# ADAPTIVE CONTROL OF AIRCRAFT WITH ACTUATOR FAILURES

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بسم الله الرحمن الرحيم

In the name of Allah, the most Merciful and the most Beneficent

# ABSTRACT

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Actuator failures in dynamic systems result in worse performance as they are uncertain in failure time and pattern. For example, many aircraft accidents can occur due to failures in the operation of actuators, like aileron, elevator, and rudder. A Direct adaptive model reference controller (D-MRAC) using Lyapunov theory is designed for an aircraft flight control system (FCS) to guarantee good performance despite the actuator's performance degradation or failures. The Direct MRAC control architecture is applied to the Boeing 747 aircraft in longitudinal maneuver. Numerical simulations are carried out to validate suggested control method performance for aircraft due to actuator lock in place, loss of actuator effectiveness and actuator hard over and hard under. The simulations are also carried out with Linear Quadratic Regulator (LQR) controller and it is seen that direct model reference adaptive controller (D-MRAC) successfully overcomes the actuator fault as compared to the non-adaptive, optimal LQR controller.

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*Dedicated To my Teachers and Parents*

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| | |
|---|---|
| $q$ | pitch rate (deg/s) |
| $\Theta$ | pitch angle (degree) |
| $\alpha$ | angle of attack (degree) |
| $V_T$ | airspeed (m/s) |
| $h$ | altitude (m) |
| $x_e$ | distance along the x-air path (m) |
| $y_e$ | distance along the y-air path (m) |
| $\delta_e$ | the elevator control input |
| $\delta_{th}$ | throttle control input |
| $\delta_s$ | stabilator control input |
| $q$ | dynamic pressure ($N/m^2$) |
| $S$ | reference surface area ($m^2$) |
| $m$ | mass of aircraft (kg) |
| $b$ | wing span (m) |
| $c$ | wing chord (m) |
| $G$ | gravity force (N) |
| $T$ | thrust force (N) |
| $D$ | drag force (N) |
| $L$ | lift force (N) |
| $Y$ | side force (N) |
| $l_{aero}$ | landing gear effects and aerodynamic loads |
| $\bar{L}$ | roll momemt (Nm) |
| $\bar{M}$ | Pitch moment (Nm) |

| | |
|---|---|
| $N$ | Yaw moment (Nm) |
| $K_{\alpha}$ | Effectiveness factor for the elevator and horizontal stabilizer |
| $c_7$ | Inertial coefficient |
| $I_{yy}$ | Mass and prodect of inertia (kgm$^2$) |
| $z_{eng}$ | Thrust component |
| $C_L$ | Lift coefficient (nondim.) |
| $C_m$ | Pitch moment coefficient (nondim.) |
| $C_D$ | Drag coefficient (nondim.) |
| $C_Y$ | Side force coefficient (nondim.) |
| $C_l$ | Rolling moment coefficient (nondim.) |
| $C_n$ | Yawing moment coefficient (nondim.) |
| $x_{c.g}$ | Centre of gravity in x axis |
| $z_{c.g.}$ | Centre of gravity in z axis |
| $d_{CL}/ d_q$ | Variation in basic lift Coefficient with pitch |
| $d_{CL}/ d\delta_e$ | Variation in basic lift coefficient with elevator angle |
| $d_{CL}/ d\delta_s$ | Variation of lift coefficient with stabilator angle |
| $d_{Cm}/ d_q$ | Variation in pitching moment coefficient with pitch |
| $d_{Cm} /d_{\delta e}$ | Variation in pitching moment coefficient with elevator angle |
| $d_{Cm}/ d\delta_s$ | Variation in pitching moment coefficient with stabilator angle $C_{DMach}$ drag coefficient at mach M |
| $C_{mbasic}$ | Basic pitching moment at zero angle of stabilizer for rigid plane with free landing gear retracted and in free air. |
| $C_{Lbasic}$ | Basic lift coefficient at zero angle of stabilizer for rigid air plane with free landing gear retracted and in free air. |

# CHAPTER 1

# INTRODUCTION

Fatal aircraft accidents is root of terrible life loss are escorted by prodigious money thrashing associated with cost of investigation, property demolition, and cause misbelief of people in traveling in airplane. Several authoritarian agencies are shoving to create technologies to extensively trim down the critical rate of aviation accident, that is currently slighter less than 2 per million departures making air journey the reliable means of transportation. In many cases, scarce airport services and individual errors are the main cause for these accidents by exploratory agencies. However, the latest several aviation accidents occurred due components malfunctioned in the control machinery. Actuators and sensors failure such as horizontal stabilizers lock in place, dysfunctional gyros, or other actuator surfaces have led to disastrous aviation accidents. For that reason, to address such failures, there is a need to design reconfigurable control mechanism in control loop. The classical controller fails to perform adequately due to variations of dynamics of system dynamics resulting in a aviation failure. When actuators stop working, they not only cause diminution of control power, but also may cause importunate turbulence, for which the compensation of serviceable actuators is needed. The mainly familiar failure of actuator occurs when actuator becomes lock in place at any position due to mechanical, hydraulic or electrical failure. A float failure causes the control surface to moves freely. One of the most disastrous failure condition occur due to actuator runaway/hardover. In a runaway situation, the failure causes the actuator surface to move at its maximum boundary rate due to overpower of actuator until it reaches its blowdown limit or maximum position limit. For example, a rudder hard over coccurs due to an electronic module malfunction making a hefty signal conveyed to actuators making rudder deflection reaching its maximum boundry of defelction. Another type of failue is loss of part/full control surface due to which the effectivenes of control surface is reduced. Due to inability of nominal controller to sensible to  variations such as envirnmnetal changes, change in dynamics, some online adaptive mechanism is needed to account for change in dynamics and thus providing adequate handling qualities despite actuator's performance degradation or failures.

## 1.1 Motivation

In 1959, the air carriers of world reached average of 100,000 plane hours of flying over loss of hull; today the average approaches approximately 800,000 hours of flying over loss of hull [1]. The record globally changes; eventhen , tavelling through air is considered securest transportation among all the foremost types of transport classes. However, the current rate of aviation accident (approximately under 2/million), is anticipated to increase three times in next 20 years [4] will be soon be ungratifying due to surmised increase in the commercial air traffic. Several agencies are making efforts to significant reduce the aviation disasters in arriving years [2,3,5]. e.g. the NASA Aviation Safety Program chief aim is to "*construct and formulate technologies which impart to trim down rate of aviation disasters* [6]." Aviation accidents main causes are due to carelessness of flight crew, stumble in maintenance, the airworthiness of the aircraft, adverse environmental conditions [1]. These diverse circumstances contribute challenges for technology to compensate. Nowadays aircraft comprises a few million parts, e.g. A Boeing 767 has approximately 3,140,000 parts. And the likelihood of malfunctioning of each of these parts is large.  The major causes of accidents are due to lapses in the power of system, failures of apparatus and some accidents are listed . Figure 1.1 tells rate of aircraft disasters from past years from 1950s. The reduction in rate of aviation disasters has made the air transportation realiable mode of transport. Disastrous aviation accidents cause extreme property ruin and life and become the headlines of newspaper all around. It takes a long time, effort and money for authoritative agencies to figure out reason of the disaster [5]. Air plane transformations have now being increased worldwide with the decrease of aviation accident rates as viewed in Figure 1.2 [1]. Figure 1.3 presents data inference from Figure 1.1 and Figure 1.2 for past years employing bland polynomial mapping. This figure verifies a powerful increment in aviation disasters in arriving years.

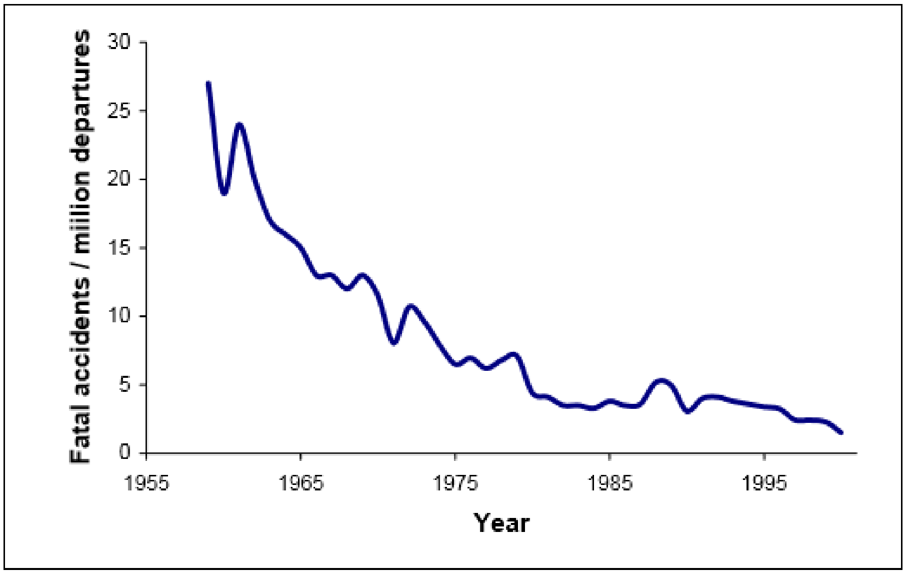


**Figure 1.1 Fatal aviation accident rate**

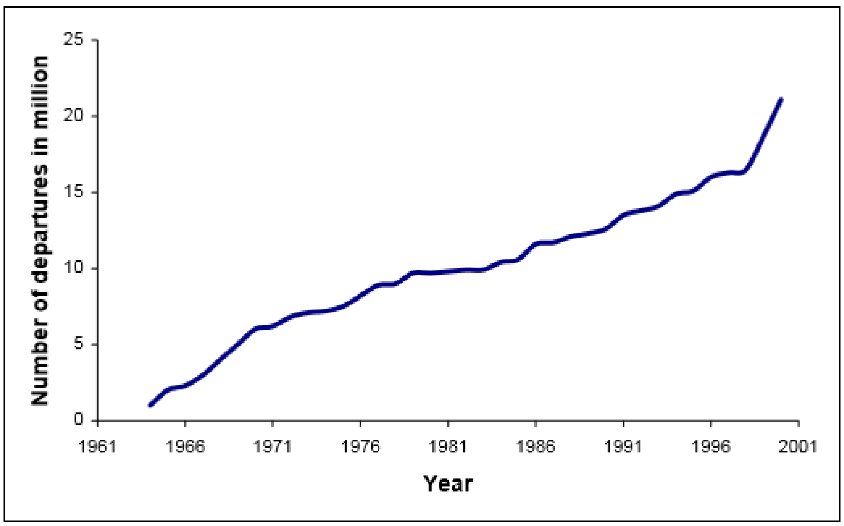


**Figure 1.2 Commercial worldwide growth of air traffic**

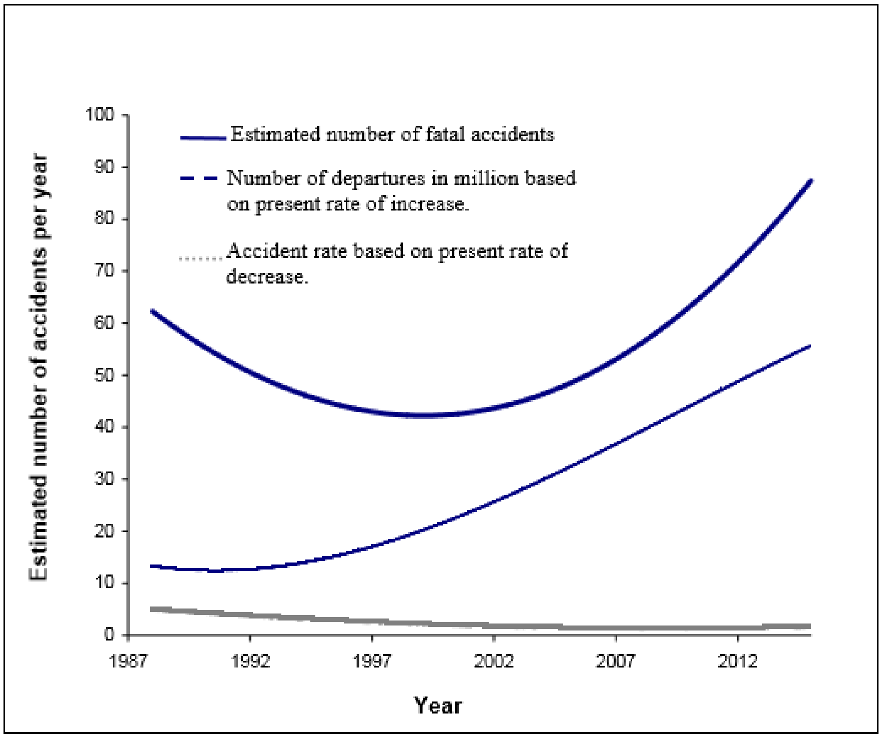


**Figure 1.3 Predicted increment of air travel and fatal aviation accidents in arriving years**

To make already safe mode of transport even safer is a significant challenge for technology. It requires months of investigation to figure out correct series of actions leading to serious aviation accident for part of the authorities. Sometimes these investigations are proven to be inconclusive and become just a riddle as what got erroneous to aircraft. Some topical examples of serious accidents are presented due to lapses in machinery of control.

1. The Alaska airline flight 261 – Jan. 2000: On January 31, the McDonnell Douglas MD-83 of the Alaska Airlines on flight 261 crashed at California about 4:20 p.m. (PST)

which was settting from Puerto Vallarta, Mexico to San Francisco, killed its eighty three people riding in airplane and all the five team members . Although, the investigation is still carried on, it shows that due to dysfunctional stabilizer, the plane memners could not manage vertical power. According to FDR, the trim of stabilizer got down trim to a full-nose and got stuck there till the plane crashed. In final tweleve minutes, the members tried to determine and debug problems of stabilizer trim but all their efforts got in vain and the MD-83 crashed deep in water just off Point Mugu, CA, 650 ft. With recovery of parts from sea with help of navy, the lapse in actuator's assemblage was verified in the ending minutes. It can thus be hypothesized that the primary reason of tragic aviation accident is due to lapse in actuation assembly of stabilizer due to its mechanical failure [7].

2. Japan Airlines (JAL) ,on 12 August, 1985 flight set off from Tokyo-Haneda at 18.12h to settle in Osaka. At 18.24h, through 23900ft with of 300kts, an abnormal vives raised a striking force causing aircraft's nose to raise which causes the pressure of hydraulic fallen and making the ailerons, elevators and yaw damper to be failed, followed by abnormal deviation of altitude and variation in speed without changing of angle of attack causing the dutch roll and plughoid oscillations. The aircraft descend below 6600ft while the crew tried to regain control with aid of engine thrust. The airspeed had fallen to 108kts upon reaching 6600ft. The airplane mount to attack to a maximum of approx. 13400ft with a 39deg. angle of and started to descend again. AL123 bursted into flames with brushe against a ridge covered with trees, stroyed again another ridge causing the

crash. The probable cause for this aviation accident was cited as: "Souring in characteristics of flight and lapse in primary power because of violation of the bulkhead of aft pressure due to left following lapse in tail, vertical fin and hydraulic power systems of flight. The repairs of the bulkhead was carried out false in 1978, form cause of propagation of the fatigue cracks and since the brags of weariness didn't come up in the later examination, this contributed to lapse." Five hundred and twenty people died. This accident was the worst aircraft disasters among all other accidents [3].

3. On January 10, 2000, a Saab 340B flight (number 498) set from Zurich-Kloten to settle in Dresden. Due to improper right aileron input soon after take off increased in rate of roll. Soon pitch quickly drop; causing the large increase in speed making the airplane to enter a very high speed which was not possible to recover causing decrease in rate of spiral making the airplane to crash killing all seven peoples on airplane and 3 memebers of airplane [5].

Ubove listing reprents the disasterous aviation accidents. It is clear from listing that above lapses are not restrained to special manufacturer of airplane or origin of world. They demand to design schemes of control that covers such failures scenerios. These devious plans are crucial and weighty regarding security issues of the people and crewmemebers. The coming portion addresses several operations of Adaptive Flight control design in such failure scenerios and a brief prospect for design techniques of control in the literature.

## 1.2 Background and literature review

Actuators may stop working during middle of system operation, and actuator behavior is vague because it is not known when an actuator will stop working and the nature and the instant of failure is unknown. [2]. The solution is to design control law to guarantee adequate handling qualities despite the actuator lapses. Adaptive control has become an efficient subroutine for dealing with systems with unconvictions. The Traditional approach is to design flight control systems utilizing models of mathematical sytem of the craft which are linearized at trim conditions, having gains or regulator parameters varied with working conditions during flight To ensure permanence and concert of consequential controllers which are gain schedeled, systematic structure were developed including practice like linear- parameter-varying (LPV) control [4], [7], and [8]. Dynamic inversion using nonlinear design techniques have been used in [1], [10], whereas hybrid technique with the purpose of wraping nonlinear adaptive neural network with model inversion control is described in [11] for robust design, and an adaptive design based on nonlinear approach employing back-stepping and neural networks is mentioned in [6]. A time-scale partition design between controller and system dynamics which is based on RBF-NN adaptive law, having applications in control of both lateral (regulation of the roll and sideslip angles) and longitudinal (angle-of-attack command systems) as well as is described in [12]. A epigrammatic “diligence viewpoint” on direct devise of flight, inclusing the practices like LPV control, robust control ($H\infty$, μ-synthesis), dynamic inversion, adaptive control, neural networks, and more, can be found in [3]. Our interest is in the design of Direct MRAC for flight dynamics of B-747 aircraft on longitudinal maenvers with actuator failures.

## 1.3 Organization of the thesis

The remaing sections of thesis is characterized in following way. In surving snippet of chapter, incentive for presented research is accompanied by snippet on history of adaptive flight control following visit of the already available methods to achieve control under faulty conditions of actuators. The working of this thesis is explained in the final snippet. In chapter 2, the aircraft dynamics describing complete longitudinal dynamics of Boeing 747 100/200 is presented. In chapter 3, a brief discussion of different types of failures is given. In chapter 4, design of LQR controller with actuator failure is offered. The LQR controller is designed using linear model at trim flight conditions. In chapter 4, we consider the design of controller for actuators lapses for linear systems. The different types of actuator failures are tested using D-MRAC for nonlinear Boeing 747 100/200 longitudinal maenver model aircraft in chapter 5. All the types of failures are easily accommodated using D-MRAC. Chapter 6 wind up the work and points suggestions for contribution to progress furthur. Through out thesis, all the reckoning are carried out along with the definitions, theorems and their proofs for augumentation of legability and the utility.

# CHAPTER 2

# AIRCRAFT DYNAMICS

This chapter presents the aircraft dynamics. In Section 2.1, model being 747 100/200 is discussed in detail. The general nonlinear equations of motion (EOM) are derived and formulated and introduces the full nonlinear equations that are included in the model. The actuators are discussed in Section 2.2. Section 2.3 covers the summary of this chapter.

## 2.1 Aircraft Modeling

In this chapter, the general six degrees of freedom (6DoF) nonlinear EOM are introduced. Boeing 747-100/200 model is used throughout the thesis as simulation test bed.

### 2.1.1 Boeing 747 100/200

The aircraft model used in this thesis is Boeing 747-100/200. The aircraft was chosen owing to its array of distinctiveness (primary and secondry edge flaps, spoilers, range of actuators, four turbo fan jet engines) makes it ideal category among business aircraft these days

The Boeing 747 is an large-scale transportation having 4 turbo fan jet engines premeditated to function internationally. a number of its presentation specifications are a range of 6,000 nautical miles, a cruise velocity larger than 965 kilometer/hour and a design top limit of 13,716 meters.

The B747 is operational with slotted trailing flaps which are three in number and Krueger style primary edge flaps to acquire neccassary high lift at low speed flight. The inboard Krueger flaps for Boeing Aircraft are unslotted. On the other hand, outboard of the inboard nacelle are made to be uneven cambered and slotted. The main landing gear is assembled through a pair of wing which is mounted on four-wheel trucks, those in the stiff situated a little left of the wing [17].


**Figure 2.1 The Boeing 747 aircraft**

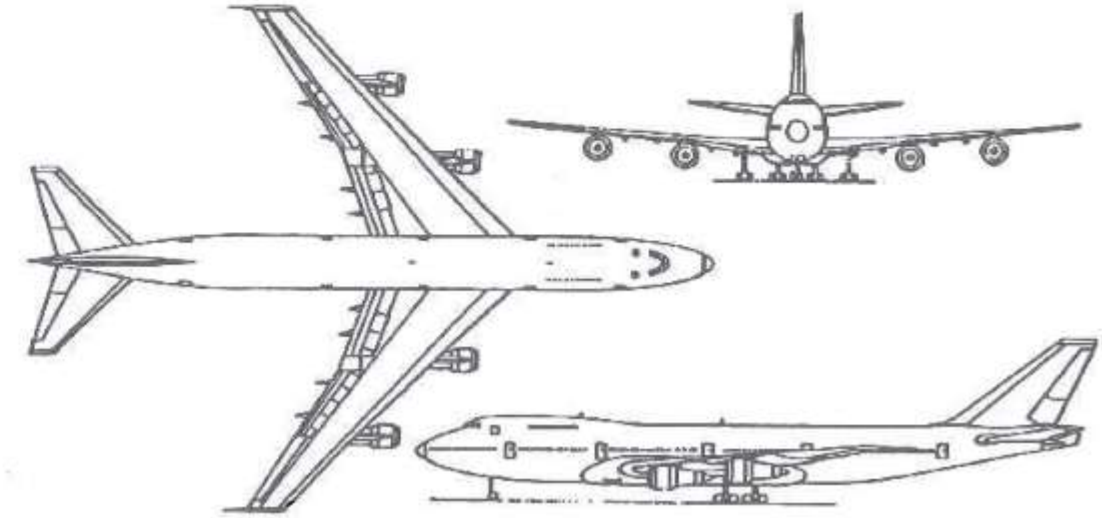

**Figure 2.2 Top, Front Side views of the Boeing 747**

A moveable horizontal stabilizer equipped with four elevator segments provides the longitudinal control. Pitch trim is provided by horizontal stabilizer, and under normal operating conditions, inboard and outboard elevators defelct mutually. An arrangement of two pairs of inboard and outboard flaps and ten spoiler panels and are employed for lateral control. The outboard ailerons are only used when flaps are down, i.e.set to a non zero deflection. The ten spoilers operate as speedbrakes when used symmetrically and other two spoiler panels are used during ground operations.

### 2.1.2 The Non-Linear General Equation of Motion (EOM)

An accurate reprentataion of the aircraft dynamics can be obtained through nonlinear, rigid body equations (note, that in this case the fexible modes are ignored). Referenes [18],[20],[19] have complete derivations of the rigid body equations for an airplane.These equations consider manipulatation of forces and moments acting upon the moving aircraft externally ,which usually results from the interaction of gravity, G, thrust, T, landing gear effects and aerodynamic loads, $l_{aero}$,(x;y;z are a evaluation of the likely moment arms that may exist).

$$F=l_{aero}+T+G \qquad (2.1.1)$$

$$M = l_{aero}.x + T.y + gear.z \qquad (2.1.2)$$

The forces and moments are determined according to dimentionless smooth coefficients ($C_D$ $C_L$ $C_Y$ $C_i$ $C_m$ $C_n$ ), dynamic pressure of flight ,q, area of refrence ,S, and moments (if the case) ,the momemt arm(either wing chord c,or wing span,b)

$$D = \bar{q} S C_D \qquad \bar{L} = \bar{q}\ S$$

(2.1.3)

$$L = \bar{q} S C_L \qquad \bar{M} = \bar{q} S \bar{c} C_m$$

(2.1.4)

$$Y = \bar{q} S C_Y \qquad N = \bar{q} S b C_n \qquad (2.1.5)$$

D denoted the force due to drag ,L the force due to lift ,Y the force due to side and $\bar{L}$ ,

$\bar{M}$ ,N the momemts of roll,pitch and yaw respectively.

The aerodynamic coefficients are referred to the stability axes.

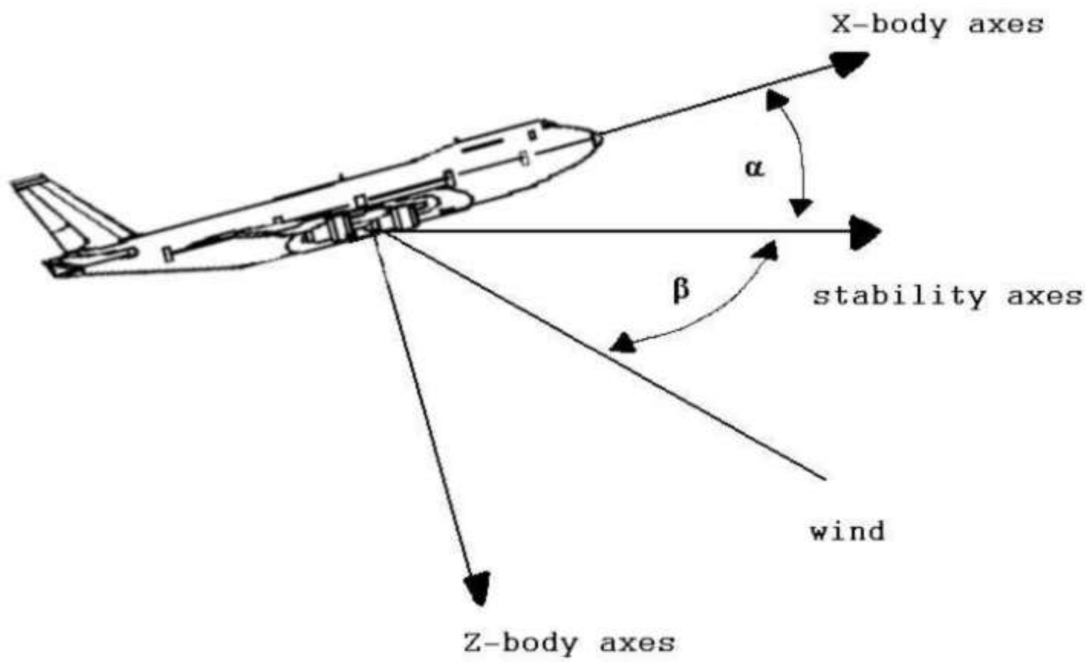


**Figure 2.3 wind, Body and Stability axis**

It is common in the aircraft community to utilize different coordinate axis for each component (aerodynamic coefficient, gravity,thrust,…..) based on the most appropriate frame for their defination and/or calulation. Before introducing the aerodynamic coefficients and/or any other component into the nonlinear equations of motion, all are firstly transformed into a common reference frame. All these transformation are carried out performing rotations with respet to three different axes by Euler angles (see

Referenes [21],[18],[20],[19]. In this thesis, the body-axes is selected as the common frame of reference . See Figure 2.2 for a graphical description of the relationship between the different coordinate frames.

The nonlinear equation of motion (EoM) or degree of freedom form the state vector which in the present case, complete nonlinear model, has 12 components corresponding to force $(\dot{\alpha}, \dot{\beta}, \dot{V}_{TAS})$, moment ( $\dot{p}$ , $\dot{q}$ , $\dot{r}$ ), kinematic $(\dot{\varphi}, \dot{\theta}, \dot{\Psi})$ and navigation equations $(\dot{h}_e, \dot{x}_e, \dot{y}_e)$. It is assumed there are no wind components in the equations presented in the main body of the thesis.

### 2.1.3 Nonlinear longitudinal model

In this section, making use of reduced order aerodynamic coefficients, the longitudinal nonlinear model of Boeing 747 is presented in detail.

The longitudinal motion of an aircraft can be defined by the following six equations: angle of attack, α, pitch rate, q,  pitch angle, $\theta$ , true air speed, $V_t$, altitude, he, and distance along the x-air path, xe (recall Earth-reference-frame).The last state is utilzed in measuring the distance the aircraft travels along the x-earth axis, though it does not have any impact on the dynamis of the airplane. The complete nonlinear equations are thus redued down to six DoF.

In Longitudinal motion, side slip angle, roll angle, roll rate nad yaw rate are considered to be zero,the 6 body axis non linear longitudinal motion of Boeing 747 are given by from reference [22].

$$\dot{\alpha} = \left[1 - \frac{\bar{q}Sc}{2mV_t^2}\left(1.45 - 1.8\,x_{c.g}\right)\frac{d_{CL}}{d_q}\right]q + \left[-\frac{\bar{q}S}{mVK_\alpha}\frac{d_{CL}}{d\delta_e}\right]\delta_e + \left[\frac{-4}{mV}\left(\sin\bar{\alpha} + 0.0436\cos\bar{\alpha}\right)\right]T + A$$

(2.1.6)

$$A = \frac{1}{V}\left(\sin\alpha\sin\alpha + \cos\alpha\cos\alpha\right)g - \frac{\bar{q}S}{mVC_{Lbasic}}$$

$$\dot{q} = \frac{c_7\bar{q}S\mathbf{c}^2}{2V_t}\left[\frac{d_{Cm}}{d_q} - \frac{1}{c}\left(1.45 - 1.8x_{c.g.}\right)\frac{d_{CL}}{d_q}\left(\cos\bar{\alpha}\,x_{c.g.} + \sin\bar{\alpha}\,z_{c.g.}\right)\right]q$$
$$+c_7\bar{q}ScK_\alpha\left[\frac{d_{Cm}}{d_{\delta e}} - \frac{1}{c}\frac{d_{CL}}{d_{\delta e}}\left(\cos\bar{\alpha}\,x_{c.g.} + \sin\bar{\alpha}\,z_{c.g.}\right)\right]\delta_e + c_7\bar{q}ScK_\alpha\frac{d_{Cm}}{d_{\delta s}}\delta_s + c_7 zengT + Q$$

(2.1.7)

$$Q = c_7\bar{q}ScC_{mbasic} + c_7\bar{q}S[C_{DMach}\left(\cos\alpha\,z_{c.g.} - \sin\alpha.x_{c.g.}\right) - C_{Lbasic}\left(\cos\alpha x_{c.g.} + \sin\alpha\,z_{c.g.}\right)]$$

$$\dot{V}t = 4\,m\left(\cos\bar{\alpha} - 0.0436\sin\bar{\alpha}\right)T + V \qquad (2.1.8)$$

$$V = \left(\sin\alpha\cos\theta - \cos\alpha\sin\theta\right)g - \frac{\bar{q}S}{mC_{DMach}}$$

$$\dot{\theta} = q \qquad (2.1.9)$$

$$\dot{h} = \left(\cos\alpha\sin\theta - \sin\alpha\cos\theta\right)Vt \qquad (2.1.10)$$

Where q is pitch rate (m/sec), Θ is pitch angle (deg), α is angle of attack (deg), $V_t$ is true airspeed (m/sec), h is altitude (m/s). Also, $\delta_e$ , $\delta_s$ and T are control inputs known as the elevator,stabilator and thrust. $\bar{q}$ is dynamic pressure ($N/m^2$), S is reference surface area

($m^2$), m is the mass of aircraft (kg), c is wing chord (m), $K_\alpha$ (effectiveness factor for the elevator and horizontal stabilizer), $c_7=1/I_{yy}$ (inertial coefficient), $z_{eng}$ (thrust component ), $C_L$ (lift coefficient ), $C_m$ (pitch moment coefficient), T thrust force (N), $x_{c.g.}$, $z_{c.g.}$centre of gravity in x, z axis. $d_{CL}/ d_q$, $d_{CL}/ d\delta_e$ ,$d_{CL}/ d\delta_s$ are variation in basic lift coefficient with change in pitch, elevator angle, and stabilator angle respectively. $d_{Cm}/ d_q$, $d_{Cm} /d_{\delta e}$ , $d_{Cm}/ d\delta_s$ represent the variation of pitching moment coefficient with pitch , elevator angle and stabilator angle respectively. $C_{DMach}$ is drag coefficient at mach m, $C_{mbasic,}$ $C_{Lbasic}$ are basic pitching moment coefficient and basic lift coefficient at angle of stabilizer at zero for rigid air craft and in free air. Note that the coefficients of aerodynamic and their derivatives are considered with look-up tables described in Ref. [22].

## 2.2 Actuator Distribution

The conventional aircraft has generally seven physical actuators which can be used to control the aircraft. These physical actuators can be mixed together to create the four virtual actuators: the throttle $\delta_{th}$, providing propulsion, the aileron $\delta_A$, controlling the rolling motions, the elevator $\delta_e$, controlling the pitching motions, and the rudder $\delta_R$, controlling the yawing motions. In Boeing aircraft, Ailerons are controlled through hydraulic systems A and/or B. If both hydraulic systems fail, both control wheels provide manual reversion. If the aileron system becomes stuck at a point, the spoiler can then be moved through the co-pilots wheel (hydraulically). Balance tabs and balance panels available on both ailerons. Flight spoilers augment the ailerons and are functioned through hydraulic system A (inboard) & B (outboard). The Spoilers will continue to function with speedbrake deployed. Hydraulic system A also consist of ground spoilers. Hydraulic system B only powers the outboard flight spoilers. On landing, if armed, all spoilers will be deployed when the thrust levers are at idle and right gear is compressed or any two wheels have spun up. If not armed, reverse thrust is selected with the speedbrakes. The rudder providing yaw is deflected through a PCU which is controlled through hydraulic system A and/or B. Pitch is provided by elevators. The elevators are deflected through control column by hydraulic systems A and/or B. If both hydraulic systems fail to operate, both control columns provide manual reversion. If the elevator system becomes stuck, the stabilizer (trim) is still available to be used. Balance tabs and balance panels are also available on both elevators.

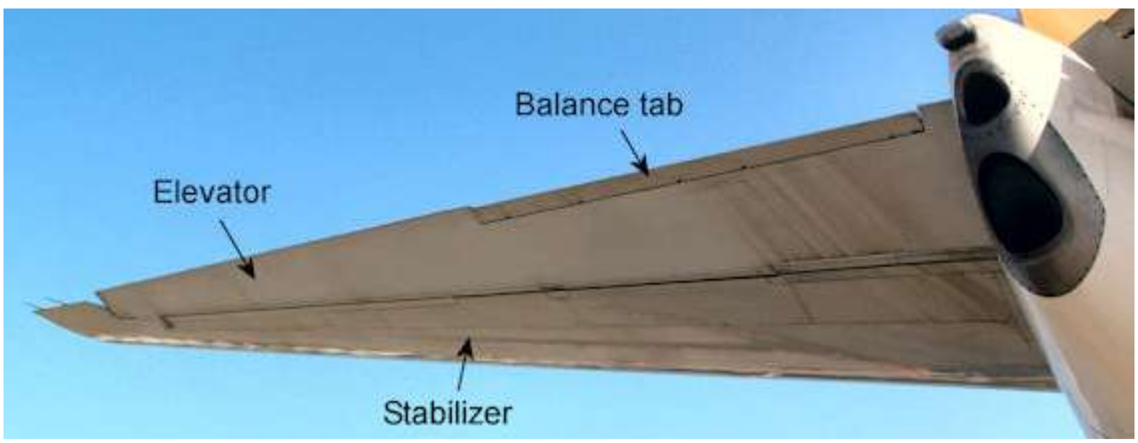


**Figure 2.4 Actuators Configuration for pitch**

## 2.3 Summary

This chapter describes complete description of Boeing 747 100/200 covering all its distinctions making it perfect model for selection in simulation bed for flight control applications. A complete nonlinear model of Boeing 747 100/200 and neccassary aerodynamic coefficients are presented in detail.

# CHAPTER 3

# FAULTS AND FAILURES

Fault is an undesired and unwanted digression of at least one constraint or attribute of the system from the tolerable/natural/customary conduct. Failure is a lasting digression from the specified conditions to perform a required function. So, failure is a stipulation which brutal than a fault. With Occurance of fault in an actuator, actuator is useable and acceptable still but become less effective and lead to slower response. But in case of failure occurance, we need a separate actuator or some control mechanism to turn out the preferred behavior.

## 3.1 Types of failure

Aircraft failures normally occur due to three major reasons: sensor, actuator and structural failures, or some mishmash of these. Table 3.1 below describes the general types of failures with their categories.

### 3.1.1 Sensor Failures

These types of failures normally occur in big airplanes and jets. The likelihood of such failure is extremely small in big airplanes, and passenger jets due to triple surplus sensors. A sensor failure essentially doesn’t contribute a menace to a big airplane due to set of backup sensors with a pilot on board. Sensor failures become noticeable in autonomous craft [24]. However, a failure in sensor can be compensated utilizing the remaining working sensors by making assumption that the the aircraft aerodynamics had not varied. So, failures due to sensors are not of considerable notice and thus not included in this research work

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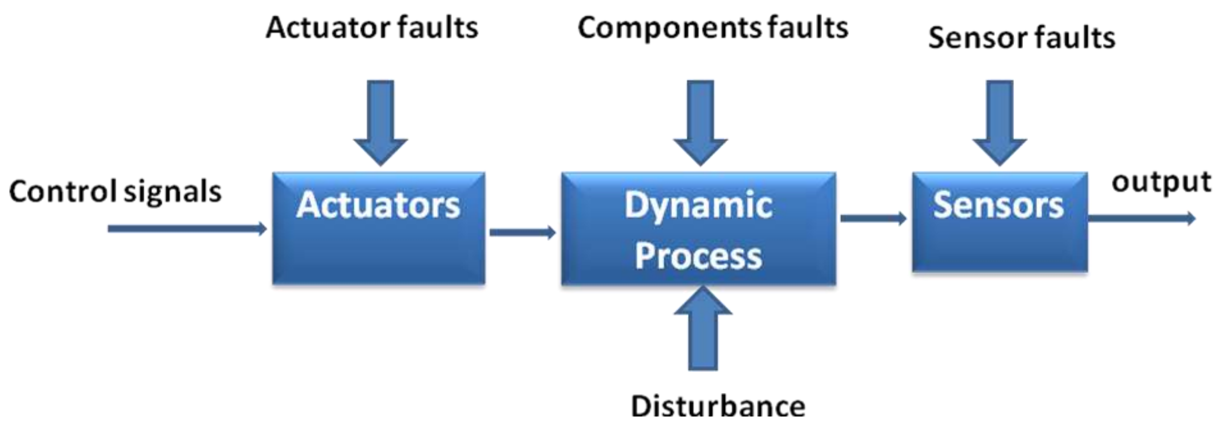


**Figure 3.1 Structure of faults**

### 3.1.2 Structural Failures

Physical damage cause the structural or aerodynamic failures and they cause change in equations of motion. The model structure remains the same during such a failure, due to general nature of equations of motion. However, the constants which govern the behavior of equations of motion can change [24]. Physical injure of airplane cause change to service circumstances of airplane because of variations in the centre of gravity or aerodynamic coefficients. This contribute to variations in the system's dynamics. Such failures examples of injure to structure are detachments of the aircraft body parts or wing battle damage [25], or e.g., the vertical fin/stabiliser (Flight 123, B-747, Japan, 1985) [26, 28] and (flight587, A300, New York, 2001) [26], wing (DHL A300B4, A300, Baghdad,

2003) [26], fuselage skin or cargo doors (flight 981, DC-10, Paris, 1974) [26]. control surfaces detachment, for example the rudder (flight 961, A310, Varadero, Cuba, 2005) [27] or engines (flight 1862, B-747, Amsterdam, 1992) [29].

| Sensor | Actuator | structural | failure | effect |
|---|---|---|---|---|
| ✓ | | | Sensor failure | insignificant in case of only malfunction |
| | ✓ | | fractional hydraulics failure | decline of maximum rate on numerous actuators |
| | ✓ | | hydraulics failre( Full) | actuator jamm at most recent location for aircraft (with hydraulic driven), or hang for small airplane |
| | ✓ | | Control failure on one or more actuators because of inner liability | Actuator jamm at most recent location |
| | ✓ | ✓ | Part of/all actuator failure | efficiency of actuator is abridged; small variation in aerodynamics |
| | ✓ | ✓ | Engine failure | huge variation in possible working area; small change in aerodynamics |
| | | ✓ | injure to surface of airplane | likely variation in possible working state; significant minor change in aerodynamics |

**Table 3.1 Aircraft Failure Modes**

### 3.1.3 Actuator Failures

Failures due to actuators are decisive because they change configuration of plant for which we designed the controller [24]. So, the failure in actuator assembly is the main heart of research work.

### 3.2 Types of Actuator Failures

Actuators may stop working during middle of system process, and they are often tentative because it is not possible to say when and how actuators failure can occur, and state of failure is unknown. Actuator failures are common in control system. They are uncertain in pattern, parameters and failure time. Actuator failure can be due to loss of effectiveness, lock in place (stuck) or hard over/runway and hard under and it may cause the maximum rate to be decreased on numerous actuators, one or more actuator jammed at last postion or float. Actuator failures are the types of failures that mostly occurrs.

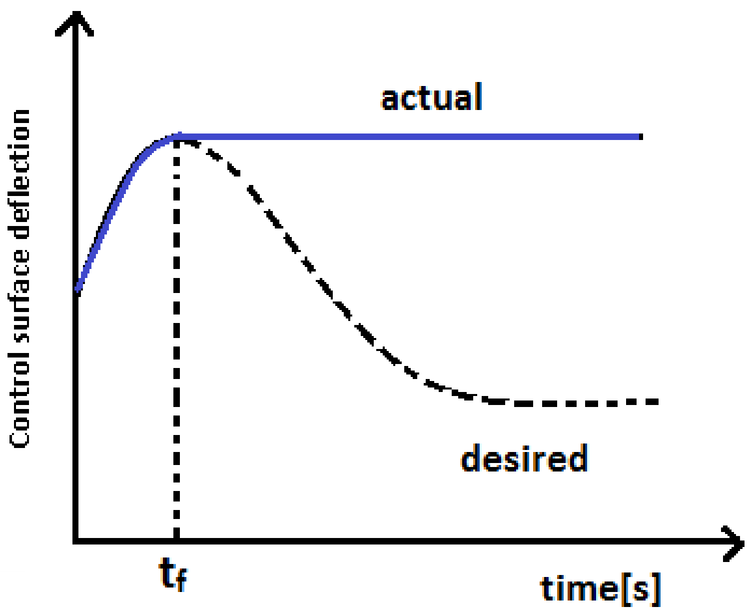


**Figure 3.2 Lock in place**

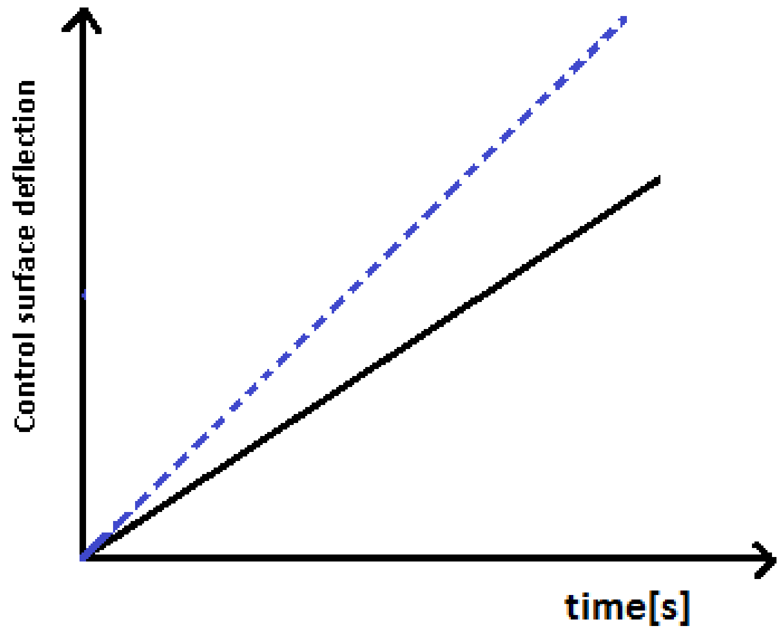


**Figure 3.3 Loss in Effectiveness**

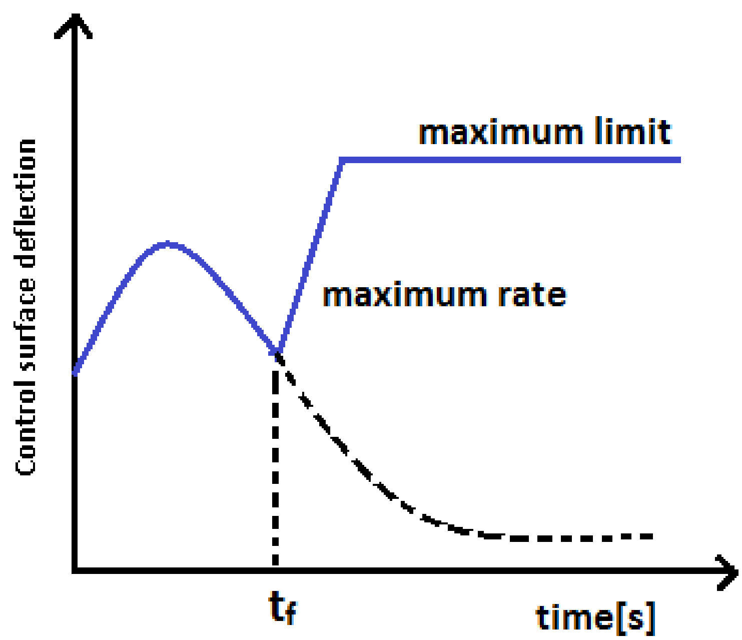


**Figure 3.4 Hard Over**

### 3.2.1 Lock in Place

A lock failure is a state of failure when an actuator becomes stuck and immovable. This might be owed to deficieny in lubrication or motorized jam. This category of failure is measured in [31, 33, 32, 34, 36] and occurred in flights for example flight 1080 (LockheedL- 1011, San Diego, 1977) [26] when one of the parallel stabilisers at complete rambling edge-up point was jammed; and flight 96 (DC-10, Windsor, Ontario, 1972) [26] in which rudder was blocked.

### 3.2.2 Loss in Effectiveness

This type of failure occurs when the loss of part/full control surface is happened.

### 3.2.3 Hard over and Hard under

Runaway or hard over is the largest part disastrous failures category. Hard over state makes actuator to shift to maximum boundary of rate. For example, electronic component failure can cause a rudder runaway giving undesired overpowered indication to actuators to deflect rudder to deflect to its maximum boundry rate. The sort of malfunction is measured in [35] and happened in incidents such as flight 427 (B-737, Near Aliquippa, Pennsylvania, 1994) [30] which suffered hard over of rudder. And Flight 85 (B-747, Anchorage, Alaska, 2002) [26] (which has hardover of a lower rudder to maximum left packed deflection, causing the aircraft making excessive rolls) .The Hard under is opposite condition of hard over.

### 3.3 Summary

In this chapter the complete description of faults and failures are presented. The different types of actuator failures such as lock in place, loss of effectiveness and hardover/hardunder is discussed. Examples of various accidents due to different actuator failures from literature are discussed in detail to address actuator failures.

# CHAPTER 4

# LINEAR QUADRATIC REGULATOR

The theory of control optimality for dynamic system is concerned with operating at minimum cost. In LQ problem, a set of linear differential equations are used to describe the system dynamics and the quadratic function is used to describe the cost. The theory main consequence is the solution given by the linear-quadratic regulator (LQR). In this, a mathematical model gives the horizon of controller prevailing whether a device or method which is based on minimizing a cost function provided by individual. The cost function is often described with computation of summations of  the deviation of major dimensions from preferred standards. So this method form basis to locate such settings of controller which is based on minimizing the undesired deviations. In effect, the LQR algorithm relies on the tiresome approach by the control systems engineer to optimize the controller. However, the control engineer still wishes to take care of the weight figures for comparison of consequences with the precise proposed aims specified. Often we say that that synthesis of controller is still an iterative progression where the formed "best possible" controllers are judged by engineer via simulation and then allows them to adjust the weighting factors accordingly in order to achieve controller in formation with the proposed goals specified.

## 4.1 LQR control method

Many systems can be defined by set of differential equations.

$$\dot{x}(t) = f(x(t),u(t))$$
$$y(t) = h(x(t),u(t)) \quad (4.1.1)$$

Where x(t) are the system's states ,an n dimention vector $x(t)=(x_1(t),\ldots\ldots x_n(t))$ , u(t) is the input to the system, an m dimention vector and y(t) is the output to the sytem ,a p dimention vector.

For LQR [37], we need a linear trim model.

### 4.1.1 Trim Condition

A trim condition is a point in the space of state and control vectors where the time derivative of the state vector is zero [38]:

$$X_t,\ U_t \Rightarrow \dot{X} = 0 \quad (4.1.2)$$

where X is a vector containing the total values of all the aircraft states, and U is a vector containing the total values of all actuators.

The parameter matrices in are the Jacobians of the state derivative function with respect to the state and control vectors, calculated at the trim condition:

$$\begin{aligned} A &= \left[\partial f\left(X,U\right)/\partial X\right]_{Xt,Ut} \\ B &= \left[\partial f\left(X,U\right)/\partial\, U\right]_{Xt,Ut} \end{aligned} \quad (4.1.3)$$

The longitudinal state and control vector is thus defined as:

$$X = \left[V\ \alpha\ q\ \theta h\right]^T$$
$$U = [\delta_e\ \delta_s\ T]^T$$

To design an LQR optimal controller, consider the system:

$$\dot{x} = Ax + Bu \quad (4.1.4)$$

And suppose we want to design state feedback control $u = Fx$ in order to stabilize the system. The design of F is a trade off among control effort and transitory reaction. The optimal control approach to this design trade off is to define the performance index (cost functional).

$$J = \int \left[ x^T(t) Q\, x(t) + u^T(t) R u(t) \right] dt$$

(4.1.5)

Where, 'J' is defined by an integral over $[0,\infty)$, Q is an $n \times n$ symmetric positive semi-definite matrix and R is an $m \times m$ symmetric positive definite matrix. The matrix Q can be written in the form such as $Q = M^T M$, where 'M' is a $p \times n$ matrix, with $p \leq n$.With this representation

$x^T Q x = x^T M^T M x = z^T z$

where z=Mx can be analyzed as a controlled output.

### 4.1.2 Optimal Control Problem

Finding $u(t) = Fx(t)$ to minimize J subject to the constraint $\dot{x} = Ax + Bu$ .

Recall that (A,B) is stabilizable if the uncontrollable eigen values of A, if any, have negative real parts.

Notice that (A,B) is stabilizable if (A,B) is controllable or $\mathrm{Re}\left[\lambda(A)\right] < 0$.

**Definition:** (A,C) is detectable if the unobservable eigen values of A, if any, have negative real parts.

**Lemma 1:** Suppose (A,B) is stabilizable, (A,M) is detectable, where $Q = M^T M$, and $u(t) = Fx(t)$. Then, J is finite for every $x(0) \in R^n$ if and only if $\text{Re}\left[\lambda(A + BF)\right] < 0$

Remarks: The need for (A,B) to be stabilizable is clear from the fact that otherwise there would be no F such that $\text{Re}\left[\lambda(A + BF)\right] < 0$.

**Lemma 2:** For any stabilizing control $u(t) = Fx(t)$, the cost is given by

$$J = x(0)^T Wx(0)$$

Where W is a symmetric positive semi definite matrix satisfying the Lyapunov equation

$$W(A + BF) + (A + BF)^T W + Q + F^T RF = 0$$

**Remark:** The control $u(t) = Fx(t)$ is stabilizing if $\text{Re}\left[\lambda(A + BF)\right] < 0$.

With (A, B) stabilizable, and (A,M) detectable,thus the optimal stabilizing control is

$$u(t) = -R^{-1}B^T Px.$$

(4.1.6)

Where ‘P’ is the symmetric positive semi-definite solution of the Algebraic Riccati Equation (ARE)

i.e.

$$PA + A^T P + Q - PBR^{-1}BP = 0.$$

(4.1.7)

**Remarks:.**

1. Since the control is stabilizing, $\text{Re}[\lambda(A - BR^{-1}B^T P)] < 0$.

2. The control is optimal among all square integrable signals u(t), not just among $u(t) = Fx(t)$

3. The Riccati equation can have more than one solution, but only one solution exist that is positive semi definite.

## 4.2 Stability Studies

### 4.2.1 Stability analysis

Given the system dynamics:

$$\dot{x}(t) = [A - Bk]x(t) \qquad x(t=0) = x_o \tag{4.2.1}$$

With $x(t)=\Re^n$ , $u(t)=\Re^m$ with a linear combination of states to keep small

$$z(t) = Cx(t) \tag{4.2.2}$$

with $z(t)=\Re^p$ .The cost functional quadratic is defined as

$$J = \int_0^\infty \left[ z^T(t) Q\, z(t) + u^T(t) Ru(t) \right] dt \tag{4.2.3}$$

In which the states's size of interest, z(t) biased through the weighting matrix in relation to quantity of action in control in u(t) if the assumptions below hold true.

(1) The entire state vector x(t) is to be available to use as feedback.

(2) [ A B] is to be stabilizable and [ A C ] is to be detectable.

(3) $R=R^T>0$

Then

(1) The linear quadratic regulator is optimal, unique and full state feedback control law can be then taken as

$$u(t) = -Kx(t) \text{ with } K = R^{-1}B^{T}S \quad (4.2.4)$$

that performs on minimizing cost function J, which imposed by open loop constraints to subject to dynamic constaints.

(2) S is the unique, symmetric, positive semidefinite matrix obtained through solution of the algebraic Riccati equation

(3) $SA + A^{T}S + C^{T}C - SBR^{-1}B^{T}S = 0$ (4.2.5)

(4) The resulting closed loop dynamics is thus obtained by substituting (4.2.4) in (4.2.1)

$$\dot{x}(t) = [A - Bk]x(t) \quad (4.2.6)$$

Are thus assured to be asymptotically stable.

## 4.3 Simulation Studies

To test the LQR controller, different actuator failures for Being 747 100/200 aircraft is considered. A trim model is obtained at the equilibrium condition, i.e. at Airspeed $V_T$ = 890 ft/s (980 km/h), Altitude h = 35,000 ft, Mass m = 1,84,000 lbs and Mach number M = 0.8.

### 4.3.1 Simulation Example: Elevator Lock In Place

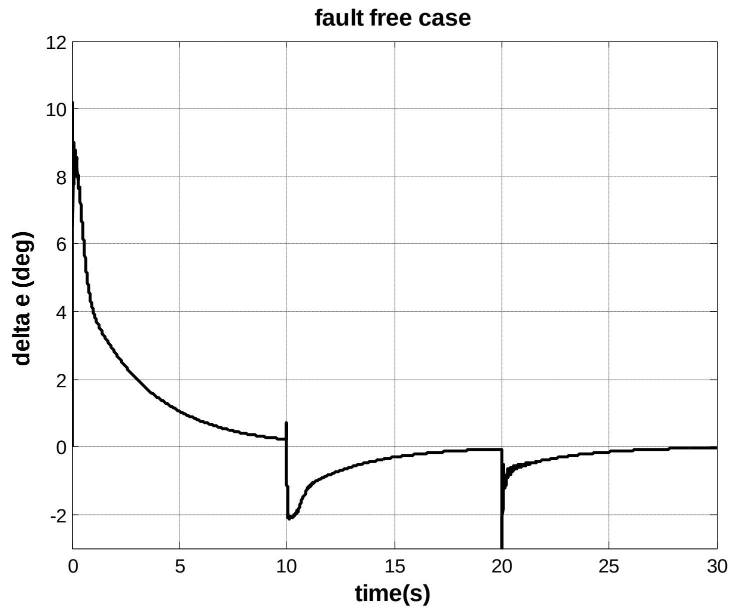


**Figure 4.1 Control surface deflection-elevator deflection fault free behavior**

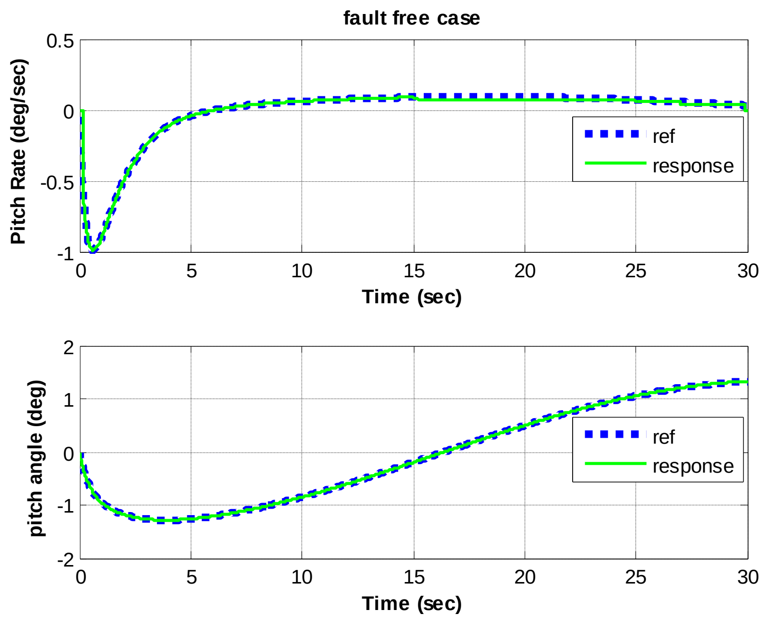


**Figure 4.2 Linear Quadratic Regulator (LQR) controller response for pitch rate and pitch angle for fault free case**

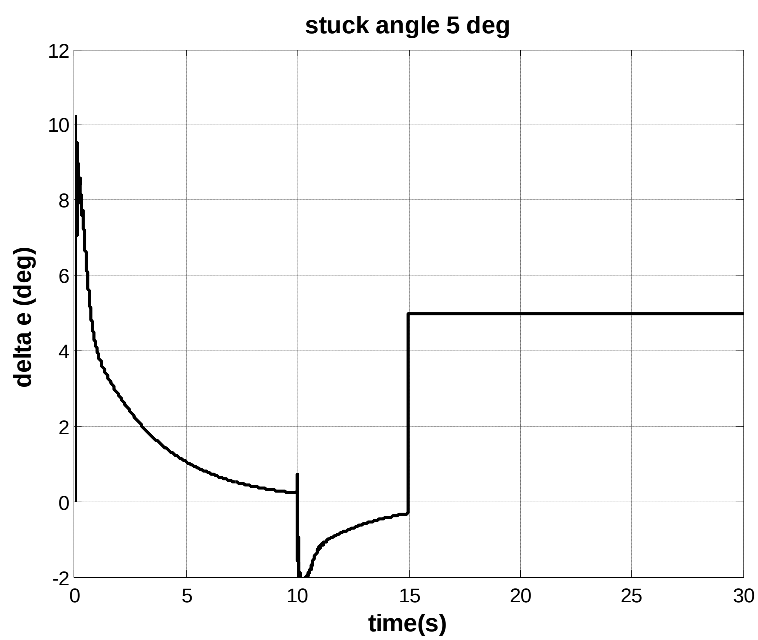


**Figure 4.3 Control surface deflection-elevator stuck at 5 degree at 15 sec**

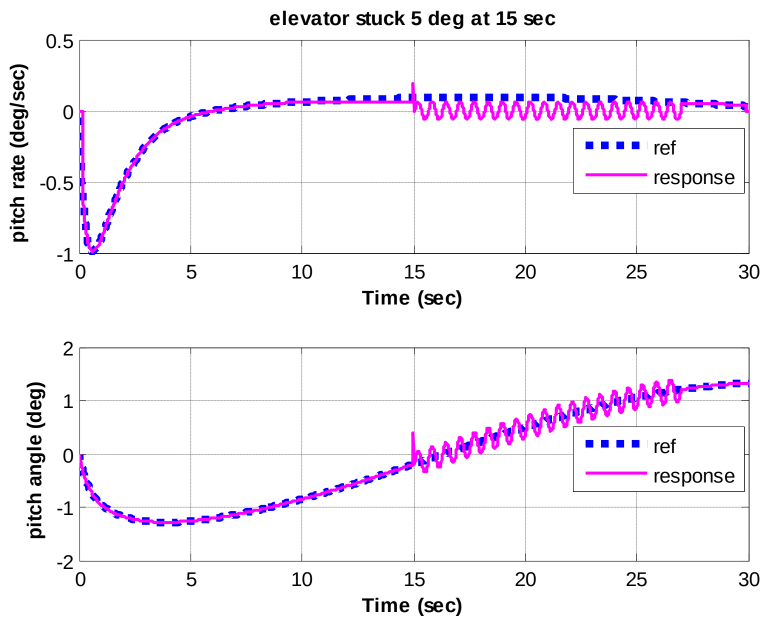


**Figure 4.4 Linear Quadratic Regulator controller (LQR) response for pitch rate and pitch angle for elevator stuck at 5 degree at 15 sec**

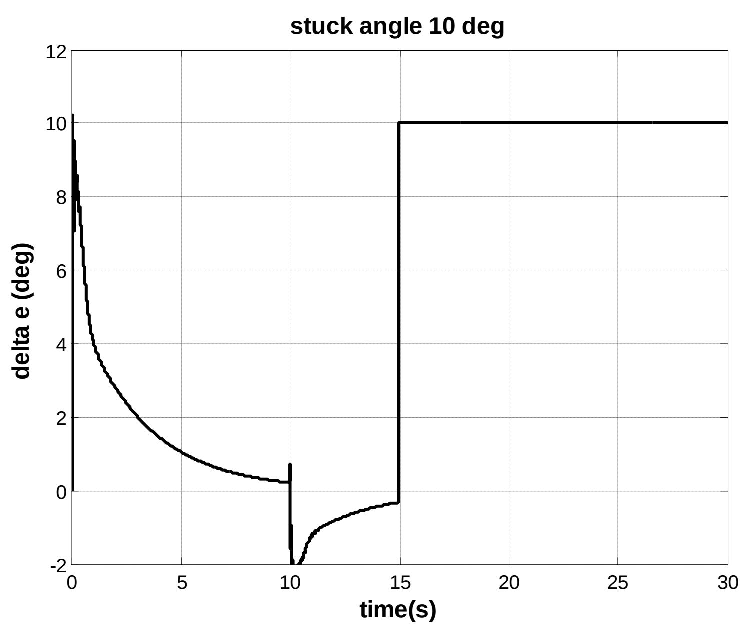


**Figure 4.5 control surface deflection-elevator stuck at 10 degree at 15 sec**

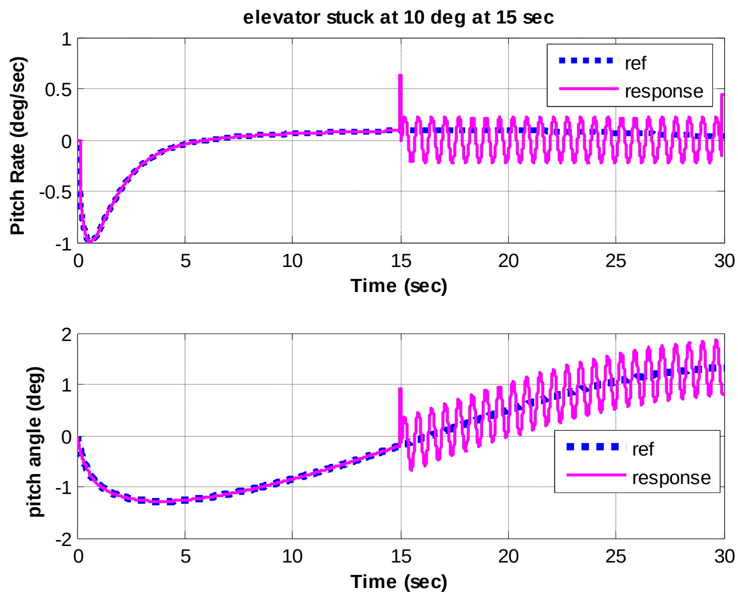


**Figure 4.6 Linear Quadratic Regulator (LQR) controller response for pitch rate and pitch angle for elevator stuck at 10 degree at 15 sec**

When no actuator fault occurs, it is seen that LQR track the pitch rate and pitch angle. Total Elevator deflection in B-747 is -17° to 20°. When the elevator operating around 5°, is stuck at the position of 5° at 15 seconds, fault is 13% of the total deflection. The performance of the LQR controller is still acceptable at 5°. Although, it oscillate but oscillations damp out and it is able to return to steady state flight. But when stuck angle of the elevator increases, the tracking error of the LQR increases. When the elevator is stuck at the position of 10° at 15 seconds, having fault 27% of the total deflection, it is observed that the aircraft of the LQR controller cannot track the pitch angle. As a result this tracking error causes a large error and an oscillation in tracking the pitch rate command. The oscillations do not damp out and aircraft become dynamically unstable.

### 4.3.2 Simulation Example: Loss Of Elevator Effectiveness

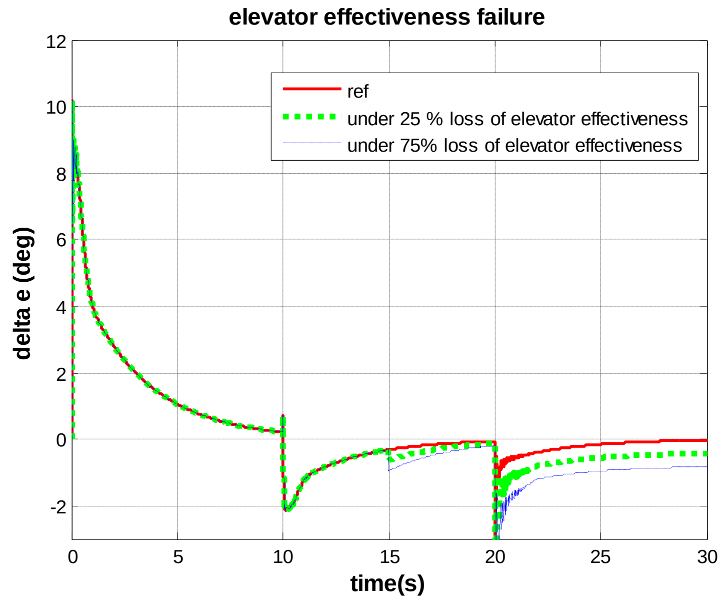


**Figure 4.7 Control Surface Deflection: Loss of Elevator Effectivenes**

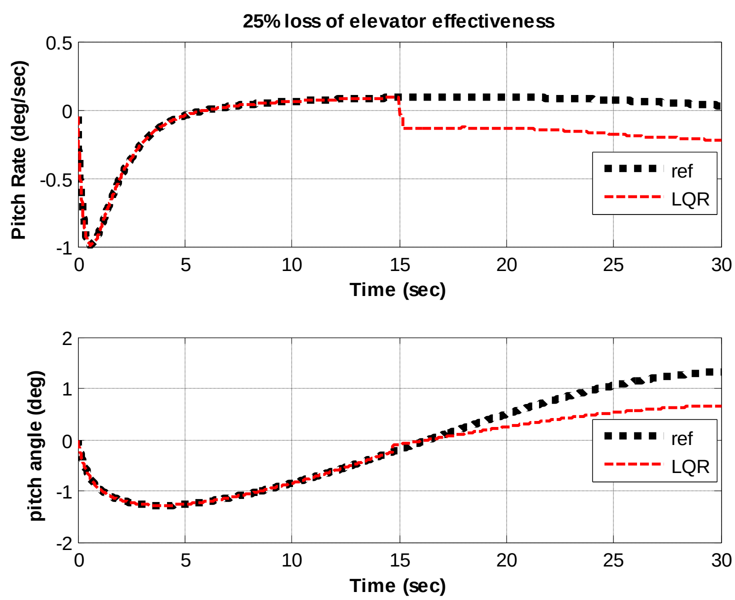


**Figure 4.8 Linear Quadratic Regulator (LQR) controller response for pitch rate and pitch angle for 25% loss of elevator Effectiveness**

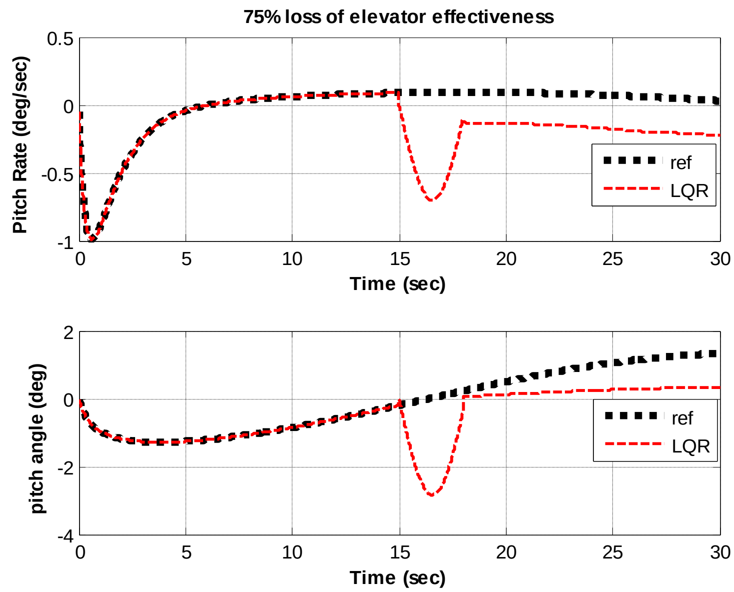


**Figure 4.9 Linear Quadratic Regulator (LQR) controller response for pitch rate and pitch angle for 75% loss of elevator Effectiveness**

Here are simulation results for elevator loss of effectivesness under 25 % loss and 75 % loss.

In first simulation we assume that after 15 seconds, the elevator experiences damage so that the elevator effectiveness is reduced to 25% of its normal effectiveness. Prior to the elevator damage, the pitch rate and pitch angle follows the command signals. After the elevator damage occurs at 15 seconds, it deviates from its commanded signal And controller is not able to scale the elevator control signal to account for the 25% drop in elevator effectiveness.

In 2nd simulation, the elevator effectiveness is assumed to reduce to 75% of its nominal effectiveness. Prior to the elevator damage, LQR drives the pitch rate and pitch angle toward the commands. After the damage, the pitch rate and pitch angle is not able to track the guidance commands as controller is not able to scale the elevator control signal to account for the 75% drop in elevator effectiveness.

### 4.3.3 Simulation Example: Stabilizer Hard Over and Hard Under

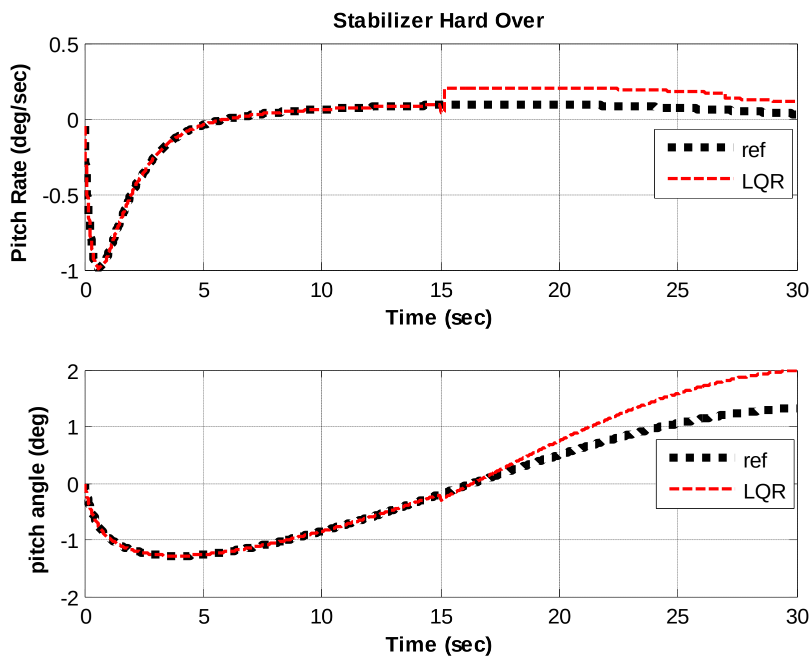


**Figure 4.10 Linear Quadratic Regulator (LQR) controller response for pitch rate and pitch angle for Stabilizer hard over**

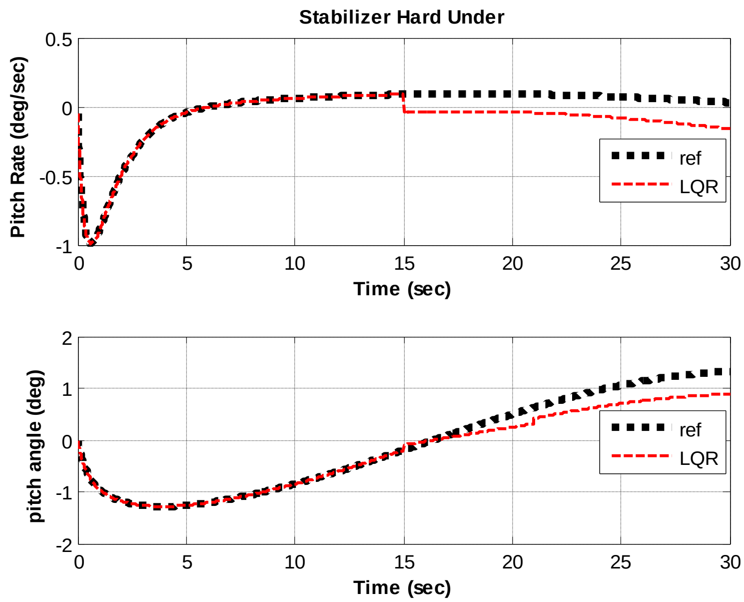


**Figure 4.11 Linear Quadratic Regulator (LQR) controller response for pitch rate and pitch angle for Stabilizer hard under**

In this failure scenario, the stabilizer proceed hard over to its maximum position 3 degrees at 15 seconds. After the fault at 15 sec, the LQR controller loses completely control. The simulation is also carried out with fault at 15 seconds with a hard under excursion of the elevator is introduced. The aircraft is lost in classical controller case.

## 4.4 Summary

In this chapter, Linear Quadratic regulator, LQR control method and its stability and optimilty analysis is discussed in detail. To check the proposed controller, elevator lock in place, elevator loss in effectiveness and elevator hard over and hard under is considered. It has been observed that LQR is not able to handle actuator failures due to its non adaptive nature.

# CHAPTER 5

# DIRECT MODEL REFERENCE ADAPTIVE CONTROLLER

Model Reference Adaptive Control (MRAC) was formerly anticipated to unravel the dilemma wherein the design specications are captured by a reference ideal model, and the controller's parameters are attuned by an adaptation mechanism/law such that the resulting dynamics of the closed loop system become same as the reference ideal model giving desired preferred response when refrence signal is applied to it. In solving this class of problems, the Lyapunov equation plays a very important role in choosing the Lyapunov function and to derive the adaptation mechanism and feedback control. In fact, the construction of Lyapunov functions is systematic and straightforward for the set of systems which can be separated to two portions: (i) a stable linear portion so that we can apply linear stability results easily, and (ii) matched nonlinear portion which can be handled using variant of techniques such as adaptive robust control techniques in different scenario. Thus, MRAC can also be seen as Lyapunov design based on Lyapunov equations.

## 5.1 Adaptive control Design

Adaptive Control wraps a range of techniques based on providing a logical advance to automatically adjust controllers in synchronized in order to accomplish or to sustain a control system performance of desired level when the dynamic model parameters are varying or unknown. An adaptive control method is based on measuring a definite performance catalog of the control system by making use of the inputs, the states, the outputs and the identified turbulence. Then contrast is then made between the measured performance index and given sets. Based on the comparison, the modification of the parameters of the controller is done by adjustable control mechanism to sustain the performance catalog of the control system close to given sets inside the tolerable restrictions and control is designed. So we can infer that the control system can be thought of as an adjustable dynamic system because its performance can be adjusted by varying the parameters of the control signal or controller.

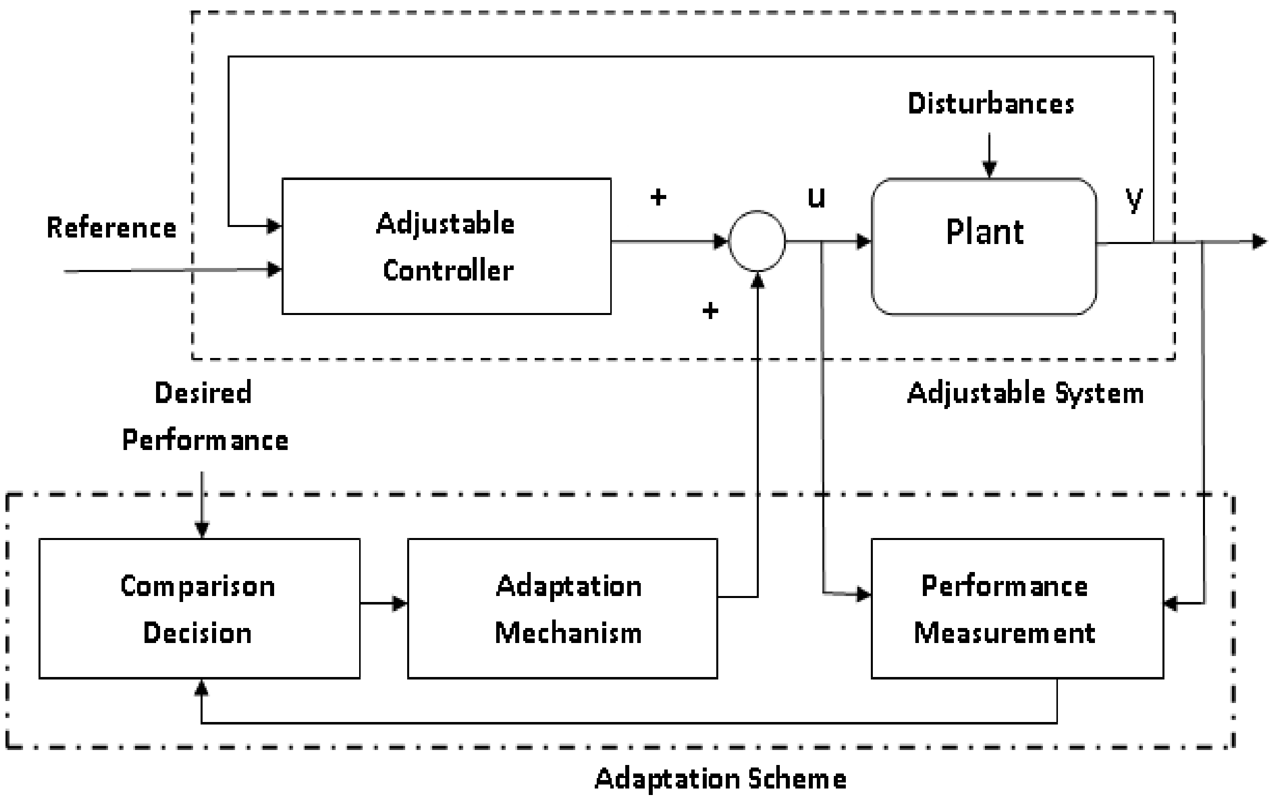


**Figure 5.1 General configuration of an Adaptive controller**

### 5.2 Direct Model Reference Adaptive controller

Here, we use Direct MRAC using Lyapunov Theory [39] to incorporate for actuator faults.

The beauty of direct MRAC is that no plant parameter estimation is required and there is only need to estimate controller parameters (gains)

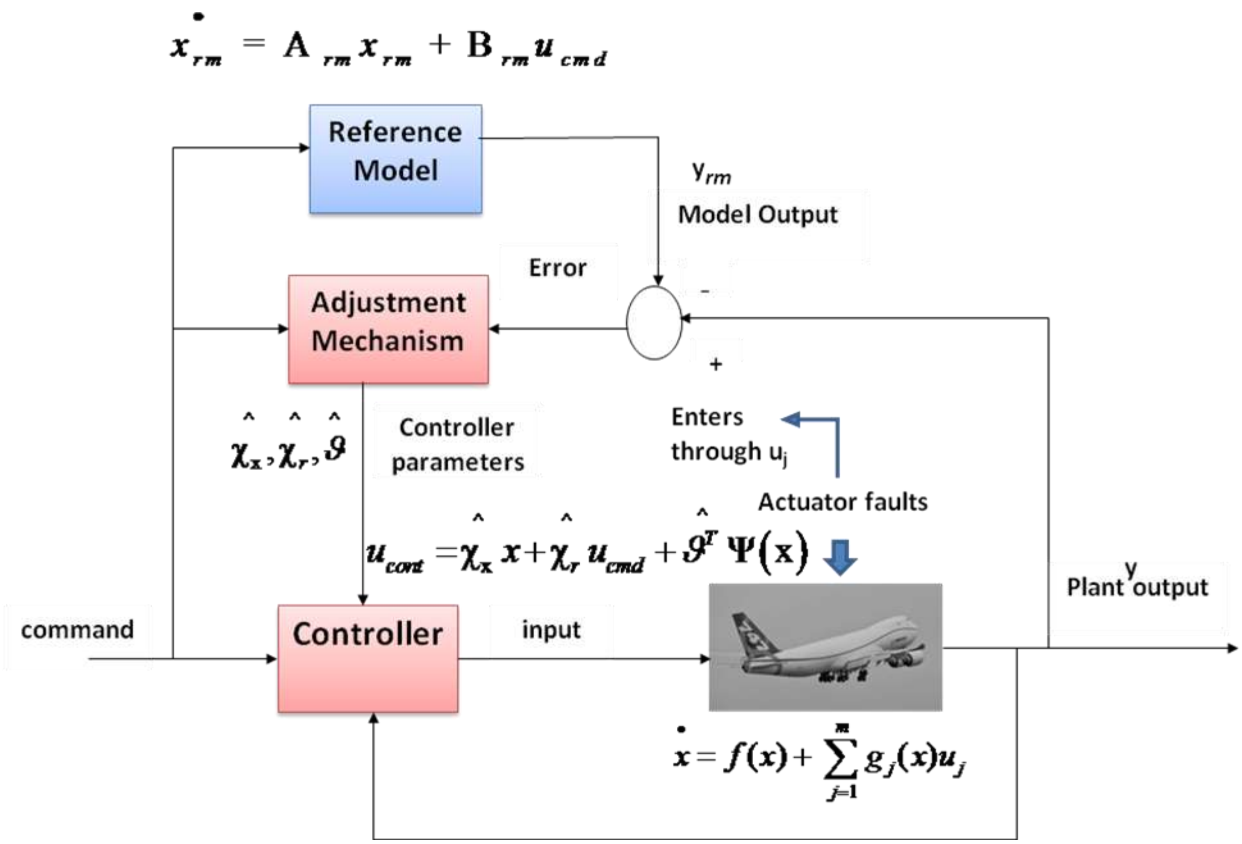


**Figure 5.2 Structure of Direct Model Reference Adaptive Control (D-MRAC)**

Generally, a model-reference adaptive control system can be described by Figure 5.2 in a schematic manner. It possess four parts: a plant which contains varying parameters, a reference model which compactly specifies the preferred response of the control system, a control law based on feedback having parameters which are adjustable, and an adjustment mechanism which updates parameters which are adaptable.

The structure of dynamic sytem or plant is known, however parameters are not known and varying. For linear plants, this implies that number of poles and the number of zeros are identified, but poles and zeros place of location are not known. In nonlinear plants, it verifies that the configuration of the dynamic equations is recognized, however several parameters are unknown.

A reference ideal model indicate the perfect response of adaptive control system to the reference control. Intuitively, it provides perfect plant response which the adjustment method need to acquire to adjust parameters. The selection of reference ideal model is based on designer design and it is one of the steps of the design of adaptive control system. On the one hand, it should replicate the recital measurement of control. It captures the desired specifications such as settling time, rise time and overshoot or frequency realm distinctiveness. On the other hand, this model performance must be attanied by adaptive control system.

The controller usually has a number of parameters which can be tuned. The controller should achieve perfect convergence of tracking. i.e. when plant parameters are found precisely, parameters of controller are able to make the output of plant matched with reference model correspondingly. Till plant parameters are not found exactly, the adjustment mechanism of adaptation will adjust the controller parameters accordingly to make the tracking perfectly achieved asymptotically.

The adaptation mechanism sets the parameters of control. In D-MRAC systems, the adjustment law looks for parameters to make the plant response employing adaptive control to become matched with reference ideal model. So, aim of the adjustment is to track exactly and make error convergence to zero. Clearly, main distinction over usual control deceit in survival the mechanism. The main problem in design of adaptation scheme is to synthesize an adjustment mechanism which will ensure that error of tracking approach zero when parameters are changed and resulting control system remains stable. Many theorems of stability are used in nonlinear control such as Lyapunov theory, hyperstability theory, and passivity theory. Here, the stability and convergence of adaptive control system can be analyzed using Lyapunov theory.

Now we proceed to mathematical formulations of MRAC.

System dynamics are of form:

$$\dot{x} = f(x) + \sum_{j=1}^{m} g_j(x) u_j \quad (5.2.1)$$

Where f(x) is nonlinear function and g(x) is input distribution matrix.

f(x) is uncertain matched nonlinear function which is given as: $f(x) = \vartheta^T \Psi(\mathrm{x})$

Where, $\vartheta$ is the matrix of constant unknown parameters and $\Psi(\mathrm{x})$ is the known non linear basis function which captures the non linearity in function.

### 5.2.1 Stable Reference Model

The stable reference model is taken as: $\dot{x}_{rm} = A_{rm}x_{rm} + B_{rm}u_{cmd}$ (5.2.2)

with $A_{rm}$ is Hurwitz and $u_{cmd} \in \Re^{m}, A_{rm} =\in \Re^{n\times n}, B_{rm} =\in \Re^{n\times m}$

the command input satisfies the actuator magnitude and rate limitations. The reference model is selected on designer choice. We want our system states to approach the desired system states. So our goal is to find command $u_{cont}$ such that the difference between desired states and system states approach zero.

$$\lim_{t\to\infty} \| x(t) - x_{rm}(t) \| = 0 \quad (5.2.3)$$

### 5.2.2 Control feedback

We need to choose control feedback such that our system behave like the desired reference model. So choosing control feedback as:

$$u_{cont} = \hat{\chi}_x x + \hat{\chi}_r u_{cmd} + \hat{\vartheta}^T \Psi(x) \quad (5.2.4)$$

so that resulting system is given by the following equation:

$$\dot{x} = (A + g\hat{\chi}_x^T)\, x + g\hat{\chi}_r^T u_{cmd} + (\hat{\vartheta}^T - \vartheta)^T \Psi(x) \quad (5.2.5)$$

### 5.2.3 Desired Dynamics

The reference model with desired dynamics is given as:

$$\dot{x}_{rm} = \mathrm{A}_{rm} x_{rm} + \mathrm{B}_{rm} u_{cmd} \tag{5.2.6}$$

where ‘$\mathrm{A}_{rm}$’ is Hurwitz and ‘$\mathrm{u}_{cmd}$’ is the reference input command.

### 5.2.4 Model Mataching Condition

In MRAC, there exist the controller gains $\chi_{\mathrm{x}}$, $\chi_r$ such that Model matching condition (5.2.7) should be satisfied.

$$\mathrm{A} + \ \mathrm{g}\chi_x^{\mathrm{T}} = \mathrm{A}_{rm} \qquad \mathrm{g}\chi_r^{\mathrm{T}} = \mathrm{B}_{rm} \tag{5.2.7}$$

### 5.2.5 Error Dynamics

The error of track is the variation between reference states and system states.

$$\mathrm{e}(\mathrm{t}) = \mathrm{x}(\mathrm{t}) - \mathrm{x}_{rm}(\mathrm{t}) \tag{5.2.8}$$

such that,

$$\dot{\mathrm{e}}(t) = \dot{\mathrm{x}}(\mathrm{t}) - \dot{\mathrm{x}}_{rm}(\mathrm{t}) \tag{5.2.9}$$

Putting the values and adding and subtracting $\mathrm{A}_{rm}$ x yields the error dynamics as:

$$\dot{e}(t) = (A + g\hat{\chi}_x)x + g\hat{\chi}_r^T u_{cmd} + (g\hat{\vartheta} - \vartheta)^T \Psi(x) - A_{rm}x_{rm} - B_{rm}u_{cmd} \pm A_{rm}x$$

$$\dot{e}(t) = A_{rm}(x - x_{rm}) + (A + g\hat{\chi}_x^T - A_{rm})x + (g\hat{\chi}_r - B_{rm})^T u_{cmd} + g\Delta\vartheta^T \Psi(x) \qquad (5.2.10)$$

$$\dot{e}(t) = A_{rm}e + g(\Delta\hat{\chi}_x x + \Delta\hat{\chi}_r u_{cmd} + \Delta\hat{\vartheta}^T \Psi(x))$$

### 5.2.6 Candidate Lyapunov Function (CLF)

To ensure system stability, we choose a suitable Lyapunov function.

$$V(e, \Delta\chi_x, \Delta\chi_r, \Delta\vartheta) = e^T P e + \text{trace}\left(\Delta\chi_x^T \Gamma_x^{-1} \Delta\chi_x |\Lambda|\right) + \text{trace}\left(\Delta\chi_r^T \Gamma_r^{-1} \Delta\chi_r |\Lambda|\right) + \text{trace}(\Delta\vartheta^T \Gamma_\vartheta^{-1} \Delta\vartheta |\Lambda|) \qquad (5.2.11)$$

Where, $\text{trace}(H) = \sum H_{ii}$ , $\Gamma_x = \Gamma_x^T > 0,\ \Gamma_r = \Gamma_r^T > 0, \Gamma_\vartheta = \Gamma_\vartheta^T > 0$, are symmetric positive definite matrices. $P=P^T>0$ is a unique symmetric positive definite matrix and obtained from solving algebraic Lyapunov equation:

$$PA_{rm} + A_{rm}^T P = -Q \qquad (5.2.12)$$

$Q=Q^T>0$ is any symmetric positive definite matrix.

### 5.2.7 Adaptive Control design

Our objective is to select adjustment laws (updatation of parameters online) such that the time derivatives of Lyapunov function decreases along error dynamics trajectories. So by taking adaptive laws as described below:

$$\dot{\hat{\chi}}_x^T = \Gamma_x x e^T PB \operatorname{sgn}(\Lambda)$$

$$\dot{\hat{\chi}}_r^T = \Gamma_r u_{cmd} e^T PB \operatorname{sgn}(\Lambda) \tag{5.2.13}$$

$$\dot{\hat{\vartheta}}^T = \Gamma_\vartheta \Psi e^T PB \operatorname{sgn}(\Lambda)$$

So, the time derivative of Lyapunov function becomes semi negative definite.

$$\dot{V}(e, \Delta\chi_x, \Delta\chi_r, \Delta\vartheta) = -e^T Q e \le 0 \tag{5.2.14}$$

Using La Salle's Lemmas such as Barbalat's and Lyapunov like lemmas: $\lim_{t \to \infty} \dot{V}(x, t) = 0$

Since, we know that $\dot{V} = -e^T Q e$. It follows that $\lim_{t \to \infty} \| e(t) = 0 \|$ (5.2.15)

So, MRAC achieves asymptotic tracking x(t) → $x_{rm}$(t) as t →∞. It means our sytem states approach desired reference model states and all signals in closed loop system are restricted.

## 5.3 Stability Studies

Lyapunov design has been a primary tool for nonlinear control system design, performance analysis and stability since its introduction in 1982. The basic idea behind this is to devise a feedback control method that translate derivative of a specied Lyapunov function candidate negative definite or negative semi-definite. Lyapunov's direct method can be viewed as mathematical interpretation of the physical property that if a system's total energy is dissipating, then the sytem's states will ultimately reach to an equilibrium point. The basic idea behind that method is that, if there exists a type of continuous scalar \energy" function such that this \energy" tapers along the system's trajectory, then resulting system is said to be asymptotically stable. it is usually referred to as the direct method beacause there is no need to solve the solution of the differential equations governing the system in determining its stability.[40]

### 5.3.1 Stability Analysis

The Direct Model Reference Adaptive Control method relies on first choosing Lyapunov function candidate and then the feedback control law can be specied such that it renders the derivative of the specied Lyapunov function candidate negative denite, or negative semi-denite while invariance principle is to be used to prove asymptotic stability [40]. This way of designing control is called Lyapunov design. Lyapunov design depends on the selection of Lyapunov function candidates.

The basic idea of Lyapunov direct method consists of:

(i) choosing a radially unbounded positive definite Lyapunov function candidate V(x), and

(ii) evaluating its derivative $\dot{V}(x)$ along system dynamics and checking for its negativeness for stability analysis.

Lyapunov design alludes the production of control methods for some desired closed-loop stability properties using Lyapunov functions for nonlinear control systems $\dot{x} = f(x,u)$ where $x = \Re^n$ is state $u = \Re^m$ is command of control and $f(x,u)$ is locally lipschitz on (x,u) and f(0,0)=0. In actual applications, Lyapunov design can be conceptually divided into two stages:

(a) choose Lyapunov function V contestant for the system, and

(b) design a controller making its derivative $\dot{V}(x)$ non positive.

Let function V (x) be a Lyapunov nominee function. Thus target is to find such u(x) certificating that, for all $x = \Re^n$, derivative w.r.t. time of V (x) alongside system gratify:

$$\dot{V}(x) = \partial V(x) / \partial x = f(x, u(x)) \leq -W(x) \tag{5.3.1}$$

A system for which a good choice of V (x) andW (x) exists is said to hold a control Lyaponuv Function. When $\dot{V}(x)$ is only negative semidenite, asymptotic stability cannot be guaranteed from Lyapunov function method directly. However, if x = 0 is shown to be the only solution for $\dot{V}(x) = 0$, then asymptotic stability can still be concluded by evoking LaSalle's Invariance Principle, Invariant Set Theorem, which basically states that, if $\dot{V}(x) \leq 0$ of a chosen Lyapunov function candidate V (x), then all solutions join to the biggest invariant set candidate in the set $\{x \mid \dot{V}(x) = 0\}$. In fact, this approach has been frequently used to proof of asymptotic stability of a closed-loop system. For our proposed Lyaponuv function candidate, $\dot{V}(x) \le 0$, $\lim_{t \to \infty} \| e(t) = 0 \|$

So, MRAC achieves asymptotic tracking $x(t) \to x_m(t)$ as $t \to \infty$ and all signals in closed loop system are bounded, already proved by (5.2.15)

## 5.4 Simulation Studies

### 5.4.1 Actuator Stuck

Actuator failures are common in control system. They are uncertain in failure time, pattern and parameters. Actuator failure can be due to loss of effectiveness, lock in place (stuck) or loss of control. Actuator stuck happens where the actuator is held at some fixed position at the time of fault occurrence

#### 5.4.1.1 Actuator Stuck Model

. Our non-linear model is described as:

$$\dot{x} = f(x) + \sum_{j=1}^{m} g_j(x) u_j \tag{5.4.1}$$

Where f(x) is smooth nonlinear function, g(x) is input distribution matrix correspond to elevator and stabilator and thrust. The lock in place of control surfaces is explained with an additive model [41].

$$u_j^f = u_o + \Delta u_j \tag{5.4.2}$$

Where $\Delta = diag\{\gamma_1, \ldots\ldots, \gamma_m\}$

$\gamma_i = 0$ at $u_o^{\ i} = \delta_{def}(t_f)$ and $\delta_{def}^{\ min} \leq u_o^{\ i} \leq \delta_{def}^{\ max}$

when the $i^{th}$ control surface is lock in place at a point , $t_f$ is the time when failure occurs.

$\gamma_i = 1$ , $u_o^{\ i} = 0$ ; when the $i^{th}$ control surface is working properly.

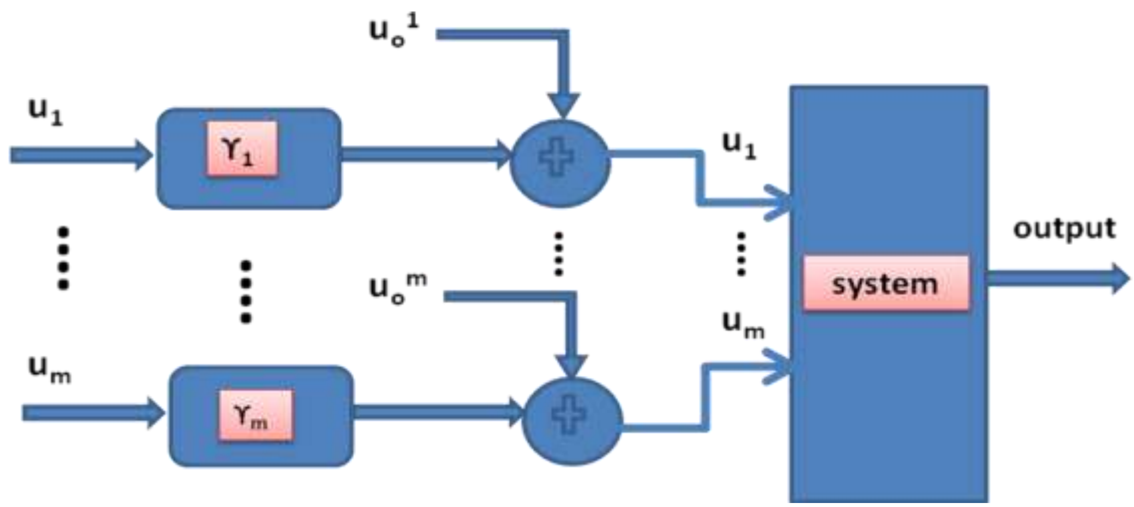


**Figure 5.3 : Lock in place Failure Model**

### 5.4.1.2 Simulation Example

To validate proposed control methodology, Boeing 747 longitudinal model simulations are performed. The proposed adaptive controller is compared with the LQR controller, the non-adaptive controller. In this simulation, the elevator is assumed to lock in place at a certain position during flight. When no actuator fault occurs, it is seen thatD-MRAC track rate of pitch and angle of pitch. Total Elevator deflection in B-747 is -17°to 20°. When the elevator operating around 5°, become jam at the position of 5° at 15 seconds, fault is 13% of the total deflection. From chapter 4, it was observed that the performance of the LQR controller was still acceptable at 5°. Although, it oscillate but oscillations damp out and it is able to return to steady state flight. On the other hand, D-MRAC has good performance and it successfully tract the pitch angle and pitch rate. As the lock in place angle of the elevator increases, the tracking error of the LQR controller becomes larger as compared to the tracking errors of D-MRAC. When the elevator jams at the position of 10° at 15 seconds, having fault 27% of the total deflection, it was observed in previous chapter that the aircraft of the LQR controller cannot track the pitch angle. As a result this tracking error causes a large error and an oscillation in tracking the rate of pitch command. The oscillations do not damp out and aircraft become dynamically unstable. On the other hand, the aircraft with D-MRAC tracks the pitch rate as well as the pitch angle commands and it is able to return to its normal stable flight.

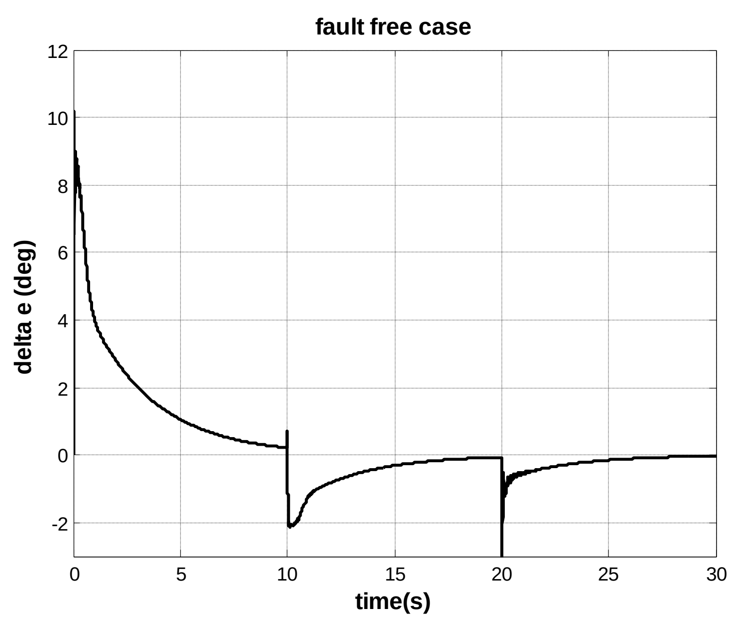


**Figure 5.4 Control surface deflection-elevator deflection fault free behavior**

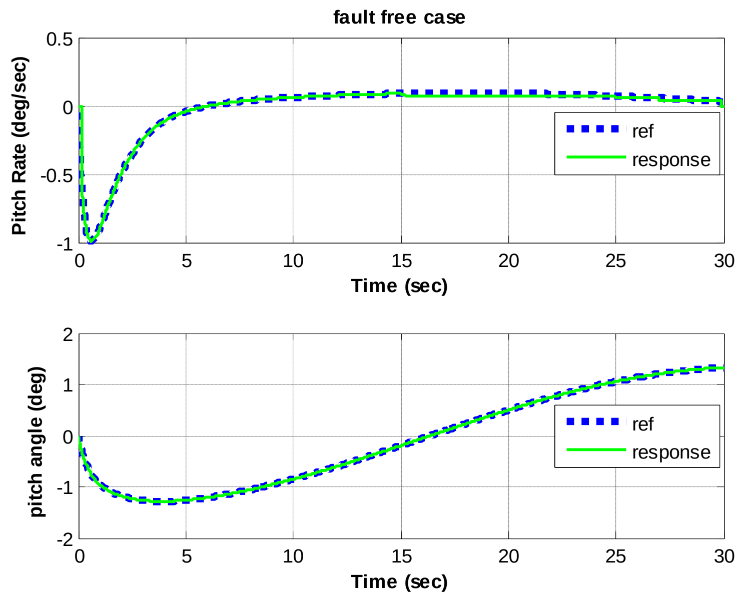


**Figure 5.5 Direct Model Reference Adaptive Controller (D-MRAC) response for pitch rate and pitch angle for fault free case**

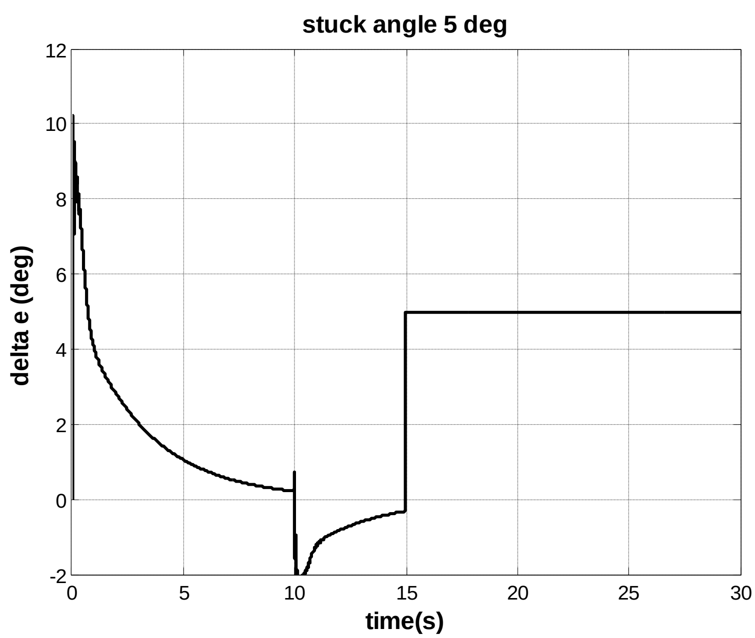


**Figure 5.6 Control surface deflection-elevator lock in place at 5 degree at 15 sec**

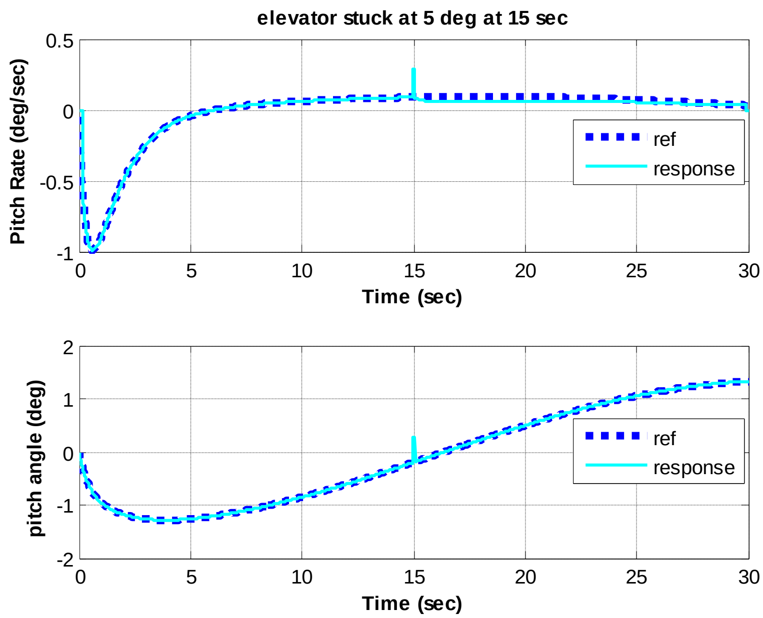


**Figure 5.7 Direct Model Reference Adaptive Controller (D-MRAC) response for pitch rate and pitch angle for elevator lock in place at 10 degree at 15 sec**

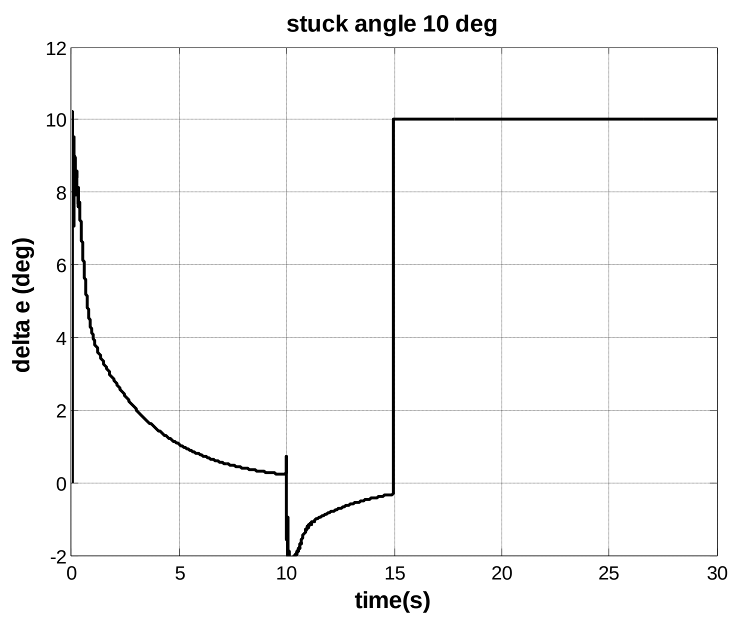


**Figure 5.8 control surface deflection-elevator lock in place at 10 degree at 15 sec**

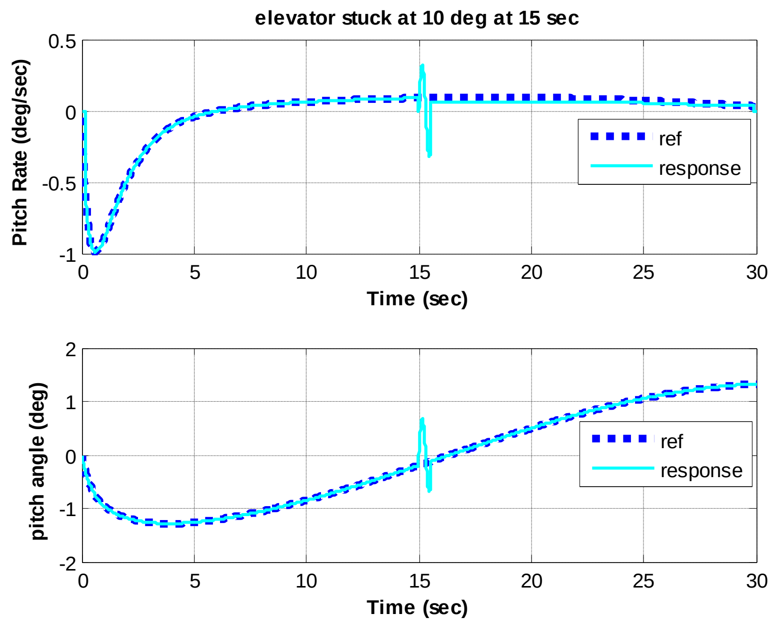


**Figure 5.9 Direct Model Reference Adaptive Controller (D-MRAC) response for pitch rate and pitch angle for elevator lock in place at 10 degree at 15 sec**

### 5.4.2 Loss Of Effectiveness

Loss of effectiveness usually occur due to loss of part/full control surface.

#### 5.4.2.1 Fault Model

Our non-linear model is again described as (5.3.1):

$$\dot{x} = f(x) + \sum_{j=1}^{m} g_j(x) u_j$$

Where f(x) is smooth nonlinear function, g(x) is input distribution matrix correspond to elevator, stabilator and thrust.

Fault model in case of control effectiveness is from [42]:

$$u_i(t) = \zeta_{ii}(t)\, u_i^*(t) \qquad \forall t > T_f \tag{5.4.3}$$

For i[th] actuator become unefficient at failure time $T_f$ .

$u_i^*(t)$ is the input to the i[th] control surface and $T_f$ is the failure time.

$\zeta_{ii}$ represent actuator loss in efficiency $\zeta_{ii} \in [\,(\zeta_{ii})_{\min}, 1]$ and $(\zeta_{ii})_{\min} \geq 0$

$\zeta_{ii} = 1$ before i[th] actuator loses efficiency

$\zeta_{ii} = [\,(\zeta_{ii})_{\min}, 1]$ after i[th] actuator loses efficiency

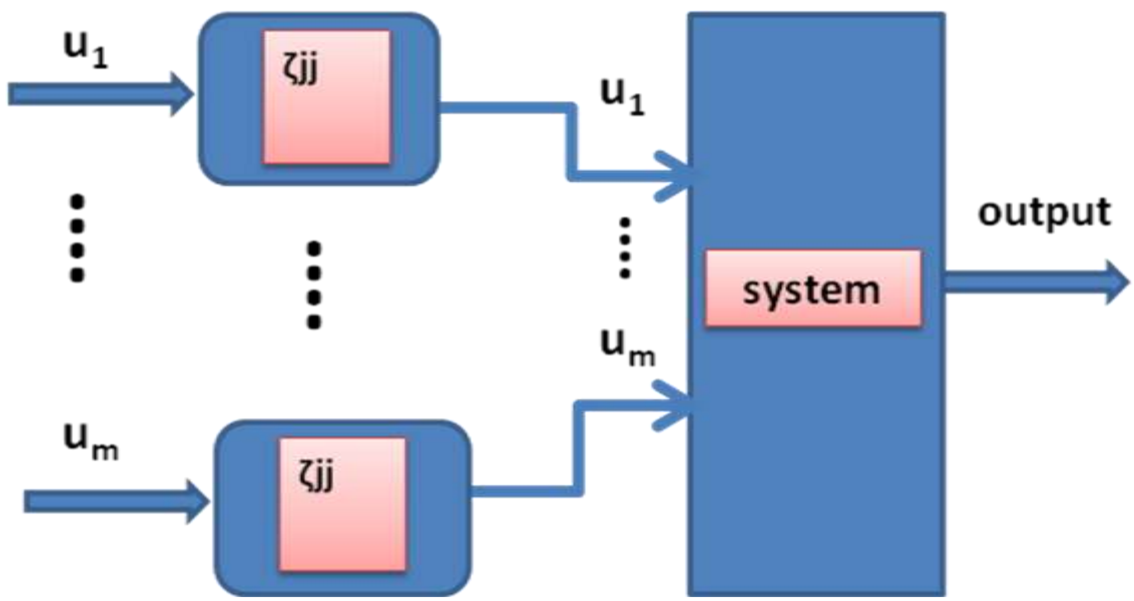


**Figure 5.10 : Loss Of Effectiveness Model**

#### 5.4.2.2 Simulation Example

Here are simulation results for elevator loss of effectivesness under 25 % loss and 75 % loss.

In simulation we assume that after 15 seconds, the elevator experiences damage so that the elevator effectiveness is reduced to 25% of its normal effectiveness. Prior to the elevator damage, the pitch rate and pitch angle follows the command signals. After the elevator damage occurs at 15 seconds, it deviates from its commanded signal. With D-MRAC, after the damage occurs, the adaptive controller correctly scales the elevator control signal to account for the 25% drop in elevator effectiveness.

Figure 5.10 describe time history of pitch rate and pitch angle under 25% elevator effectiveness failure and configuration with D-MRAC.

In 2nd simulation, furthermore assumption is made that the elevator experiences unknown damage at 15 seconds such that the elevator effectiveness is reduced to 75% of its nominal effectiveness. Prior to the elevator damage, D-MRAC drives the pitch rate and pitch angle toward the commands. After the damage, the pitch rate and pitch angle exhibit transient responses before D-MRAC drives the pitch rate and pitch toward the commands. Note that the elevator control surface never exceeds ±20 degrees.

Figure 5.11 shows the time history of the commands (i.e., the outputs of the reference models) and the closed-loop pitch angle and pitch rate.

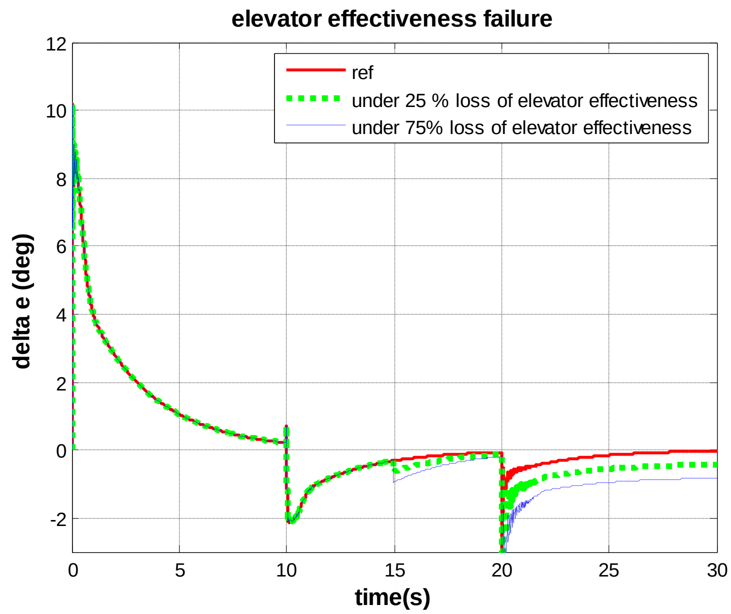


**Figure 5.11 Elevator Deflection: Loss Of Elevator Effectiveness**

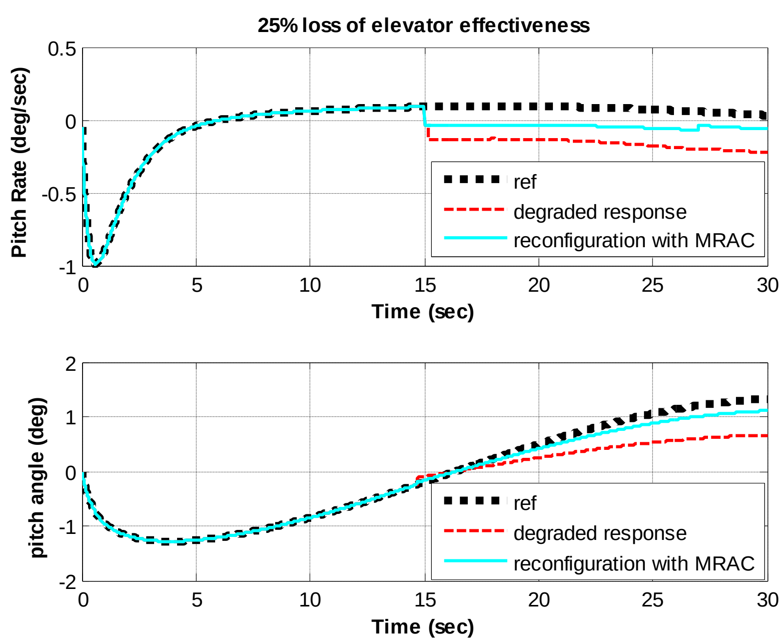


**Figure 5.12 Direct Model Reference Adaptive Controller (D-MRAC) response of pitch rate and pitch angle under 25% loss of elevator effectiveness**

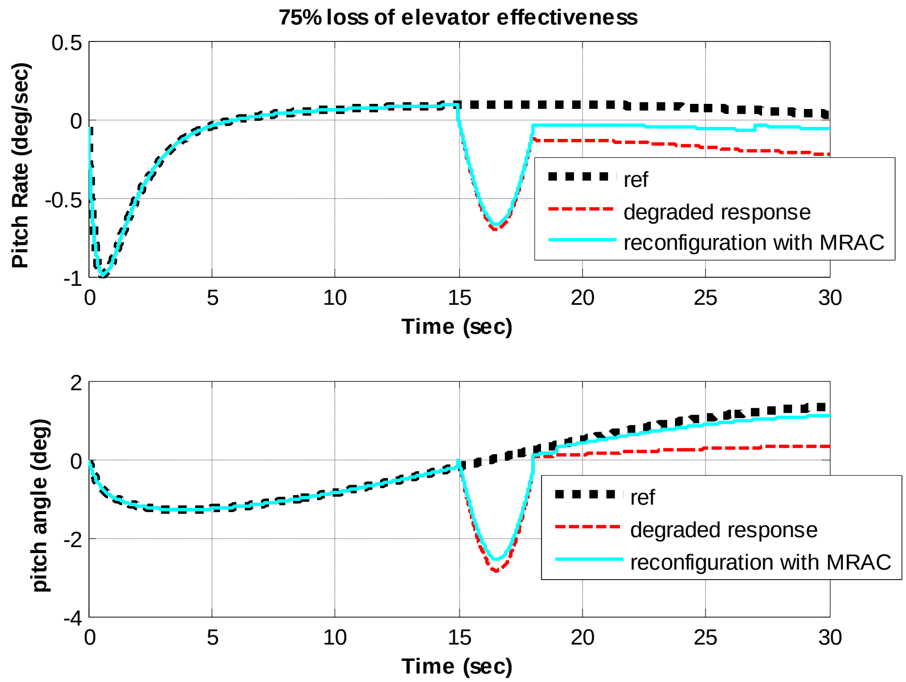


**Figure 5.13 Direct Model Reference Adaptive Controller (D-MRAC) response for pitch rate and pitch angle under 75% loss of elevator effectiveness**

### 5.4.3 Hard Over and Hard Under

This type of failure occurs when the actuator crosses the maximum defelction limit.When actuator posses to cross the maximum positive deflection, it is hard over. And when actuator proceed to cross the negative deflection limit, the type of failure is hard under. Hard over and hard under are the results of partial power loss. Below is the table of Boeing 747 control limits.

| Control Surface | Maximum Deflection |
|---|---|
| Elevators | -23,17 degrees |
| Horizontal Stablizers | -12,3 degrees |
| Inboard Ailerons | -20,20 degrees |
| Outboard Ailerons | -25,+15 degrees |
| Spoilers 1,2,3,4,9,10,11,12 | 0,25 degrees |
| Spoilers 5,8 | 0,20 degrees |
| Spoilers 6,7(only speed brakes) | 0,20 degrees |
| Rudder | -25,25 degrees |

**Table 5.1 Boeing 747 control surfaces limits**

#### 5.4.3.1 Fault Model

A partial power loss causes a change in the actuator dynamics, to be more specific on its maximum rate. So this can be simulated, increasing the settling time to a maximum. To model hard over and hard under, we use following equations:

$$u_j^f = ş_i + u_j \tag{5.4.4}$$

In presence of actuator partial loss of power causing the actuator to proceed to hard over or hard under, $ş_i(t)$ can be expressed as

$$ş_i(t) = Þ_i(t)\left\{\frac{t_m}{t_s^*}\right\} \quad (5.4.5)$$

$$\begin{aligned} \text{Where} \quad Þ_i(t) &= 1 \text{ if the ith actuator fails at } t_j \\ &= 0 \text{ otherwise} \end{aligned} \quad (5.4.6)$$

Where $t_s^*$ is the settling time for fault actuator. For simulation purposes $Þ_i$ is a step signal, reaching 1 at $t_j$ actuator failure time. $t_m$ is the maximum settling time to which it is increased to introduce partial power loss causing hard over.

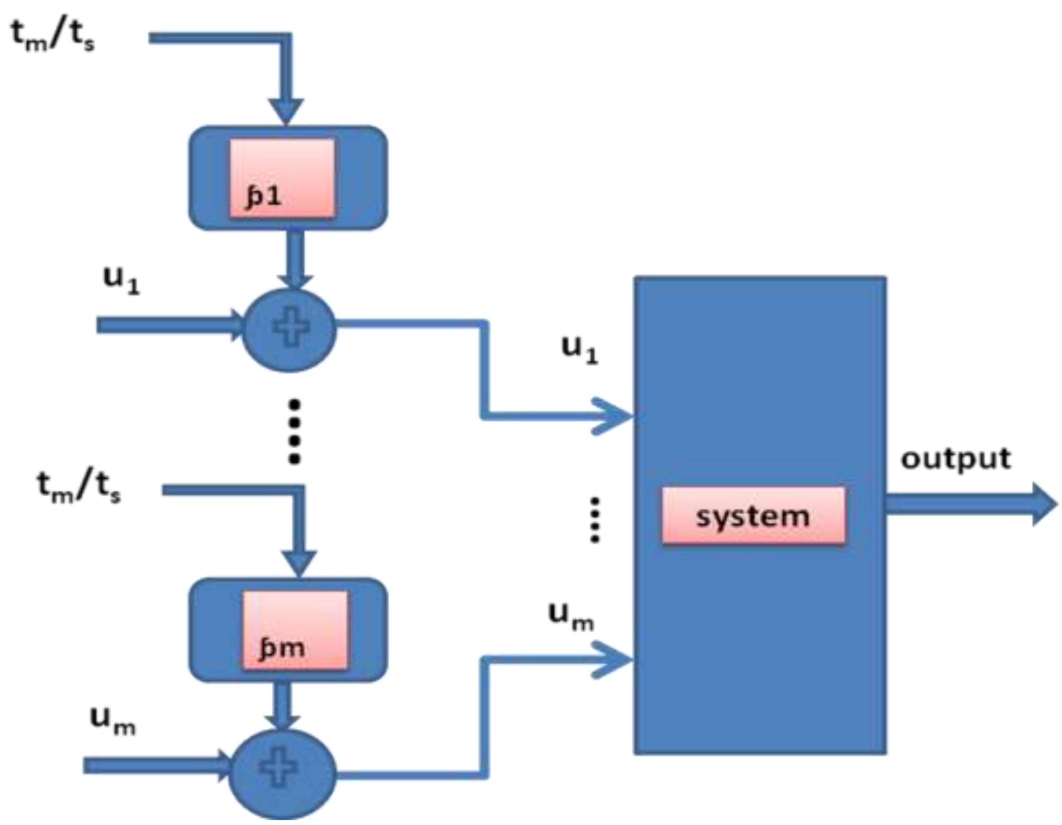


**Figure 5.14 : Hard over Model**

### 5.4.3.2 Simulation Example

In this failure scenario, the stabilizer proceed hard over to its maximum position 3 degrees at 15 seconds. From figure it is seen that Direct Model Reference Adaptive Controller (D-MRAC) is able to maintain the pitch angle and pitch rate reference value. Without D-MRAC, pitch angle possess the response changes, because after the fault, the Classic controller loses completely control. On other hand however it is possible to notice a small steady state error in D-MRAC, but it is able get good results. The simulation is also carried out with fault at 15 seconds with a hard under excursion of stabilizer is introduced. The scenario is similar to the previous one, good pitch angle and pitch rate control is achieved. The aircraft is lost in classical controller case, but D-MRAC totally recovers it.

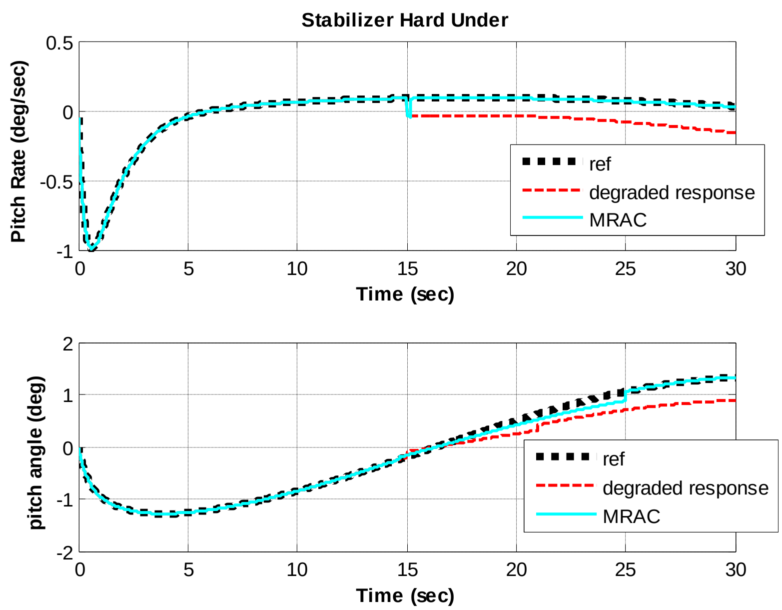


**Figure 5.15 DMRAC response of pitch rate and pitch angle under stabilizer hard under**

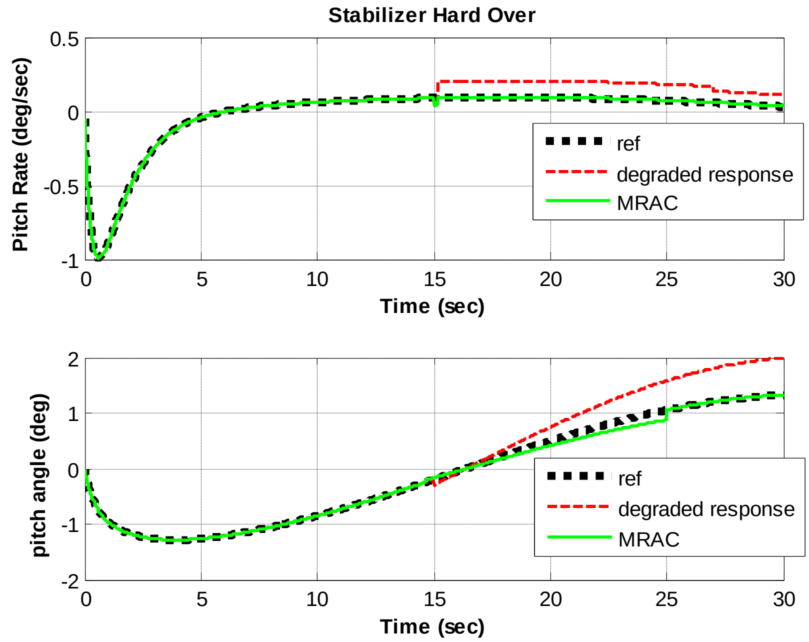


**Figure 5.16 DMRAC response of pitch rate and pitch angle under stabilizer hardover**

### 5.4.4 Multiple Actuator Failures

Here we consider multiple actuator failures case, where at one instant elevator jams and at other instant the thrust is lost. It is observed that D-MRAC correctly compensate for the both failures. All the states track accurately. Although elevator jams at 15 sec, the pitch rate, pitch angle and angle of attack suffers a minor tracking error but D-MRAC correctly brings back to command signal. Similar is case for thrust loss at 20 sec. again the D-MRAC suffer low tracking error but brings back all the states to reference commands. The airspeed is not affected for elevator jam however the speed abruptly drops with thrust loss at 20 sec and classical controller loses control. but D-MRAC is able to maintain the speed to its original speed.

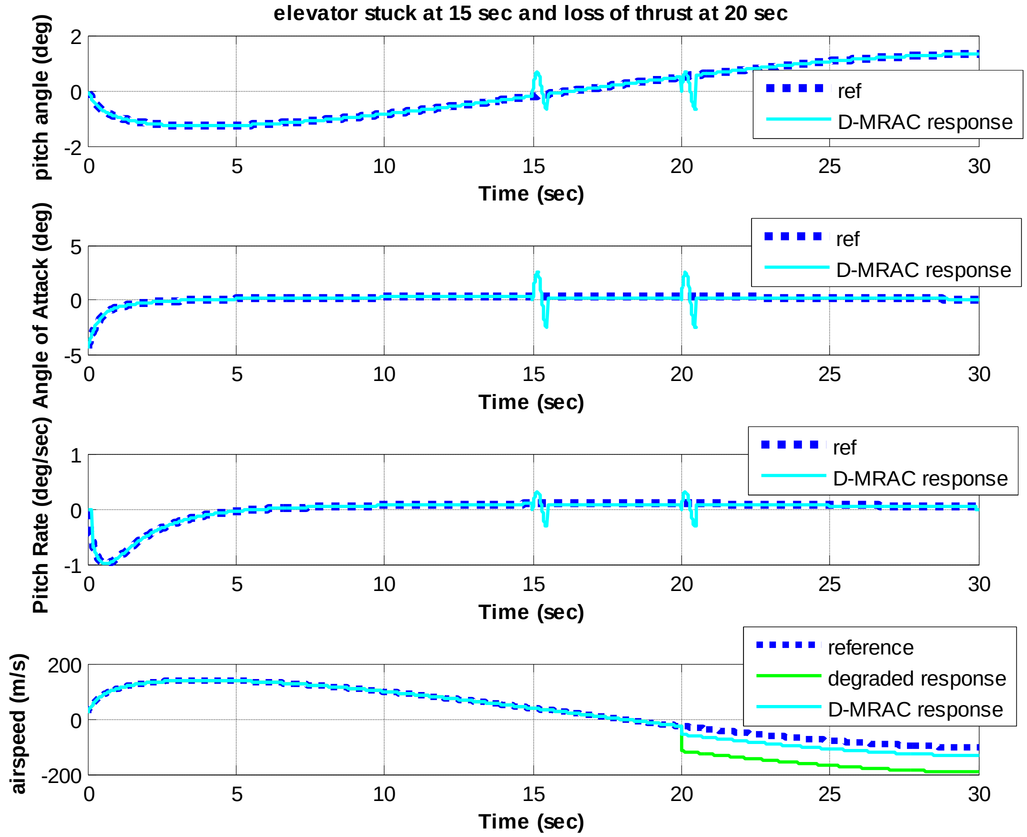


**Figure 5.18 D-MRAC response for Multiple Actuator Failures**

## 5.5 Summary

In this chapter, different failures are taken into account such as elevator stuck, loss of elevator effectiveness, and elevator run away to test the proposed D-MRAC controller. It is seen that for every type of failure, D-MRAC is able to handle the failures successfully and achieves the desired performance despite actuator performance degradation.

# CHAPTER 6

# CONCLUSIONS AND FUTURE SUGGESTIONS

## 6.1 Conclusions

Direct Model Reference adaptive controller (D-MRAC) is used to handle different failures. Model Reference Adaptive Control derives the response of the controlled dynamic syetem to draw near to reference ideal model asymptotically. Different types of faults and failures such as sensor failures, actuator failures and aerodynamics or structural failures are discussed in details. Actuator failures are particularly presented in detail in this thesis work. And it is seen that the model reference adaptive control minimize the effect of fault on the system's behavior. The Adaptive control approach is compared to non adaptive approach. Different actuator failures were tested to check the performance of proposed D-MRAC controller such as the actuator stuck, loss of actuator effectivesness, hardover/runaway and hardunder and it is observed that for each of lapse, it is able to handle the failure scenerios successfully. The each type of failure is checked by increasing error and it is seen that the proposed controller is able to handle it successfully and thus achieving the desired system performance despite the actuator degraded performance. The results are also carried out for non adaptive controller: the LQR controller, but it is observed that the non adaptive controller is not able to handle the situation when the failure occurs due to its non adaptive nature and become worse when the failure increases. So, the proposed adaptive controller, Direct Model Reference Adaptive Controller (D-MRAC) using Lyaponuv theory performs better than the non adaptive classical controller, the LQR controller. It is able to handle the aircraft even in the worst conditions and successfully bring back the aircraft into safe flight condition ensuring the aircraft dynamically stable.

### 6.2 Future Suggestions

The Direct Model Reference Adaptive Controller (D-MRAC) controller guarantee good performance despite the actuator failure or actuator performance degradation while the aircraft with LQR controller oscillates as failure occur and become dynamically unstable as the failure increases. As a future work, we can use dead zone modification with Model Reference Adaptive Controller ( MRAC) for dealing with noise due to small tracking error and thus to produce more bounded signals. The present approach can be used to incorporate structural faults. Apart we can use the hybrid approach to augument the Adaptive controller with the nominal controller due to fact that Adaptive control is improved to robust control to deal with uncertainties or parameters varying gradually and Robust control has advantages in dealing with disturbances, rapidly varying parameters, and unmodeled dynamics. So to ensure good performance we can suggest the Adaptive augmentation of a Robust Baseline controller. And we can also perform hardware implementation of DMRAC algorithm for real time applications

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