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Final Year Project Report

Development of an Automatic Flight Control System

A project report submitted to the

School of Science and Technology

in partial fulfilment of the requirements for the

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In

Aeronautical Electronics Engineering (Avionics)

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Abstract

In the aviation industry, aircraft manufactures are constantly striving to fully understand the essentials required to successfully design an AFCS, based on dynamic response of the aircraft in relation to atmospheric turbulence and structural flexibility.

This report is about the analysis of the longitudinal and lateral dynamics stability of an aircraft, based on the data of CHARLIE. Applying Stability Augmentation System (SAS), Attitude and Flight Path Control System.

Furthermore, to achieve the control system in a good response it was simulated using Matlab environment.

Acknowledgements

I would like to give a huge and sincere thank you to my project supervisor Dr Zoubir Zouaoui, who was the only one who supported me to do this project, and more importantly teached and guided me throughout the year, giving so much time and support solving problems, helping me along the way with helpful advice, and inspiring me to continue in the near future in this topic area.

I would also like to give a special thank to my family and boyfriend for all their help and support.

Contents

1. Introduction

To design successfully any automatic flight control system (AFCS) you need a general understanding of an aircraft dynamic response, nevertheless this knowledge is not satisfactory. It is also important to know the quality of aircraft response which can result in the aircraft being regarded as a satisfactory to fly. Among the different methods of designing control systems, the most common one in AFCS is a self contained account of the main methods of design linear control systems.

When you have to control a commercial aircraft, the pilot can primarily command rates of rotation in any or all 3 axes: pitch, roll and yaw. Nevertheless to make the pilot's workload smaller, especially on long haul flights, most aircrafts have some AFCS or an autopilot system. The AFCS will have a complexity and a capability to control the airplane according to the customer's needs and the investments they want to make

Many autopilot systems have the minimum capacity to control or hold certain path parameters. Modern aircrafts which do medium or long range transportation have an autopilot system that can do navigational tasks too.

The autopilot can be designed to hold and control. Some longitudinal parameters in a flight parameter control or hold mode. So, the autopilot must be designed in a way that the pilot cannot select conflicting modes. When pilots want to communicate with the AFCS they make an input into the computer and it will follow through the command.

Because of the natural split of the airplane equations of motion into longitudinal and lateral-directional equations, this is the order which the fundamentals of autopilot system follow:

- Basic longitudinal Autopilot mode.
- Basic lateral-Directional Autopilot mode.
- Longitudinal navigation modes.
- Lateral-Directional navigation modes.

It is very important to get flying stability of an aircraft when a control system in an aircraft is designed and modeled. In the first place finding a way of modeling the dynamic motion of aircrafts must be considered. We have efficient modeling systems which can describe all the flying state within the engineering field. For each case, we can use various methods to model the dynamic motion, and this will also simplify the problem.

2. Aims and objectives

2.1 Aim: To develop an autopilot or an AFCS for an aircraft. The Design is conducted in the Matlab environment and will be tested in the Flight Simulator.

2.2 Objectives: to develop an AFCS understanding of the principles of small perturbation model for aircraft flight dynamics, the use of the model for the assessment of aircraft flying qualities. Later develop control laws for proposed stability augmentation systems (SAS) for aircraft longitudinal and lateral movements. Furthermore, develop the design according to the analysis result in MATLAB and later test it in the flight simulator.

2.3 Summary of tasks to be completed: In order to achieve the aim, a set of objectives have been carried out as follows:

1. Know concepts about control systems designs.
2. Develop flight dynamic models for small aircraft perturbation motion.
3. Design state space models and transfer function models (longitudinal and lateral).
4. Research flying qualities analysis.
5. Develop and enhances handling and flying qualities.
6. Understand Control Law design (SAS).
7. Close-loop system performance analysis.
8. Understand and choose which control system design approaches is better.
9. Be familiar with advanced programming techniques using in Matlab (Simulink).
10. To test the design in the flight simulator and made modifications accordingly.
11. Conclusions and discussions of the analysis.

2.4 List of equipment/facilities needed:

1. Library – to carry out research using books, journals and the internet.
2. Software – Matlab is used for designing the flight control sketches and solving the complex matrix.

3. Review of Literature

3.1 AIRCRAFT CONTROL SYSTEM

3.1.1 Introduction

The value of any vehicle to its user will always depend on the effectiveness to proceed in the time allowed on an exactly controllable path between its departure point and its final destination.

The velocity vector, which can be denoted as x, is affected by the position, *x*, of the vehicle in space by any kind of control, *u*, can be used, by any disturbance, *ξ*, and by time, *t*. Thus, the motion of the vehicle can be represented in the most general way by the vector differential equation:

$$X=f(x, u, \xi, t) \tag{3.1.1.1}$$

Where *f* is some vector function .The methods by which the path of any vehicle can be controlled vary widely, depending mainly on the physical constraints which obtain. Due to the bigger freedom of the motion, aircraft control problems are normally more complicated than other vehicles.

The features of an aircraft which tend to make it resist any change of its velocity vector, either in its direction or its magnitude, or in both, are what constitute its stability

The path of any aircraft is never stable on itself; aircrafts have only neutral stability in heading. If there is no control, aircraft tend to fly in a constant turn. So as to fly a straight and level course continuously-controlling we must make corrections, either through a human pilot, or with an automatic flight control system (AFCS). In aircraft, such AFCS employ feedback control to get these advantages:

1. The velocity of response is better than from the aircraft without closed loop control.
2. The accuracy when it follows commands is better.
3. The system is able to suppress, to some degree, unwanted effects which have arisen due to disturbances affecting the aircraft´s flight.

Nevertheless, under certain conditions such feedback control systems tend to oscillate; the AFCS then has less stability. Although the use of high values of gain in the feedback loops can help to achieve fast and accurate dynamic response, their use is invariably inimical to good stability. Therefore, designers of AFCS must strike an acceptable, but delicate, balance between the requirements for stability and for control.

When it tends initially to deviate further from its equilibrium flight path, it is statically unstable. When an aircraft is put in a state of equilibrium flight path, and stays at that position, for some time, we can say that the aircraft is dynamically stable.

3.1.2 Control surfaces

All aircrafts have control surfaces or other means that we can use to generate the forces and moments necessary to produce the accelerations which cause the aircraft to be steered along its three-dimensional flight path to its specified destination.

In figure 3.1.2.1, a conventional aircraft is represented. It has the usual control surfaces, namely elevator, ailerons, and rudder. This conventional aircraft has a fourth control, the change in thrust, which is obtained from the engines.

One characteristic of flight control is that the required motion often needs a number of control surfaces to be used simultaneously. When more than one control surface is deployed simultaneously, there often results considerable coupling and interaction between motion variables. It is this physical situation which makes AFCS designs both fascinating and difficult.

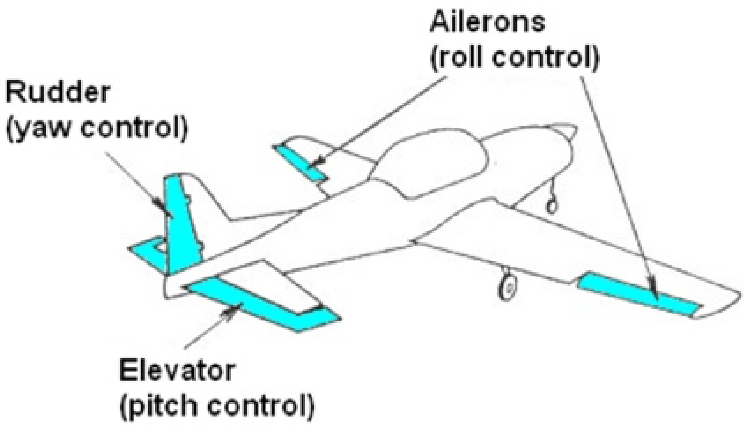

Figure 3.1.2.1 Conventional aircraft (McLean 1990, p. 6)

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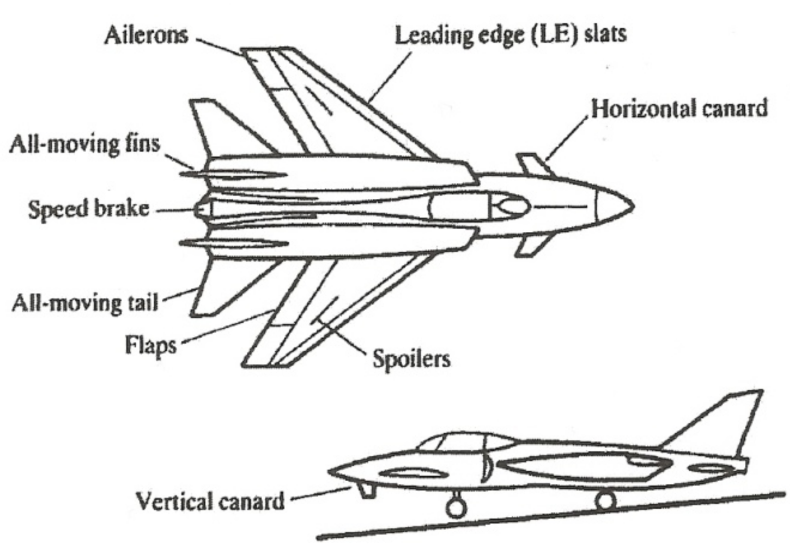


Figure 3.1.2.2 A proposed control configured aircraft (McLean 1990, p. 4)

A sketch of a proposed CCV is illustrated in Figured 1.2 where several extra and unconventional control surfaces are shown.

We move the surfaces by actuators that are signaled electrically (fly-by-wire) or by through fiber optic paths (fly-by-light). However, in a conventional aircraft, the pilot has direct mechanical links to the surfaces, and how he manages the deflections, or changes, he requires from the controls is by means of the so called primary flying controls.

3.1.3 Primary Flying Controls

The primary flying controls belong to the flight control system and are defined as the input elements which a human pilot moves directly cause an operation of the control surfaces. The most important primary flying controls are pitch control, roll control and yaw control. The use of these flight controls affects motion especially about the transverse, the longitudinal, and the normal axes respectively. However, each may affect motion about the other axes.

Figure 3.1.3.1, is the cockpit layout in a typical, twin engine, general aviation aircraft. The yoke is the primary flying control which is used for pitch and roll control.

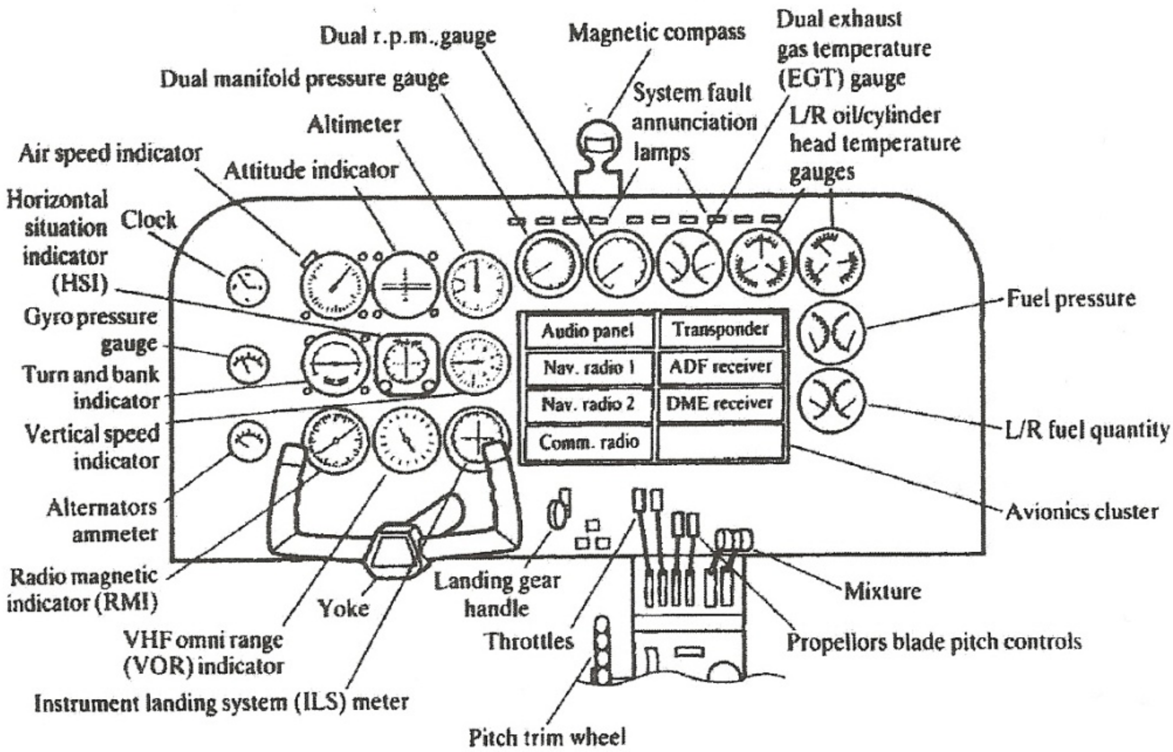


Figure 3.1.3.1 Cockpit layout (McLean 1990, p. 5)

A pilot pushes the pedal left or right with his feet to move the rudder and this affects yaw control. In the type of aircraft with the type of cockpit illustrated, the link between these primary flying controls and the control surfaces is through cables and pulleys.

There are trim wheels for pitch, roll and yaw, which is often called a ‘nose trim’.

When an electrical or hydraulic failure happens, such a powered flying control system stops working, so that the control surface could not be moved: then, the aircraft would be out of control. So that this won’t happen, most civilian and military aircraft keep a direct, mechanical connection from the primary flying control. When this is done, it is said that the control system has ´manual revision ´. Fly-by-wire (and fly-by-light) aircraft have basically the same type of flight control system; (FBW) aircraft have flight control systems which are triplicate, sometimes quadruplicated, to accomplish this stringent reliability requirement.

Modern aircraft are being equipped with side arm controllers that provide signals corresponding to the forces that the pilot applies.

3.1.4 Flight Control System

Apart from the control surfaces that are used for steering, every aircraft has motion sensors which provide measures of changes in motion variables that happen when the aircraft responds to the pilot´s commands or as it finds any disturbance.

We can represent the general structure of an AFCS as the block schematic of Figure 3.1.4.1.

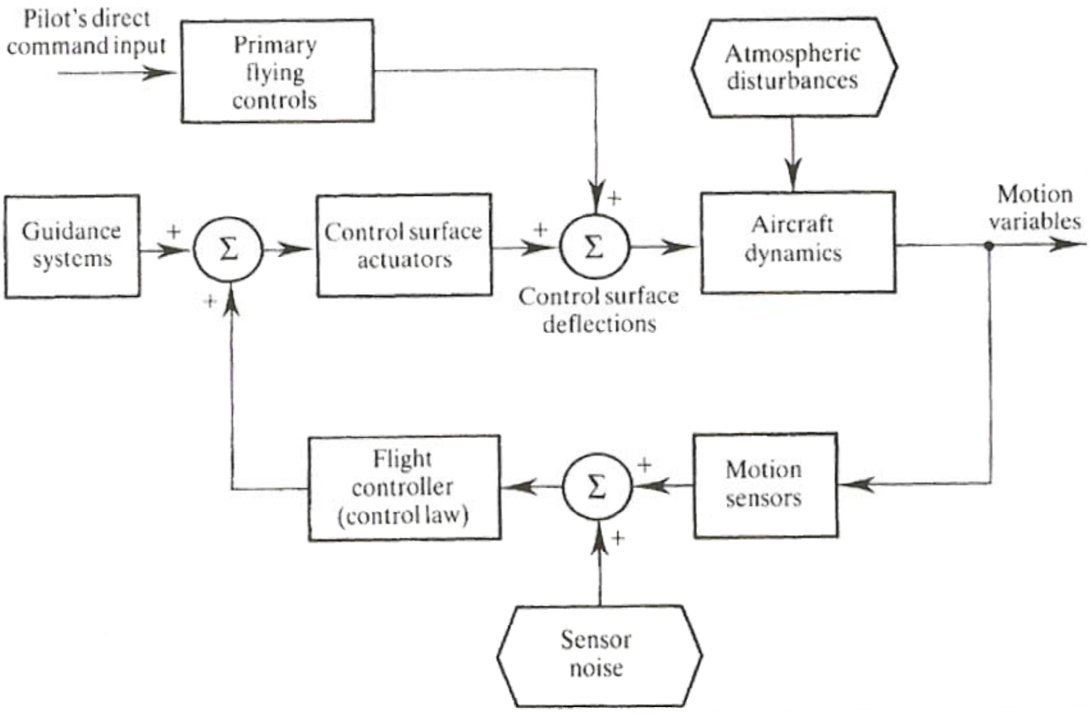


Figure 3.1.4.1 General structure of an AFCS (McLean 1990, p. 8)

3.2 EQUATIONS OF MOTION OF AN AIRCRAFT

3.2.1 Introduction

The problems involving AFCS are generally related to events which do not persist, because of that is considered an Earth axis system as a basic frame of reference, to which any other axis frames employed in the study are referred, so the aircraft itself must have a suitable axis system, as shown in Figure 3.2.1.1.

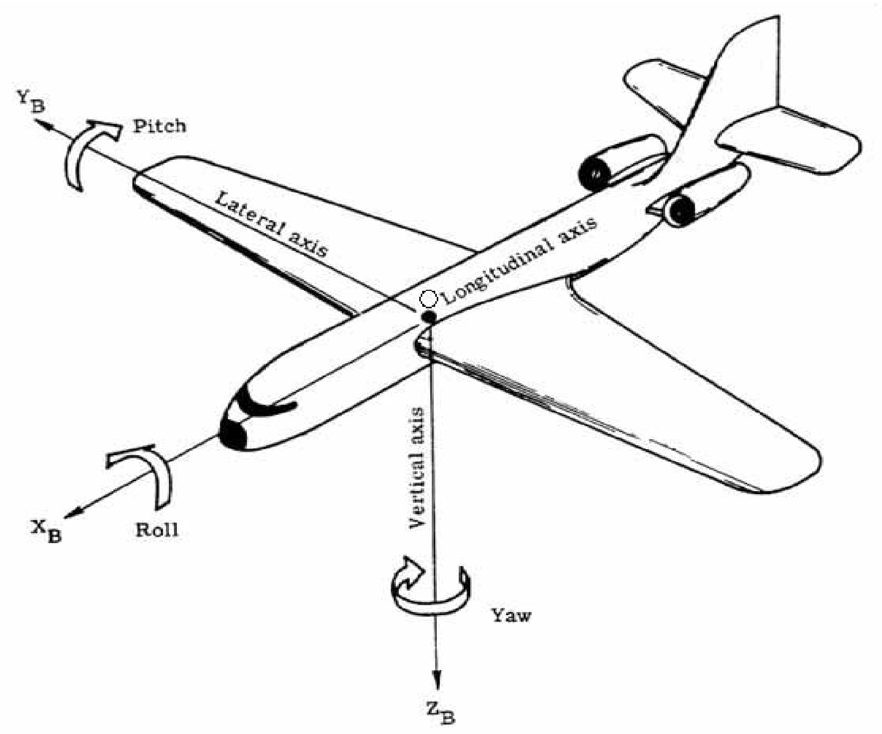


Figure 3.2.1.1 System of axis

The origin of the axes, O, is located at the centre of gravity (CG) of the aircraft. $OX_BY_BZ_B$ is a set of rectangular axes and moves with the aircraft. It's plane of symmetry is OX_BZ_B .

Table 3.2.1.1 Essential elements of the modeling

Axes	OX_B	OY_B	OZ_B
Force components	X	Y	Z
Moment components	L (rolling)	M (pitching)	N (yawing)
Angles of rotation	Φ (bank)	Θ (pitch)	Ψ (yaw)
Linear velocity of components	U	V	W
Angular velocity components	P (roll)	Q (pitch)	R (yaw)

of the aircraft			

From Table 3.2.1.1, all the essential elements which are used to describe the dynamic motion of an aircraft are set up a relationship between them. In other words, they can be applied to make equations for modelling.

Components of linear velocity and force are positive in the positive direction of the corresponding axis. Components of angular and its velocity as well as moment are positive in the clockwise sense about the axes.

According to Donald McLean's work [2] (p27), a set of equations below can describe the resultant of gravitational, aerodynamic and thrust forces in the 'six degrees of freedom', which were developed from Newton's second law:

$$\begin{aligned} X &= m(\dot{U} + QW - RV + g\sin\Theta) \\ Y &= m(\dot{V} + RU - PW - g\cos\Theta\sin\Phi) \\ Z &= m(\dot{W} + PV - QU - g\cos\Theta\sin\Phi) \end{aligned} \qquad (3.2.1.1)$$

Similarly, the moments of an aircraft can be expressed in another set of equations below:

$$\begin{aligned} L &= \dot{P}I_{xx} - I_{xz}(\dot{R} + PQ) + (I_{zz} - I_{yy})QR \\ M &= \dot{Q}I_{yy} + I_{xz}(P^2 - Q^2) + (I_{xx} - I_{zz})PR \\ N &= \dot{R}I_{zz} - I_{xz}\dot{P} + PQ(I_{yy} - I_{xx}) + I_{xz}QR \end{aligned} \qquad (3.2.1.2)$$

3.2.2 Equations for perturbed longitudinal motion:

$$\dot{u} = X_u u + X_w U_0 \alpha - g\cos\gamma_0 \theta$$

$$\dot{\alpha} = \frac{Z_u}{U_0}u + Z_w\alpha + q - \frac{g\sin\gamma_0}{U_0}\theta + \frac{Z_{\delta_E}}{U_0}\delta_E \qquad (3.2.2.1)$$

$$\dot{q} = M_u u + M_w U_0 \alpha + M_{\dot{w}} U_0 \dot{\alpha} + M_q q + M_{\delta_E}\delta_E$$

$$\dot{\theta} = q$$

Where there are some substitutions as: $X_x = \frac{1}{m}\frac{\partial X}{\partial x}$, $Z_x = \frac{1}{m}\frac{\partial Z}{\partial x}$ and $M_x = \frac{1}{I_{yy}}\frac{\partial M}{\partial x}$ [2] (p32) combined with some ignorance of insignificant parameters [2] (p33). Moreover, a new variable, δ_E, the deflection of elevator, is introduced in (3.2.2.1). In fact, $\Theta_0 = \gamma_0 + \alpha_0$, where γ_0 is the initial flight path angle and α_0 is the initial angle of attack. In the stability axis system [2] (p36), W_0 and α_0 are considered as zero, so $\Theta_0 = \gamma_0$.

Sometimes when the aircraft is disturbed from its trim condition, the stability axes rotate with the airframe and, consequently, the perturbed X_s may or may not be parallel to the relative wind while the aircraft is with some disturbance. This situation is illustrated in Figure(3.2.2.1).

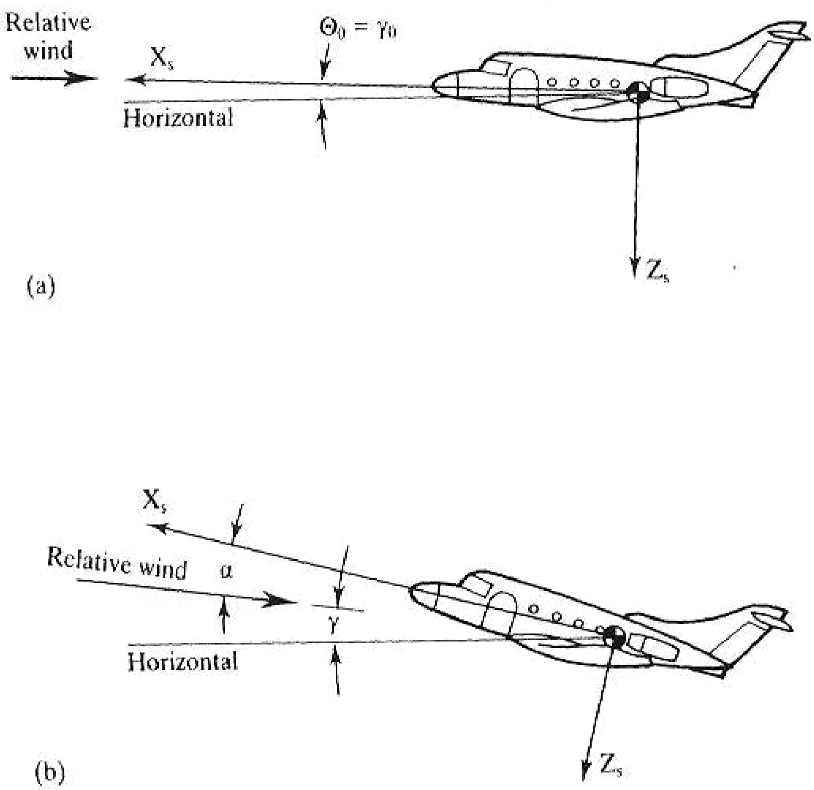


Figure 3.2.2.1 Directional stability axes with respect to the relative wind: a) Steady Flight. b) Perturbed Flight (McLean 1990, p. 36).

3.2.3 Equations for perturbed lateral motion:

$$\dot{v} = Y_v\beta - r - \frac{g}{U_0}cos\theta_0\Phi + Y^*_{\delta_a}\delta_a + Y_{*\delta_r}\delta_r$$

$$\dot{p} = L'_\beta\beta - L'_p p + L'_r r + L'\delta_a + L'_{\delta_r}\delta_r \qquad (3.2.3.1)$$

$$\dot{r} = N'_\beta\beta + N'_p p + N'_r r + N'\delta_a + N'_{\delta_r}\delta_r$$

$$\dot{\Phi} = p$$

$$\dot{\psi} = r\sec\gamma_0$$

Where, δ_a, is the deflection of the ailerons, and δ_r is the deflection of the rudder.

3.3 FLYING AND HANDLING QUALITIES

3.3.1 Introduction

A number of parameters related to the complex frequency domain, for example, the damping ratio and undamped natural frequency of the short period longitudinal motion of the aircraft, feature aircraft flying qualities. Knowing the parameters the designer can image the aircraft´s response to any command or disturbance; it gives a general orientation of the way the aircraft will fly in a controlled way.

Handling qualities show the ease with which a pilot can do a determinate mission with an aircraft which has its own set of flying qualities. But, handling qualities depend on flying qualities and also the primary flying controls, the visual and motion cues available, and the display of flight information in the cockpit. We must realize that a human pilot is changeable, active factor closing an external loop in the region of an AFCS.

As similar missions can be carried out by different kinds of aircraft, it can be said that the required handling qualities also depend on the kind of aircraft.

''The UK laid specifications which were in a number of respects different in expression from the ones of the American Authorities, the UK specifications (MoD, 1983) were decided by 1978 to correspond wherever with those used by the American authorities.

For most classes of fixed wing aircraft, the most significant of these specifications is MIL-F-8785(ASG), Military Specification-Flying Qualities of Piloted Airplanes published in 1980" (McLean 1990, p.151)

3.3.2 Definitions necessary for flying qualities specifications.

3.3.2.1 Aircraft classes

We consider that an aircraft belongs to one of these four classes:

Table 3.3.2.1 Aircraft classification (McLean 1990, p. 152)

Class	*Aircraft characteristics*
I	Small, light aircraft (max. weight ≃ 5 000 kg)
II	Aircraft of medium weight and moderate manoeuvrability (weight between 5 000 and 30 000 kg)
III	Large, heavy aircraft with moderate manoeuvrability (30 000+ kg)
IV	Aircraft with high manoeuvrability

3.3.2.2 Flight Phases

We can divide any mission an aircraft is use to accomplish into three off light in this ways:

In phase A we include all the non- terminal phases of a flight for example those which involve fast manoeuvring, accuracy tracking, or precise control of the flight path. Integrated in phase A would be flight phases like: air-to-air combat (CO), ground attack (GA), weapon delivery (WD), reconnaissance (RC), air-to-air refuelling when the aircraft acts as the receiver (RR), terrain following (TF), maritime search and rescue (MS), close formation flying (FF),and aerobatics (AB).

In phase B there are the non-terminal phases of flight which we get by regular manoeuvres that don't need accurate tracking, although accurate flight path control may be needed. We include in the phase: climbing (CL), cruising (CR), loitering (LO), descending (D), aerial delivery (AD) and air-to-air refuelling when the aircraft acts a tanker (RT).

''Phase C are terminal flight phases, usually accomplished with gradual manoeuvres, which require accurate flight path control. This phase would include: take-off, landing (L), overshoot (OS) and powered approach (including instrument approach) (PA).''

3.3.2.3 Levels of Acceptability

We start the requirements for airworthiness in terms of three different, specific values of control (o stability) parameter. The levels have relation with the capability to finish the mission intended for the aircraft. These levels are defined in Table

Table 3.3.2.3.1 Flying level Specification (McLean 1990, p. 153)

Level	*Definition*
1	The flying qualities are completely adequate for the particular flight phase being considered.
2	The flying qualities are adequate for the particular phase being considered, but there is either some loss in the effectiveness of the mission, or there is a corresponding increase in the workload imposed upon the pilot to achieve the mission, or both.
3	The flying qualities are such that the aircraft can be controlled, but either the effectiveness of the mission is gravely impaired, or the total workload imposed upon the pilot to accomplish the mission is so great that it approaches the limit of his capacity.

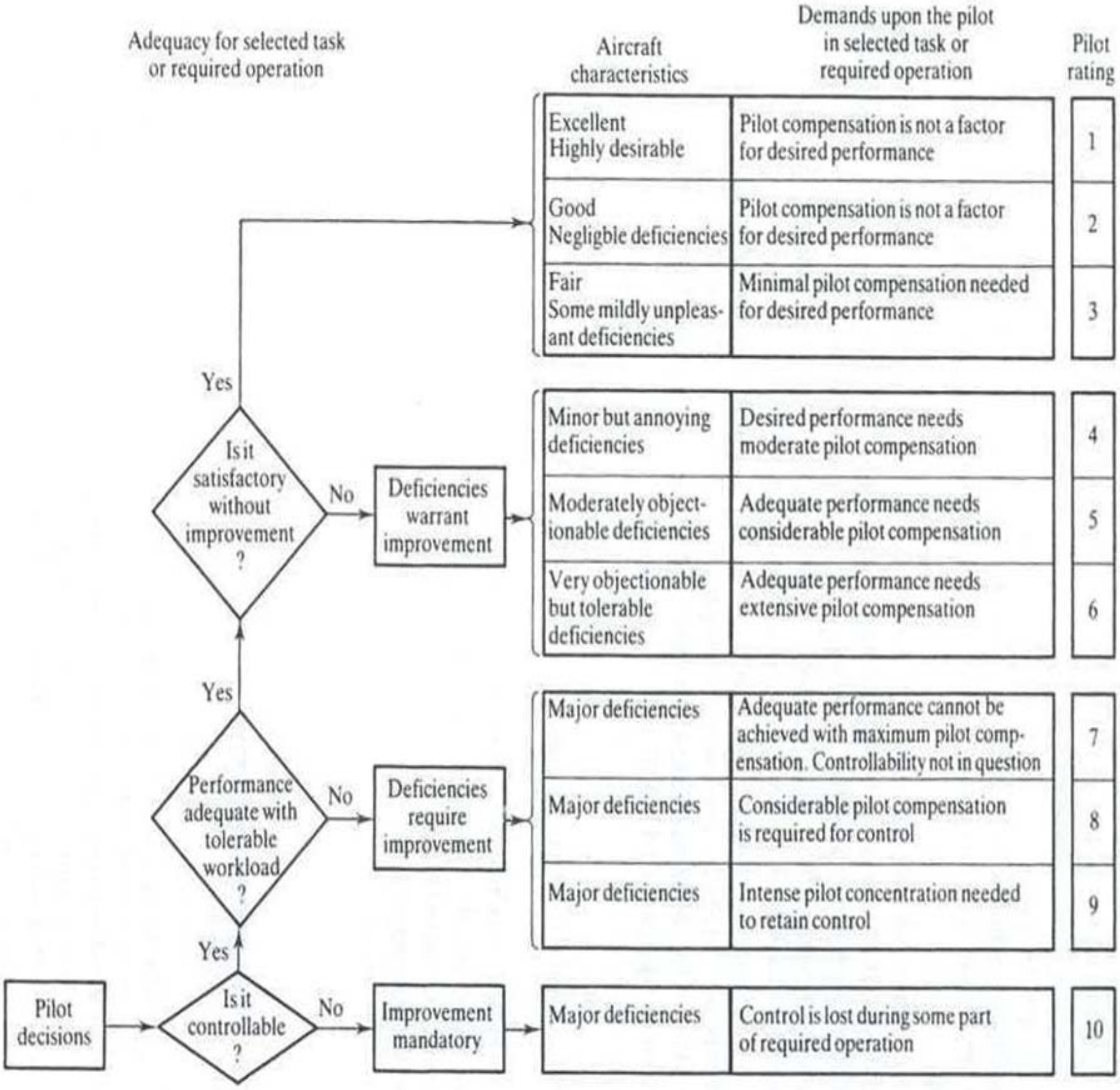
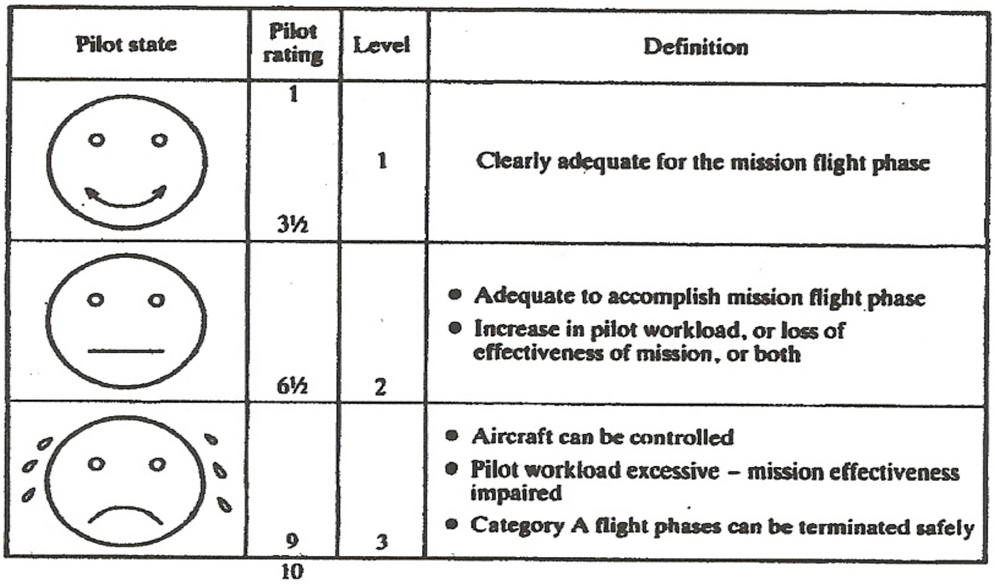


Figure 3.3.2.3.1 Cooper-Harper rating chart (McLean 1990, p. 154)

***"**There is a direct relationship between these levels of acceptability and the pilot rating developed by Cooper and Harper (1986). The rating scale is shown in Figure 3.3.2.3.1 and a representation of the relationship between the rating scale and the levels of acceptability is illustrated in Figure 3.3.2.3.2."* (McLean 1990, p.155)



Pilot state	Pilot rating	Level	Definition
	1 3½	1	Clearly adequate for the mission flight phase
	6½	2	• Adequate to accomplish mission flight phase • Increase in pilot workload, or loss of effectiveness of mission, or both
	9 10	3	• Aircraft can be controlled • Pilot workload excessive – mission effectiveness impaired • Category A flight phases can be terminated safely

Figure 3.3.2.3.2 Acceptable levels of flying qualities (McLean 1990, p. 155)

3.3.3 Longitudinal Flying Qualities

3.3.3.1 Static Response

The airspeed of an aircraft should not tend to diverge periodically when it is disturbed from its trim condition and with its free or fixed pitch control.

3.3.3.2 Phugoid Response

When the frequencies of the phugoid and the short period modes of motion are separated, and the pitch control is free or fixed, we must get to the values of damping ratio which are in Table 3.3.3.2.1.

Table 3.3.3.2.1 Phugoid mode flying qualities (McLean 1990, p. 156)

Level	*Damping ratio of phugoid mode*
1	$\geqq 0.04$
2	$\gtrapprox 0.0$
3	An undamped oscillatory mode having a period of at least 55 s

3.3.3.3 Short Period Response

The flying qualities to the short period are defined by the parameters, ζ_{sp}, the short period damping ratio, and ω_{sp}/n_{z_α} ,where n_{z_α} is the aircraft acceleration sensitivity. In Table 3.3.3.3.1, the specified values of damping ratio are shown.

Table 3.3.3.3.1 Short period mode damping ratio specification (McLean 1990, p. 156)

Flight phase category	*Level 1*		*Level 2*		*Level 3*	
	Min.	*Max.*	*Min.*	*Max.*	*Min.*	*Max.*
A	0.35	1.3	0.25	2.0	0.1	—
B	0.3	2.0	0.2	2.0	0.1	—
C	0.5	—	0.35	2.0	0.25	—

When the short period oscillations are non-linear with amplitude, the flying qualities parameters have to apply to each oscillation.

3.3.4 Lateral flying qualities

The flying qualities for lateral/directional motion is more complicated than for longitudinal motion and for this reason they need more parameters.

3.3.4.1 Rolling Motion

''The time constant of the roll subsidence mode, Tr has to be less than the maximum values found in Table 3.3.4.1.1. It is customary to specify roll performance in terms of the change of bank angle achieved in a given time responding to a step function in roll command. In Table 3.3.4.1.2 we can see the required bank angles and time.'' (McLean 1990, p.159)

Table 3.3.4.1.1 Short period mode damping ratio specification (McLean 1990, p. 161)

Flight phase category	*Class*	*T_R (seconds)*		
		Level 1	*Level 2*	*Level 3*
A	I, IV	1.0	1.4	Not specified – limit is believed to lie within range 6–8 s
A	II, III	1.4	3.0	
B	All	1.4	3.0	
C	I, IV	1.0	1.4	
C	II, III	1.4	3.0	

Table 3.3.4.1.2 Bank angle specification (McLean 1990, p. 160)

Class	Flight phase category	Bank angle in fixed time		
		Level 1	Level 2	Level 3
I	A	60° in 1.3 s	60° in 1.7 s	60° in 2.6 s
	B	60° in 1.7 s	60° in 2.5 s	60° in 3.4 s
	C	30° in 1.3 s	30° in 1.8 s	30° in 2.6 s
II	A	45° in 1.4 s	45° in 1.9 s	45° in 2.8 s
	B	45° in 1.9 s	45° in 2.8 s	45° in 3.0 s
	C	30° in 2.5 s	30° in 3.5 s	30° in 5.0 s
III	A	30° in 1.5 s	30° in 2.0 s	30° in 3.0 s
	B	30° in 2.0 s	30° in 3.0 s	30° in 4.0 s
	C	30° in 3.0 s	30° in 4.0 s	30° in 6.0 s
IV	A	90° in 1.3 s	90° in 1.7 s	90° in 2.6 s
	B	60° in 1.7 s	60° in 2.5 s	60° in 3.4 s
	C	30° in 1.0 s	30° in 1.3 s	30° in 2.0 s

''For class IV aircraft, for level 1,the yaw control should be free for class IV for level1. For other aircraft and levels, the yaw control can be used to reduce any sideslip which tends to retard roll rate.'' (McLean 1990, p.160)

3.3.4.2 Spiral Stability

''When the specifying spiral stability itis assumed that the aircraft is trimmed for straight and level flight, with no bank angle, no yaw rate and with the flying controls free. The specification is given in terms of the time taken for the bank angle to double following an initial disturbance in bank angle of up to 20°" (McLean 1990, p.160).The time must exceed the values given in Table 3.3.4.2.1.

Table 3.3.4.2.1 Spiral mode stability specification (McLean 1990, p.160)

Flight phase category	Level		
	1	2	3
A and C	12 s	8 s	5 s
B	20 s	8 s	5 s

3.3.4.3 Lateral /directional Oscillations-Dutch Roll

The dutch roll mode has a little practical role in the control of an aircraft, but it has important bother value. The values of the important dutch roll parameters are: damping ratio(ζ_D) and the dutch roll frequency(ω_D).

Table 3.3.4.2.2 Dutch roll mode specification (McLean 1990, p. 161)

Flight phase category	*Class*	*Level*								
		1			*2*			*3*		
		ζ_D	$\zeta_D\omega_D$	ω_D	ζ_D	$\zeta_D\omega_D$	ω_D	ζ_D	$\zeta_D\omega_D$	ω_D
A	I, IV	0.19	0.35	1.0	0.02	0.05	0.5	0	—	0.4
A	II, III	0.19	0.35	0.5	0.02	0.05	0.5	0	—	0.4
B	All	0.08	0.15	0.5	0.02	0.05	0.5	0	—	0.4
C	I, IV	0.08	0.15	1.0	0.02	0.05	0.5	0	—	0.4
C	II, III	0.08	0.1	0.5	0.02	0.05	0.5	0	—	0.4

3.4 Control System Design Methods I

3.4.1 AFCS AS A CONTROL PROBLEM

Despite the difficulty in defining control systems, we can say that its objective is to alter the dynamical performance of a physical process and then the response from the controlled system meets the user´s requirements. Both when the disturbances are extraneous to the aircraft, if the AFCS has been specially arranged to reduce the effects of unnecessary inputs, the desired dynamic performance of the closed loop system to command inputs will be inevitably impaired.

‘´´Consequently, when an aircraft has to fly on some particular mission, through the extreme regions of its flight envelope, say recourse is frequently taken to either gain-scheduling or self-adaptive control schemes trying to retain some measure of the compromise solution at every flight condition to be encountered.’´´

3.4.2 CONVENTIONAL CONTROL METHODS

3.4.2.1 Introduction

Conventional control methods as those considered appropriate to time-invariant, linear s.i.s.o system are usually assessed by considering the nature of the roots of its characteristic equation. When we know the values of the roots, insight into the nature of the corresponding dynamic response can be acquired by a designer. The best way to present information about the response of a system is to show in a s-plane diagram the location of the zeros of the system. When designing AFCS, It is attractive to consider the use of methods that give the designer the possibility of precisely locating the poles of the resulting closed loop system in a way that they communicate with the specified locations.

3.4.2.2 Simple Pole Placement Method

A method which is very simple means the use of the specified values of the poles to form a characteristic polynomial that the closed loop system must have. Then, we use negative feedback in the dynamics of the aircraft to change the coefficients of the polynomial to those of each characteristic polynomial.

3.4.2.3 Method of Eigenvalues Assignment

When we consider a linear and multivariable system, the we have more trouble with eigenvalue assignment, than to pole placement. If a transfer function of a system a system has a pole, λ, then λ is also an eigenvalue of the coefficient matrix, A, of that same system. Nevertheless, some eigenvalues of A could not be poles of the associated of the system. Those will be the resulting eigenvalues when the appropriate feedback control has been applied to the controllable, and observable, basic system.

Let's consider that system is defined by the vector equations:

$$\dot{x} = Ax + Bu \tag{3.4.2.3.1}$$

$$y = x \tag{3.4.2.3.2}$$

Where x є R^n, u є R^m and є R^n.

In here there is a problem when the values of some r all of the eigenvalues of matrix A, the dynamic response of the system is not acceptable. A feedback gain matrix, K, should be found to use in the control law:

$$u=Kx \tag{3.4.2.3.3}$$

Such the eigenvalues, γ_i, of the closed loop system will be placed precisely at specified locations.

The closed loop system is defined by:

$$\dot{x} = (A + BK)X = \bar{A}x$$

(3.4.2.3.4)

The eigenvalues, λ_i, are determined from the characteristic polynomial, f(γ):

$$f\ (\gamma\ ) \cong |\gamma I - A| = |\gamma I - A - BK|$$

(3.4.2.3.5)

''We can evaluate the polynomial of Ec (3.4.2.3.5), and the coefficients of that polynomial can then be equated with those of the desired polynomial, which is formed from the specified eigenvalues.''

3.5 Control System Design Method II

3.5.1 INTRODUCTION

There will be an AFCS problem if a set of specifications don't exist in the dynamic behavior of an aircraft. To achieve the required dynamic performance, we must use additional equipment together with the basic aircraft, in the most insignificant cases. All of these control system designs depend upon the interpretation of the dynamic response.

3.5.2 CONTROLLABILITY

Controllability is a property which described the effects of the control inputs to the changes in the state variables of some mathematical representation of the aircraft dynamics. It is said that the model of the aircraft dynamics is completely controllable if any and only if all initial state variables, $\mathbf{x}_i(0)$, are transferred in finite time, with the application of some control function, u(t), to any final state, x(T).

3.5.3 STABILIZABILITY

It is essential for AFCS that, if an aircraft is to be completely controlled, any unstable subspace must lie in a controllable subspace.

A system represented by $\dot{x} = Ax + Bu$ is considered as stabilizable if any vector, *x*, contained in its unstable subspace is also contained in its controllable subspace.

Any asymptotically stable system is obviously stabilizable, and any completely system must be stabilizable. For this reason, the pair (A, B) is stabilizable when $\dot{\mathbf{x}} = A\mathbf{x} + B\mathbf{u}$ is stabilizable."

3.5.4 RECONSTRUCTIBILITY AND OBSERVABILITY

A fundamental property of a linear system is whether, knowing the output from the system, the behavior of the state of the system can be determined. Knowing that the output is from Ec(3.4.2.3.2).

Where y є R^p, then the system will be completely reconstructibility as long as the row vectors of the reconstructibility matrix $\bar{R}$, order (n x'np), spans the space R^n, that is, $\bar{R}$ has rank n.

$$\bar{R} = \begin{bmatrix} C \\ CA \\ CA^2 \\ \cdot \\ \cdot \\ CA^{n-1} \end{bmatrix} \tag{3.5.4.1}$$

Equation (3.5.4.1) is the transpose of the observability matrix, namely:

$$V = R' = [C' : A'C' : ([A'])^2 C' : \ldots : (A')^{n-1} C'] \tag{3.5.4.2}$$

Observability means that we can determine the state vector x(0), at time t=0, from the output variables that happen in the future. In AFCS only output signals from the past are available.

For systems where only a single output is considered, R and V are squared matrices, of order (n x n).The conditions for reconstructibility and observability simply require that R and V be non-singular, which is equivalent to the condition that C[Si-A]' has no pole-zero cancellations.

3.5.5 THEORY OF LINEAR QUADRATIC PROBLEM

We have to determine an optimal control, u, to minimize the performance index, J, given by:

$$J = \frac{1}{2}\int_0^{\infty} (x'Qx + u'Gu)dt \qquad (3.5.5.1)$$

To control the aircraft whose dynamics are described by equation (3.4.2.3.1).

Where x ϵ R^n and u ϵ R^m. The matrices A and Q are of order (n x n), B is of order n $\times$ m, and G is of order (m$\times m$).

3.6 Stability Augmentation Systems

3.6.1 INTRODUCTION

About 1950, the term ‘stability augmentation system’ came into use the United States. At that moment, there was an American manufacturer called Northrop who was famous for its ‘flying wing’ aircraft, and was the best of those designs.

All similar systems have been called stability augmentation system (SASs) ever since, However; their purpose is still the same as it was at first: the values of a number of specific stability derivates of an aircraft must be increased through negative feedback control. Basically, the values of the stability derivates are calculated more effectively by using AFCs than by physical surfaces.

In general, SASs has to be with the control of a single mode of an aircraft’s motion. The general structure of such a SAS is shown in the block diagram of figure1, where we can see that there are four principal elements: aircraft dynamics, actuator dynamics and flight controller.

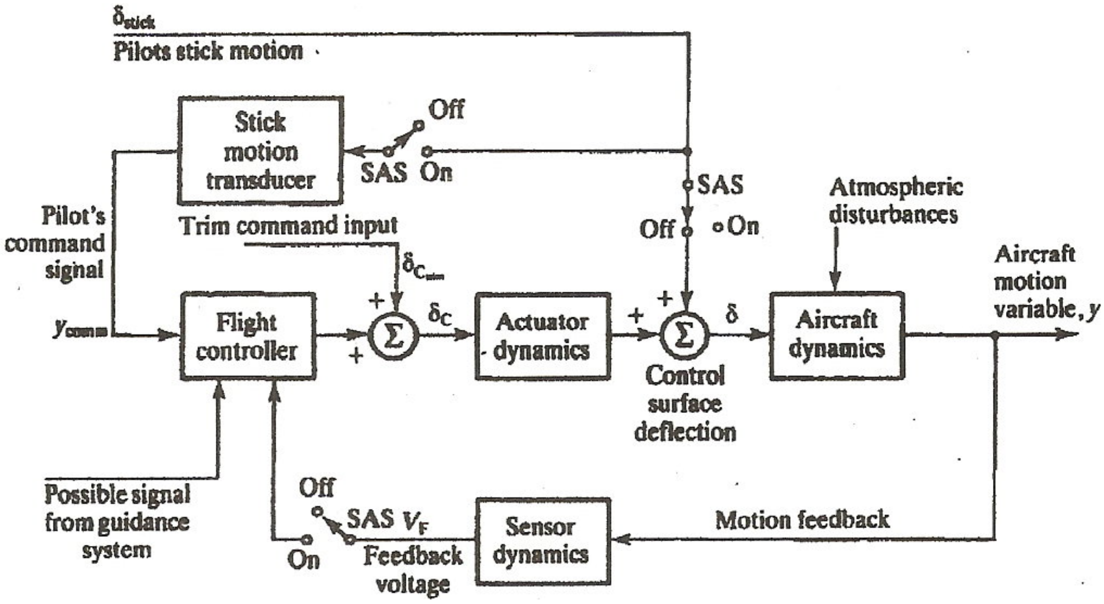


Figure 3.6.1.1 Stability Augmentation System (McLean 1990, p. 271)

If the SASs is switched off, the pilot can control the aircraft directly moving the appropriate control surface through his/her cockpit controls. The flight controller is not then active.

If the SAS is switched on, the control surface is driven by its actuator which the flight controller will be controlling.

"In SAS studies, command inputs are normally considered only secondarily; the flying qualities of the aircraft are enhanced by the control action of the feedback control system in a way that the effects of the atmospheric or other disturbances upon the aircraft's motion are suppressed. Sensor noise also affects the quality of control." (McLean 1990, p.270)

Nevertheless, a lot of types of SAS cannot be switched off by a pilot; they are active all the time, since the electrical system is on. If any failure occurs to the SAS, the aircraft has to be controlled from the pilot's cockpit controls. The main SAS function that we can find on modern aircraft is:

- Pitch rate SAS,
- Roll rate damper
- Yaw damper

3.6.2 ACTUATOR DYNAMICS

The Actuator which is used in combat and transport aircraft is electrohydraulic. Electric actuators can be used sometimes in commercial aviation. These actuators systems have their own dynamic characteristics which affect the performance of the closed loop SAS.

The characteristic of the actuator dynamics, which can sometimes have very important effects on the SAS performance, is the existence of non-linearities.

AFCS are not allowed to make use of the full range deflection of the control surfaces: it is usually limited to deflections $\mp$10%. Then the SAS is said to have 10% control authority.

3.6.3 SENSOR DYNAMICS

All sensors used in an AFCS are transducers. In modern aircraft, its objective is to measure motion variables and to produce output voltages currents corresponding to these motion variables. The sensors that are used in AFCS are chosen to have bandwidths and damping so that they can be instantaneous in their action, so the performance of the SAS is more affected by its location on the fuselage than by a sensor's dynamics.

3.6.4 LONGITUDINAL CONTROL (Use of elevator only)

In SASs, the controller output is the command voltage to the control surface actuator that gives the correct deflection.

The customary forms of feedback for AFCSs, and hence an SAS, are linear, i.e the control takes the form:

$$\boldsymbol{u} = K\,\boldsymbol{y} \qquad (3.6.4.1)$$

When the output vector y is defined as the state vector,

$$\boldsymbol{y} = \boldsymbol{x} \qquad (3.6.4.2)$$

(where C=Identity matrix)

Then, full state variable is involved

3.6.4.1 Pitch Rate SAS

The stability derivate that these systems try to augment is M_Q (change in pitching moment caused by a change in pitch rate). The block diagram of a typical, conventional pitch rate SAS is shown in Figure 2.

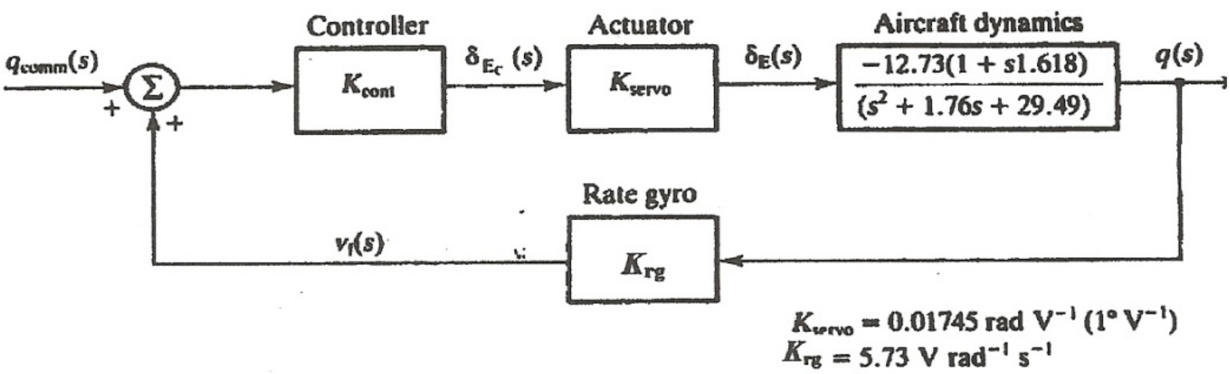


Figure 3.6.4.1.1 Stability Augmentation System (McLean 1990, p. 276)

''The feedback signal can be obtained from rate gyro used to measure the pitch rate q. AS there is a sign change present inherently in the aircraft dynamics associated with the relationship of pitch rate to elevator deflection, we add the feedback signal to the command signal, q_{comm} *."* (McLean 1990, p.276)

3.6.5 LATERAL CONTROL

As the simultaneous use of two independent control surfaces- the ailerons and the rudder – control lateral motion in conventional aircraft, lateral motion studies are more involved than those involving longitudinal motion only.

5.4.1The Yaw Damper

Not many aircrafts have such a degree of inherent damping of the dutch roll motion adequate to assure the handling qualities. That is why, when their rudders are used, the lack gives rise to oscillatory yawing motion, with some coupling into the rolling motion.

We show on figure 11 a block diagram of this yaw damper, using proportional feedback.

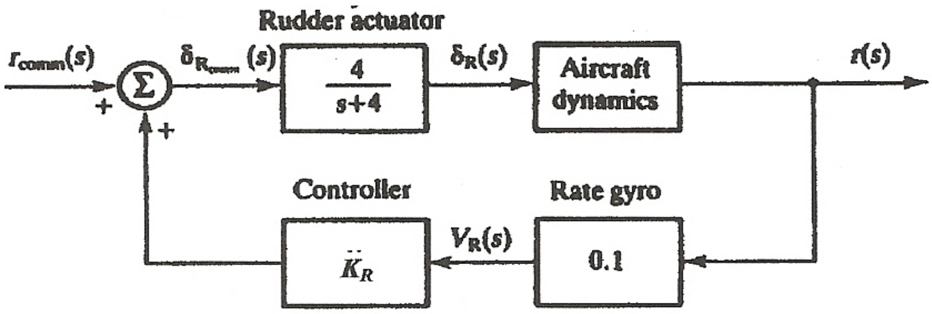


Figure 3.6.5.1.1 Yaw damper block diagram
(McLean 1990, p. 300)

This approximation is not as satisfactory as in the case of the pitch rate SAS, for instance, because the response of the rudder actuator is slower than the control surface actuators.

'The dynamic response of the uncontrolled aircraft Charlie-4 to an initial disturbance in roll rate of 1$^{\circ}s^{-1}$ is shown in Figure 3.6.5.1.2.

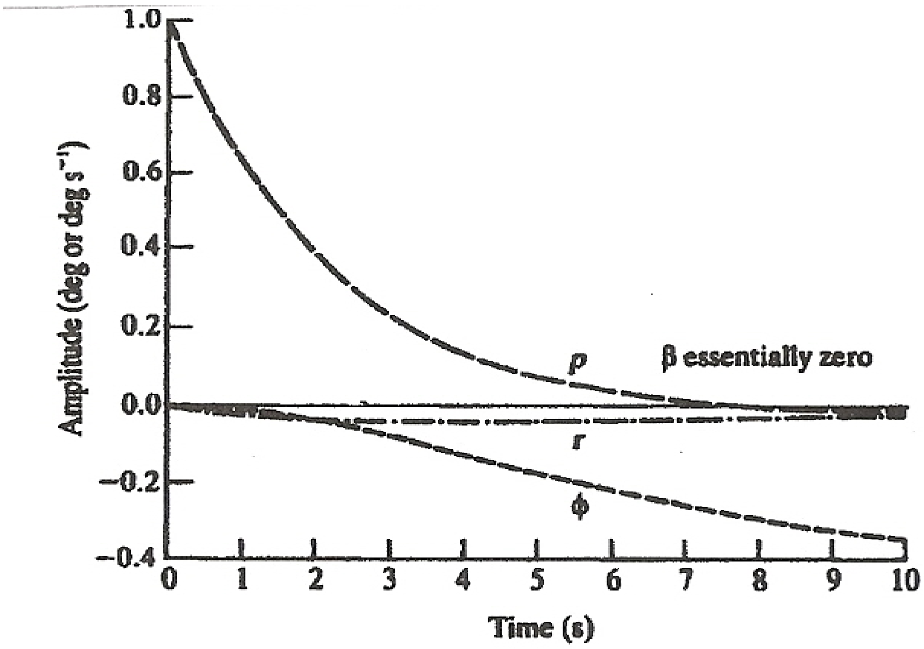


Figure 3.6.5.1.2 Response of uncontrolled aircraft to p(0)=1$^{\circ}s^{-1}$ (McLean 1990, p. 301)

3.6.5.2 The roll Rate Damper

When the roll performance of an aircraft is considered to be inadequate, we usually fit this type of AFCS, and this case is when the roll rate, in time, is too long. Consequently, the aileron actuator have a gain, k_{act}, and the rate gyro a sensitivity of k_p.

A block diagram of a typical roll rate damper is shown in Figure 3.6.5.2.1.

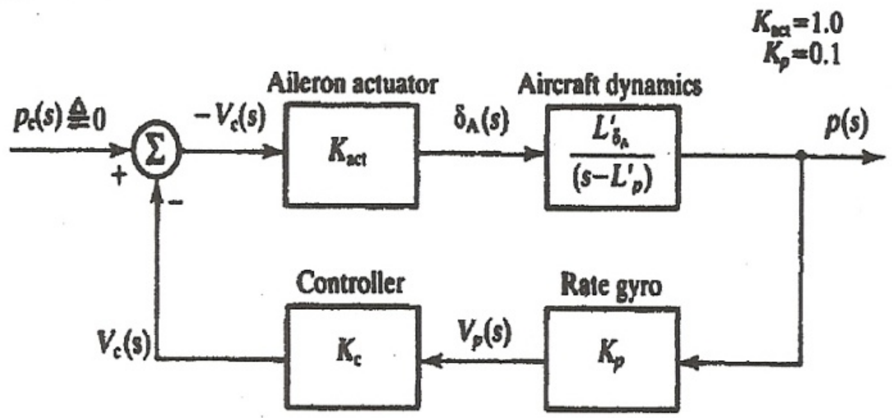


Figure 3.6.5.2.1 Roll rate damper block diagram. (McLean 1990, p. 307)

Figure 3.6.5.2.2 shows the corresponding roll rate response of a CHARLIE-4 obtained from the roll rate damper, with a value of controller gain of 30.0.

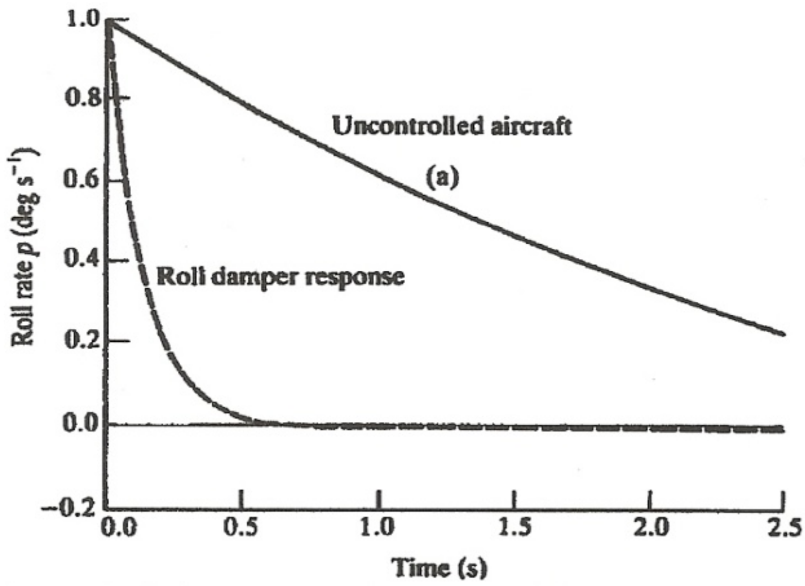


Figure 3.6.5.2.2 Roll damper and uncontrolled roll rate response block diagram (McLean 1990, p. 309)

It must be appreciated, however, that the roll rate damper does not affect the initial rolling acceleration available, although it does reduce the maximum roll rate which the aircraft can produce.

3.7 Attitude Control Systems

On modern aircraft, attitude control systems find extensive employment. They form the main functions of any AFCS, which allow an aircraft to be placed and maintained in any required, specified orientation in space, both in direct response to a pilot's command, and in response to command signals from an aircraft 's guidance.

3.7.1 PITCH ATTITUDE CONTROL SYSTEMS

Pitch attitude control systems have the use of elevator only as the control in the system. In figure 3.7.1.1, a block diagram of a typical system is shown.

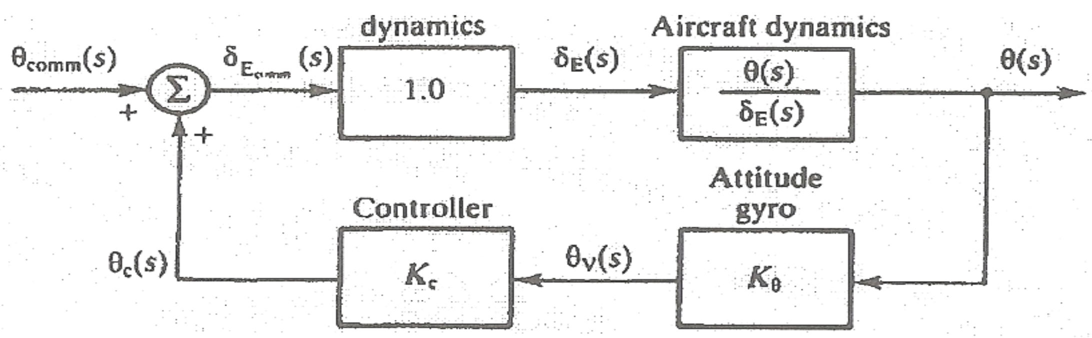


Figure 3.7.1.1 Block diagram of a Pitch Attitude Control System
(McLean 1990, p. 318)

The feedback gain, $k_\theta k_q$, is increased. The feedback signals, which are used in an AFCS for longitudinal motion, depend solely on motion variables derivatives $X_u, Z_w, M_\alpha, or\ M_q,$ then the total damping of the system does not change with the application of feedback. The response of a pitch attitude control system used for FOXTROT-2 is shown in figure2.

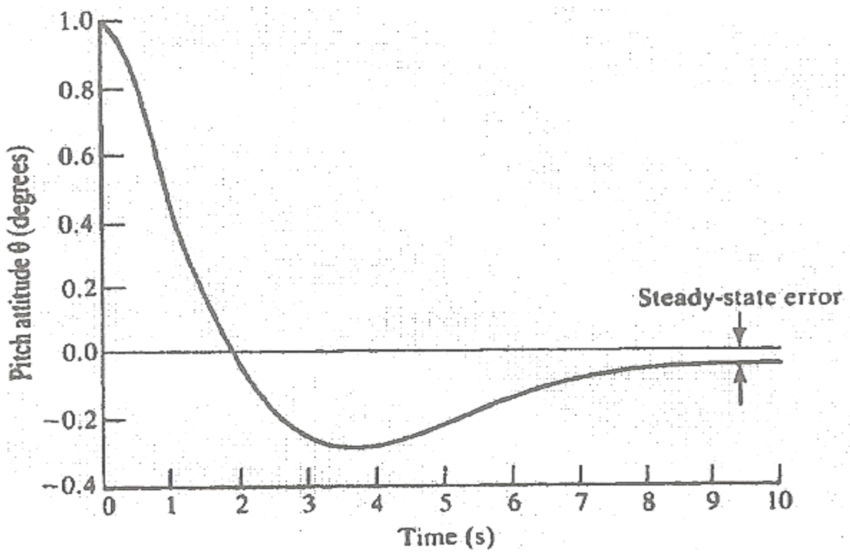


Figure 3.7.1.2 Response of a Pitch Attitude Control System(FOXTROT-2). (McLean 1990, p. 319)

3.7.2 ROLL ANGLE CONTROL SYSTEM

We control roll angle by the ailerons at low-to-medium velocity on all types of aircraft; At high speed (spoilers are used on military aircraft).

''Roll control for swing-wing aircraft is generally produced by controlling surfaces, moving differentially, and located at the tail. Swing –wings usually contain spoilers to increase the roll control power of the tail surfaces.'' (McLean 1990, p.323)

Getting the degree of dynamic stability desired in roll requires the use of roll attitude control system, which is a feedback control system that control the roll attitude in the presence of disturbances to have a better response from the pilot or guidance system. A block diagram which represents a typical system is shown a Figure 3.7.2.1.

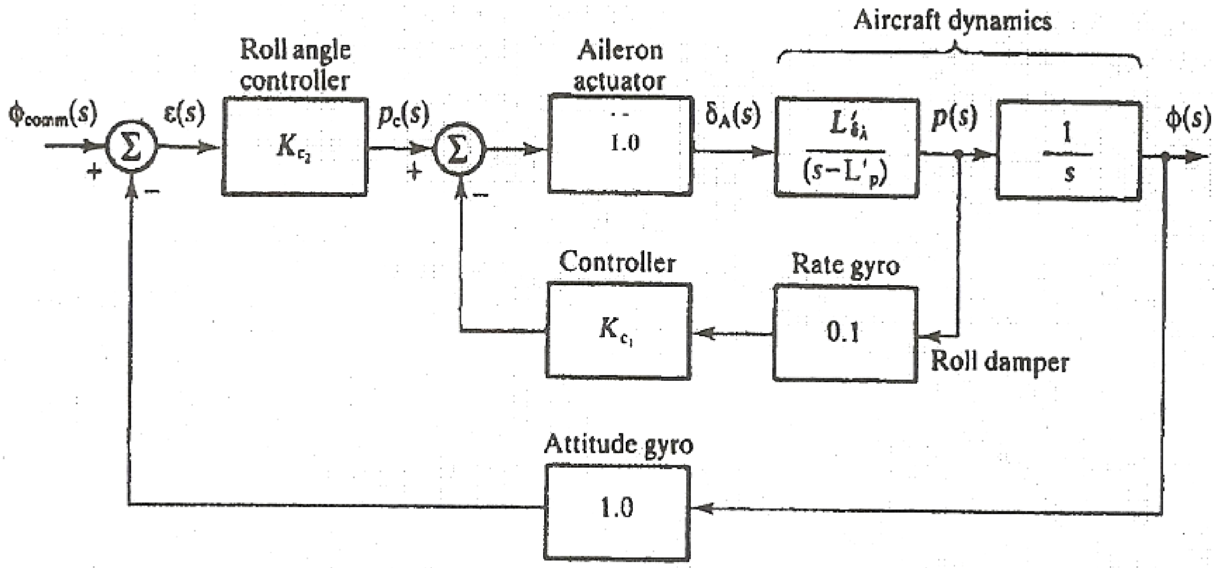


Figure 3.7.2.1 Bank angle control system with roll rate inner loop damper. (McLean 1990, p. 329)

The actuator dynamics are represented by a transfer function of unity. With this method we require a rate gyroscope, to use in the roll damper.

3.7.2.1 Some Problems arising with Roll control

"In fighter aircraft, the pilot usually controls the roll angle indirectly through a CSAS, a commanded role damper, since such CSASs are necessary to assist the aircraft to provide the rapid roll performance which is essential for modern aerial combat, or for evasive manouvres during low level strike missions. To achieve the performance required inevitably means the use of high loop gains. Such high values of gain cause a number of problems, although it is worth noting that the gains of such CSASs are often fixed throughout the flight envelope. Among the problems are the following:

1. *The command signal from the pilot must usually be 'damped'. If the input signal to the CSAS corresponding to a small deflection of the pilot's stick is too large them the pilot-induced oscillations may result.*
2. *When the speed of the aircraft is low, and the dynamic pressure is relatively small, such as during a landing approach, the response of the aircraft is sluggish.*
3. *A system with a too high value of loop gain precludes control of bank angle by use of the rudder, which is a technique used by pilots in making S-turns during landing, or during manoeuvres in aerial combat.*
4. *On swept-wing aircraft, as the stall condition is approached, it is essential to reduce the value of the loop gain by a substantial amount to avoid very large deflections of the control surfaces."* (McLean 1990, p.330)

3.8 Flight Path Control System

Some flight missions require that an aircraft is made to follow with great precision some specially defined path. For fixed-wing aircraft there are four positioning tasks that have to be performed vey accurately. These tasks are: air-to-ground weapons delivery, air-to-air combat, in flight refueling and all-weather landing.

Conventional aircraft' have no special control surfaces to allow the control of translation in both the normal and lateral directions. Thus, the reduction of an inadvertent lateral displacement from some desired track, for instance, has to be achieved indirectly by means of a controlled change of aircraft heading.

3.8.1 HEIGHT CONTROL SYSTEMS

''When a system is used to control the height at which an aircraft is flying, it acts as a feedback regulator to maintain the aircraft's height at a reference (or set) value, even in the presence of disturbances. The pilot can either fly the aircraft by manual control or use the pitch attitude control system to control the climb (or descent) of the aircraft until it has reached the required height. When that height has been reached, the height control system is selected to maintain that height thereafter. For a Supersonic transport (SST) aircraft, a height system is a necessity.'' (McLean 1990, p.359)

3.8.1.1 Height Hold system

Figure 3.8.1.1.1 shows the block diagram of a height hold system.

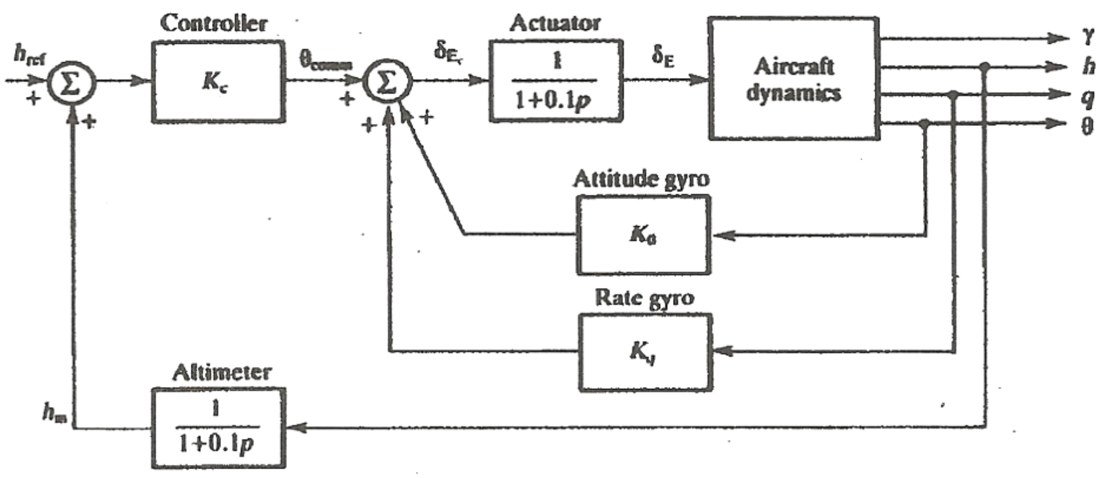


Figure 3.8.1.1.1 Height hold System. (McLean 1990, p. 362)

It represents a pitch attitude control system; with a pitch rate SAS as its inner loop. An outer loop, which means the use of an altimeter to give a feedback signal proportional to height, is used to get the height hold function. The good dynamic can be seen from Figure 3.8.1.1.2, and trying other values of the controller gain to improved dynamic response, is shown in Figure 3.8.1.1.3.

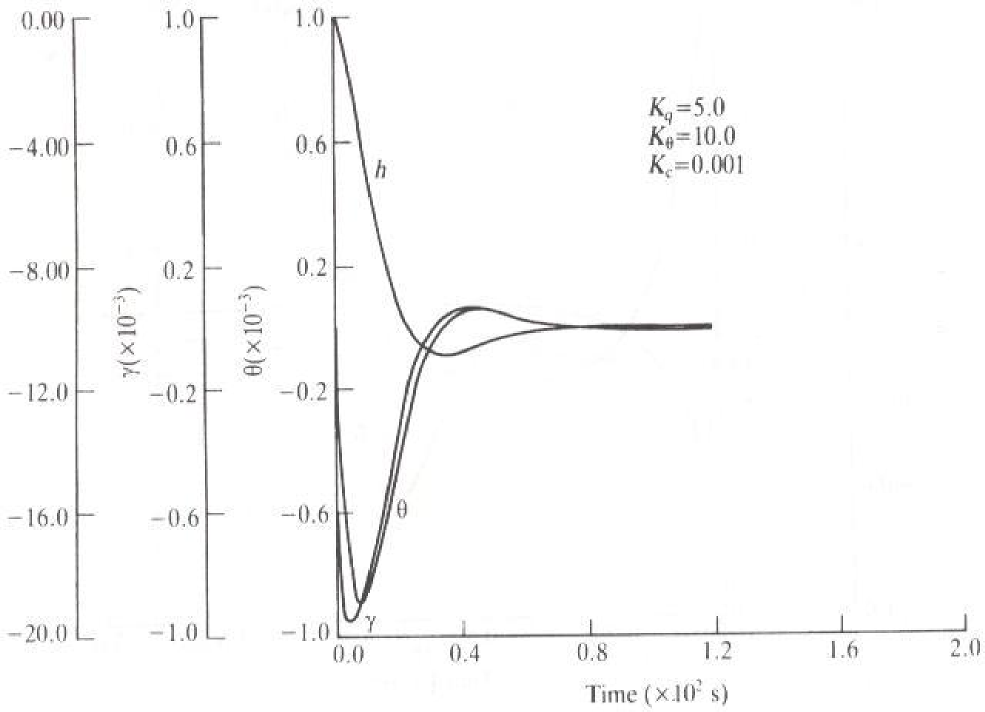


Figure 3.8.1.1.2 Response of height hold system (McLean 1990, p. 362)

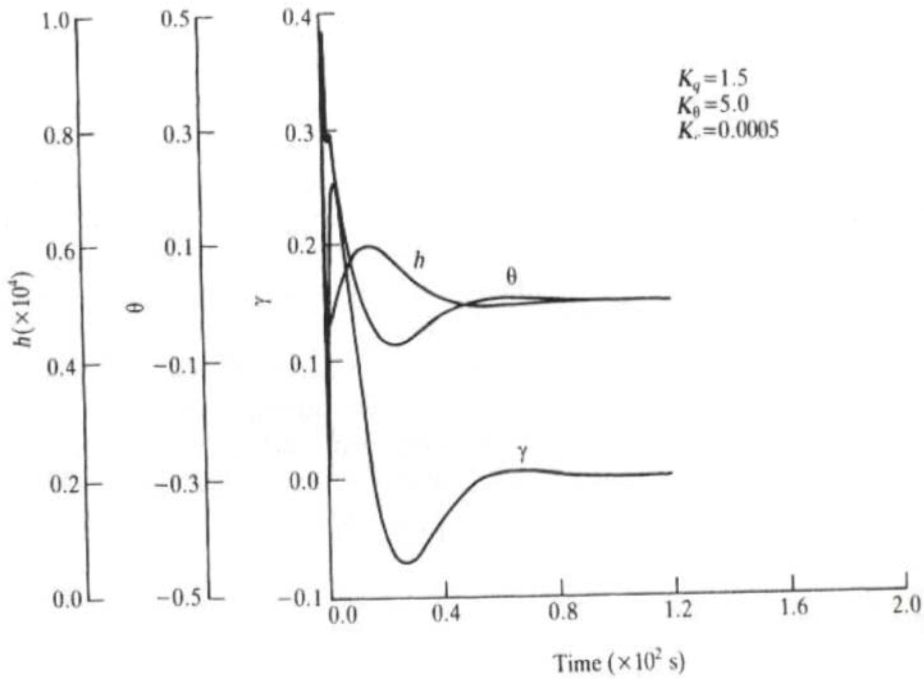


Figure 3.8.1.1.3 Commanded step response of height hold system (McLean 1990, p. 363)

3.8.1.2 Some problem using the Height Control Systems

Once of the most difficulty design problems likely to meet in this type of the system relates to the 'backside' parameter, a, namely:

$$a = \frac{1}{m}\left(\frac{\partial D}{\partial u} - \frac{\partial T}{\partial u}\right) \qquad (3.8.1.2.1)$$

Where *D* represents the aircraft's drag force, and *T* is its thrust.

The parameter *a* is one of the zeros of the transfer function relating height to elevator deflection. In certain aircraft, a performance reversal can arise (on backside of the power curve) in which $\frac{\delta T}{\partial u} \gg \frac{\partial D}{\partial u}$; *a* is then negative. When this happens it is difficult to find a suitable value for the gain of the controller to assure stability of the height hold system. In that case a more complex form of control law than the simple proportional feedback control being used in these two systems is required.'''

3.8.2 HEADING CONTROL SYSTEM

The heading angle, of an aircraft is defined by:

$$\lambda = \beta + \psi \qquad (3.8.1)$$

The direction control system operated by means of coordinated turns, thereby ensuring that the sideslip angle, β, was effectively zero. Nevertheless, the turning manoeuvre requires the use of the ailerons. When there is rudder use, then it would seem that the yaw angle, ψ, could be controlled by means of a yaw damper system, and with enough sideslip suppression could give the basis of a heading control system.

"A heading control system, with a block diagram like the one in Figure 3.8.2.1, can be considered . For CHARLIE-4, and using the yaw damper we can show that if the state vector is defined as" (McLean 1990, p.377)

$X' = [\beta \;\; p \;\; r \;\; \varphi \;\; \psi \;\; \delta_R \;\; e_{WO}]$

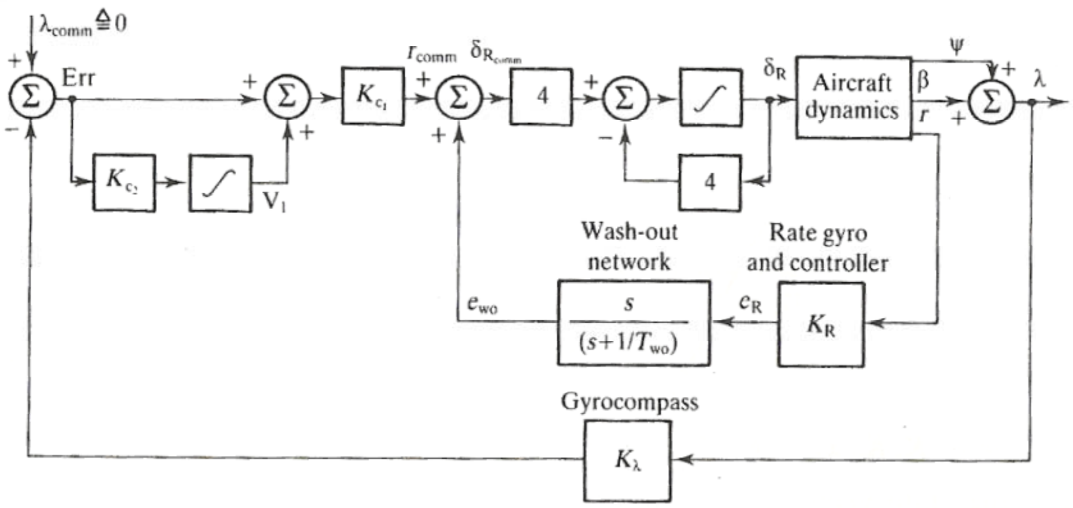


Figure 3.8.2.1 Heading Control System (McLean 1990, p. 377)

3.8.3 VOR-COUPLED AUTOMATIC TRACKING SYSTEM

Getting automatic tracking of an aircraft's lateral path needs the use of navigation systems. The VOR system is one of the most popular and effective of these systems. It is used together with DME (distance measure equipment) and with both working, rho/theta navigation system. VOR gives the bearing (θ) information.

The VOR system operates in a range of 108-135 MHz; DME operates at UHF, in a range 960-1215 MHz.

The beams width of the VOR transmissions is relatively coarse, being about $\mp$ 10º.

The VOR guidance can be regarded as a accurate for the reception range which, because the transmission is VHF, is line-of-sight, i.e. about one hundred miles. But as an aircraft nears a particular transmitter the system inherently becomes more sensitive. It can easily be understood, from studying Figure 11.30(a) why this comes about. Obviously the greater the displacement, d, from the beam's centre-line the greater is the error, г, as range reduces.

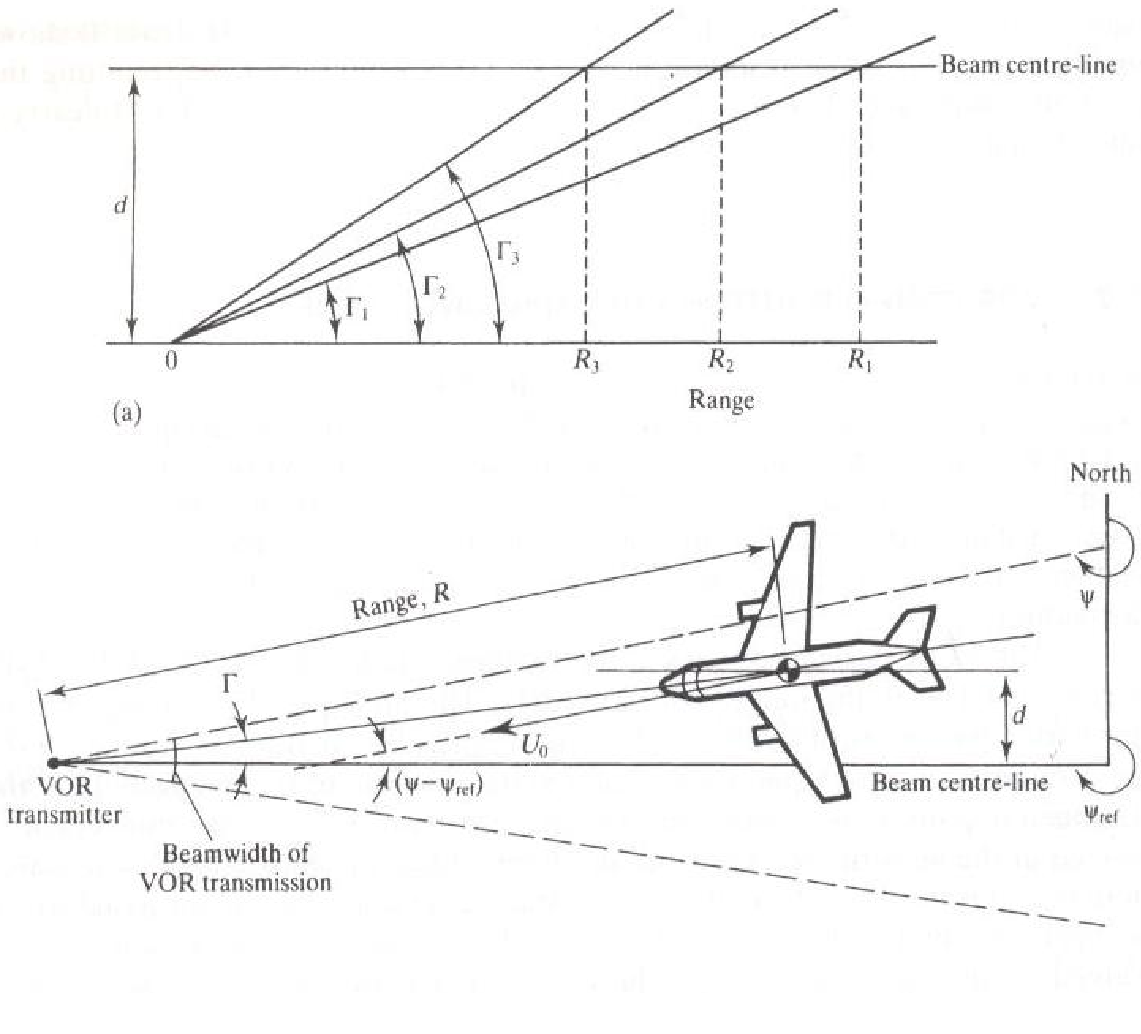


Figure 3.8.3.1 a) Change of error angle with range for fixed displacement.
b) Geometry of VOR system (McLean 1990, p. 382)

A block diagram of VOR- coupled system is shown in figure 11.32

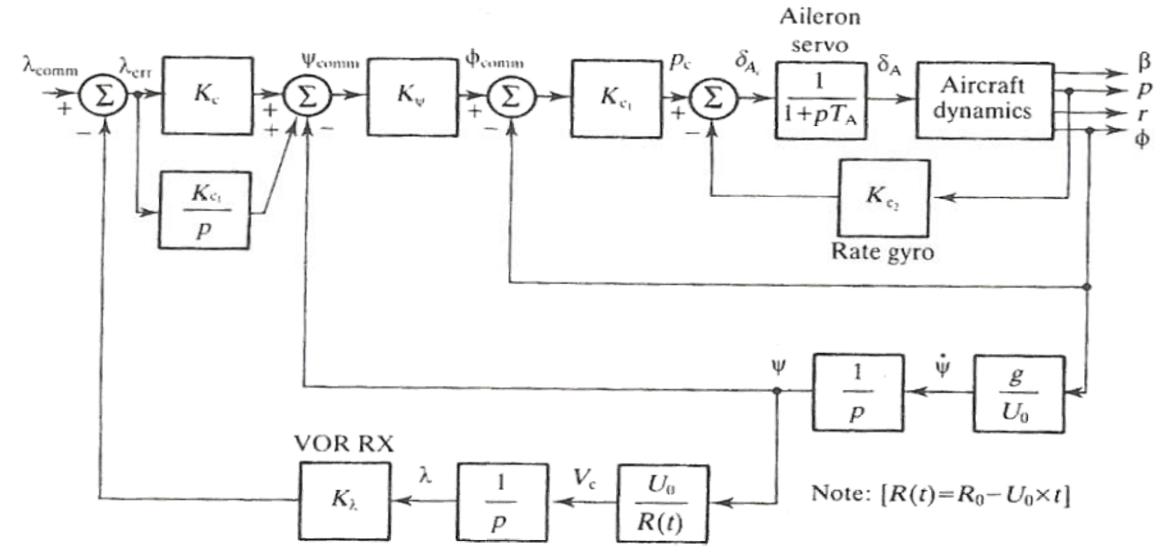


Figure 3.8.3.2 Block diagram of VOR- coupled system. (McLean 1990, p. 384)

3.8.4 ILS LOCALIZER-COUPLED CONTROL SYSTEM

''ILS equipment is located only at airports where the runway length is greater than 1800 m. The ILS involves a lot of of independent low-power radio transmissions:

1. The localizer which gives information to an aircraft and tells it whether it is flying to the left or the right of the centre-line of the runway towards which it is heading

2. The glide path which gives an aircraft information about whether it is flying above or below a preferred decent path.

3. Marker beacons which indicate to an aircraft its precise situation at fixed points from the runway threshold.'' (McLean 1990, p.386)

A representation of the transmission features of the ILS localizer and glide path systems is shown as Figure 9 and the situations of the transmitter and aerial systems related to the runway are shown in Figure 10.

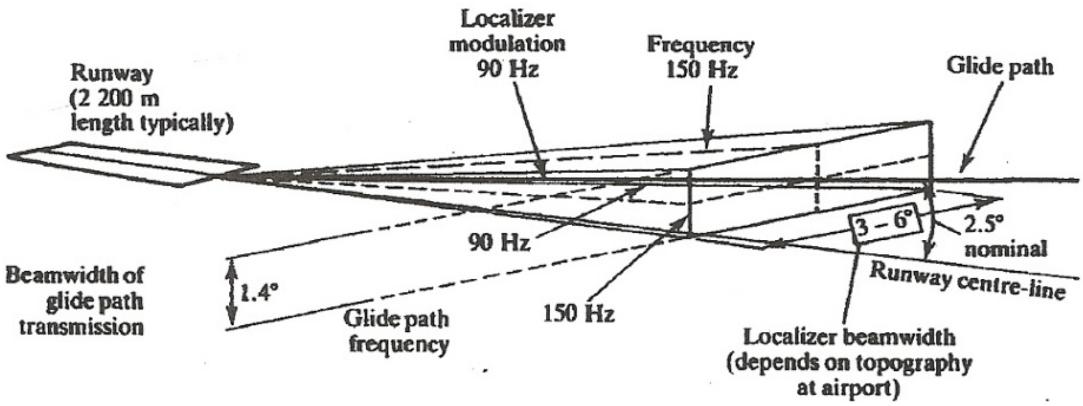


Figure 3.8.4.1 ILS localizer and glide slope transmissions. (McLean 1990, p. 387)

3.8.5 ILS GLIDE-PATH-COUPLED CONTROL SYSTEM

This is a system which uses the output signal from the airborne glide path receiver as a guidance command to the attitude control system of the aircraft. The loop is closed via the aircraft kinematics which changes the pitch attitude of the aircraft into a displacement to the preferred descent path (the glide path) into the airport.

To understand better in geometric terms, in order to'' the situation shown in Figure 11.38 the aircraft's flight path angle is less than 2.5º, therefore r is positive(note that r= $\gamma + 2.5°$),so is representing the case when the aircraft is approaching the glide path.

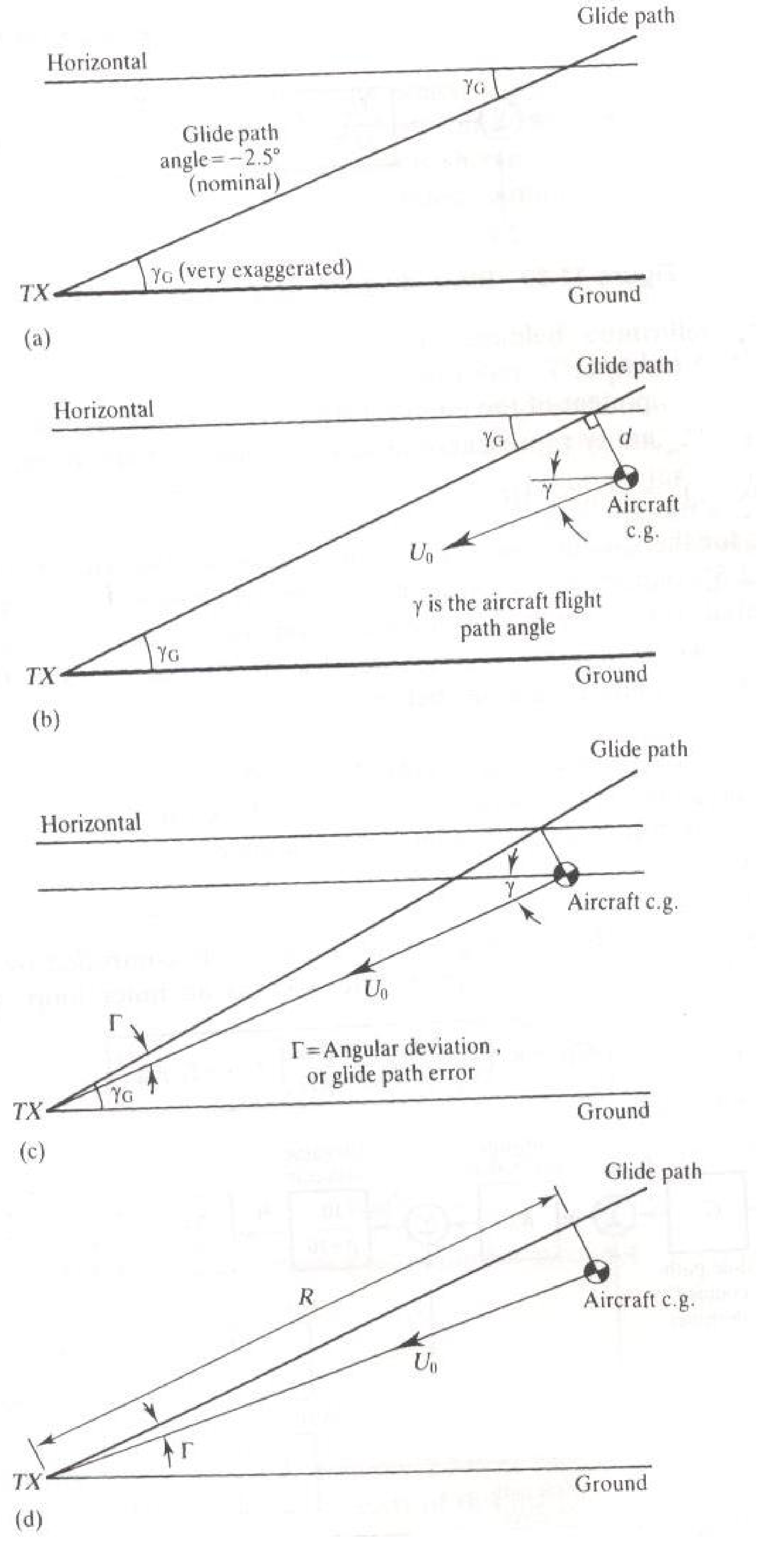


Figure 3.8.5.1 a) The glide path geometry.
b) Aircraft below glide path-geometry.
c) Angular deviation from the glide path-geometry.
d) Slant range definition (McLean 1990, p. 391)

The aircraft flight path angle γ, is defined by:

$$\gamma = (\theta - \alpha) \qquad (3.8.5.1)$$

Thus, the flight path angle is most effectively controlled by using a pitch attitude control system with a pitch rate SAS as an inner loop.

The block diagram of a typical glide path control systems is shown in Figure 3.8.5.2.

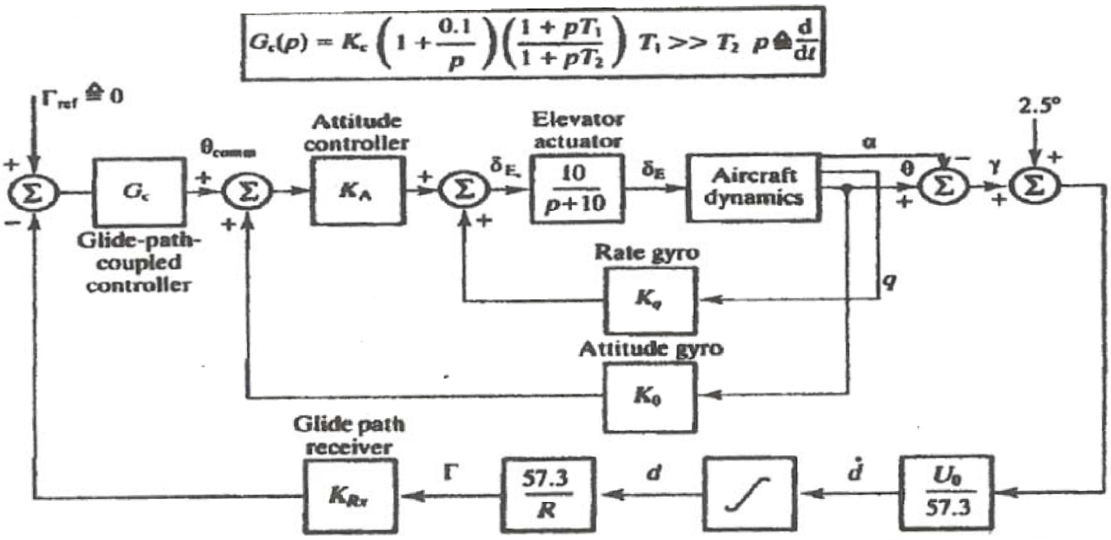


Figure 3.8.5.2 Glide-Path –Coupled control system (McLean 1990, p. 392)

A typical set of parameters for a CHARLIE-1 is shown in Figure 3.8.5.3.

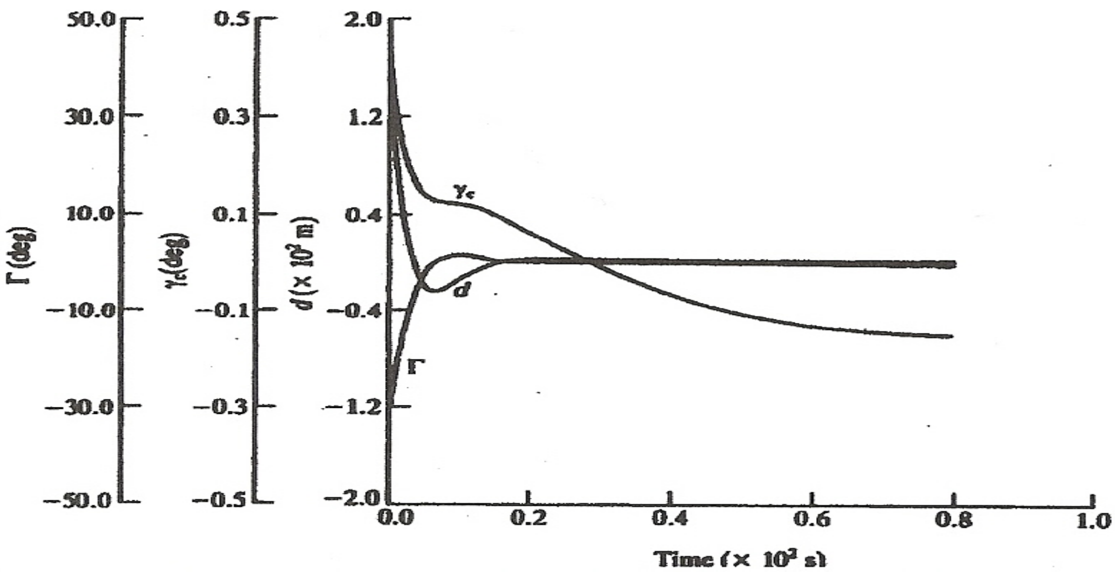


Figure 3.8.5.3 Response of Glide-Path –Coupled control system (McLean 1990, p. 394)

It is clear the Effectiveness of the system restoring the aircraft.

The treatment above supposes that the glide slope receiver is located at c.g. of the aircraft, and measures the aircraft's angular derivation from the glide path at c.g. However, if the aircraft receiver is installed at, say, the nose of the aircraft, the dynamics of the system are affected, as follows. From Figure 3.8.5.4, the height measured at the receiver is:

$$h_A \approx h + x_A\theta \qquad (3.8.5.2)$$

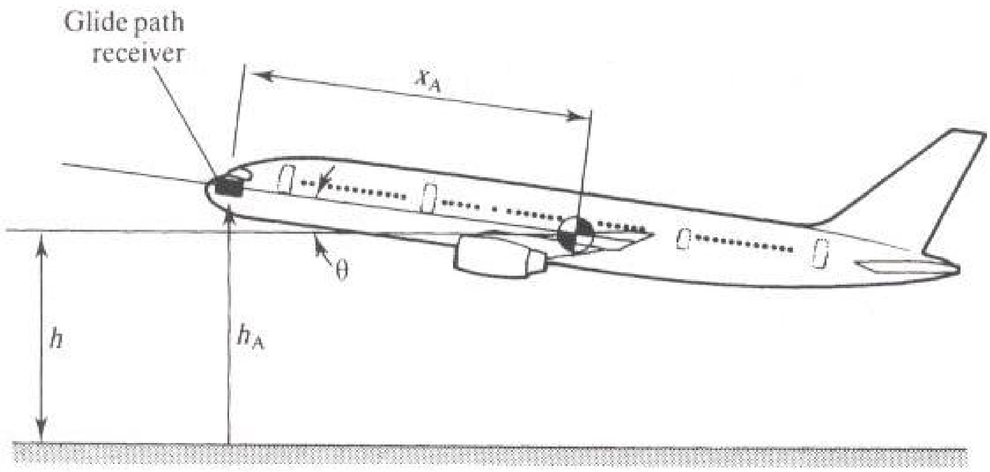


Figure 3.8.5.4 Glide path receiver located in aircraft nose. (McLean 1990, p. 398)

3.8.6 AUTOMATIC LANDING SYSTEM

It uses the ILS, and the whole automatic landing segment is made up of a number of phases which are shown in Figure 16.

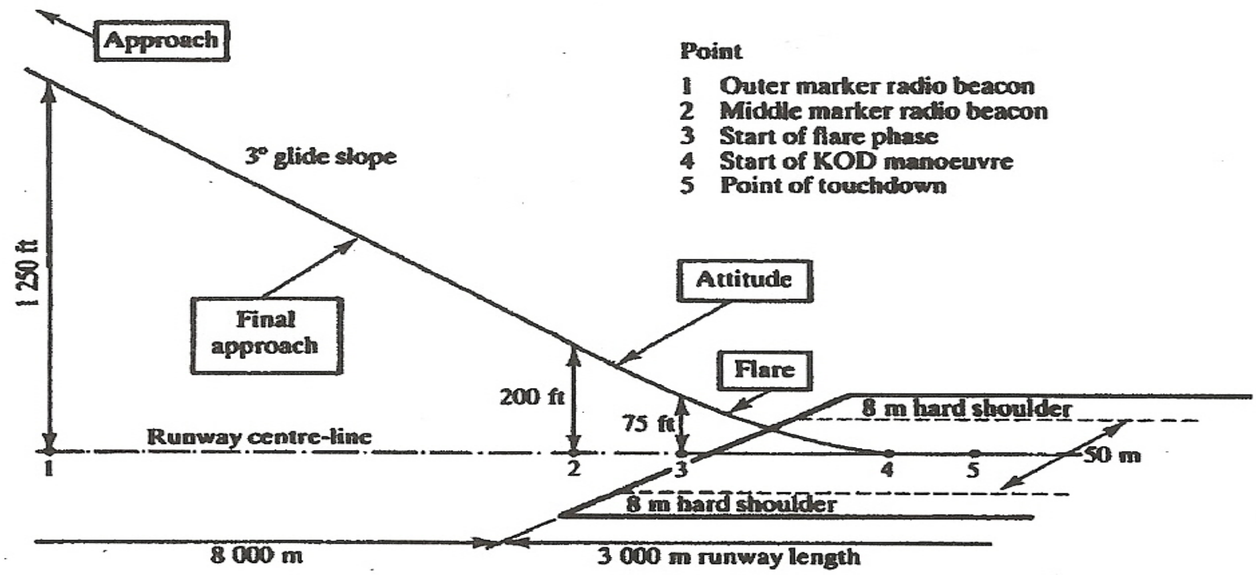


Figure 3.8.6.1 BLEU aircraft automatic landing (McLean 1990, p. 399)

What distinguish landings into the different categories are the conditions of visibility, which are summarized in Figure 3.8.6.2.

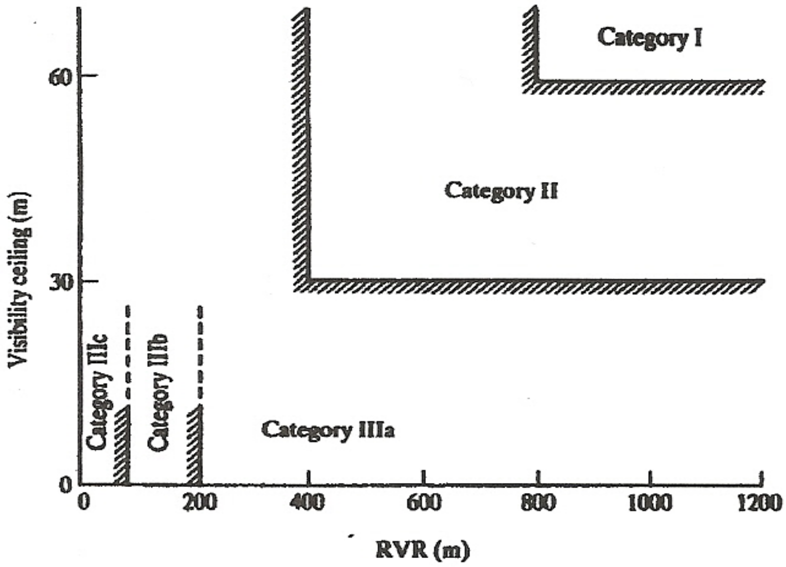


Figure 3.8.6.2 Definition of landing categories. (McLean 1990, p. 400)

4. Design Process I (previous calculations)

4.1 EQUATION OF MOTION

First of all in a design of an AFCS for an aircraft in this case a CHARLIE (a very large, four-engined, passenger jet aircraft), we have some specific parameters that is shown in Appendix A for this aircraft.

4.1.1 Longitudinal motion

In the next equations has been in account the first flight condition for the longitudinal motion, based in Equation (3.2.2.1) but we can easily calculate for each condition:

$$\dot{u} = 0.021u + 0.122\alpha - 9.8\theta$$

$$\dot{\alpha} = -2.98 \times 10^{-8}u - 0.512\alpha + q - 0.029\delta_E$$

$$\dot{q} = 0.000036u - 0.402\alpha - 0.0536\dot{\alpha} - 0.357q - 0.378\delta_E$$

$$\dot{\theta} = q$$

So we have the next states spaces matrices for the longitudinal motion:

$$A = \begin{bmatrix} X_u & X_w & 0 & -g\cos\gamma_0 \\ Z_u & Z_w & U_0 & -g\sin_0 \\ M_u + M_w & M_w + M_w & M_q + U_0 M_w & -g\sin\gamma_0 \\ 0 & 0 & 1 & 0 \end{bmatrix}; B = \begin{bmatrix} 0 \\ Z_{\delta E} \\ M_{\delta E} + Z_{\delta E} M_w \\ 0 \end{bmatrix}$$

Later it has to be substituting the values in the first condition from appendix A:

$$A = \begin{bmatrix} -0.021 & 0.122 & 0 & -9.8 \\ -0.2 & -0.512 & 67 & 0 \\ 1.96 \times 10^{-4} & -5.59 \times 10^{-4} & -0.4106 & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix}; B = \begin{bmatrix} 0 \\ -1.96 \\ -0.3764 \\ 0 \end{bmatrix} \qquad (4.1.1.1)$$

4.1.2 Lateral motion

In the lateral equations we have the next equations for the first condition:

$$\dot{v} = -0.089\beta - r - 0.146c\theta + Y^*_{\delta_a}\,\delta_a + 0.015\delta_r$$

$$\dot{p} = -1.33\beta - 0.98p + 0.33r + 0.23\delta_a + 0.06\delta_r$$

$$\dot{r} = 0.17\beta - 0.17p - 0.217r + 0.026\delta_a - 0.15\delta_r$$

$$\dot{\Phi} = p$$

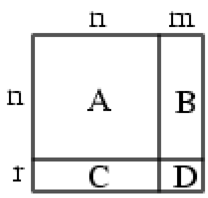
Consequently we have:

$$A = \begin{bmatrix} Y_v & 0 & -1 & -\frac{g}{U_0 cos\theta_0} \\ L'_\beta & L'_p & L'_r & 0 \\ N'_\beta & N'_p & N'_r & 0 \\ 0 & 1 & 0 & 0 \end{bmatrix}; B = \begin{bmatrix} 0 & 0 \\ L'_{\delta A} & L'_{\delta R} \\ N'_{\delta A} & N'_{\delta R} \\ 0 & 0 \end{bmatrix}$$

So if we substituting the values for the first condition we have:

$$A = \begin{bmatrix} -0.089 & 0 & -1 & 0.146 \\ -1.33 & -0.98 & 0.33 & 0 \\ -0.17 & -0.17 & -0.217 & 0 \\ 0 & 1 & 0 & 0 \end{bmatrix}; \quad B = \begin{bmatrix} 0 & 0.015 \\ 0.23 & 0.06 \\ 0.026 & -0.15 \\ 0 & 0 \end{bmatrix} \qquad (4.1.2.1)$$

Once we have the A matrix and the B matrix we can calculate C in order of the outputs that we can have in the simulation, and the D which shows the number of inputs in relation with the B matrix, the next figure is a help to understand the process;



4.2 DEVELOP SUITABLE MODELS (STATE SPACE AND TRANSFER FUNCTIONS) TO DESCRIBE THE LONGITUDINAL FLIGHT DYNAMICS OF THE AIRCARFT.

In small perturbation flying, the angle of attack is considered very small, so the relationship between α and w is easy to find:

$$\alpha = {}^{w}/_{U_0} \quad \text{and} \quad \dot{\alpha} = {}^{\dot{w}}/_{U_0} \tag{4.2.1}$$

The short-period case implies $u = \dot{u} = 0$, and $\gamma_0 = 0$. Therefore, can be shown as an easy-looking version, assuming $Z_w = Z_\alpha$ and $M_w U_0 = M_\alpha$:

$$\dot{\alpha} = Z_\alpha \alpha + q + \frac{Z_{\delta_E}}{U_0}\delta_E \tag{4.2.2}$$

$$\dot{q} = (M_\alpha + M_{\dot{\alpha}} Z_\alpha)\alpha + (M_{\dot{\alpha}} + M_q)q + \left(M_{\dot{\alpha}}\frac{Z_{\delta_E}}{U_0} + M_{\delta_E}\right)\delta_E \tag{4.2.3}$$

Generally, **the state-variable model** of a system is defined as below

$$\dot{\mathbf{x}} = A\mathbf{x} + B\mathbf{u} \tag{4.2.4}$$

$$\mathbf{y} = C\mathbf{x} + D\mathbf{u} \tag{4.2.5}$$

Where *A*, *B*, *C* and *D* are matrixes. For **u** is the input vector while $\dot{\mathbf{x}}$ and **x** are state-variable, so is called state equation. While **y** is the output, so is defined as output equation

Which can also be written as below:

$$\begin{bmatrix}\dot{\alpha}\\ \dot{q}\end{bmatrix} = \begin{bmatrix} Z_\alpha & 1 \\ M_\alpha + M_{\dot{\alpha}} Z_\alpha & M_{\dot{\alpha}} + M_q\end{bmatrix}\begin{bmatrix}\alpha\\ q\end{bmatrix} + \begin{bmatrix}\frac{Z_{\delta_E}}{U_0}\\ M_{\dot{\alpha}}\frac{Z_{\delta_E}}{U_0} + M_{\delta_E}\end{bmatrix}\delta_E$$

Where;

$$\dot{\mathbf{x}} = \begin{bmatrix}\dot{\alpha}\\ \dot{q}\end{bmatrix}, \quad \mathbf{x} = \begin{bmatrix}\alpha\\ q\end{bmatrix}, \quad A = \begin{bmatrix} Z_\alpha & 1 \\ M_\alpha + M_{\dot{\alpha}} Z_\alpha & M_{\dot{\alpha}} + M_q\end{bmatrix} \text{ and } \quad B = \begin{bmatrix}\frac{Z_{\delta_E}}{U_0}\\ M_{\dot{\alpha}}\frac{Z_{\delta_E}}{U_0} + M_{\delta_E}\end{bmatrix}$$

While for the input vector, $\mathbf{u} \triangleq \delta_E$. So is the state equation of the longitudinal small perturbed motion.

In this particular task, to describe the longitudinal flight dynamics effectively, transfer functions in Laplace domain are essential. Generally, it can be expressed as below:

$$G(s) = {}^{y(s)}/_{U(s)} = C[sI - A]^{-1}B \tag{4.2.6}$$

Where $I = \begin{bmatrix} 1 & 0 \\ 0 & 1 \end{bmatrix}$.

So, finally, we the transfer function of angle of attack is:

$$\frac{\alpha(s)}{\delta_E(s)} = \frac{\frac{Z_{\delta_E}}{U_0} s + M_{\delta_E} - \frac{Z_{\delta_E}}{U_0} M_q}{s^2 - (Z_\alpha + M_{\dot{\alpha}} + M_q)s + (M_q Z_\alpha - M_\alpha)}$$

With the same principle, the transfer function for the pitch rate is shown as below:

$$\frac{q(s)}{\delta_E(s)} = \frac{\left(M_{\dot{\alpha}} \frac{Z_{\delta_E}}{U_0} + M_{\delta_E}\right)s - Z_\alpha M_{\delta_E} + \frac{Z_{\delta_E}}{U_0} M_\alpha}{s^2 - (Z_\alpha + M_{\dot{\alpha}} + M_q)s + (M_q Z_\alpha - M_\alpha)}$$

4.3 LONGITUDINAL SHORT-PERIOD FLYING QUALITIES

To understand better how the flying qualities affect in the calculations of motions it is good to see in two degrees (short-period).

The dynamic stability of small perturbation case of longitudinal motion is analyzed from the eigenvalues λ_i of the coefficient matrix *A*. The aircraft is dynamic stable if all the eigenvalues of *A* are real and negative, or complex and their real part negative (see section 3.4.2.3). As long as there is one eigenvalue of *A* is zero or positive for a real number, or its real part for a complex number, the system is unstable. Fortunately, we can use a MATLAB command 'eig(A)'.

4.3.1 For the 1st condition:

With all the parameters from the appendix, we get:

$$A = \begin{bmatrix} Z_\alpha & 1 \\ M_\alpha + M_{\dot{\alpha}} Z_\alpha & M_{\dot{\alpha}} + M_q \end{bmatrix} = \begin{bmatrix} -0.0272 & 1 \\ 0.0132 & -1.149 \end{bmatrix}$$

Then we get the eigenvalues by MATLAB:

$$\lambda_1 = -0.0156 \quad \text{and} \quad \lambda_2 = -1.1606$$

Both λ_1 and λ_2 are less than zero, so in this condition, aircraft is stable for short-period longitudinal motion. We can get the natural frequency and damping ratio:

$$\omega_{sp} = 0.1344 \text{ rad s}^{-1} \quad \text{and} \quad \zeta_{sp} = 4.3757$$

4.3.2 For the 2nd condition:

$$A = \begin{bmatrix} Z_\alpha & 1 \\ M_\alpha + M_\alpha Z_\alpha & M_\alpha + M_q \end{bmatrix} = \begin{bmatrix} -0.023 & 1 \\ 0.0115 & -1.3 \end{bmatrix}$$

Then the eigenvalues are:

$$\lambda_1 = -0.0141 \quad \text{and} \quad \lambda_2 = -1.3089$$

Since they are all negative, so this condition is also stable, and ω_{sp} as well as ζ_{sp} are:

$$\omega_{sp} = 0.1356 \text{ rad s}^{-1} \quad \text{and} \quad \zeta_{sp} = 4.8783$$

4.3.3 For the 3rd condition:

$$A = \begin{bmatrix} Z_\alpha & 1 \\ M_\alpha + M_\alpha Z_\alpha & M_\alpha + M_q \end{bmatrix} = \begin{bmatrix} -0.016 & 1 \\ 0.0053 & -1.231 \end{bmatrix}$$

Consequently, the eigenvalues are:

$$\lambda_1 = -0.0117 \quad \text{and} \quad \lambda_2 = -1.2353$$

As shown, the 3rd condition is stable. Then we can get ω_{sp} and ζ_{sp} correspondingly:

$$\omega_{sp} = 0.1200 \text{ rad s}^{-1} \quad \text{and} \quad \zeta_{sp} = 5.1958$$

4.3.4 For the 4th condition:

$$A = \begin{bmatrix} Z_\alpha & 1 \\ M_\alpha + M_\alpha Z_\alpha & M_\alpha + M_q \end{bmatrix} = \begin{bmatrix} -0.008 & 1 \\ -0.0066 & -0.315 \end{bmatrix}$$

According to this, the eigenvalues are:

$$\lambda_1 = -0.0313 \quad \text{and} \quad \lambda_2 = -0.2917$$

For the natural frequency and damping ratio, it can be make:

$$\omega_{sp} = 0.0955 \quad \text{and} \quad \zeta_{sp} = 1.6911$$

A diagram below can show the poles of four conditions (from section 3.4.2.2):

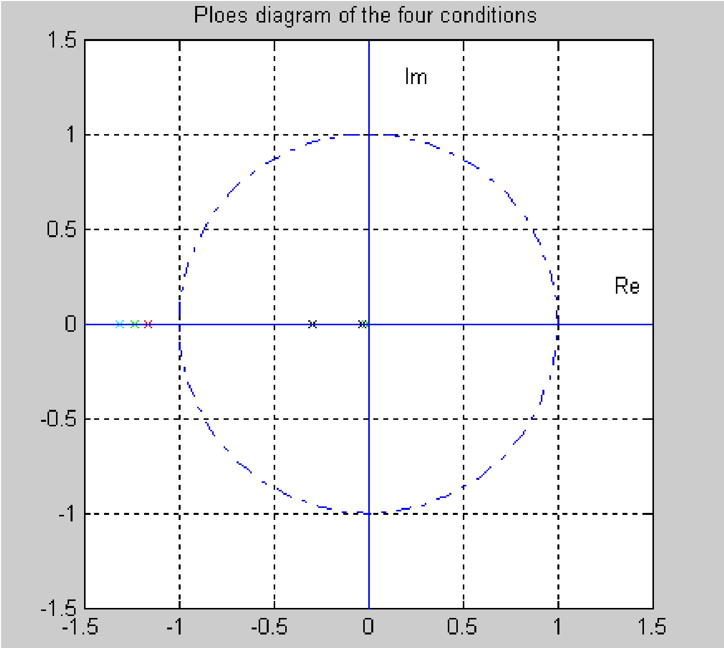


Figure 4.3.1 Poles placement diagram

Note: one of the poles in each condition is too close to the origin, so the system cannot guarantee the stability.

4.3.5 Flying quality analysis:

According to the specifications of the levels of acceptability (looking the damping ratio) it is clear that all of them are in Level 3.And the definition of that level is:

"The flying qualities are such that the aircraft can be controlled, but either the effectiveness of the mission is gravely impaired, or the total workload imposed upon the pilot to accomplish the mission is so great that it approaches the limit of his capacity".(see section 3.3)

4.4 DESIGN STABILITY AUGMENETATION SYSTEM

4.4.1 Calculation of the transfer fuction

In longitudinal control (use elevator only) design, pitch rate SAS is one important facet. It can help to augment M_q, which leads to improvement of the damping ratio. For simplification, only rate gyro signal is considered.

Substituting the parameters of the first condition, we can get:

$$\frac{q(s)}{\delta_E(s)} = \frac{-0.3635s - 0.0099}{s^2 + 1.1762s + 0.0181}$$

And its eigen equation is:

$$\Delta_{sp}(s) = s^2 + 1.1762s + 0.0181$$

If we assume that $\zeta_{sp} = 0.7$ and $\omega_{sp}^2 / n_{z\alpha} = 2.0\ (\omega_{sp} = 1.0719)$, which meets the standard of CAT A, Level 1. According to (E.22) and (E.23), a new eigen equation is founded:

$$\Delta'_{sp}(s) = s^2 + 1.5007s + 0.1149$$

Where the roots of this equation is $s_1 = -0.0809$ and $s_2 = -1.4198$, which are also the new poles of the system. Compared with the previous poles, the new ones are not only stable, but more desirable. Then we have to find the value of K_q. The key step is to establish the feedback equation.

Then, we can make the new eigen equation $|sI - (A - BK)| = 0$ .So substituting my parameters my values are: $K_q = [-0.2394 \quad -0.8904]$.

4.4.2 Simulation:

With MATLAB simulink, it can be seen the response from the transfer function, as follows:

$$\delta_z(s) \longrightarrow \boxed{\dfrac{-0.3635s - 0.0099}{s^2 + 1.1762s + 0.0181}} \longrightarrow q(s)$$

Figure 4.4.2.1. Block diagram of pitch rate and elevator

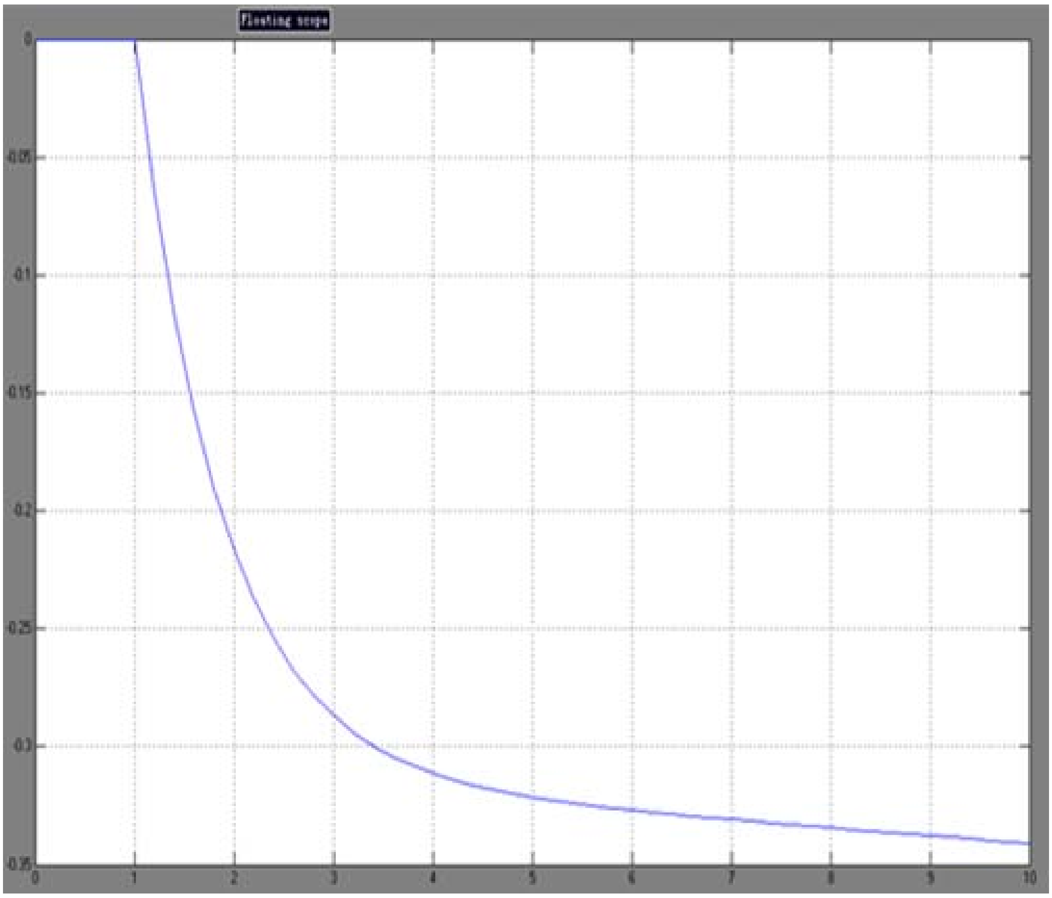


Figure 4.4.2.2. Simulation of the previous system

The new transfer function can be written as below:

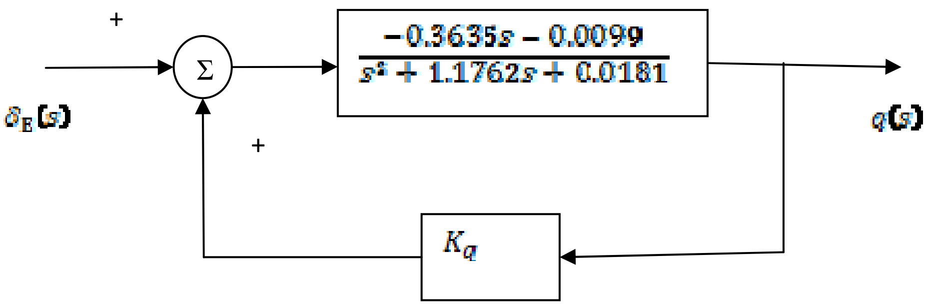


Figure 4.4.2.2 The block of the new system

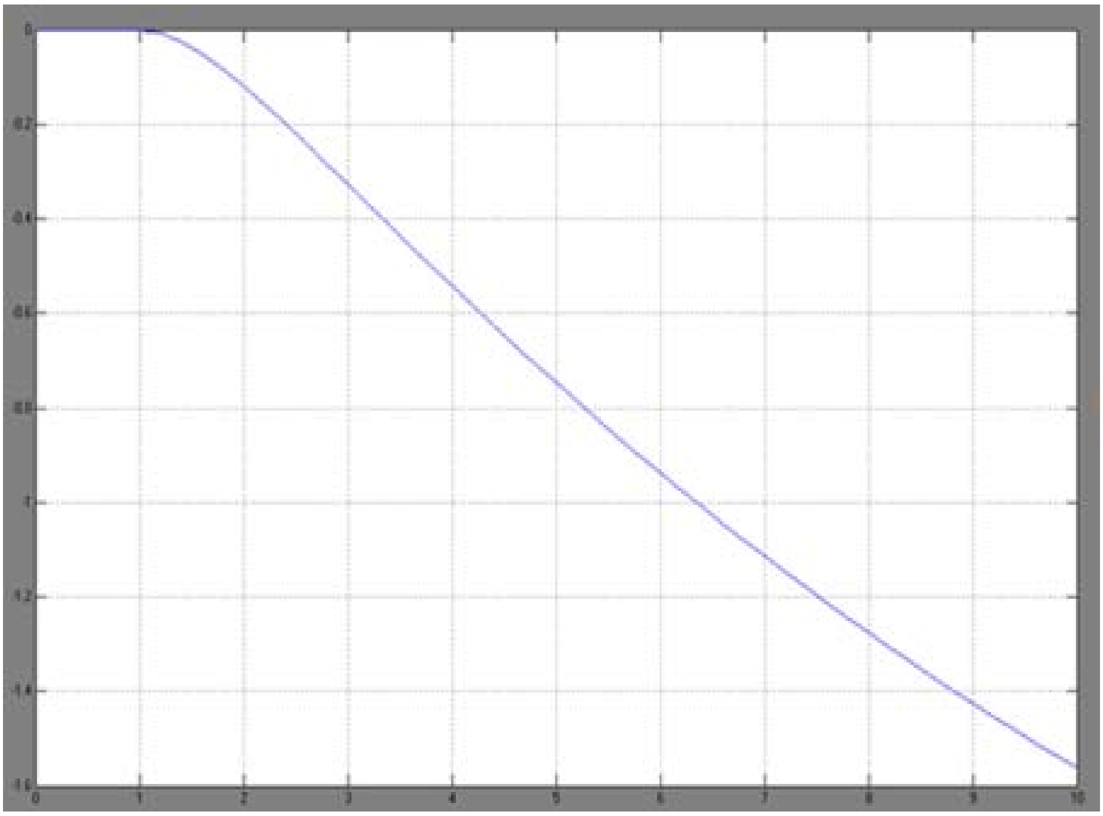


Figure 4.4.2.3. Simulation of the new system

It is obvious that the response is slower after making improvements(with the new poles), therefore it may be interesting when it comes to saving fuel, which is one of the most important objectives for the airlines.

4.5 METHOD OF THE OBSERVER

Another method to determine the error is the method of the **observer**. Finding the matrix "C"($y = Cx$) and later the tranfer function of each condition:

$$G(s) = \frac{C * Adj\,(SI - A) * B}{|SI - A|} \qquad C = (w\ q) \qquad B = \begin{bmatrix} Z_{\delta E} \\ M_{\delta E} \end{bmatrix}$$

4.5.1. Find the transfer function depending on the conditions

4.5.1.1 For case 1

C= (202.5 3460) $\qquad B = \begin{bmatrix} -2.9 \\ -0.6313 \end{bmatrix}$

$$G(S) = (-2771.55s - 754426.77))/(s2 + 1.469s - 0.363)$$

4.5.1.2 For case 2

C = (418 11730) $B = \begin{bmatrix} -6.83 \\ -1.25 \end{bmatrix}$

$$G(S) = \frac{-17517.4s - 15.469 * 10exp6}{s2 + 1.578s - 0.095}$$

4.5.1.3 For case 3

C = (25.3 20900) $B = \begin{bmatrix} -9.51 \\ -1.5 \end{bmatrix}$

$$G(S) = \frac{-31590.6s - 50.33 * 10exp6}{s2 + 2.198s - 0.9688}$$

4.5.1.4 For case 4

C =[1274 10100] $B = \begin{bmatrix} -5.18 \\ -0.92 \end{bmatrix}$

$$G(S) = \frac{-15,8913\,\delta - 13.624 * 10exp6}{s2 + 1.07s + 0.994}$$

4.5.2 Characteristic equation for the closed loop system.

Finding the characteristic equation we can now the values of $\mathbf{K_1}$ and $\mathbf{K_2}$ in order that the desired poles that we choose. And later calculate the K_s values ($|SI - (A - BK_s)|$=0) so we have our rate gyro feedback.

4.5.2.1 For case 1

$$|SI - (A - BK)| = 0$$

The equation is:

$$\mathbf{S^2 + S\,(1.469 - 2.9k_1 - 0.6315k_2) + (-0.3631 - k_2(0.435) - k_1(49.77)) = 0}$$

$N_{z\alpha} = -Z_w * \frac{Vo}{g} = \frac{0.634 * 75}{10} = 4.755$

Desired Poles; $P_{1,2}$ = (-0.7+0.7j) and (-0.7-0.7j)

Then, we have:

$\mathbf{S^2 + S\,(1.4) + 0.98 = 0}$

Resolving both equations, we obtain: $K_1 = -0.029$ $K_2 = 0.242$

According with:

$$|SI - (A - BK_e)| = 0$$

with Desired Poles; $P_{1,2}$ = (-150+150j) and (150+150j),

we obtain: K_{e1} =0.097 K_{e2} =-0.174

4.5.2.2 For case 2

The equation is: $\mathbf{S^2+S(0.96 - 6.83k_1 - 1.25k_2)+(-0.097 - k_2(0.748) - k_1(230.94))=0}$

$$N_{z\alpha} = -Z_w * \frac{Vo}{g} = \frac{0.618 * 190}{10} = 11.742$$

Desired Poles; $P_{1,2}$ = (-1.85 + 1.85j) and (-1.85 – 1.85j)

We obtain:

$\mathbf{S^2+S(3.7) + 6.84 = 0}$

Resolving both equations, we obtain: K_1= -0.025 K_2 = -1.561

4.5.2.3 For case 3

The equation is: $\mathbf{S^2+s(2.198 - 9.5k_1 - 1.5k_2)+(0.968 - k_2(1.391) - k_1(391.6))=0}$

$$N_{z\alpha} = -Z_w * \frac{Vo}{g} = \frac{0.925 * 253}{10} = 23.4$$

Desired Poles; $P_{1,2}$ = (-2.6 + 2.6j) and (-2.6 - 2.6j)

We have:

$\mathbf{S^2+S(5.2) + 13.52 = 0}$

Resolving both equations, we obtain: K_1= -0.026 K_2 = -1.838

4.5.2.4 For case 4

The equation is: $\mathbf{S^2+S(1.067 - 5.18k_1 - 0.92k_2) + (0.993 - k_2(0.341) - k_1(242.72))=0}$

$N_{z\alpha} = -Z_w * \frac{Vo}{g} = \frac{0.387 * 260}{10} = 10.062$

Desired Poles; $P_{1,2}$ = (-1.25 + 1.25j) and (-1.25 – 1.25j)

So, we have:

$$\mathbf{S^2+S\,(2.5) + 3.125 = 0}$$

Resolving both equations, we obtain: K_1= -0.0066 K_2 = -1.52

4.5.3 The controller without the observer (using Matlab) is:

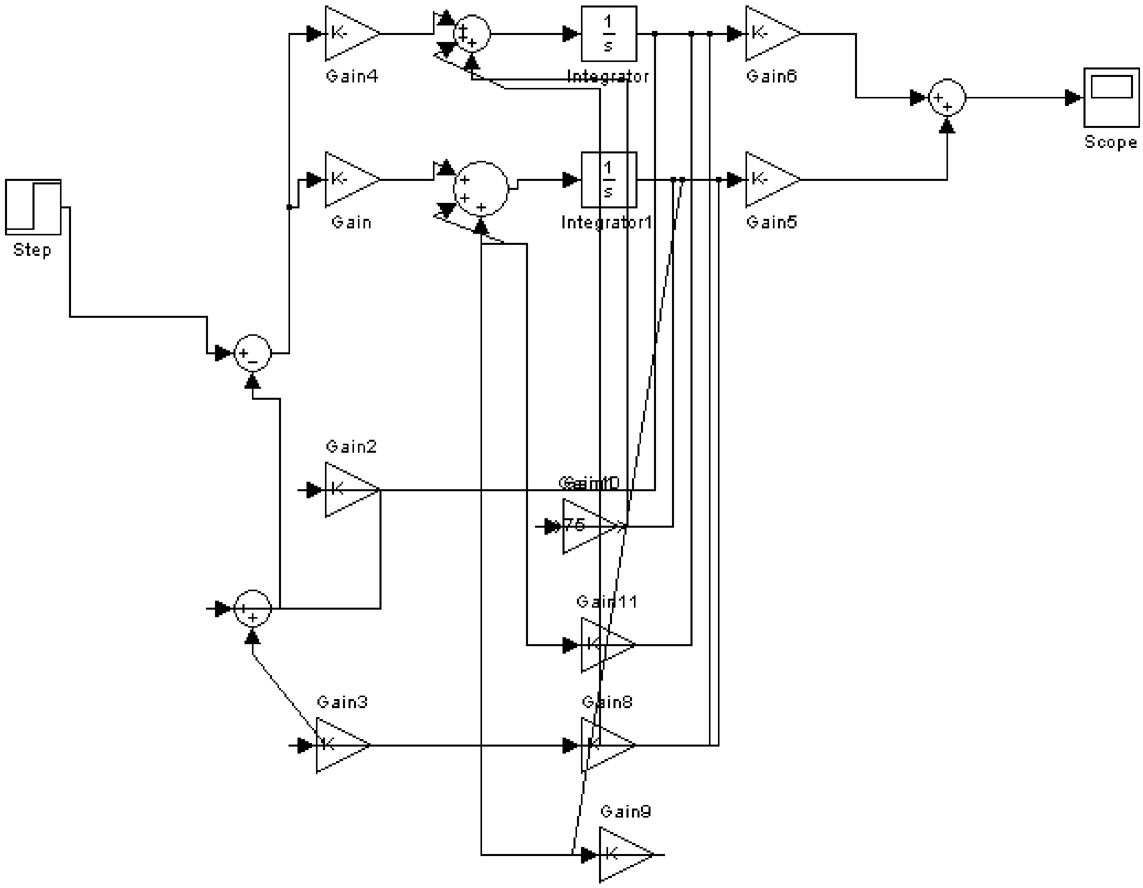


Figure4.5.3.1 Controller without the observer

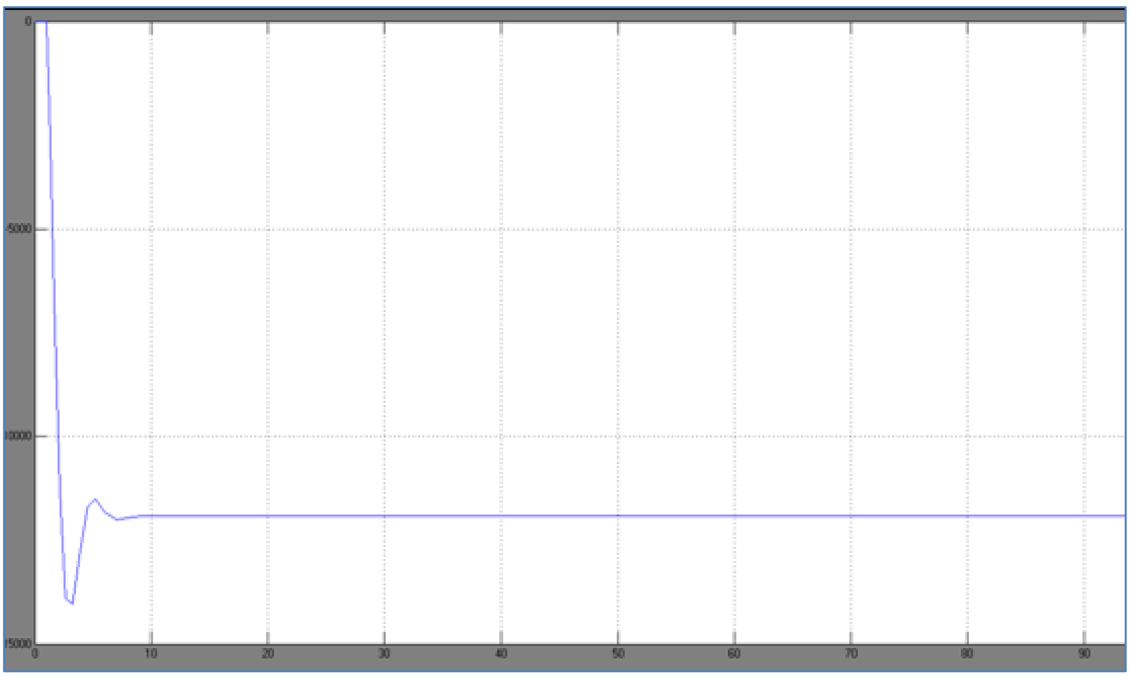
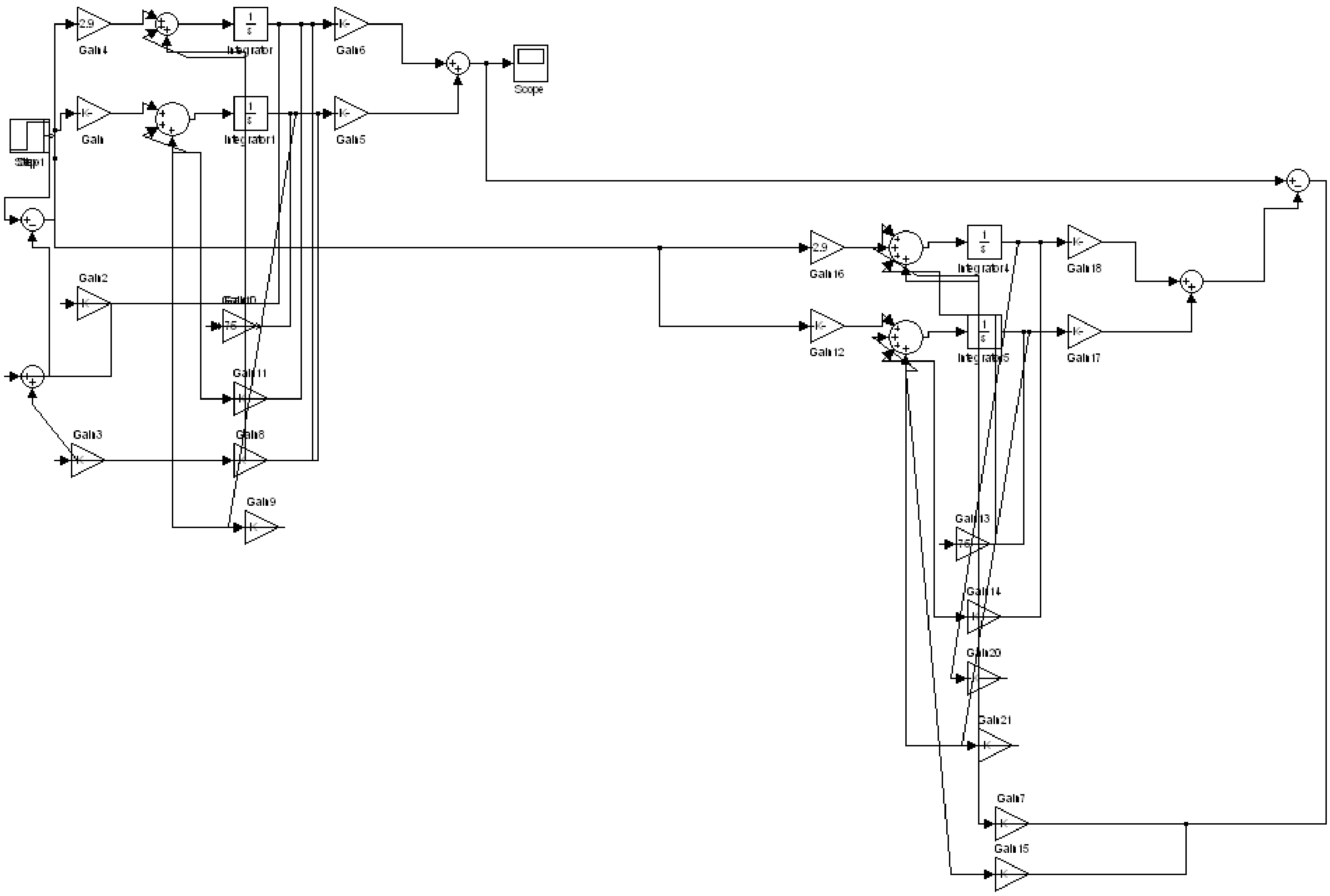


Figure 4.5.3.2 Simulation with the first condition



4.5.4The controller with the observer (using Matlab) is:

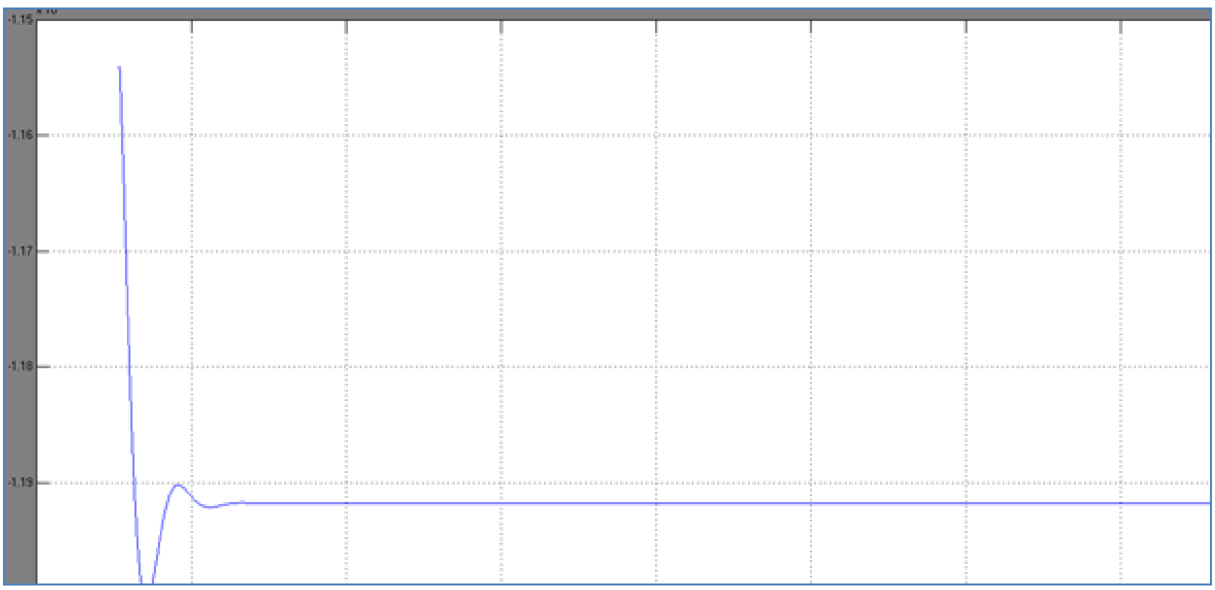


Figure 4.5.4.2 Simulation with observer

According with the second simulation my conclusion is that I´ve chose the values of the poles too high because there is a jump at the start of the signal, so I have to try with other values. But the response is smoother in figure 12 which means that there is an improvement.

5. Design Process II (Final results and Discussions)

5.1 INTRODUCTION

After calculate the equations of motion and the state space model for the aircraft, and understand how a Stability Augmentation works it can be design the Attitude and Flight Path Control Systems.

In the way of designing an Automatic Flight Control System (AFCs), it can be difference three kinds of Control System which are individually the inner loop from the other ones, as it is shown in the next Figure

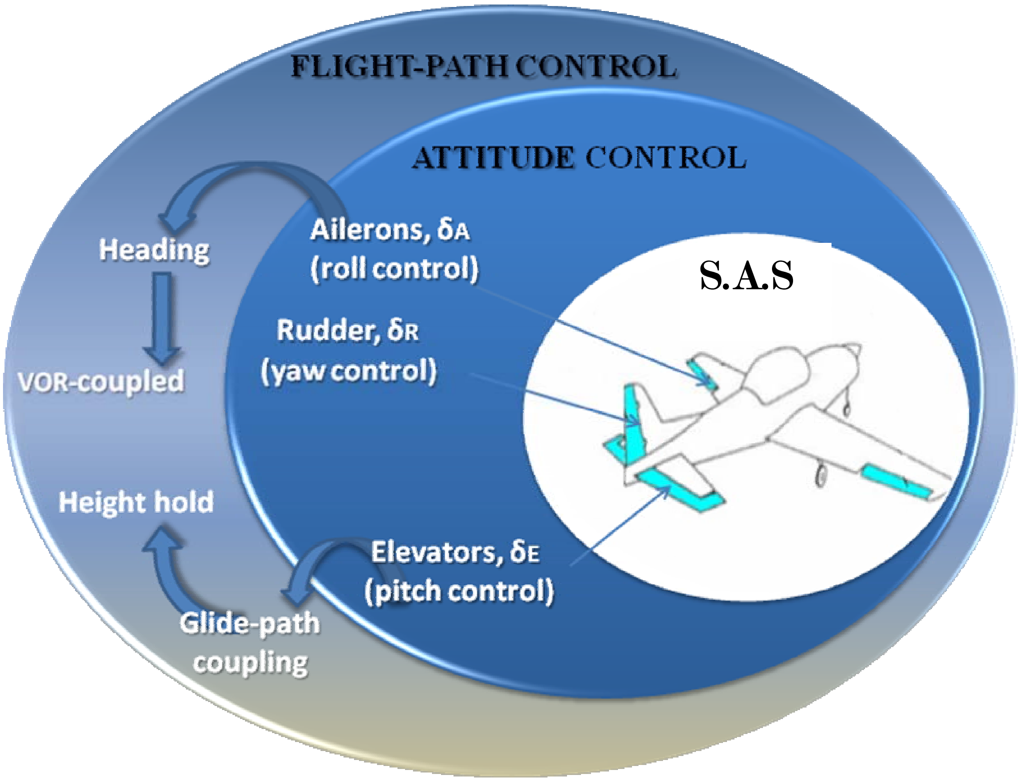


Figure 5.1.1 Diagram of Control System Design

The most basic system is the SAS, which are the closed loop of the system used to remedy the deficiencies in flying qualities, as a feedback to augment the stability.

SAS are the inner loop of the Attitude control system, which are the essential function of any AFCS, to maintain the aircraft in any required, specified orientation in the space, either in direct response to a pilot's command. It can be seen that in Attitude control there are three primary motions to be controlled: pitch attitude, which involves the use of elevators as a control system; the roll caused by the deflection of the ailerons; and the yaw, as a result of the rudder deflection.

The Attitude Control are the inner loop of the Flight Path Control, which are designed because there are a number of missions which require that the aircraft are made to follow with great precision some specially defined path. In longitudinal motion, from the pitch attitude as an inner loop it can be designed the Glide-Path-Coupled and the Height hold. On the other hand, in lateral motion, we it can be calculated from roll and yaw motion acting as an inner loop the heading and the VOR-coupled.

5.2 ATTITUDE CONTROL SYSTEMS

5.2.1 Pitch Attitude

The first of the primary motions that has to be controlled is the pitch control, as a pitch attitude.

The pitching is an up or down movement caused by the deflection of the elevators. To control it, it is necessary the State Space model with longitudinal parameters (from equation 4.1.1.1), and in this case from CHARLIE-1, which is a very large, four-engined aircraft, as we can see the derivatives from Appendix.

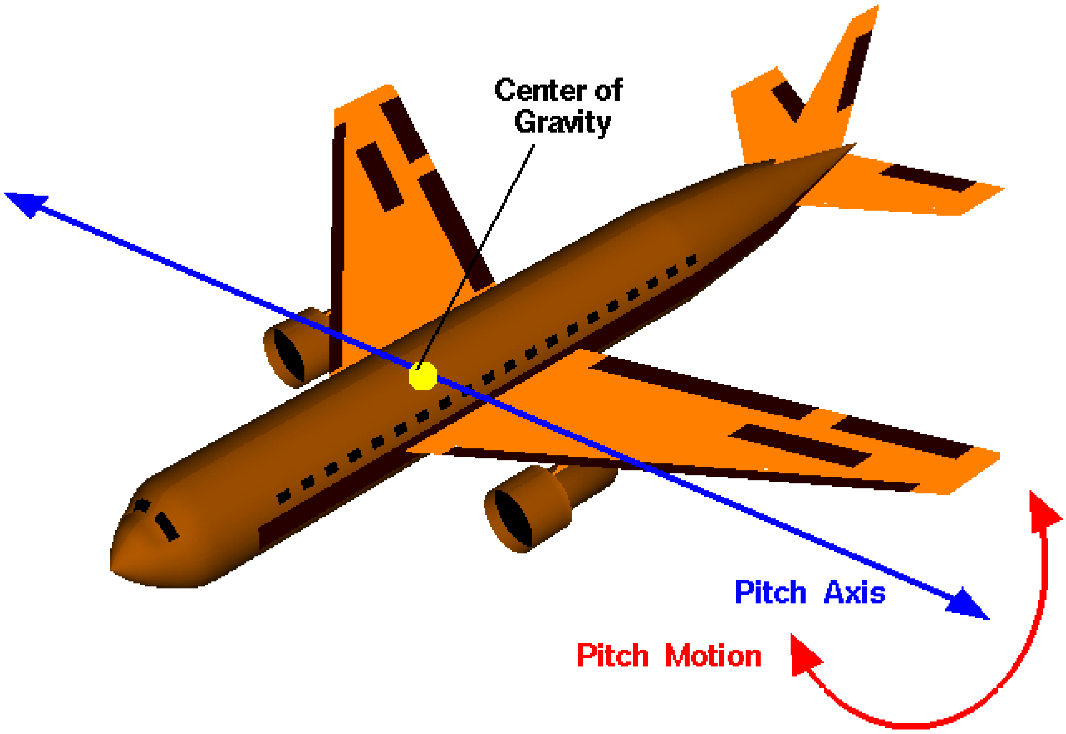


Figure 5.2.1.1 Axis system of pitching motion.

The damping movement caused by the elevators is controlled by two feedbacks: rate gyro (K_q), and attitude gyro (K_θ).As it shown in the next control system.

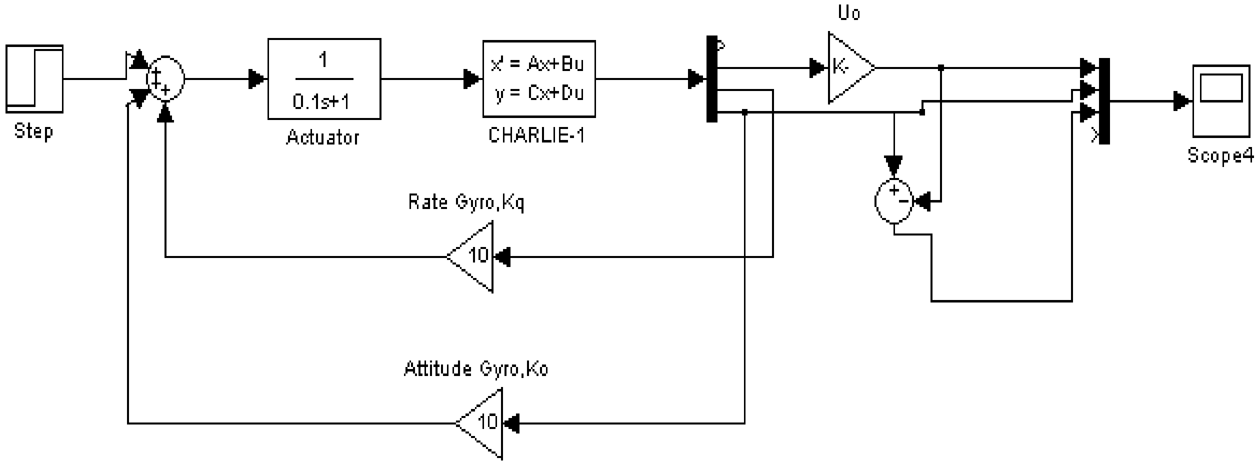


Figure 5.2.1.2 Control system for Pitch Attitude.

Once controlled, it has to be analyzing in the range of the flying qualities.

The pitch attitude works with the actuator and the feedbacks in such a manner that it can control the SAS close loop first, in this particular case (CHARLIE-1), the chosen values of controller gains are: K_q =10, and K_θ =10, and $U_0 = \frac{1}{U_0} = \frac{1}{67}$.

This pitch attitude has been designed to control the height hold and the glide-coupled, and because of that it used γ, as the difference between *u* and *θ.*

It can be seen the response in Figure 5.2.3.

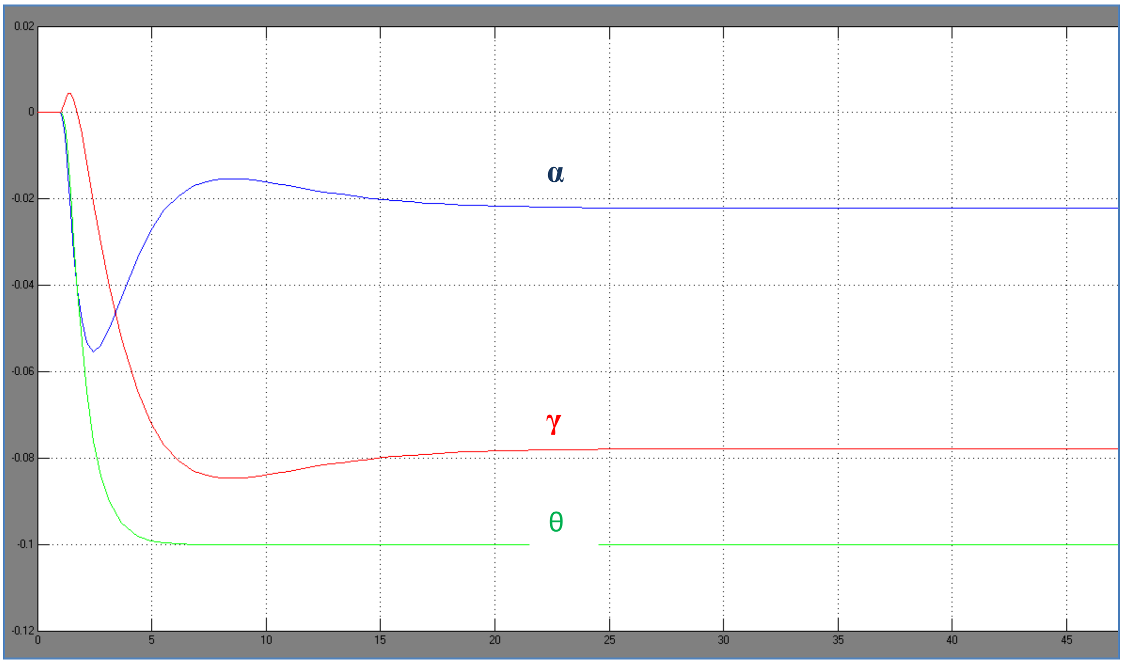


Figure 5.2.1.3 Response for pitch Attitude Control System

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The most important thing that it can be seen in the simulation is that θ is dropping to zero in a time of 5 to 10 seconds, which this means that the pitch attitude is controlled in this time.

It is important to know that the pitch attitude can be disadvantageous in presence of atmospheric turbulence because of the pitch attitude in this case tends to hold the pitch angle at a constant value. So, when the aircraft has a problem because of the pitch attitude it can be arise again when is banked at some large angle. This, depends of the rate and attitude gyros, because at some large bank angles the signals from q and θ cannot be zero simultaneously. For this reason in turning flight the system performance depend upon the gyro; if the vertical one is used then the bank angles has to be restricted in a range, as it shows as follows controlling the roll system.

5.2.2 ROLL CONTROL SYSTEM

The second primary control system that has been controlled is the roll control.

The roll angle axis lie in along the aircraft certain line. Is an up or down movement of the wings caused because of the deflections of the ailerons.

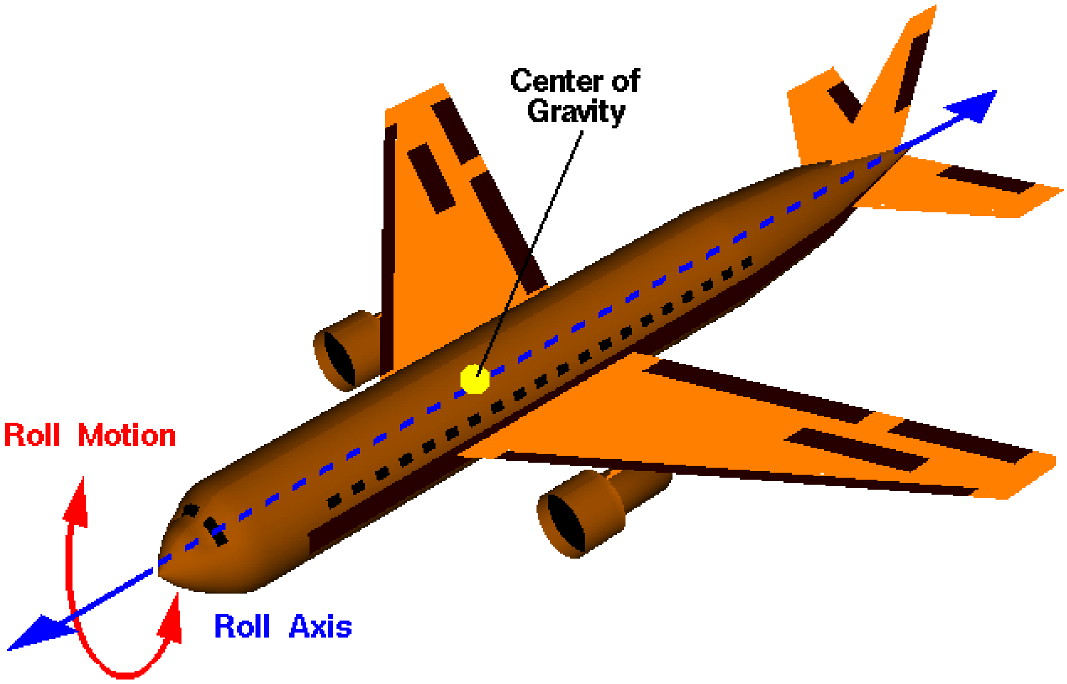


Figure 5.2.2.1 Axis system of rolling motion

The roll angle is generally controlled simply and effectively by the ailerons at low–to-medium speeds on all types of aircrafts (in military the spoilers are used).

The roll angle is a lateral motion, so it is necessary the lateral parameters in the State Space models (from equation 4.1.2.1), while it is calculate with the same values for CHARLIE-1.

Next step is to find a control system design for this particular case. The chosen design is the "Bank angle control system with roll rate inner loop damper". This means that the roll damper has been used as a inner loop, which permits a designer to use considerable freedom in arriving at the required dynamic performance of the roll angle, as it can be seen in the next Figure 5.2.2.2.

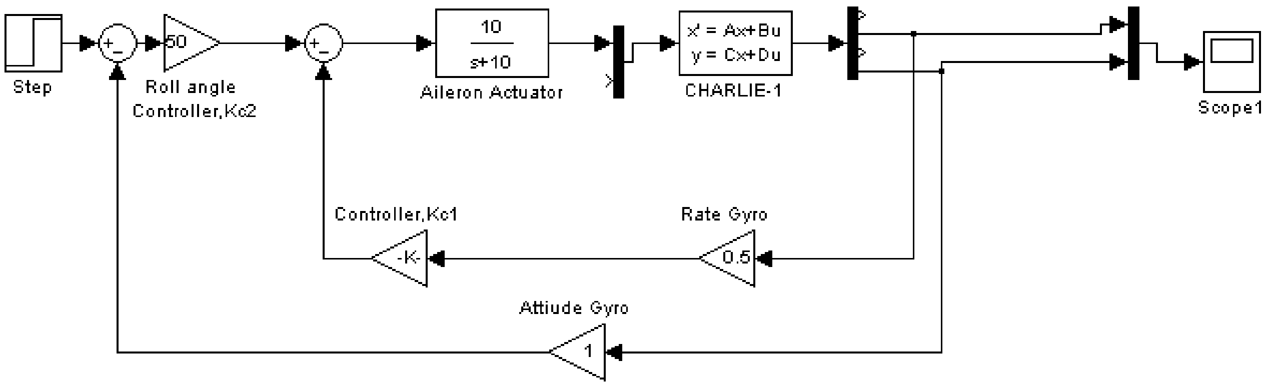


Figure 5.2.2.2 Control System for Bank angle control system with roll rate inner loop damper

The actuator dynamics are represented as a simple first order lag. Then using the roll damper as a inner loop, the frequency of the roll angle system can be controlled by K_{c2} and the damping by 0.5 , K_{θ} =1.

The step response for the system is shown in Figure 5.2.2.3.

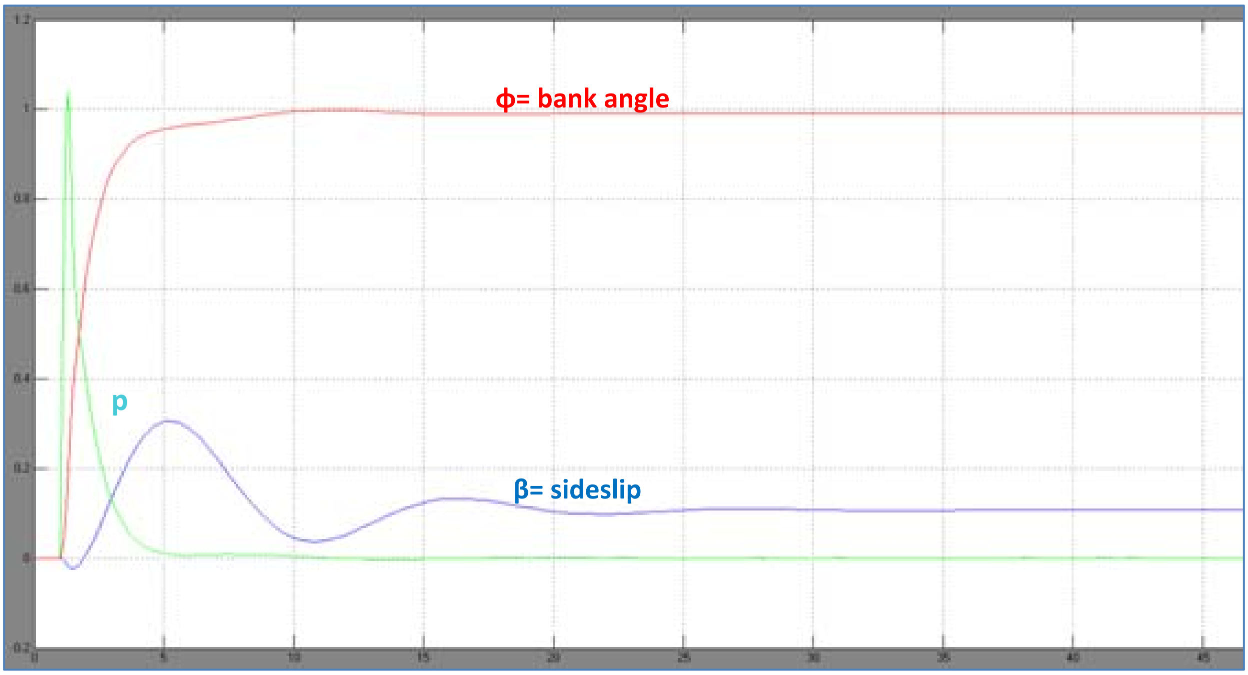


Figure 5.2.2.3 Response for Bank angle control system with roll rate inner loop damper

It can be seen how effective is the response of the bank angle, because of it is controlled in a time of 10 seconds for 1° in the roll angle φ.

However some problems arise with the Roll control (see in section 3.7.2.1).

5.2.3 YAW CONTROL

This is the third primary control that has been designed, which a side to side movement of the nose is caused by the deflection of the rudder. To control it is necessary to have in account that it is a lateral movement, so we need the lateral parameters (from equation 4.1.2.1) to design the State Space model for CHARLIE-1.

In the case of the yaw it has been used a control system design called ARI (aileron-to-rudder), it can be said that is a “mixed-control”, because it controls the yaw and the roll. This means that it maintains the sideslip to zero, which is produced by the yawing movement caused by the deflection of the ailerons, while it controls in the same time the roll rate.

This ARI system has been designed especially for the moment when an engine fail so the pilot will need to command a constant aileron deflection to know the yawing motion caused by engine failure.

The control system design is shown next:

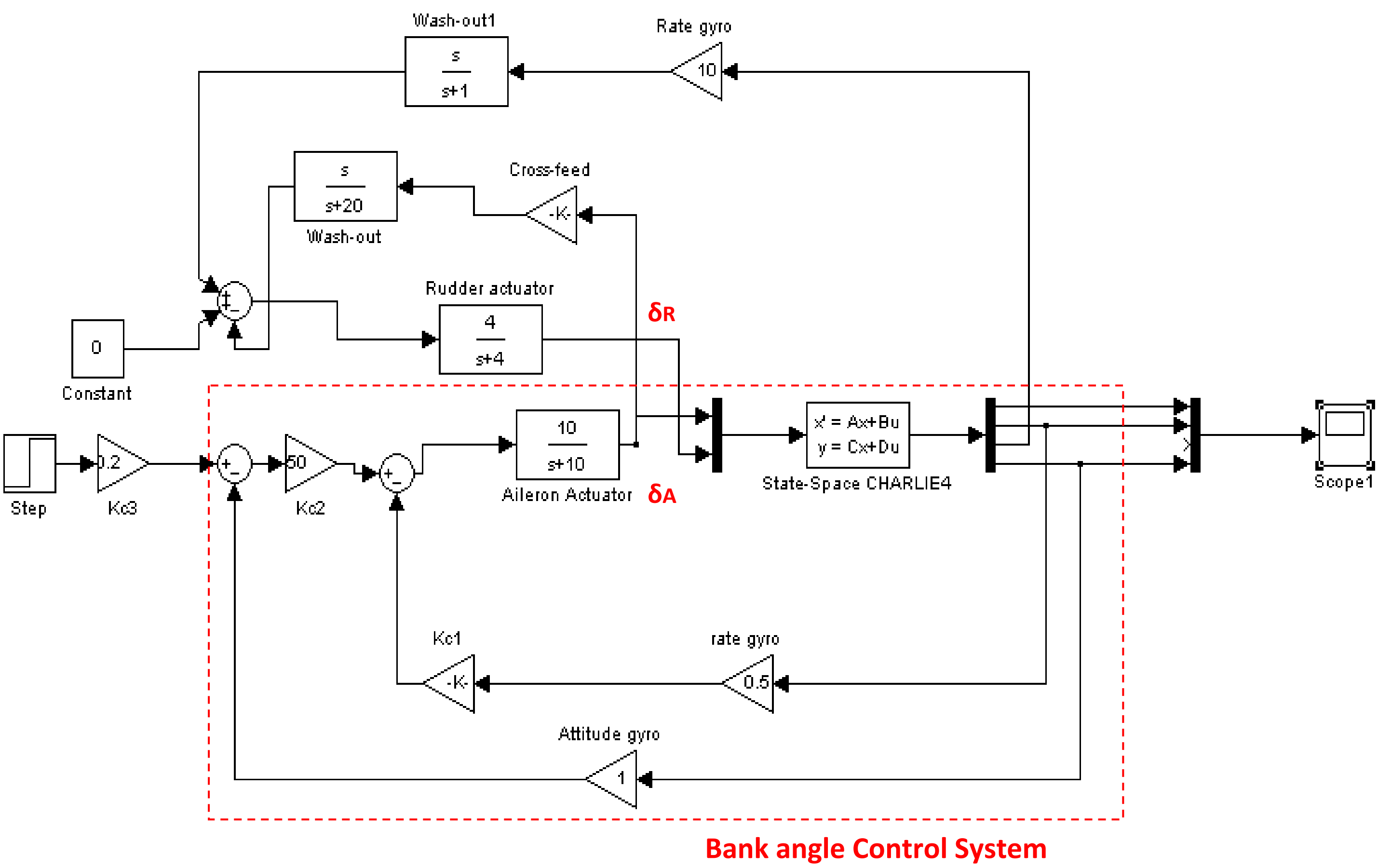


Figure 5.2.3.1 ARI Control System

It can be seen that the bank angle control system with roll rate inner loop damper acts as an inner loop to control the deflection of the ailerons helped by a crossfeed gain, which is necessary in real flights in the moment when the flaps are deployed, or when the landing gear is lowered, which it should vary with the forward speed of the aircraft. The crossfeed gain is $K_{cf} = 0.035$, has been found from flights studies(see McRuer and Johnston, 1975). Additionally, the presence of the wash-out following the crossfeed it is necessary to permit that the aircraft produce some sideslipping manoeuvres, which are the most common in cross-wing landings, in this case the values for the wash-out, $\frac{s}{s+\frac{1}{T_2}} = \frac{s}{s+20}$, so this means that $T_2 = 0.05s.$

The response depends seriously upon the values of T_2 and K_{cf}, as it can be seen in the simulation, in Figure 5.2.3.2.

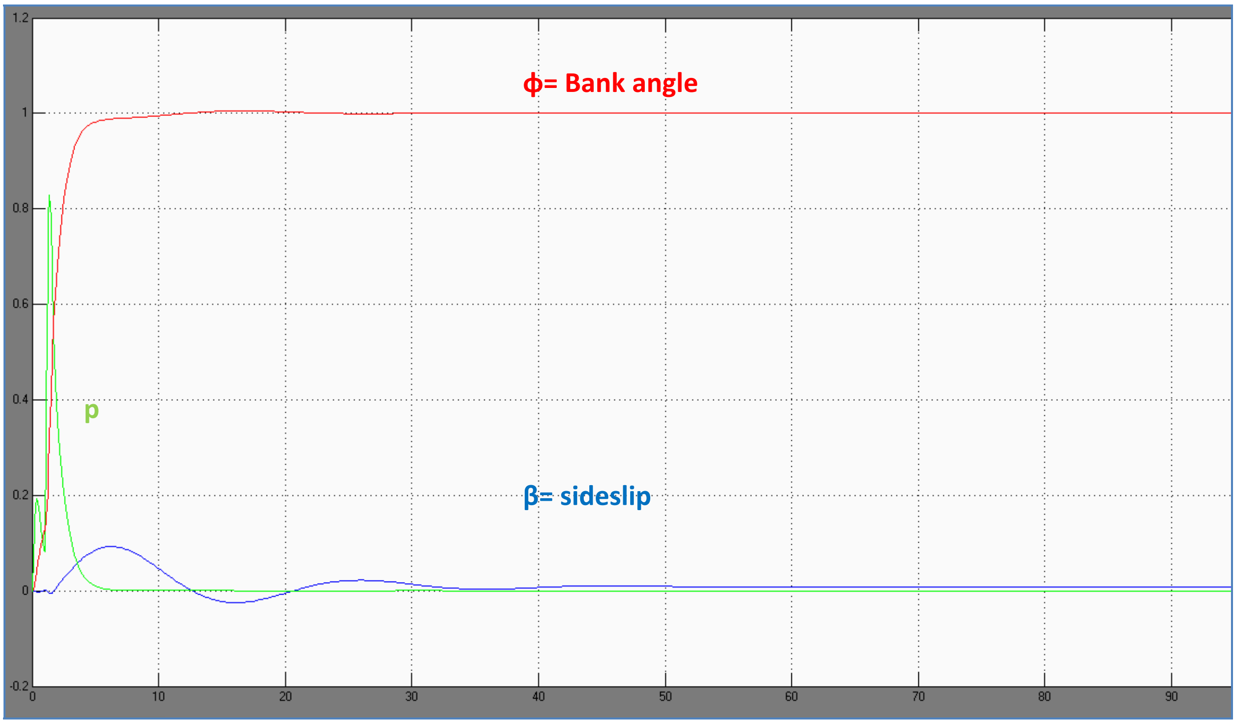


Figure 5.2.3.2 Response for ARI Control System

It can be seen that the bank angle has not been affected for in the presence of the crossfeed but it is evident how effective the sideslip has been suppressed.

5.3 FLIGHT PATH CONTROL SYSTEMS

While the control of the attitudes angles of an aircraft is the special function of Flight Control, the control of path through space is more properly a guidance function. Because of that we presume that an aircraft has to follow automatically a set of directions.

5.3.1 Heading Control System

From the idea that an aircraft need to be automatically steered along a set of directions, the heading control is when the aircraft it's taken as its yaw angle, since it is assumed that any turn the aircraft makes under automatic control will be coordinated. So the heading controls the yaw moment in coordinate turns and the bank angle. Consequently, it is a lateral movement which needs the lateral parameters (from equation 4.1.2.1) to be controlled.

The system shown in Figure 5.3.1.1 relates to CHARLIE-1.

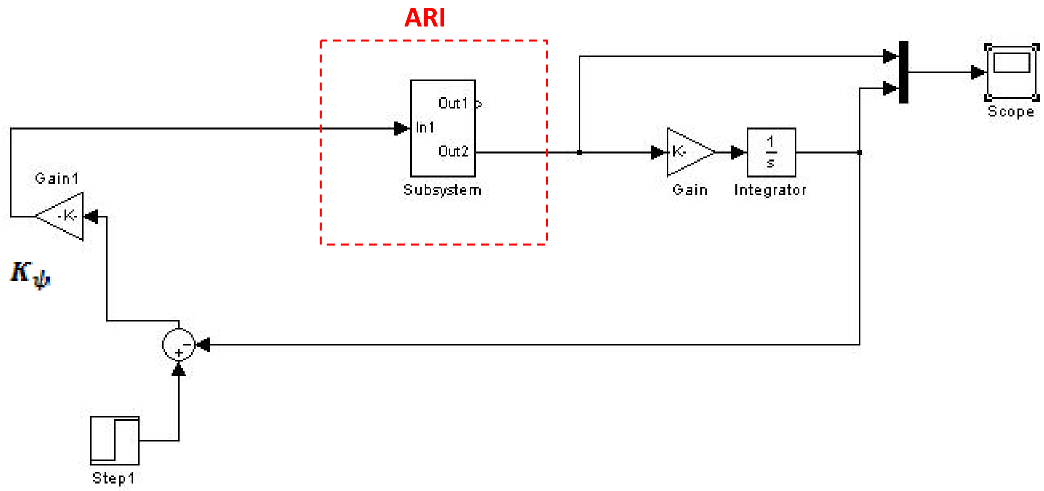


Figure 5.3.1.1 Control System for Heading.

The ARI (aileron-to-rudder) system acts as an inner loop to control the deflection of the ailerons followed by an aircraft kinematics, consisting of a feedback ($\frac{g}{U_0} = \frac{9.8}{67} = 0.146$), which it change the input from φ to ψ . And it can be seen that a new feedback path signal has been introduced to have a direction control, in this case is $K_\psi = \frac{120}{x}$, the variable x is the degrees that the system turns in relation to the reference. For a x=15° for the next simulation.

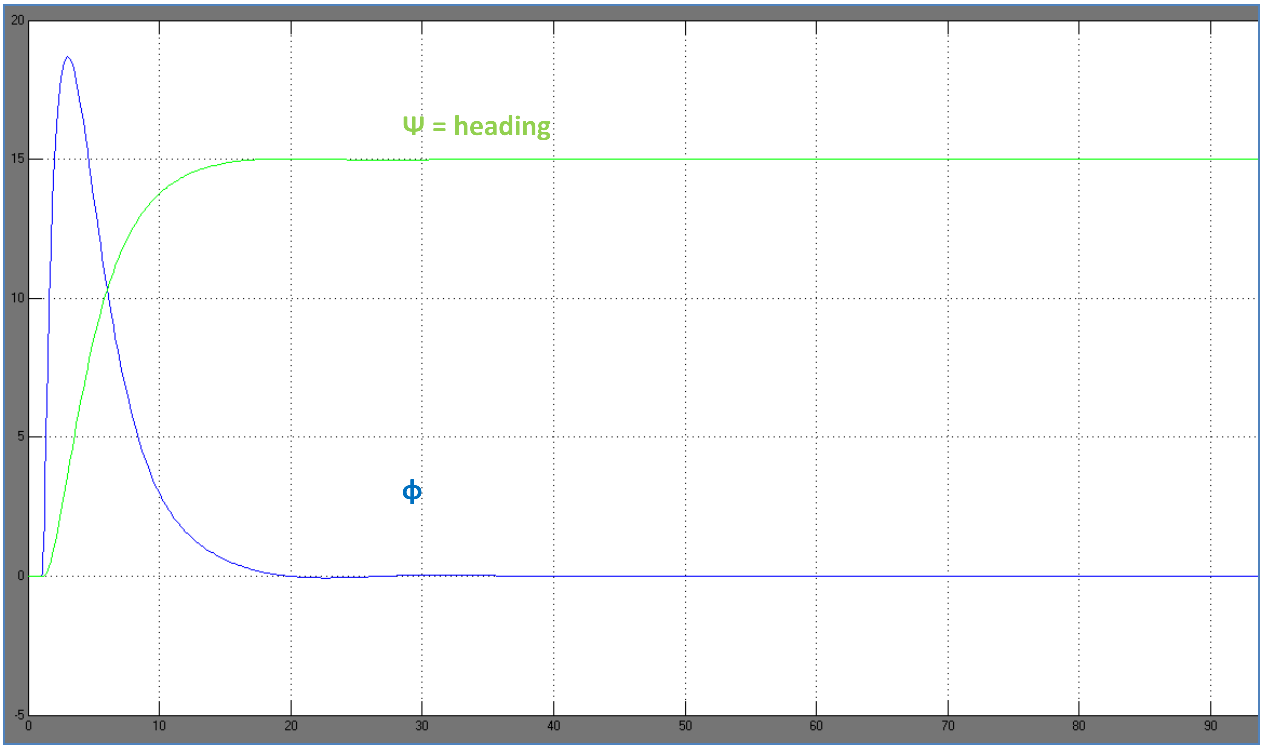


Figure 5.3.1.2 Response for Heading.

In this simulation it is evident that the heading is controlled in a time of 20 seconds at 15° , and in the same time the bank angle has a pick at the star but is not too important because while the roll/bank angle is in this situation the heading has the control of the inner loop.

5.3.2 Height hold Control System

Other directional system that it has to be controlled is the height hold which is used to control the height acting like a feedback regulator to maintain the aircraft height as a reference. The pilot's can even fly the aircraft controlling the pitch attitude control system to control the climb (or descent) of the aircraft until it has reached the required height. As it is shown in Figure 5.3.2.1.

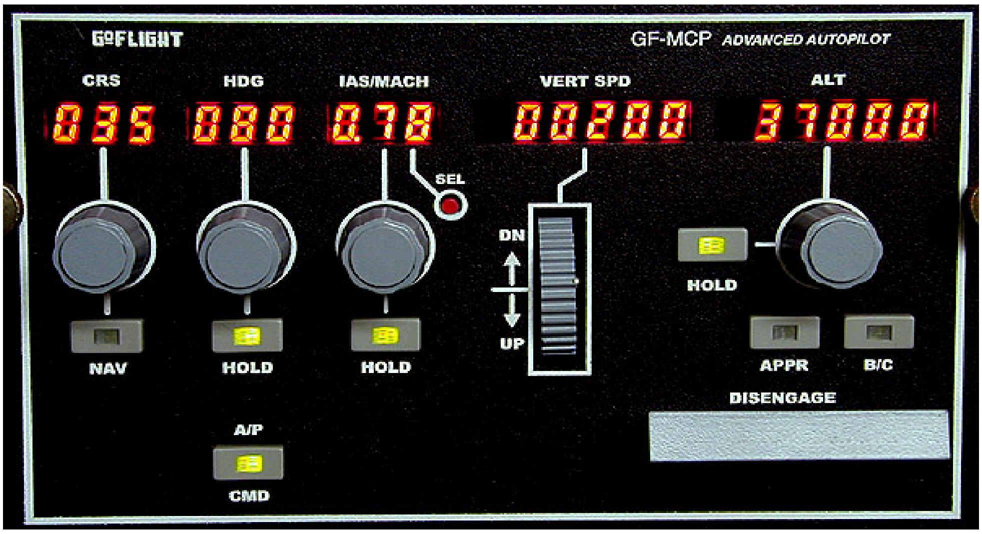


Figure 5.3.2.1 Interface panel of an autopilot

Is the best thing to know how the heading works in the interface panel, where the pilot controls the height. It works once the pilot put the height and the aircraft's path will follow closely.

In this case the movement, caused by the deflection of the elevators is controlled by longitudinal parameters (for CHARLIE-1), because in this case the mean inner loop is the Pitch Attitude with pitch rate SAS as the inner loop, as is shown in the next control system.**

The values for K_{c1} =0.2 is used because of the input signal to be reduced in order to the outputs, the $K_c = 0.03$ is introduced to reduce the dynamic instability. Followed by the feedbacks gain for the pitch attitude: attitude gyro, $K_\theta = 10$, and the rate gyro, $K_q = 5$.

Pitch Attitude

In this case *w* has to be transform in *h* to have an output for the altimeter so:

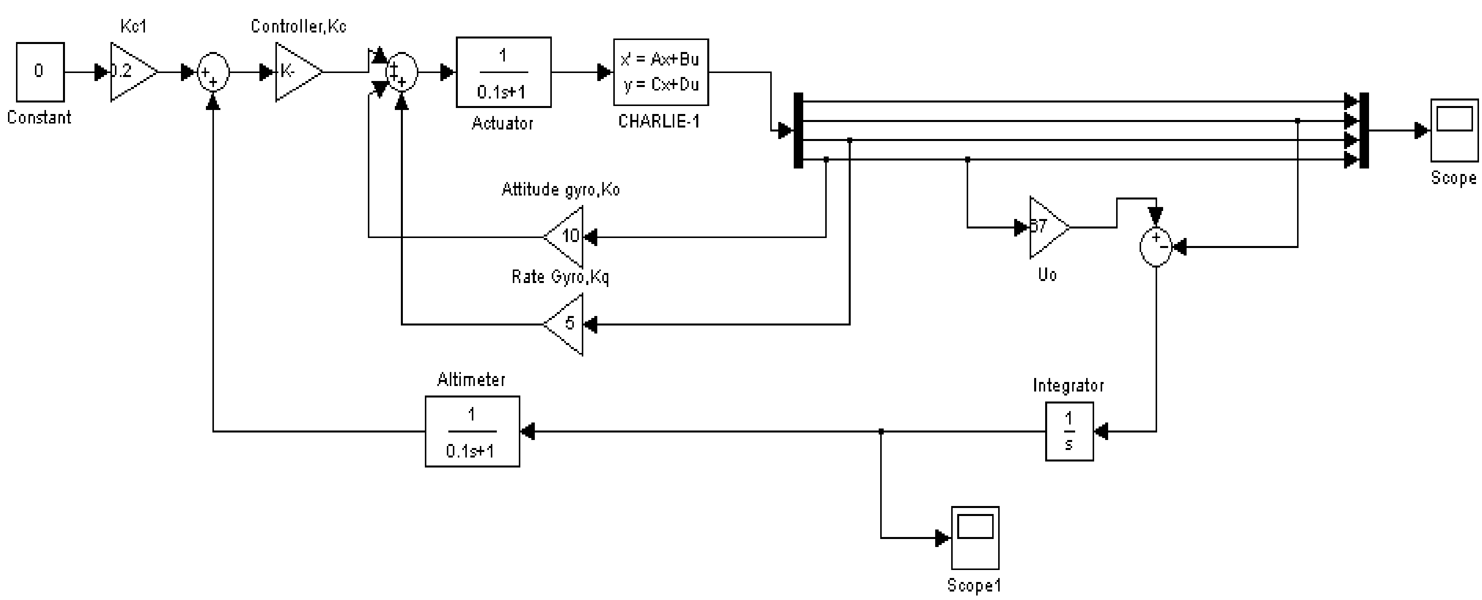
$$-\ddot{h} = \dot{w} - U_0 q$$

$$\ddot{h} = U_0 q - \dot{w}$$

$$\dot{h} = U_0 \theta - w$$

So with this expressions it can be understood that before the integrator we have $\dot{h}$, to have after *h* acting properly to control the altimeter in the outer loop to provide a feedback signal proportional to the height, and is used to achieve the height hold function.

Figure 5.3.2.2 Control System for Height hold.



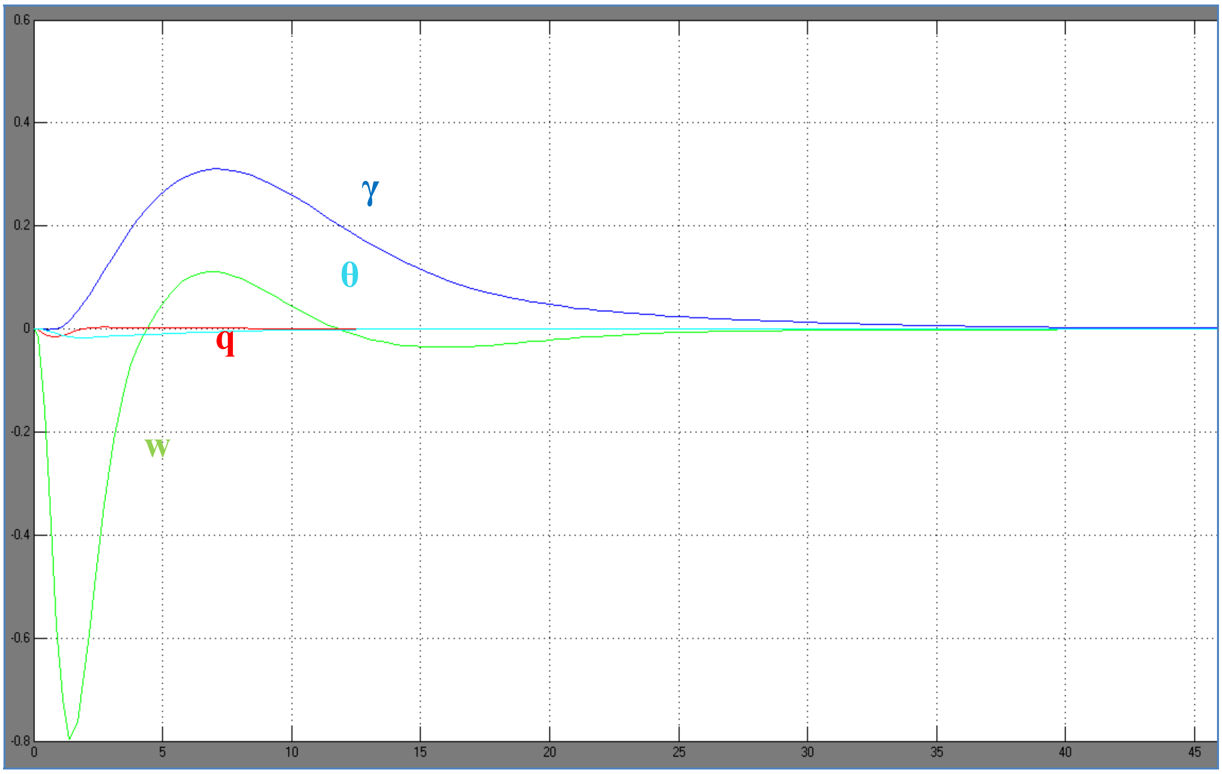


Figure 5.3.2.3 Response of Height hold for the variables

It can be seen how the behavior of the variables from the “Scope” are.

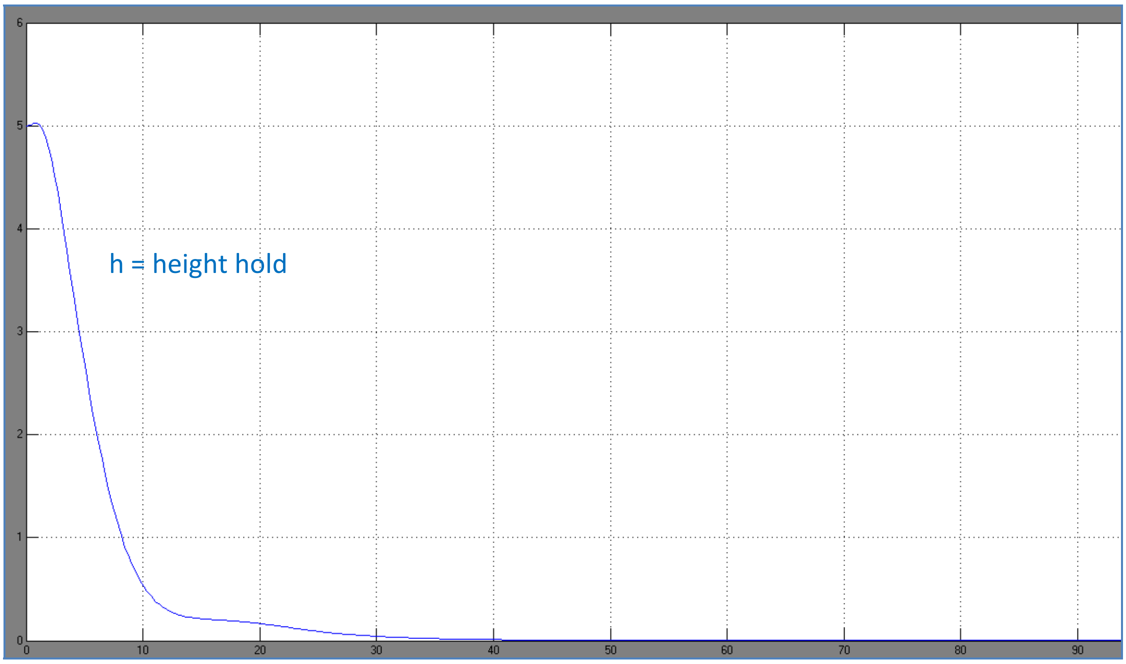


Figure 5.3.2.4 Response of height hold

It can be seen the controlled gain lead to improved the dynamic response, because is non-oscillatory, smooth and rapid response.

See in section 3.8.1.2 the problems of using this type of systems.

5.3.3 VOR-Coupled Automatic Tracking System

From the section 7.3 I made my own control system, based on the theory. The mean thing that must be taken into account is that the heading is going to be the principal subsystem in the VOR control and later we have to add the navigation system K_λ, and the function $R(t) = R_0 - U_0(t)$, which is shown in figure

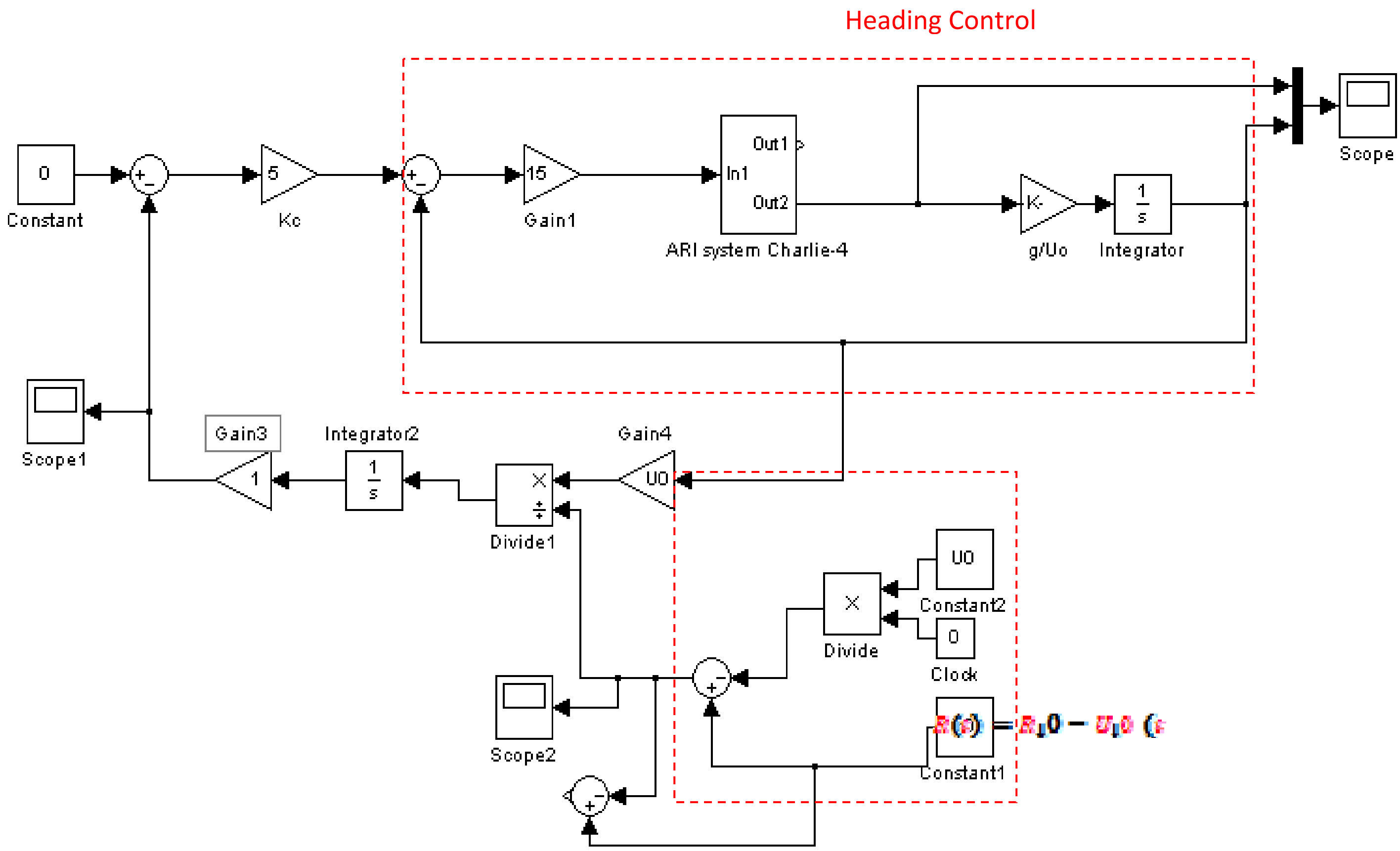


Figure 5.3.3.1 VOR-Coupled Control System

The values are taken are:

$$U_0 = 250\frac{m}{s}$$

$$R_0 = 45000\, m$$

We start the integrator2 with 3º for the initial condition, so we run the simulation to have in The Scope for the heading and the roll as follows:

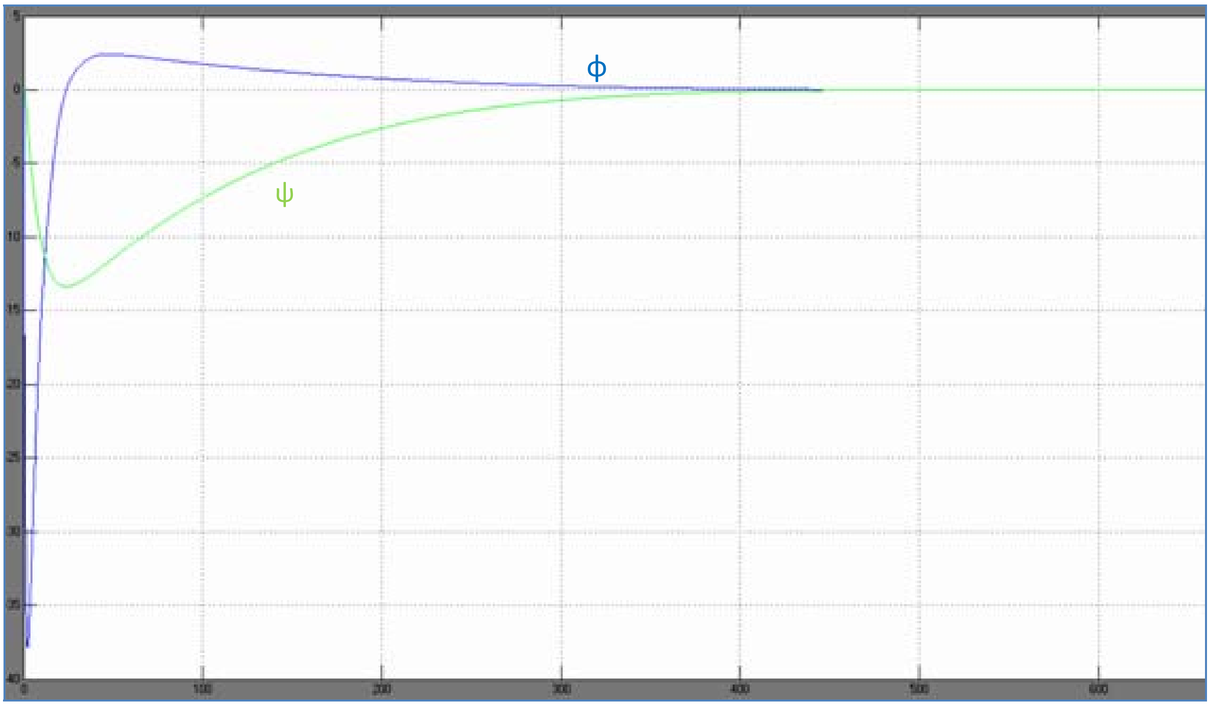


Figure5.3.3.2 Simulation for the roll and heading lateral (Scope)

As we can see here the roll is unstable at the start but in the same time is the heading control stable which it does mean that is stable because in this position of the plane the roll has less importance than the heading.

In the other hand we have a Scope2 that shows the behavior of R in order to:

$R(t) = R_0 - U_0(t)$,so we have the next simulation:

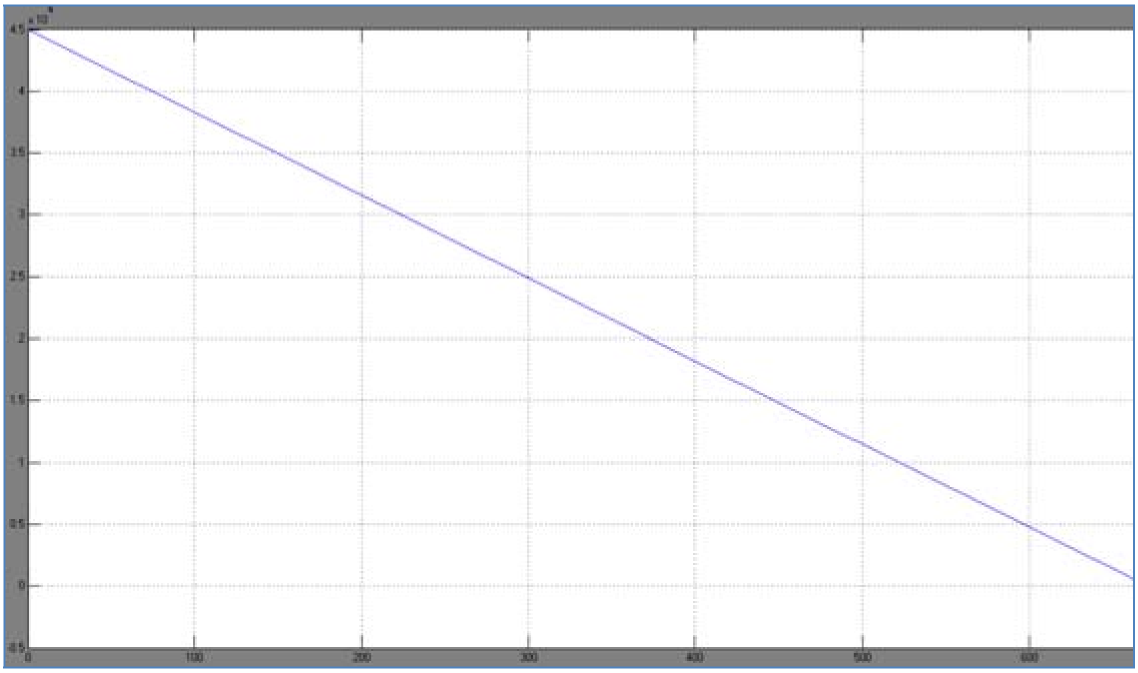


Figure 5.3.3.3 Simulation of R function lateral (Scope1)

As we can see, is the response of R so is going down as we wanted probe it because is decreasing in order to $R_0 - U_0$, respecting to t..

We can see when is intercepted the VOR, so as we can see in the next figure is around 300-400 metres of the runway.

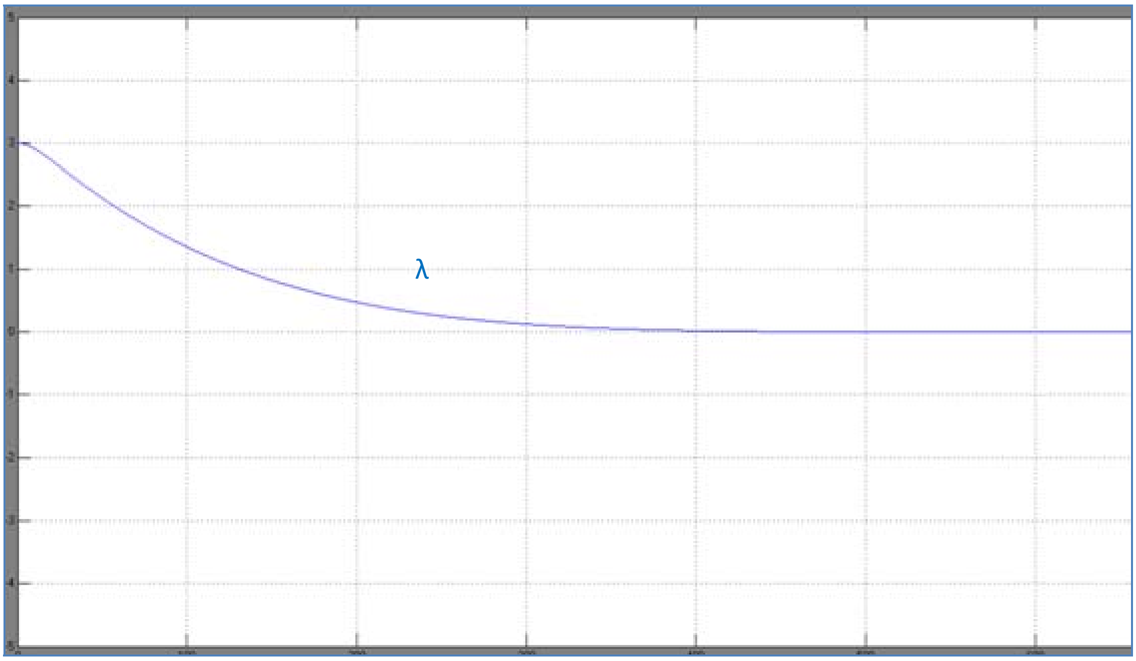


Figure 5.3.3.4 Simulation of λ function lateral (Scope1)

5.3.4 Glide-Path-Coupled Control System

In order to the theory showed in the section 7.4.1 I based my conclusions to design the glide-path-coupled control system.

Knowing that the mean subsystem of the glide-path-coupled is the pitch attitude control system, which is means that the states spaces parameters are from the longitudinal motion, and the flight path angle is $\gamma = (\theta - \alpha)$, I made it as follows:

As we can see pitch attitude, which is the pitch rate as a inner loop has the control, ''effectively control any changes in the angle of attack which may arise as a result of the elevator's being used to drive the aircraft back into the glide path. The block diagram of a typical control system is shown in the Figure****. The gain of the glide path receiver, K_{Rx}, can be considered, without loss generality.''

Glide-path-coupled control system

Pitch Attitude

''The transfer function of the glide-path-coupled controlled represents essentially a proportional plus integral term controller. The phase advance term has been added to provide extra stabilization, if required.

So far it has been presumed that the airspeed, U_0, is constant throughout the coupled trajectory, but this is never the case. A speed control system, used in conjunction with the glide-path-coupled system, is essential to ensure that the aircraft's flight path angle, γ, in the as the steady state, has the same sign as the commanded pitch angle. ''

The Simulation for this control system it is shown in figure

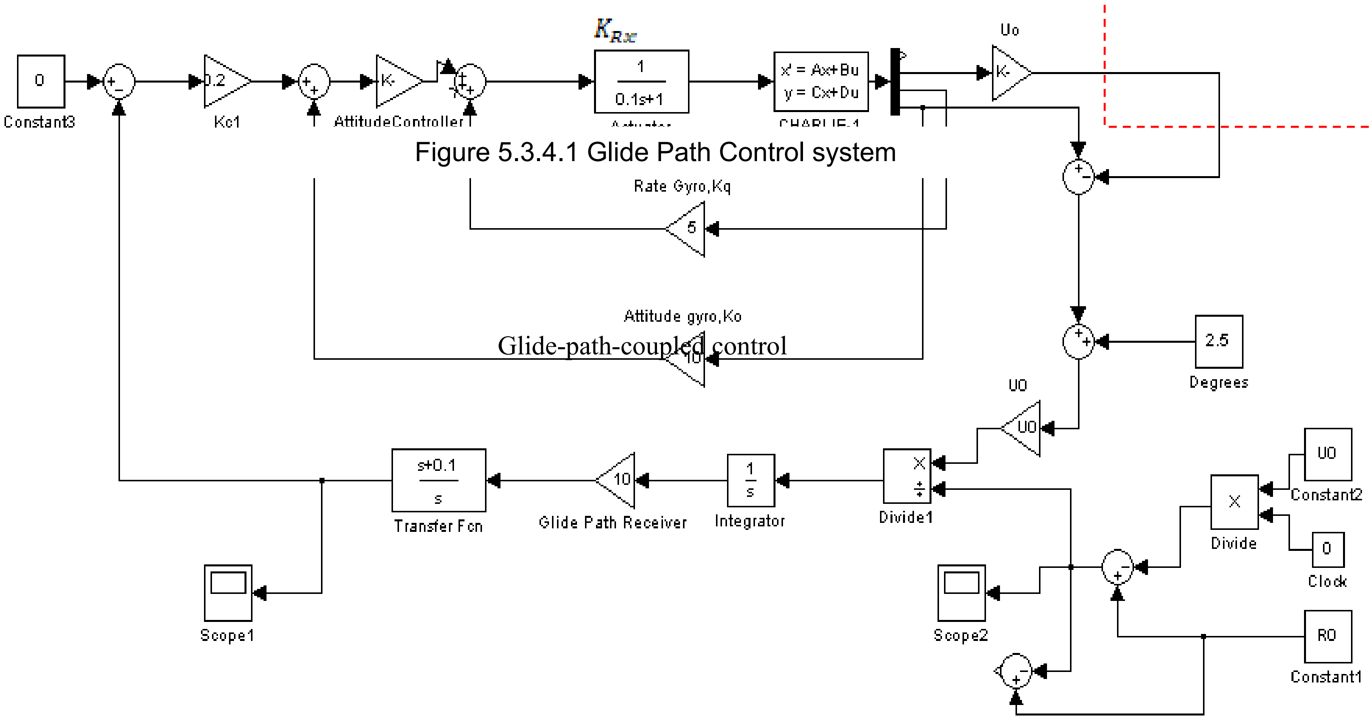


Figure 5.3.4.1 Glide Path Control system

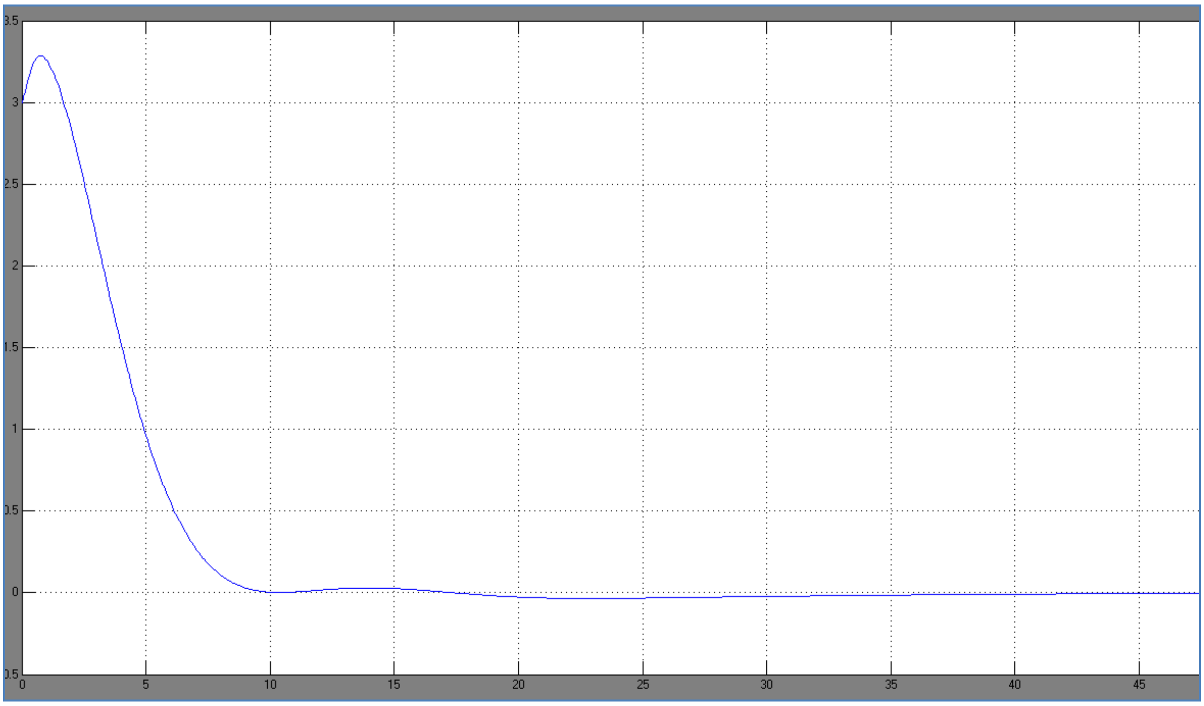


Figure 5.3.4.2 Response of Glide Path Control system

6.1 CONCLUSION

6.1 Design Process I (previous work)

This section introduces the AFCS as a control problem, because when the linear feedback is used is impossible to have all the requirements satisfactory in the same time like: stability tracking, performance and disturbance. So, in this section is introduced a set of conventional methods used to contain the dynamic elements in a linear and time-invariant response.

6.2 Design Process II (final results and discussions)

6.2.1 Attitude Control Systems

This area is related to the capability of the Automatic Flight Control has in order to control and maintain the attitudes angles of an aircraft in direct response to a variable pilot's command.

One of the most commons of the use of negative feedback, which is the principle of a SAS, is shown in the design process of Pitch Attitude, which is in longitudinal motion. Followed by a Roll and Yaw control, which are in the lateral motion, and consequently the ARI systems to control both in the moment of a failure in the aircraft is introduced.

6.2.2 Flight Path Control Systems

While the attitudes angles is a function from the Flight Control, the control of the path through the space is more properly to a guidance commands. So, this means that we presume that an aircraft has to follow with great precision a set of direction. In order to achieve that a set of control system designs with directional control has been introduced with heading or height hold System.

Other step to improve the tracking of an aircraft's lateral path is the introduction of navigation system, in specially during the approach. To design them it is necessary the use of Attitude Control as a inner loops, using directional system like coordinated turns and helped with radio transmissions and appropriate airborne receivers.

In conclusion we need a set of elements to provide an aircraft with great tracking control, especially during the approach where is essential the use of VOR receivers, ILS localizers and Glide path to obtain the appropriate guidance commands.

6.3 CONCLUSION OVERALL

When we consider the design of an AFCS an engineer will do it successfully when he can not only establish an right model representing the appropriate dynamical behavior of the aircraft to be controlled but also recognize how an effective control system design can be made.

The subjects of the flying and handling qualities of an aircraft are essential. Because these qualities govern the ease, the accuracy, and the precision with which a pilot can manage his/her flying task it is especially important for the designer of an AFCS to understand them, their specifications and how they can be measured. If an aircraft has poor handling qualities, it is necessary to improve the failure by introducing a control system.

To conclude an AFCS needs to be equipped whit SAS, attitude and path control systems. Some being active all the time in a flight, and others being switched in by the pilot only when required for a particular phase or flight.

7. RECOMENDATIONS

It would be appropriated to finish these set of objectives with:

- Finishing landing approach with automatic landing system.
- Test the values in Flight Simulator.

8. REFERENCES

McLean, Donald 1990, *Automatic Flight Control Systems* (Hertfordshire, Prentice Hall International)

APPENDIX

CHARLIE - A very large, four-engined, passenger jet aircraft.

General parameters;

Wing area (m^2)	510
Aspect ratio	7.0
Chord, $\bar{c}$ (m)	8.3
Total related thrust (kN)	900
Centre of gravity (c.g.)	$0.25\bar{c}$
Pilot's location (m) – relative to c.g.	
l_{xp}	26.2
l_{zp}	-3.05
Weight (kg)	250000
Inertias (kg m^2)	
I_{xx}	18.6×10^6
I_{yy}	41.35×10^6
I_{zz}	58×10^6
I_{xz}	1.2×10^{64}

Flight condition				
	Flight condition			
Parameter	1	2	3	4
Height (m)	Sea level	6100	6100	12200
Mach No.	0.198	0.5	0.8	0.8
U_0 (m s^{-1})	67	158	250	250
$\bar{q}$ (N m^{-2})	2810	8667	24420	9911
α_0 (degree)	8.5	+6.8	0	4.6
γ_0 (degree)	0	0	0	0
Stability derivatives(longitudinal motion)				
X_u	-0.021	-0.003	-0.0002	-0.0002
X_w	0.122	0.078	0.026	0.039
X_{δ_E}	0.292	0.616	0.0	0.44
$X_{\delta_{th}}$	3.88×10^{-6}	3.434×10^{-6}	3.434×10^{-6}	3.434×10^{-6}
Z_u	-0.2	-0.07	-0.09	-0.007
Z_w	-0.512	-0.433	-0.624	-0.317
Z_q	-1.9	-1.95	-3.04	-1.57
Z_{δ_E}	-1.96	-5.15	-8.05	-5.46
$Z_{\delta_{th}}$	-1.69×10^{-7}	-1.5×10^{-7}	-1.5×10^{-7}	-1.5×10^{-7}

M_u	0.000036	0.00008	-0.00007	0.00004
M_w	-0.006	-0.006	-0.005	-0.003
$M_{\dot{w}}$	-0.0008	-0.0004	-0.0007	-0.0004
M_q	-357	-0.421	-0.668	-0.339
M_{δ_E}	-0.378	-1.09	-2.08	-1.16
M_{δ_F}	0.7×10^{-7}	0.67×10^{-7}	-0.67×10^{-6}	0.67×10^{-7}