s, the Standard Oil tanker J.A Moffet became the first ship to use autopilot. The early aircrafts were primarily designed to maintain the attitude and heading of the aircraft. With the advent of the high performance aircrafts, new problems have arisen i.e. unsatisfactory dynamic characteristics. To obtain the more performance, several improvements have been done. Now days modern autopilots are dominating. Modern autopilots generally divide a flight into take-off, ascent, level, descent, approach, landing, and taxi phases. Modern autopilots use computer software to control the aircraft. The software reads the aircraft's current position, and controls a flight control system to guide the aircraft. In such a system, besides classic flight controls, many autopilots incorporate thrust control capabilities that can control throttles to optimize the air-speed, and move fuel to different tanks to balance the aircraft in an optimal attitude in the air. Although autopilots handle new or dangerous situations inflexibly, they generally fly an aircraft with a lower fuel-consumption than all but a few of the best pilots.

2 Techniques Available

The basic aim of an autopilot is to track the desired goal. Autopilot can be displacement type or pitch type. There are different techniques available to design an autopilot like model-following control, sliding mode control, model predictive control, robust control, lyapunov based control, adaptive control and dynamic inversion control. Every technique has its disadvantages, so to improve the disadvantages new techniques are proposed.

Dynamic Inversion is a design technique which is based on feedback linearization, basically it achieves the effect of gain scheduling. This approach transforms the nonlinear system a linear time invariant form. The transformation is then inverted to obtain a nonlinear control law, but the limitation (1) of this technique is that it can not be applied to nonminimum phase plants. In this technique, the set of existing dynamics are canceled out and replaced by a designer selected set of desired dynamics. Dynamic inversion is similar to model following control, in that both methodologies invert dynamical equations of the plant and an appropriate co-ordinate transformation is carried out to make the system look linear so that any known linear controller design can be used.

Lyapunov based control design techniques (3; 5) use the Lyapunov's stability theorem for nonlinear systems and come up with adaptive control solutions that guarantee stability of the error dynamics (i.e. the tracking error remains bounded in a small neighborhood about zero) or sometimes assure asymptotic stability (i.e. the tracking error goes to zero). In sliding mode control (5; 8), the essential idea is to first lay out a path for the error signal that leads to zero. Then the control solution is found in such a way that the error follows this path, finally approaching zero. In doing so, however, the usual problems encountered are high magnitudes of control and control chattering. In Predictive control (9), the error signal is first predicted for some future time (based on a model that may or may not be updated in parallel). The control solution at the current time step is then obtained from an error minimization algorithm that minimizes a cost function, which is a weighted average of the error signal. In this study of autopilots, classical methods are used to find the response when the command is given.

3 Components of Autopilot

The basic components of the autopilot is shown in the figure 5 Aircraft mo-

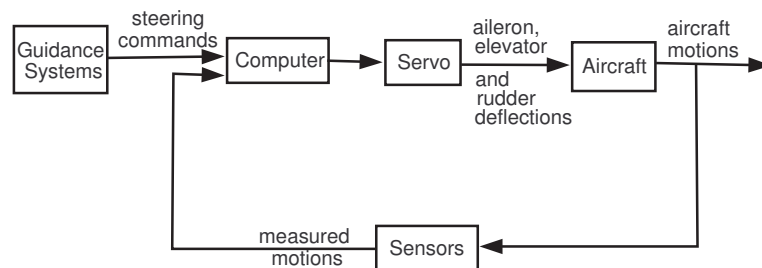


Figure 5: Basic Components of an Autopilot

tion is usually sensed by a gyro, which transmits a signal to a computer (see illustration). The computer commands a control servo to produce aerodynamic

forces to remove the sensed motion. The computer may be a complex digital computer, an analog computer (electrical or mechanical), or a simple summing amplifier, depending on the complexity of the autopilot. The control servo can be a hydraulically powered actuator or an electromechanical type of surface actuation. Signals can be added to the computers that supply altitude commands or steering commands. For a simple autopilot, the pitch loop controls the elevators and the roll loop controls the aileron. A directional loop controlling the rudder may be added to provide coordinated turns.

4 Aircraft Dynamics

4.1 Equation of motion

The assumption taken is that the X and Z axes lie in the plane of symmetry, origin of axis system is the center of gravity of the aircraft and perturbations from equilibrium are small. Also assuming the aircraft to be rigid body, constant mass and quasisteadyflow, the longitudinal equations ?? of motion for an aircraft can be written as:

$$\left(\frac{mV_T}{Sq}\dot{u} - C_{x_u}u\right) + \left(-\frac{c}{2V_T}C_{x_\alpha}\dot{\alpha} - C_{x_\alpha}\alpha\right) + \left(-\frac{c}{2V_T}C_{x_q}\dot{\theta} - C_w(\cos\Theta)\theta\right) = C_{F_{x_a}} \quad (1)$$

$$-(C_{z_u}u) + \left(\frac{mV_T}{Sq} - \frac{c}{2V_T}C_{z_\alpha}\right)\dot{\alpha} - C_{z_\alpha}\alpha + \left(-\frac{mV_T}{Sq} - \frac{c}{2V_T}C_{z_q}\right)\dot{\theta} - C_w(\sin\Theta)\theta = C_{F_{z_a}} \quad (2)$$

$$-(C_{m_u}u) + \left(\frac{c}{2V_T}C_{m_\alpha}\dot{\alpha} - C_{m_\alpha}\alpha\right) + \left(\frac{I_y}{Sq c}\ddot{\theta} - \frac{c}{2V_T}C_{m_q}\dot{\theta}\right) = C_{m_a} \quad (3)$$

where u , α and θ represents vaiations in the forward velocity, angle of attack and the pitch angle respectively. m , c , q , S represents mass, mean aerodynamic chord, dynamic pressure and span area respectively. V_T is the total forward velocity in the longitudinal direction. C_{x_u} , C_{x_α} , C_{x_α} , C_{x_q} , C_w , C_{z_u} , C_{z_α} , C_{z_α} , C_{z_q} , C_{m_u} , C_{m_α} , C_{m_α} and C_{m_q} are the stability derivatives taken from the reference (7). In solving the equation of motion to obtain the transient solution, which is obtained from the homogenous equations, that is with no external inputs: $C_{F_{x_a}} = C_{F_{z_a}} = C_{m_a} = 0$. Taking the laplace transform of Eqs. 1, 2 and 3 with initial condition zero yields:

$$\left(\frac{mV_T}{Sq}s - C_{x_u}\right)u + \left(-\frac{c}{2V_T}C_{x_\alpha}s - C_{x_\alpha}\right)\alpha + \left(-\frac{c}{2V_T}C_{x_q}s - C_w(\cos\Theta)\right)\theta = 0 \quad (4)$$

$$-(C_{z_u}u) + \left[\left(\frac{mV_T}{Sq} - \frac{c}{2V_T}C_{z_\alpha}\right)s - C_{z_\alpha}\right]\alpha + \left[\left(-\frac{mV_T}{Sq} - \frac{c}{2V_T}C_{z_q}\right)s - C_w(\sin\Theta)\right]\theta = 0 \quad (5)$$

$$-(C_{m_u}u) + \left(\frac{c}{2V_T}C_{m_\alpha}s - C_{m_\alpha}\right)\alpha + \left(\frac{I_y}{Sq c}s^2 - \frac{c}{2V_T}C_{m_q}s\right)\theta = 0 \quad (6)$$

Solving the above Eqs. 4, 5 and 6, a quartic equation is obtained. The solution of this equation will give the two quadratic factors, one is short period mode and second is phugoid mode. Short period mode is more important, because if the mode has high frequency and is heavily damped, then the aircraft will

respond rapidly to an elevator input without any desirable overshoot. When the short period mode is lightly damped or has a low frequency, the airplane will be difficult to control and in some cases it may be dangerous to fly, but in phugoid mode it occurs slowly so that pilot can easily negate the disturbance by small control movements.

4.2 Short Period Approximation

The short period oscillation occurs at almost constant forward speed, therefore substituting $u = 0$ in the longitudinal equation of motion. The equation in the direction of X axis does not contribute to the short period oscillation as forces in the X direction contributes to the forward speed. With these assumptions and neglecting $C_{z\alpha}$ the equations becomes:

$$\left(\frac{mV_T}{Sq}s - C_{z\alpha}\right)\alpha + \left[-\frac{mV_T}{Sq} - \frac{c}{2V_T}C_{zq}\right]s - C_w(\sin\Theta)]\theta = C_{z\delta_e}\delta_e \quad (7)$$

$$\left(\frac{c}{2V_T}C_{m\dot{\alpha}}s - C_{m\alpha}\right)\alpha + \left(\frac{I_y}{Sq}c^2s^2 - \frac{c}{2V_T}C_{mq}s\right)\theta = C_{m\delta_e}\delta_e \quad (8)$$

where δ_e is the elevator deflection, here elevator command is given and $C_{z\delta_e} = (c/l_t)C_{m\delta_e}$. Value of c/l_t can be calculated from the geometrical figures of the aircraft which is shown at the last. Solving the above equation the transfer function (i.e. $\frac{\theta(s)}{\delta_e(s)}$) is obtained which will be further used with the autopilot to control the system.

4.3 Type1 and Type0 System

Block diagram of the type0 and type1 system are shown in figures 6 and 7 respectively. Difference between the type0 and type1 system is in the steady state error. In type0 system there is steady state error for unit step input, but in type1 system there is no such type of steady state error. In the type1

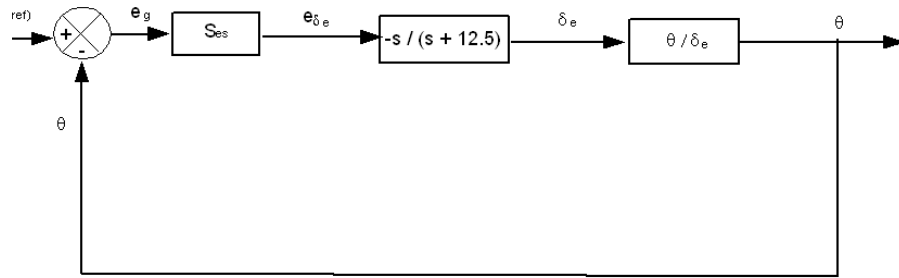


Figure 6: Block Diagram for Type0 System

system, there is one inner loop, this inner loop contains two blocks, one for the aircraft dynamics and other for the combined amplifier and elevator servo. For the internal loop, S_g is used as feedback whose value is tuned. In the outer loop, S_{es} is the outer loop gain with unity feedback. In the type0 system there is no need of the internal loop. To make the system type0 a differentiator is used

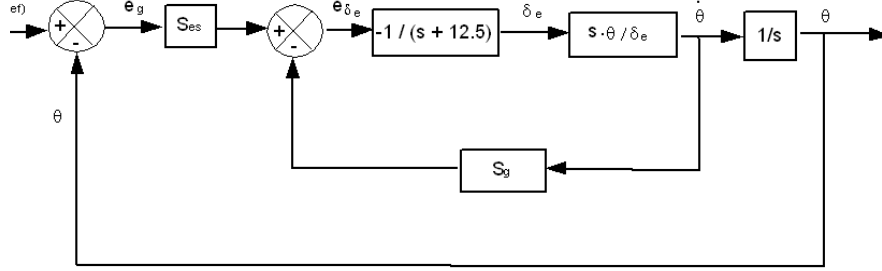


Figure 7: Block Diagram for Type1 System

with the aircraft dynamics, so in this case there is no need of S_g , but the same autopilot is used for the both type of systems. This autopilot shown in the block diagrams is used for general aviation aircraft which is taken from reference (2), further this same autopilot is used for other type of aircrafts i.e. fighter aircraft, jet aircraft and large commercial jet aircraft.

5 Numerical Experiments

All the data used in the simulations for different aircrafts i.e. general aviation aircraft, fighter aircraft, business aircraft and large commercial jet aircraft is taken from the reference (7). For simulation purpose step size of 50 msec is used. In the trim condition parameters like α and Θ are chosen as 0.

5.1 Transfer Function Calculation

The transfer function for all of the aircrafts is calculated using the short period approximation as discussed in section 4.2 are described below:

1. General Aviation Aircraft

$$TransferFunction = \frac{\theta(s)}{\delta_e(s)} = \frac{-11.8(s + 1.97)}{s(s^2 + 5s + 12.96)} \quad (9)$$

2. Fighter Aircraft

$$TransferFunction = \frac{\theta(s)}{\delta_e(s)} = \frac{-4.51(s + 0.44)}{s(s^2 + 0.873s + 2.13)} \quad (10)$$

3. Business Jet Aircraft

$$TransferFunction = \frac{\theta(s)}{\delta_e(s)} = \frac{-2.08(s + 0.5727)}{s(s^2 + 1.3125s + 2.384)} \quad (11)$$

4. Large Commercial Jet Aircraft

$$TransferFunction = \frac{\theta(s)}{\delta_e(s)} = \frac{-5.364(s + 5.2)}{s(s^2 + 10.09s + 26.68)} \quad (12)$$

In case of short period mode the natural frequency, damping ratio and time required to decay the amplitude to half of its value for all the aircrafts are shown in table 1

Table 1: Short Period

Type of Aircraft	w_n	ζ	$t_{1/2}$
General Aviation	3.6	0.694	0.277
Fighter	1.46	0.3	1.58
Business Jet	1.54	0.42	1.07
Large Commercial Jet	5.16	0.976	0.137

5.2 Selection of the Gain Parameters

After some trials and error, the gain parameters are selected which are shown in table 2. Using these gain values, best response is obtained and the comparison

Table 2: Gain Parameters

Type of Aircraft	Type1 System	Type0 System	
	S_g	S_{es}	S_{es}
General Aviation	1.20	5.00	100.00
Fighter	0.15	0.22	0.55
Business Jet	1.00	1.00	1.50
Large Commercial Jet	2.00	1.70	1.80

of different aircraft autopilots are done, which helps in analysing the controller for different type of aircrafts.

5.3 Result analysis

In this study, basic goal of autopilot is to track the given command for 50 sec in the longitudinal case for all four aircrafts. For comparison study, command is square wave having same magnitude in all cases i.e. in both type1 and type0 systems.

5.3.1 General Aviation Aircraft

In the general aviation aircraft the simulation results are shown in figure 12 and figure 13. As it is shown in figure 12, command of $\delta_e = 6$ (*deg/sec*) is given for $t = 5 - 10$ sec and $\delta_e = 0$ (*deg/sec*) for rest of the time. From figure 8, it is observed that all of the poles and zeros lie in left half of s plane and system is quite stable. With gain upto 4.63 system is stable, and further increase of gain shows instability. For justifying stability of the type1 system bode plot is also drawn which is shown in figure 10, in this study gain margin is more and also phase margin is more which is advantageous for stability point of view. In figure 9, poles and zeros always lie in left half of plane whatever is the value of gain, so type0 system for General Aviation Aircraft is always stable which is also justified from bode plot as shown in figure 11. In this gain margin comes out to be infinite which shows its stability. In the figure 12, it is shown clearly that in type1 system response is fast and also there are no oscillations to track the desired variable. But as seen from figure 13 response in type0 system is still more faster from type1 system and response is smoother. So this autopilot actually suits well with the general aviation aircraft. Further analysis will show the response of same autopilot with different aircrafts.

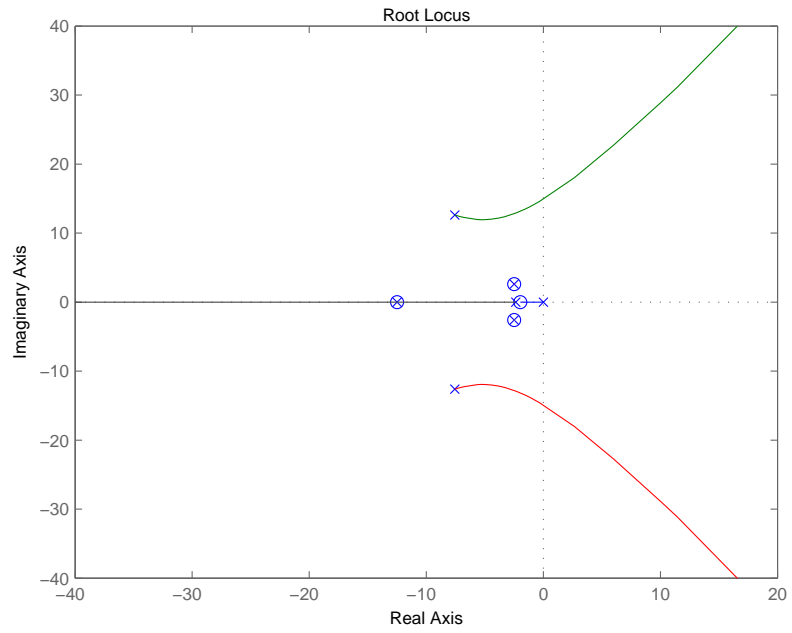


Figure 8: Root Locus for General Aviation Aircraft in Type1 System

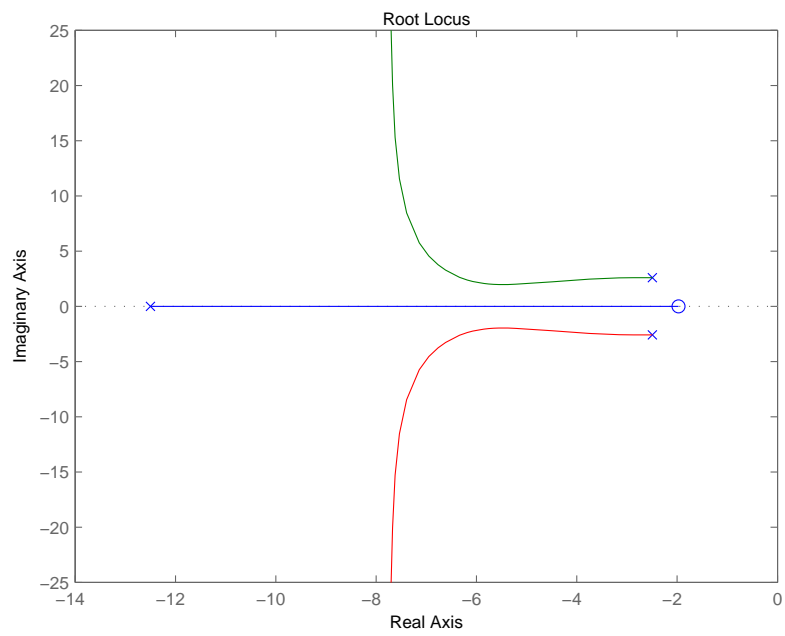


Figure 9: Root Locus for General Aviation Aircraft in Type0 System

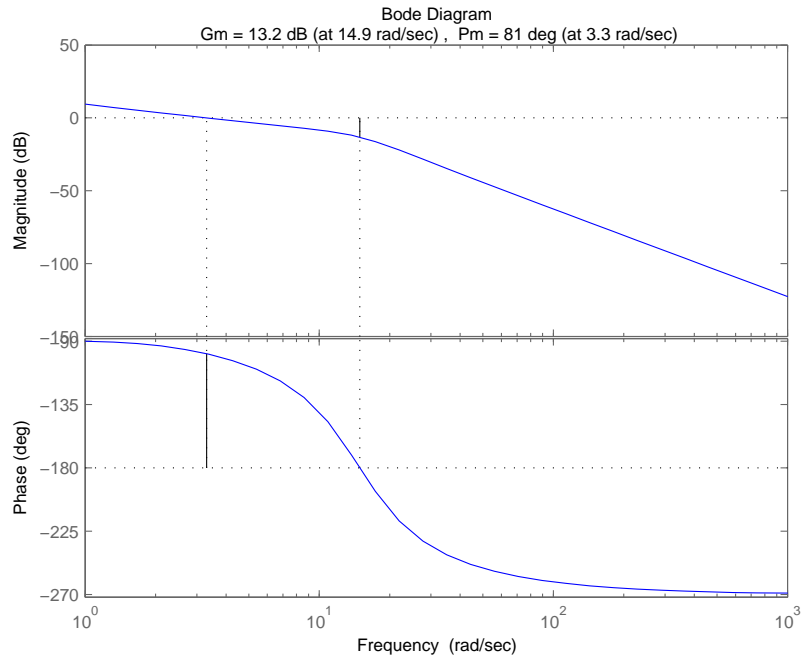


Figure 10: Bode Plot for General Aviation Aircraft in Type1 System

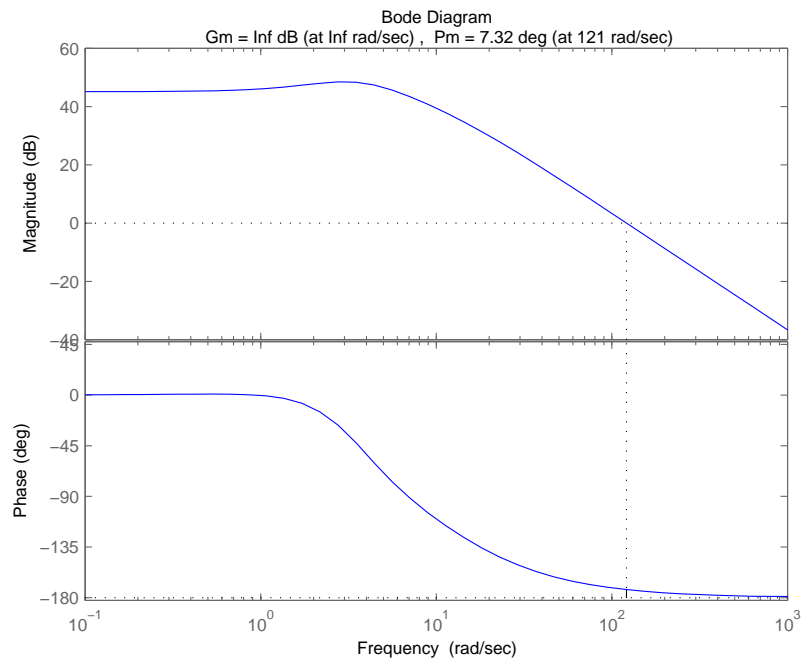


Figure 11: Bode Plot for General Aviation Aircraft in Type0 System

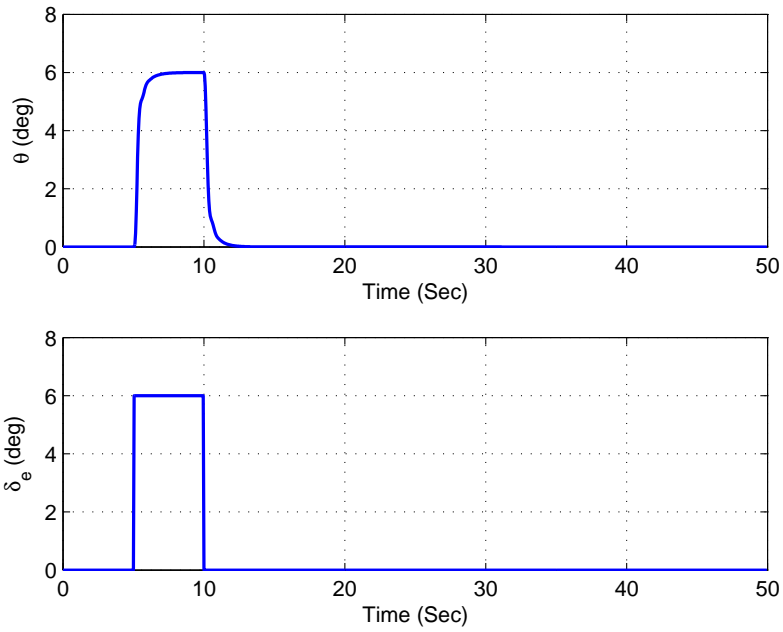


Figure 12: Pitch angle and elevator command respectively for general aviation aircraft in Type1 System

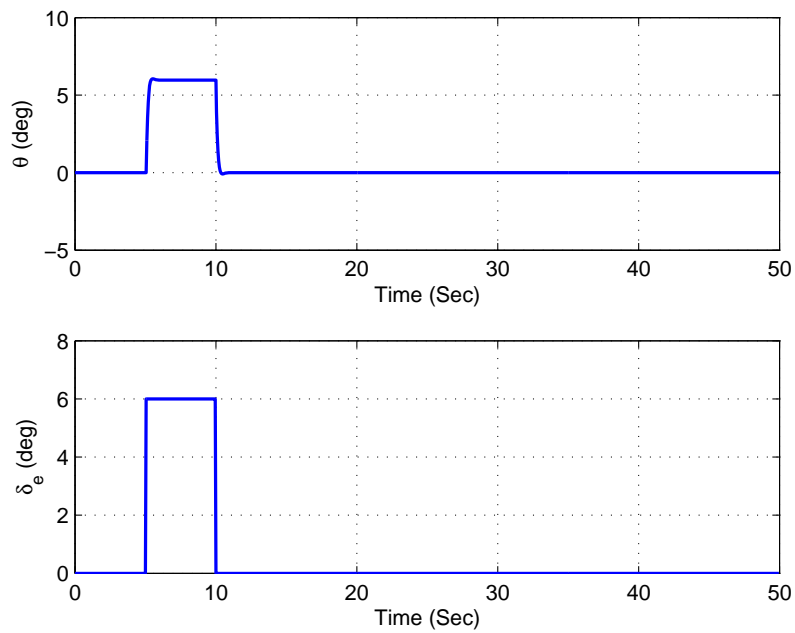


Figure 13: Pitch angle and elevator command respectively for general aviation aircraft in Type0 System

5.3.2 Fighter Aircraft

It is well known that autopilot is different for different aircraft. Now same autopilot used in above aircraft is used for fighter aircraft. Fighter aircraft is high performance aircraft and also its response should be quick, fast and smooth. In the figure 14 root locus for type1 system is shown in which initially all the pole and zeros lie in right half of the s plane, but their location is very close to origin. The gain value till 1.69 (which is less) system is stable, if it is increased further it becomes unstable. This stability also depends on the selection of gain values which are chosen such that system is stable with better response.

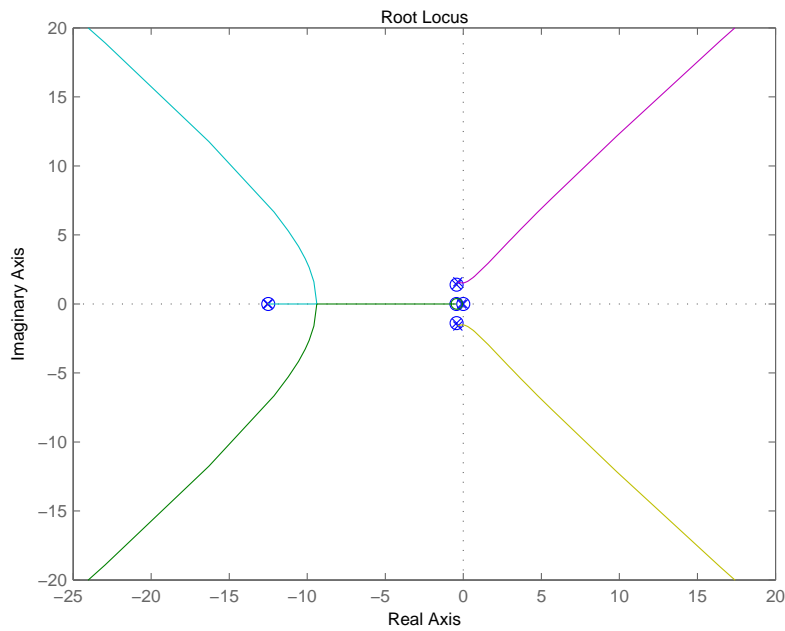


Figure 14: Root Locus for Fighter Aircraft in Type1 System

Bode plot drawn in figure 16 for type1 system shows the gain margin and phase margin magnitudes which are well satisfactory with stability analysis. In case of type0 system also similar type of trend is followed which is totally different from general aviation aircraft. In figure 15 location of poles and zeros shows stable system, but the degree of stability is determined by the bode diagram which is shown in figure 17. Gain margin in type0 system is good but phase margin is less comparatively. Now the main difference comes in the simulation results shown in figure 18 for type1 system and in figure 19 for type0 system. In type1 system the response is oscillatory and also response time is very slow and even it is not able to track the desired goal in 15 sec which is much more for fighter aircraft, usually fighter aircraft should have less response time and the variable should track the desired goal smoothly and quickly which is not happening in this case. Similarly in type0 system also, oscillations are more and also settling time is more. This leads to the poor handling of the aircraft and this should be preferably avoided. So this autopilot is not suitable for the fighter aircrafts as

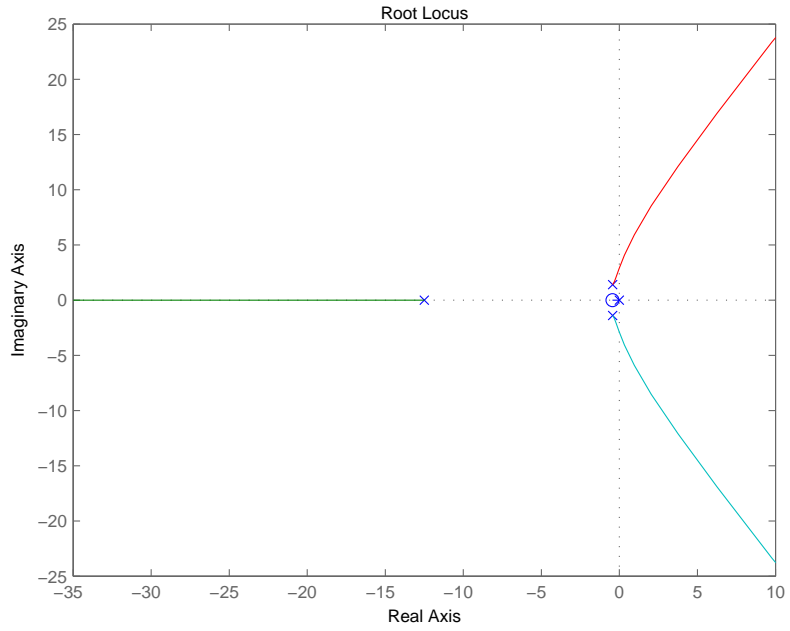


Figure 15: Root Locus for Fighter in Type0 System

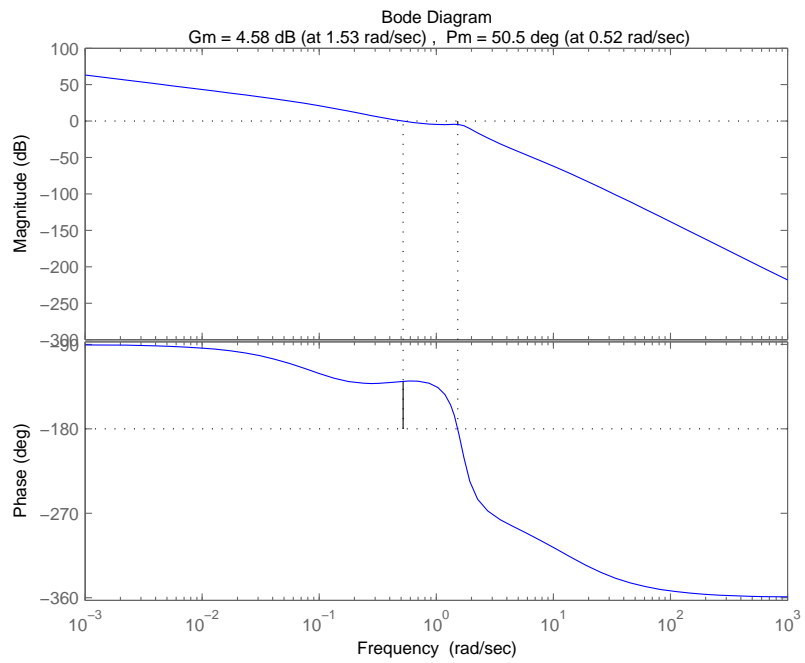


Figure 16: Bode Plot for Fighter in Type1 System

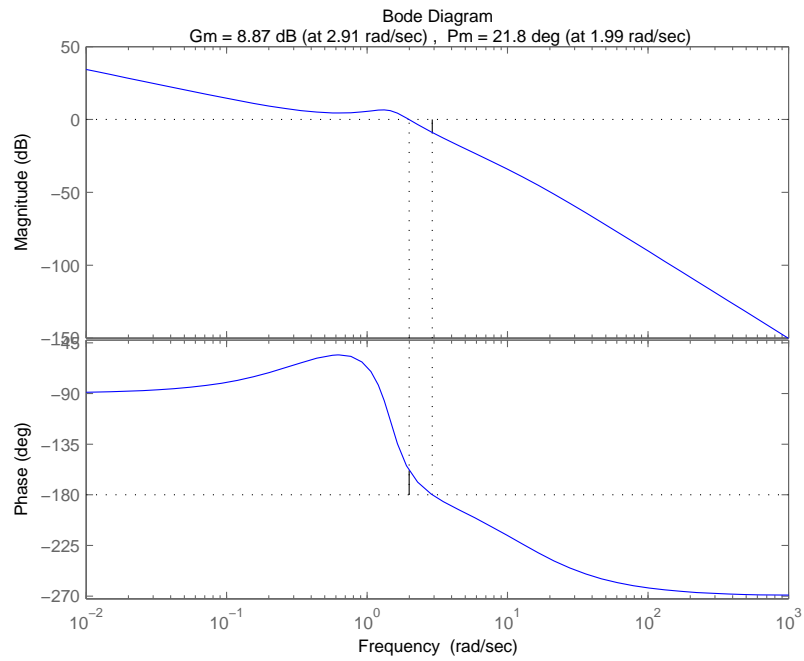


Figure 17: Bode Plot for Fighter in Type0 System

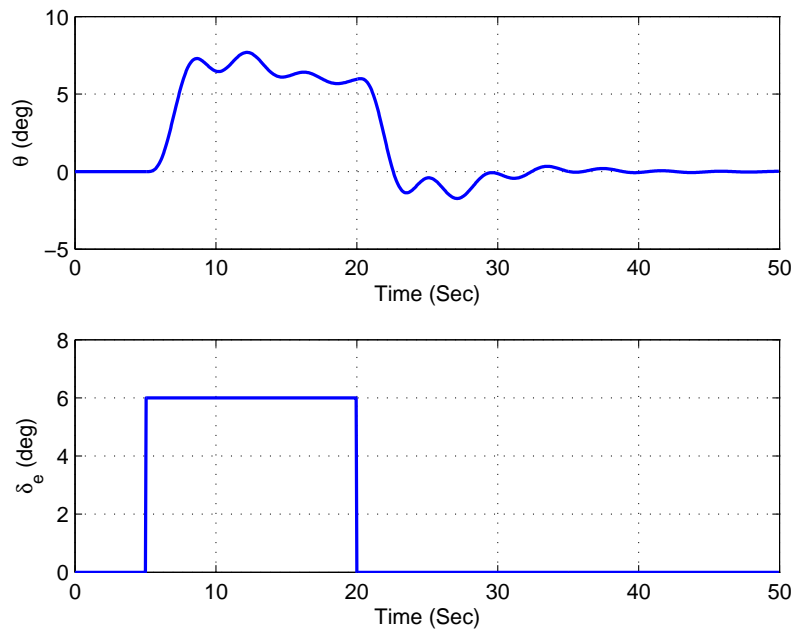


Figure 18: Pitch angle and elevator command respectively for Fighter in Type1 System

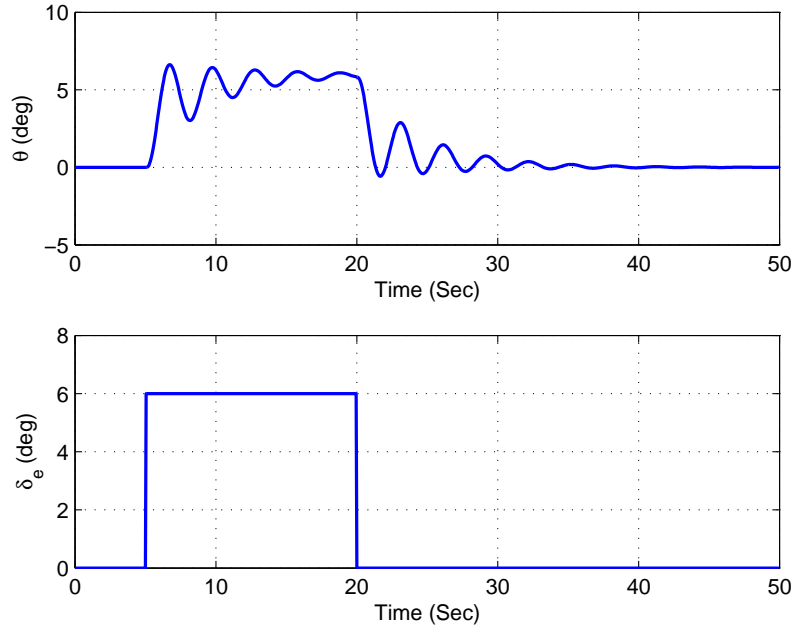


Figure 19: Pitch angle and elevator command respectively for Fighter in Type0 System

they require fast, smooth and better tracking response than other aircrafts.

5.3.3 Business Jet Aircraft

Weight of business jet aircraft is twice than the fighter aircraft. So autopilot for this aircraft should also different than the general aviation aircraft. Detailed study will justify all the points. In figure 20 in case of type1 system it is shown clearly that the poles and zeros lie in left half of the s plane, so system is stable.

But when the gain value is increased beyond the value 1.52 (after this point locus comes in right half plane). Its stability is also justified from the bode plot as shown in figure 22, gain margin and phase margins (3.63 GM and PM is 60.8 deg) are quite satisfactory from the stability point of view. Similar is case with type0 system which is shown in figure 21, but there is difference between the gain margin and phase margin of the type1 system. In type0 system shown in figure 23 phase margin is quite less than type1 system, but still it is quite good to stable the system and correspondingly its gain margin is more. Simulation results for type1 system is shown in figure 24 and for type0 it is shown in figure 25. In type1 system the response time to reach the desired command (6 deg) is more (i.e. about 2.2 sec) as compared to the general aviation aircraft in which response time is less than 1.5 sec and also in general aviation aircraft it settles down early, but here it does not settles down for 15 sec and also it has very oscillatory response which is not required. In case of type0 system the response type is less as compared to type1 system (i.e about 1.4 sec), but when it is compared with the general aviation aircraft then this response time is very

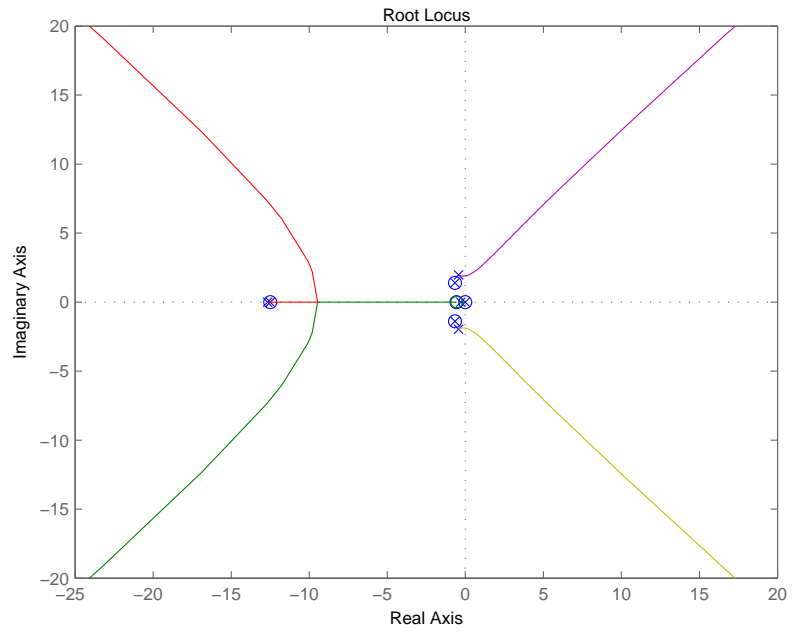


Figure 20: Root Locus for Business Jet Aircraft in Type1 System

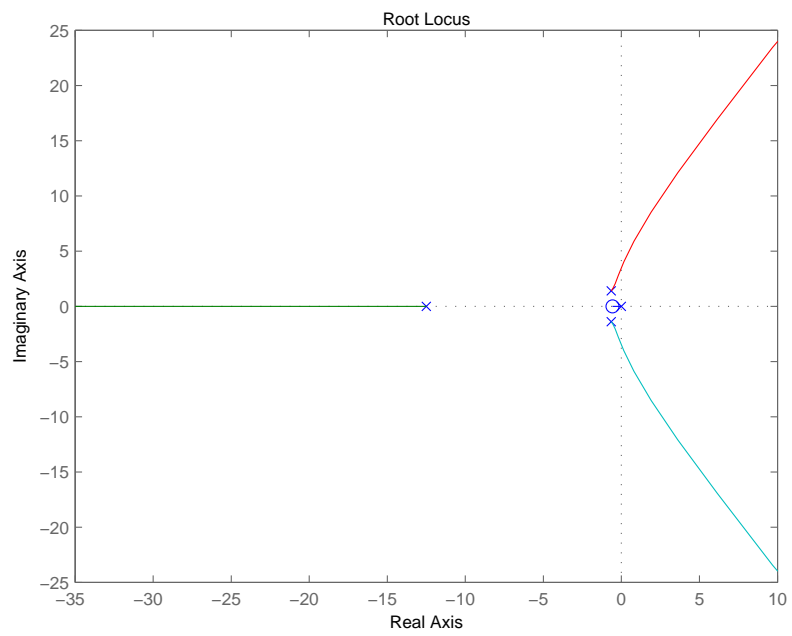


Figure 21: Root Locus for Business Jet Aircraft in Type0 System

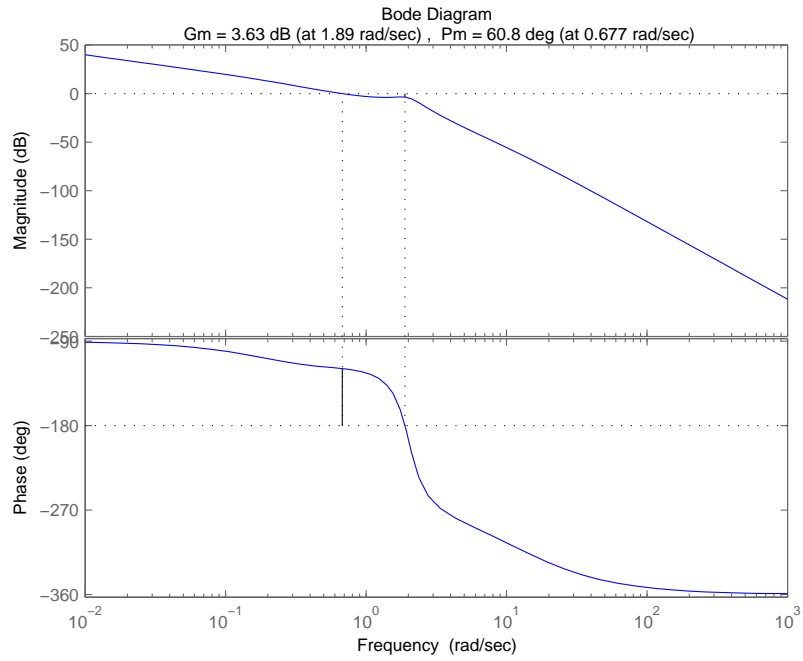


Figure 22: Bode Plot for Business Jet Aircraft in Type1 System

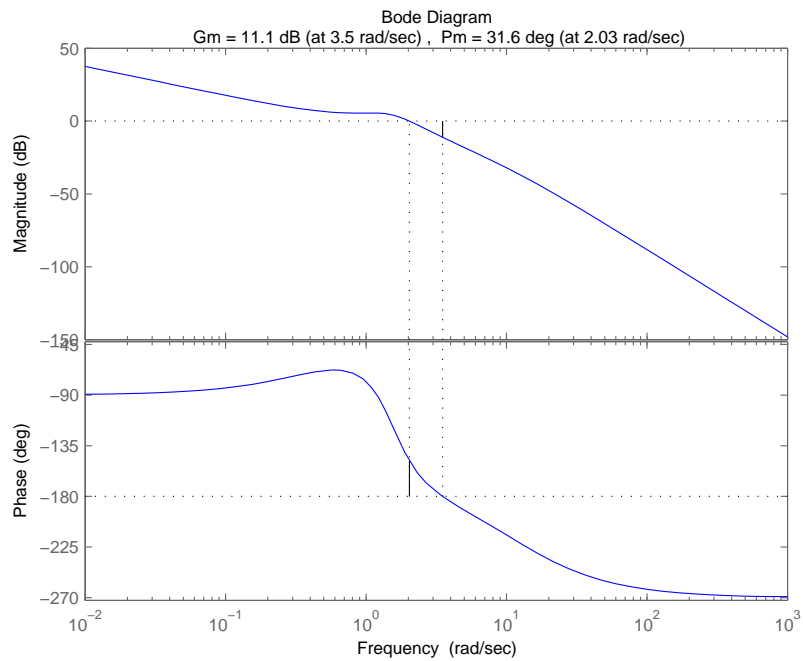


Figure 23: Bode Plot for Business Jet Aircraft in Type0 System

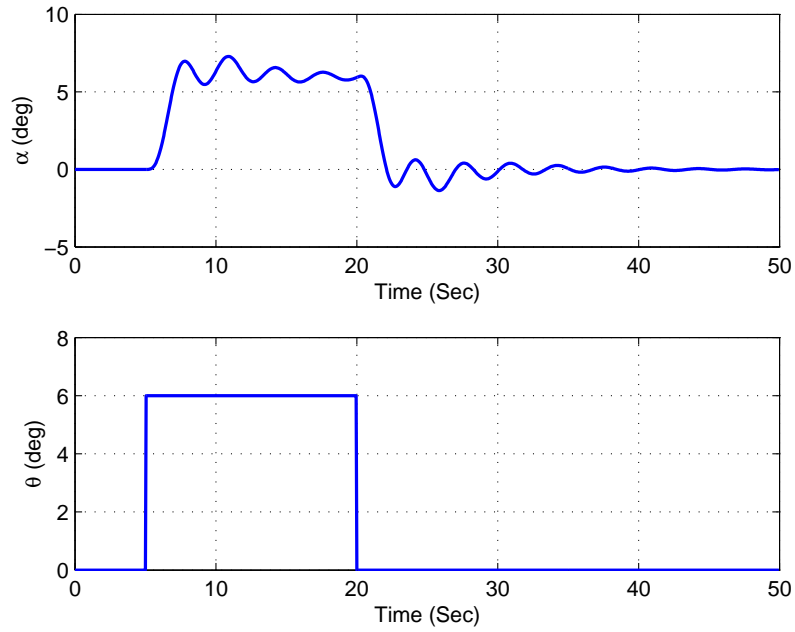


Figure 24: Pitch angle and elevator command respectively for Business Jet Aircraft in Type1 System

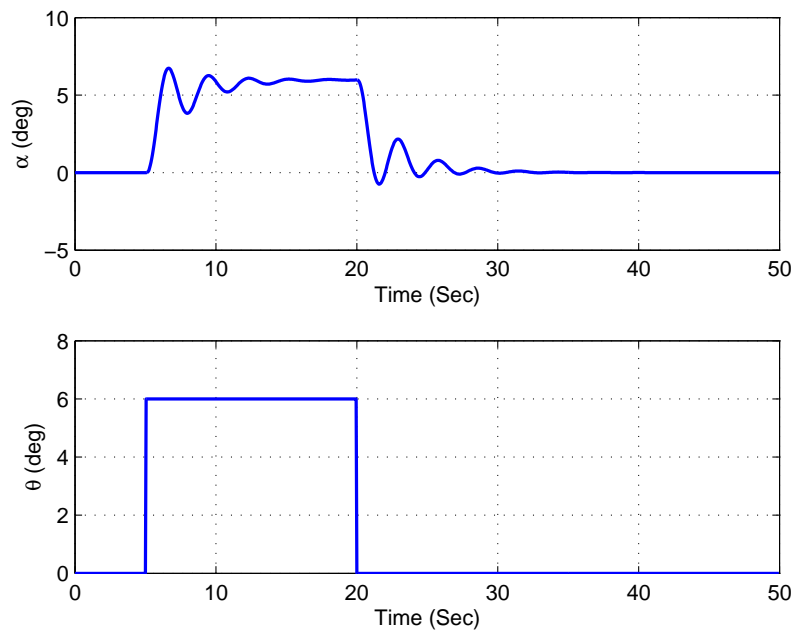


Figure 25: Pitch angle and elevator command respectively for Business Jet Aircraft in Type0 System

large as in general aviation aircraft response time is about 0.4 sec and settling time is about 0.6 sec which is very less as compared to jet aircraft (settling time is about 10 sec). This autopilot is stable for the jet aircraft but its oscillatory and dynamics characteristics are not suitable for this type of the aircraft.

5.3.4 Large Commercial Jet Aircraft

After getting the transfer function and tuning the gain values the root locus is shown in figures 26 and 27. In this aircraft, location of zeros and poles are

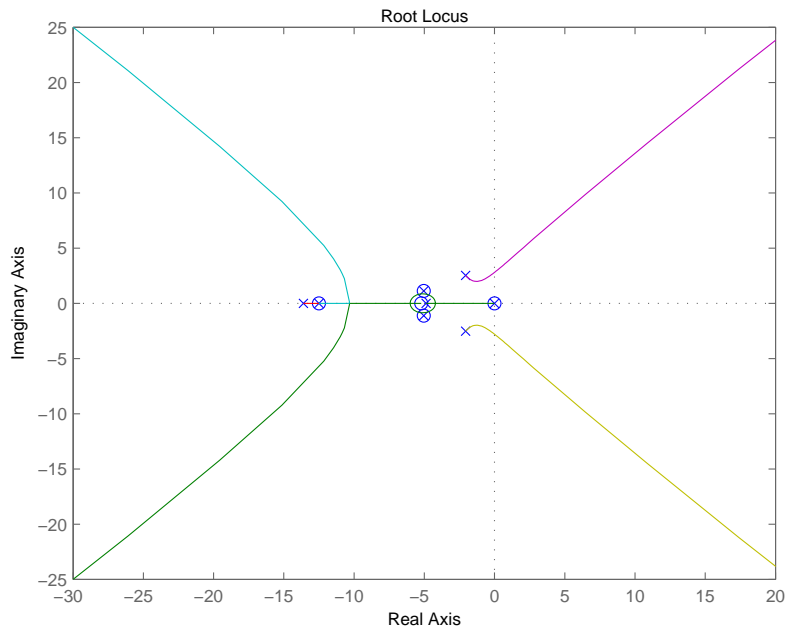


Figure 26: Root Locus for Large Commercial Jet Aircraft in Type1 System

more left as compared to fighter and business jet aircraft which shows that this system is more stable than those systems. This stability is also justified from the bode plots for both type1 and type0 system which is shown in figure 28 and 29. Simulation results are shown in figures 30 and 31 for type1 and type0 system. This autopilot shows the settling time about 1.5 sec for both of the cases and the response time is still more in type0 as compared to general aviation aircraft. So this response turns out to be sluggish which should be preferably avoided. This autopilot is not acceptable because of its slow response.

6 Conclusions

Output response has been studied using the same autopilot for different types of aircrafts with the same step input given. The autopilot pertains to general aviation aircraft. In fighter aircraft, response is oscillatory with more rising and settling time. But in fighter aircraft response should be quick. In jet

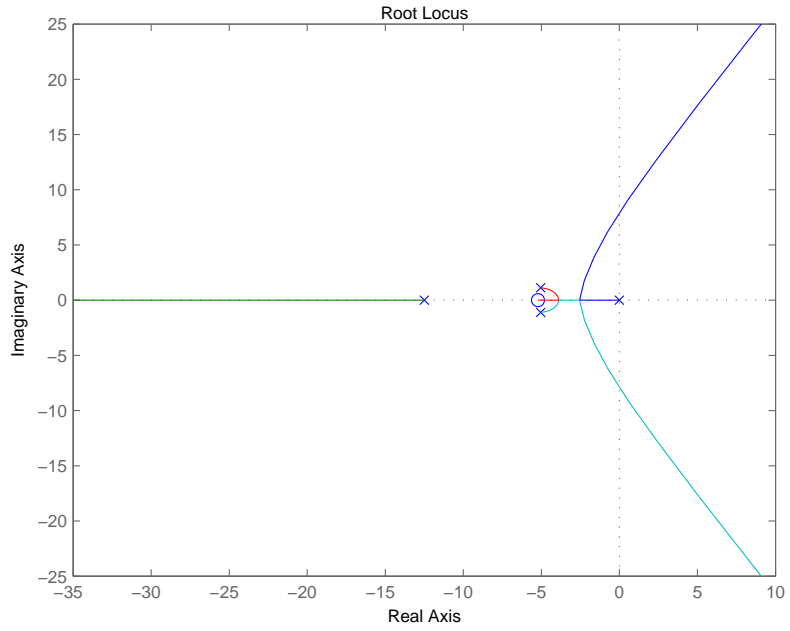


Figure 27: Root Locus for Large Commercial Jet Aircraft in Type0 System

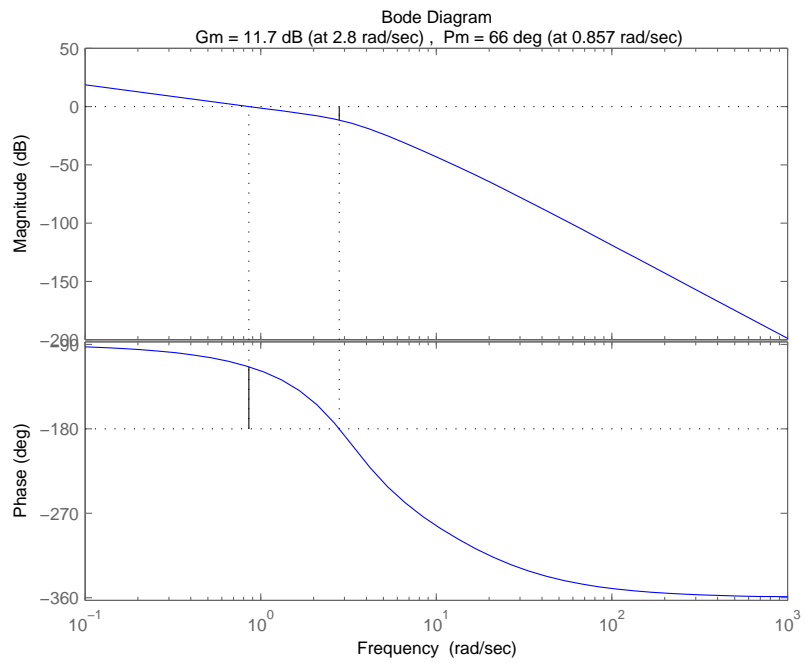


Figure 28: Bode Plot for Large Commercial Jet Aircraft in Type1 System

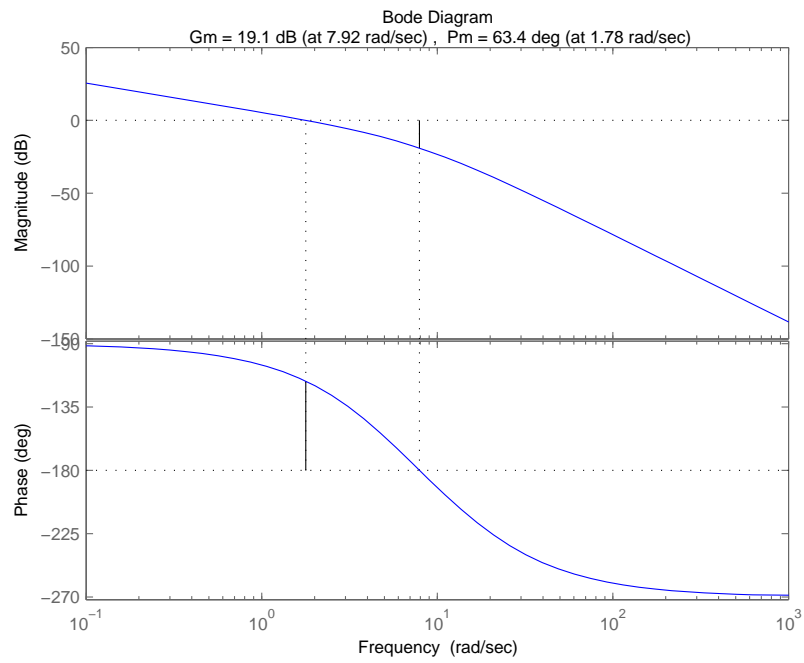


Figure 29: Bode Plot for Large Commercial Jet Aircraft in Type0 System

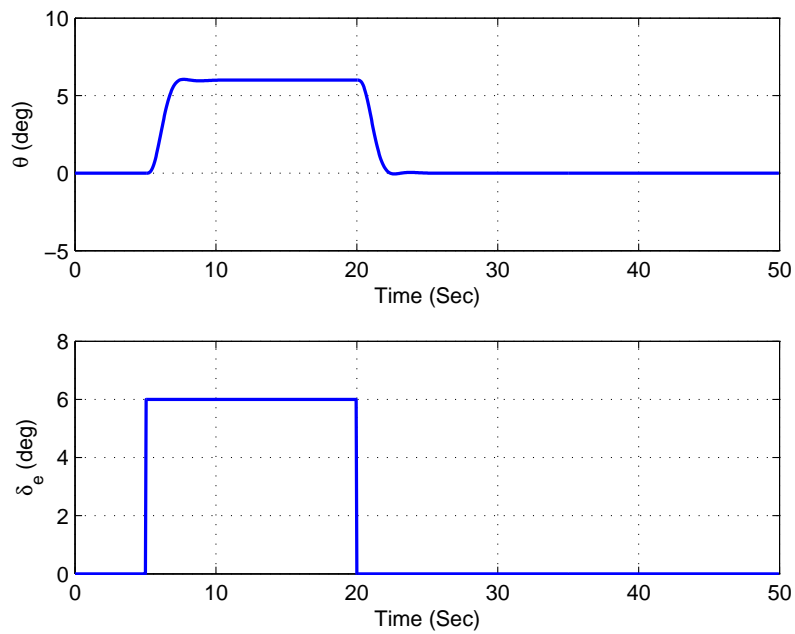


Figure 30: Pitch angle and elevator command respectively for Large Commercial Jet Aircraft in Type1 System

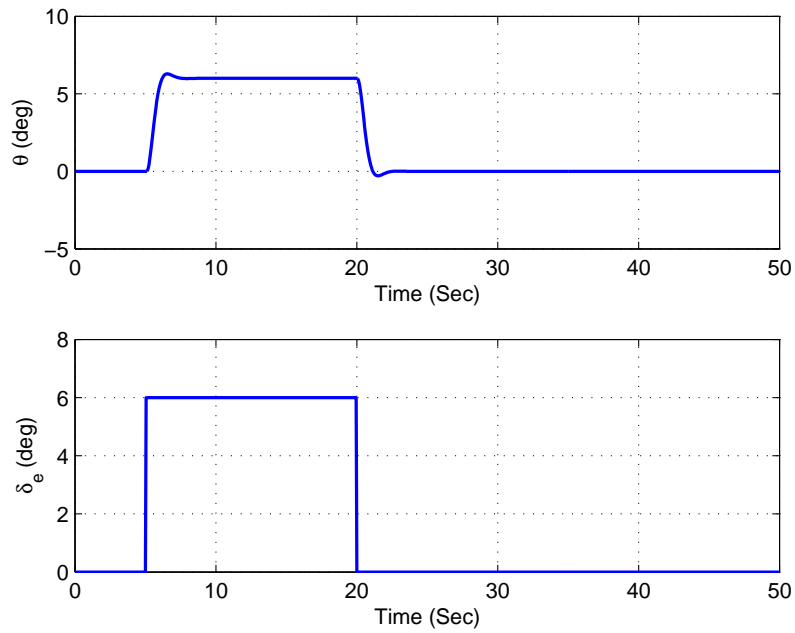


Figure 31: Pitch angle and elevator command respectively for Large Commercial Jet Aircraft in Type0 System

aircraft similar type of response is obtained. Response in large commercial jet is still better than other aircrafts, but still not satisfactory. Every aircraft has its own stability derivatives and mass characteristics, so to control an aircraft its autopilot has to be designed. One's autopilot can not be used in different aircraft. If some of the characteristics are good then there are other factors which might not give proper response. Every aircraft must have its own autopilot.

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Matlab Program

In this section matlab files are presented for all four type of aircrafts. The main file for all of the aircrafts is shown below, where the comments are shown in italics.

Main file for the simulation results

```
clear all;
close all;
clc;
d2r = pi/180;          degree to radian conversion factor
r2d = 180/pi;        radian to degree conversion factor
stabderivatives;
```

Calculating the aircraft transfer function

```
-----
num1 = [-(E*A1-A*E1) -(E*B1-B*E1)];
den1 = [(-A*C1) (C*A1-B*C1-A*D1) (D*A1+C*B1-B*D1) (D*B1)];
thetatf = tf(num1,den1)
amplifiertf = tf([12.5],[1 12.5]); autopilot transfer function
integraltf = tf([1],[1 0]);
```

Calculating the system open loop and closed loop transfer function

```
-----
innerlooptf = thetatf*amplifiertf/(1+Sg*thetatf*amplifiertf);
openlooptftype1 = Sestype1*innerlooptf*integraltf;
outerlooptftype1 = openlooptftype1/(1+openlooptftype1);
openlooptftype0 = Sestype0*thetatf*amplifiertf;
outerlooptftype0 = Sestype0*thetatf*amplifiertf/(1+Sestype0*thetatf*amplifiertf);
[numtype1,dentype1] = tfdata(outerlooptftype1,'v');
[numtype0,dentype0] = tfdata(outerlooptftype0,'v');
[Atf Btf Ctf Dtf] = tf2ss(numtype1,dentype1);
systype1 = ss(Atf,Btf,Ctf,Dtf);
[Atf Btf Ctf Dtf] = tf2ss(numtype0,dentype0);
systype0 = ss(Atf,Btf,Ctf,Dtf);
t0 = 0;
tf = 50;
dt = 0.05;
N = floor((tf - t0)/dt) + 1;
```

```
t(1) = t0;
kdisp = 1;
count = 1;
for k = 1:N-1
if (k == kdisp)
disp(t(k) ='), disp(t(k))
kdisp = kdisp + 100;
end
```

Command input

```
-----
```

```
if(t(k) <=5)
```

```

inputdelta(k) = 0;
elseif(t(k)<=20)
inputdelta(k) = 6*d2r;
else
inputdelta(k) = 0;
end

```

Time update

```

-----
t(k+1) = t(k) + dt;

```

Counter update

```

-----
count = count + 1;
end
tu = t(1:N-1);

```

Plotting the simulation results along with root locus and bode plot

```

-----
figure(1);
rlocus(openlooptftype1)

```

```

figure(2);
rlocus(openlooptftype0)

```

```

figure(3);
margin(openlooptftype1)

```

```

figure(4);
margin(openlooptftype0)

```

```

figure(5);
[y,t,x]= lsim(systype1,inputdelta,tu);
subplot(2,1,1);plot(tu, y*r2d,'LineWidth',1.5);
xlabel('Time (Sec)');
ylabel('theta (deg)');
grid on;
subplot(2,1,2);plot(tu, inputdelta*r2d,'LineWidth',1.5);
xlabel('Time (Sec)');
ylabel('deltae (deg)');
grid on;

```

```

figure(6);
[y,t,x]= lsim(systype0,inputdelta,tu);
subplot(2,1,1);plot(tu, y*r2d,'LineWidth',1.5);
xlabel('Time (Sec)');
ylabel('theta (deg)');
grid on;
subplot(2,1,2);plot(tu, inputdelta*r2d,'LineWidth',1.5);

```

```

xlabel('Time (Sec)');
ylabel('deltae (deg)');
grid on;

```

End of the main file -----

This above main file is same for all the four types of aircrafts, but the function stabderivatives will be different in all the cases. The matlab files stabderivatives for all the four aircrafts are stated below:

General Aviation Aircraft

Matlab program stabderivatives for General Aviation Aircraft

```

VT0 = 176;           Calculating the long. velocity from mach no.
lcratio = 4.89;      ratio of lt and c which is calculated from the geometricl figures
Sg = 1.2;           outer loop gain
Sestype1 = 5.0;      inner loop gain for type1 system
Sestype0 = 100.0;    inner loop gain for type0 system
Stability derivatives
-----

```

```

CL = 0.41;
CD = 0.05;
CLalpha = 4.44;
CDalpha = 0.33;
Cmalpha = -0.673;
CLalphadot = 0;
Cmalphadot = -4.36;
CLq = 3.8;
Cmq = -9.96;
CLM = 0;
CDM = 0;
CmM = 0;
CLdeltae = 0.355;
Cmdeltae = -0.923;

```

Center of gravity and mass characteristics

```

Weight = 2750;
m = Weight/32.2;
IX = 1048;
IY = 3000;
IZ = 3530;
IXZ = 0;

S = 184;
b = 33.4;
c = 5.7;
rho0 = 0.002377;

```

Calculating Stability Derivatives

CZalpha = -CD - CLalpha;
CZalphadot = 0;
CZq = 0;

q = 0.5*rho0*VT0*VT0; *Calculating the dynamic pressure*

Calculating the coefficients attached with alpha, velocity, pitch angle and their derivatives in short period approximation

A = ((m*VT0/(S*q)) - (0.5*c/VT0)*CZalphadot);
B = -CZalpha;
C = (-(m*VT0/(S*q)) - (0.5*c/VT0)*CZq);
D = 0;
E = Cmdeltae/ltratio;
A1 = (-(0.5*c/VT0)*Cmalphadot);
B1 = -Cmalpha;
C1 = IY/(S*q*c);
D1 = -0.5*c*Cmq/VT0;
E1 = Cmdeltae;
End of stabderivatives file for general aviation aircraft

Fighter Aircraft

Matlab program stabderivatives for Fighter Aircraft

VT0 = 286; *Calculating the long. velocity from mach no.*
ltratio = 19.65/9.55; *ratio of lt and c which is calculated from the geometrical figures*
Sg = 0.15; *outer loop gain*
Sestype1 = 0.22; *inner loop gain for type1 system*
Sestype0 = 0.55; *inner loop gain for type0 system*
Stability derivatives

CL = 0.735;
CD = 0.263;
CLalpha = 3.44;
CDalpha = 0.45;
Cmalpha = -0.64;
CLalphadot = 0;
Cmalphadot = -1.6;
CLq = 0;
Cmq = -5.8;
CLM = 0;
CDM = 0;
CmM = 0;
CLdeltae = 0.68;

Cmdeltae = -1.46;

Center of gravity and mass characteristics

Weight = 16300;
m = Weight/32.2;
IX = 3549;
IY = 58611;
IZ = 59669;
IXZ = 0;

S = 196.1;
b = 21.94;
c = 9.55;
rho0 = 0.002377;

Calculating Stability Derivatives

CZalpha = -CD - CLalpha;
CZalphadot = 0;
CZq = 0;

q = 0.5*rho0*VT0*VT0; *Calculating the dynamic pressure*

Calculating the coefficients attached with alpha, velocity, pitch angle and their derivatives in short period approximation

A = ((m*VT0/(S*q)) - (0.5*c/VT0)*CZalphadot);
B = -CZalpha;
C = (-(m*VT0/(S*q)) - (0.5*c/VT0)*CZq);
D = 0;
E = Cmdeltae/ltratio;
A1 = (-(0.5*c/VT0)*Cmalphadot);
B1 = -Cmalpha;
C1 = IY/(S*q*c);
D1 = -0.5*c*Cmq/VT0;
E1 = Cmdeltae;

End of stabderivatives file for fighter aircraft

Jet Aircraft

Matlab program stabderivatives for Jet Aircraft

VT0 = 223; *Calculating the long. velocity from mach no.*
ltratio = 22.4/10.93; *ratio of lt and c which is calculated from the geometrical figures*
Sg = 1.0; *outer loop gain*
Sestype1 = 1.0; *inner loop gain for type1 system*
Sestype0 = 1.5; *inner loop gain for type0 system*

Stability derivatives

CL = 0.737;
CD = 0.095;
CLalpha = 5.0;
CDalpha = 0.75;
Cmalph = -0.80;
CLalphadot = 0;
Cmalphadot = -3.0;
CLq = 0;
Cmq = -8.0;
CLM = 0;
CDM = 0;
CmM = -0.05;
CLdeltae = 0.4;
Cmdeltae = -0.81;

Center of gravity and mass characteristics

Weight = 38200;
m = Weight/32.2;
IX = 118773;
IY = 135869;
IZ = 243504;
IXZ = 5061;

S = 542.5;
b = 53.75;
c = 10.93;
rho0 = 0.002377;

Calculating Stability Derivatives

CZalpha = -CD - CLalpha;
CZalphadot = 0;
CZq = 0;

$q = 0.5 \cdot \rho_0 \cdot V_{T0} \cdot V_{T0}$; *Calculating the dynamic pressure*

Calculating the coefficients attached with alpha, velocity, pitch angle and their derivatives in short period approximation

A = $((m \cdot V_{T0} / (S \cdot q)) - (0.5 \cdot c / V_{T0}) \cdot CZalphadot)$;
B = -CZalpha;
C = $-(m \cdot V_{T0} / (S \cdot q)) - (0.5 \cdot c / V_{T0}) \cdot CZq$;
D = 0;
E = Cmdeltae/ltratio;

```

A1 = -(0.5*c/VT0)*Cmalphadot);
B1 = -Cmalpha;
C1 = IY/(S*q*c);
D1 = -0.5*c*Cmq/VT0;
E1 = Cmdeltae;
End of stabderivatives file for jet aircraft
-----

```

Large Commercial Jet Aircraft

Matlab program stabderivatives for Large Commercial Jet Aircraft

```

VT0 = 278;           Calculating the long. velocity from mach no.
lcratio = 98/27.31;  ratio of lt and c which is calculated from the geometrical figures
Sg = 2.0;           outer loop gain
Sestype1 = 1.7;     inner loop gain for type1 system
Sestype0 = 1.8;     inner loop gain for type0 system
Stability derivatives
-----

```

```

CL = 1.11;
CD = 0.102;
CLalpha = 5.70;
CDalpha = 0.66;
Cmalpha = -1.26;
CLalphadot = 6.7;
Cmalphadot = -3.2;
CLq = 5.4;
Cmq = -20.8;
CLM = 0.81;
CDM = 0.0;
CmM = -0.27;
CLdeltae = 0.338;
Cmdeltae = -1.34;

```

Center of gravity and mass characteristics

```

Weight = 636600;
m = Weight/32.2;
IX = 18200000;
IY = 33100000;
IZ = 49700000;
IXZ = 970000;

```

```

S = 55000;
b = 195.68;
c = 27.31;
rho0 = 0.002377;

```

Calculating Stability Derivatives

CZalpha = -CD - CLalpha;
CZalphadot = 0;
CZq = 0;

q = 0.5*rho0*VT0*VT0; *Calculating the dynamic pressure*

Calculating the coefficients attached with alpha, velocity, pitch angle and their derivatives in short period approximation

A = ((m*VT0/(S*q)) - (0.5*c/VT0)*CZalphadot);
B = -CZalpha;
C = (-(m*VT0/(S*q)) - (0.5*c/VT0)*CZq);
D = 0;
E = Cmdeltae/ltratio;
A1 = (-(0.5*c/VT0)*Cmalphadot);
B1 = -Cmalpha;
C1 = IY/(S*q*c);
D1 = -0.5*c*Cmq/VT0;
E1 = Cmdeltae;

End of stabderivatives file for largecommercialjet aircraft
