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Research, Design, Simulation And Implementation Of An Automatic Flight Control System For A Real-Time Flight Simulator.

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Science, Freedom, Beauty, Adventure... Aviation Offers It All.
- Charles Lindbergh (1902 – 1974)

SUMMARY

Vertigo Flight Simulation is a company located in The Netherlands with its mission being to design, produce and sell low cost flight simulation devices to all levels of the flight training market. One of the products which is developed by Vertigo Flight Simulation is the Vertigo P-2 Trainer. The P-2 Trainer is a fixed-base flight simulator capable of simulating single and twin engine piston aircraft with Flight & Navigation Procedures Trainer Type II (FNPT II) approved aerodynamic and engine models.

One important feature which is not yet available for the P-2 Trainer is an Automatic Flight Control System (AFCS) which contains an AutoPilot (AP) and a Flight Director (FD). This system is developed, which includes research, design and implementation, in this thesis by the author as its Master of Science (MSc) Thesis Project for obtaining his Engineering Degree in Electrical Engineering, specialization Avionics.

An Automatic Flight Control System has two main functions: the first is to control the aircraft without the need for the pilot to fly the aircraft and the second function is to present the pilot with suggestions how he or she should control the aircraft to follow a certain, by the pilot chosen, attitude, course, altitude or flight plan / flight path.

During research, it was found that Automatic Flight Control Systems are basically all designed and based on classical control system theory: inner-loops are used to control the aircraft (these are called the control loops) and outer-loops are used to guide the aircraft (these are called the guidance loops). The problem initially found in implementation of the system was to tune the various inner- and outer-loop gains to give desired aircraft-responses. This problem was overcome by approaching the problem from the classical control system theory point of view.

Because the P-2 Trainer is developed with flexibility in mind, it uses a modular design to be able to easily change the aerodynamic aircraft-model and instruments to simulate a new type of aircraft. Because of this design, the Automatic Flight Control System must also be a flexible system. Since the method of simulation, testing and tuning the system is extremely time absorbing, a method will be presented to let the system adapt itself every time a new aerodynamic model is loaded in the flight simulator.

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NOTATIONS

General Aviation Notations

ϕ (phi)	Roll Angle / Bank Angle	[deg] or [rad]
θ (theta)	Pitch Angle	[deg] or [rad]
δ_r (delta_r)	Rudder Deflection	[deg]
δ_e (delta_e)	Elevator Deflection	[deg]
δ_a (delta_a)	Aileron Deflection	[deg]
α (alfa)	Angle Of Attack (AoA)	[deg] or [rad]
q	Pitch-Rate ($\theta_{\dot{}}$)	[deg/sec] or [rad/sec]
p	Roll-Rate ($\phi_{\dot{}}$)	[deg/sec] or [rad/sec]
$H_{\dot{}}$	Vertical Speed (V/S)	[ft/sec] or [m/sec]
H	Altitude (ALT)	[ft] or [m]
Ψ (psi; uppercase)	Heading (HDG)	[deg] or [rad]
ψ (psi, lowercase)	Yaw Angle	[deg] or [rad]
γ (gamma)	Flight Path Angle	[deg] or [rad]
r	Yaw Rate ($\psi_{\dot{}}$)	[deg/sec] or [rad/sec]

ABBREVIATIONS AND ACRONYMS

ACO	Aircraft Certification Office
ADC	Analogue To Digital Converter
AFCS	Automatic Flight Control System
AI	Attitude Indicator
ALT	Altitude
AOA	Angle Of Attack
AP	AutoPilot
ATS	Auto Throttle System
DAC	Digital To Analogue Converter
DIO	Digital Input / Output Port
FAA	Federal Aviation Administration
FD	Flight Director
FNPT II	Flight & Navigation Procedures Trainer Type II
fpm	Feet Per Minute
HDG	Heading
HSI	Horizontal Situation Indicator
IAS	Indicated Air Speed
MAP	Mode Annunciator Panel
MCP	Mode Control Panel
MFD	Multi Function Display
PCB	Printed Circuit Board
PFD	Primary Flight Display
STC	Supplemental Type Certificate
V/S	Vertical Speed

1 INTRODUCTION

1.1 Vertigo Flight Simulation

Vertigo Flight Simulation (www.vertigosim.com) is a company located in The Netherlands with its mission being to design, produce and sell low cost flight simulation devices to all levels of the flight training market. One of the products which is developed by Vertigo Flight Simulation is the Vertigo P-2 Trainer (Fig. 1.1 & Fig. 1.2).



Fig. 1.1: The Front Of The Vertigo P-2 Trainer.



Fig. 1.2: The Cockpit Of The Vertigo P-2 Trainer.

The P-2 Trainer is a fixed-base flight simulator capable of simulating single and twin engine piston aircraft with Flight & Navigation Procedures Trainer Type II (FNPT II) approved aerodynamic and engine models. Due to its use of advanced glass cockpit and instrument techniques, the simulator is extremely flexible. The instrument panel can therefore be modified very easily if the customer wants its trainer to simulate another type of aircraft or the same type of aircraft with a different instrument layout.

1.2 The Automatic Flight Control System

One important feature which is not yet available for the P-2 Trainer is an Automatic Flight Control System (AFCS) which includes an AutoPilot (AP) and a Flight Director (FD). The development of this AFCS, which includes research, design, implementation and testing, is done by the author as its Master of Science (MSc) Thesis Project. The project was supervised by two mentors:

- *dr.ir. T.J.J. van den Boom* - member of the scientific staff of the Delft Center For Systems And Control (DCSC) department of the Delft University of Technology (TU Delft).
- *ir. A. Levy* - CEO and founder of Vertigo Flight Simulation and as well an airline pilot by the Royal Dutch Airlines KLM.

1.3 Description Of The Development

The title of this thesis is: 'Research, Design, Simulation And Implementation Of An Automatic Flight Control System For A Real-Time Flight Simulator.' In this title four important aspects of this project can be found:

- Research
- Design
- Simulation
- Implementation

→ Research

The research started with a comprehensive study of literature. During this research, only basic theory of AFCS's was found. Because more info was needed for the design of an AFCS for the Vertigo Trainer, an avionics repair and maintenance company at Schiphol Airport in The Netherlands was found and was willing to cooperate to provide the author access to their extensive library with avionics manuals. These manuals provided the author with data from 'real' AFCS's. In chapter 3.5 and appendix B, this data is presented.

In pursuit of more information about AFCS's, an aviation engineer at The Boeing Company (www.boeing.com) was contacted. However, due the 9/11/2001 incident in New York, USA, the information export laws were enforced rigorously and the engineer was restricted to provide the author with the requested information.

→ Design

From a comprehensive review of the literature, the inner structure of an AFCS was found. The problem with the information found in the literature, is the fact that all this information refers to 'real' AFCS's which are used in 'real' aircrafts. Since our project is part of a flight simulator project and not of a real aircraft project, our goal is not to develop a real AFCS, but rather a simulation of a real AFCS. During the process of designing the AFCS, we started with the simplest AFCS which has only a few basic functions / modes. Then this simple AFCS was implemented and tested. Due to time constraints, it was unfortunately not possible to expand the design to a more complex AFCS with more complicated functions and modes.

→ Simulation

Before implementation, the designed system was first analysed and simulated with The MathWorks Matlab & Simulink Release 13[®]. During these analysis and simulations, the designed system was evaluated and extra feedback-loop(s) were added to improve the aircraft-responses.

→ Implementation

When the basic system was completely designed, the design was implemented into a computer program / routine. The computer language C++ was used and the program Borland C++ v4.52[®] was utilized for the implementation. The generated code will be merged in the future into the main program of the P-2 trainer.

Due to time constraints, the implemented design could not be tested in the P-2 Trainer or even on the stand-alone computer. The plan was to first test the implemented design on the stand-alone computer which runs the entire aerodynamic and navigation models. This approach was chosen to work more efficiently, save time and save developing costs. The stand-alone computer will work identical to the main computer of the P-2 with one input (the keyboard) and one output (the monitor screen). Because of this approach, there is a difference between the real trainer and the stand-alone computer. In the real trainer, when the main computer wants to position the steering gear at a certain point, it sends a signal to a servo-motor which moves the whole steering gear. A force sensor on the steering gear detects the movement of the steering gear and sends a feedback signal to the main computer. Because of the mass inertia of the whole system, it takes some time for the steering gear to arrive at the position where the main computer wants it to be. This delay will not be present on the stand-alone computer because on this computer, the position output of the AFCS is also the position input to the ailerons and elevators. As a result, on the stand-alone computer, the program must simulate the mass inertia delay which is present on the real trainer. If this is not done, the simulated aircraft will be controlled much too quickly and hence the system will be unstable very rapidly and it will not be a good simulation of how the AFCS will work on the real trainer.

Another aspect of this project which had to be cancelled due to time constraint, was the design, implementation and testing of an auto-tuning AFCS. Because the flight trainers of Vertigo Flight Simulation are developed with flexibility in mind, the designed AFCS has to be developed in the same way. Flexibility in this context means the ability to use the same AFCS for simulating AFCS's for different types of simulated aircraft. The P-2 trainer simulates single and twin engine piston aircraft and the future trainers will simulate jet engine aircraft. Therefore the developed AFCS has to be able to be used for simulation of a wide range of aircrafts. In other words: the developed AFCS has to be reconfigurable without much effort. This is an important design aspect which has to be kept in mind during the whole process of design. It would be superb if the system could completely adjust itself when a new aircraft is simulated without the need for the flight simulator engineer to adjust the program. But because the simulated aircrafts are of such different types, a completely adaptive AFCS will probably not be possible. The best possible alternative is to let the AFCS adjust its own variables based on a list of aircraft-specific parameters. Research has to be done to know exactly which aircraft-specific parameters will have great influence on the performance of the AFCS. The goal will have to be that the list of aircraft-specific parameters will have to be entered only one time when the aerodynamic model of the simulator is changed. The ideas the author has on this design aspect, will be described very shortly in chapter 6.

1.4 Structure Of The Vertigo P-2 Trainer

For the design of the AFCS, it is essential to have knowledge of the structure of the Vertigo P-2 Trainer and the place of the AFCS in this structure. In Fig. 1.3, a block diagram is shown of the complete simulator:

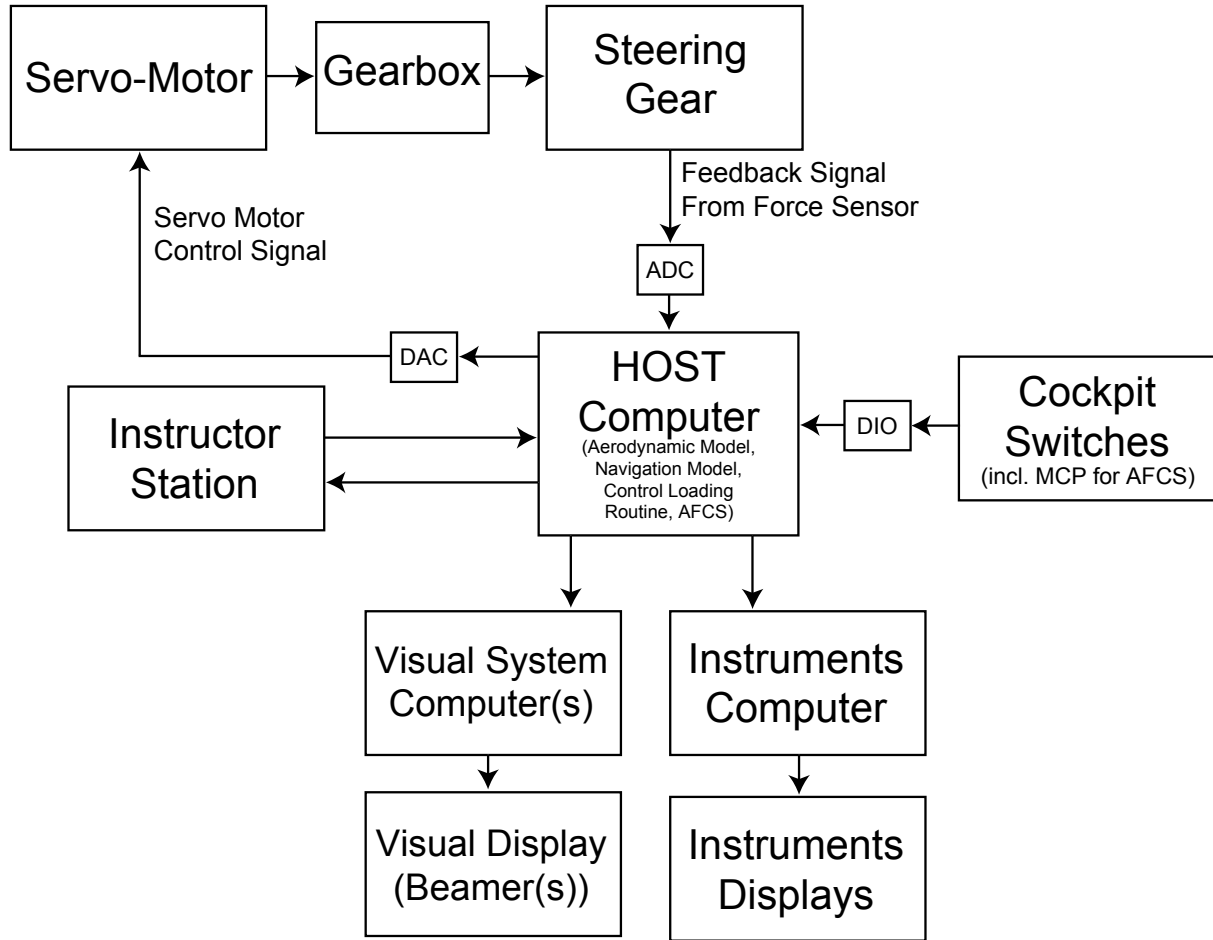


Fig. 1.3: A Block Diagram Of The Vertigo P-2 Trainer.

The Servo-Motors are driven by Servo-Amplifiers which are not drawn in Fig. 1.3.

The AFCS which was designed and implemented is a routine / program which runs on the HOST computer. This computer is the core of the flight simulator because it does all the important calculations (aerodynamic model calculations, navigation model calculations, control loading calculations and the AFCS calculations). The HOST computer is connected via a network to the other computers and it sends and receives data from the other modules via digital-to-analogue converters (DAC's), analogue-to-digital converters (ADC's) and digital I/O ports (DIO's). In Fig. 1.4, an inside look of the HOST computer is drawn.

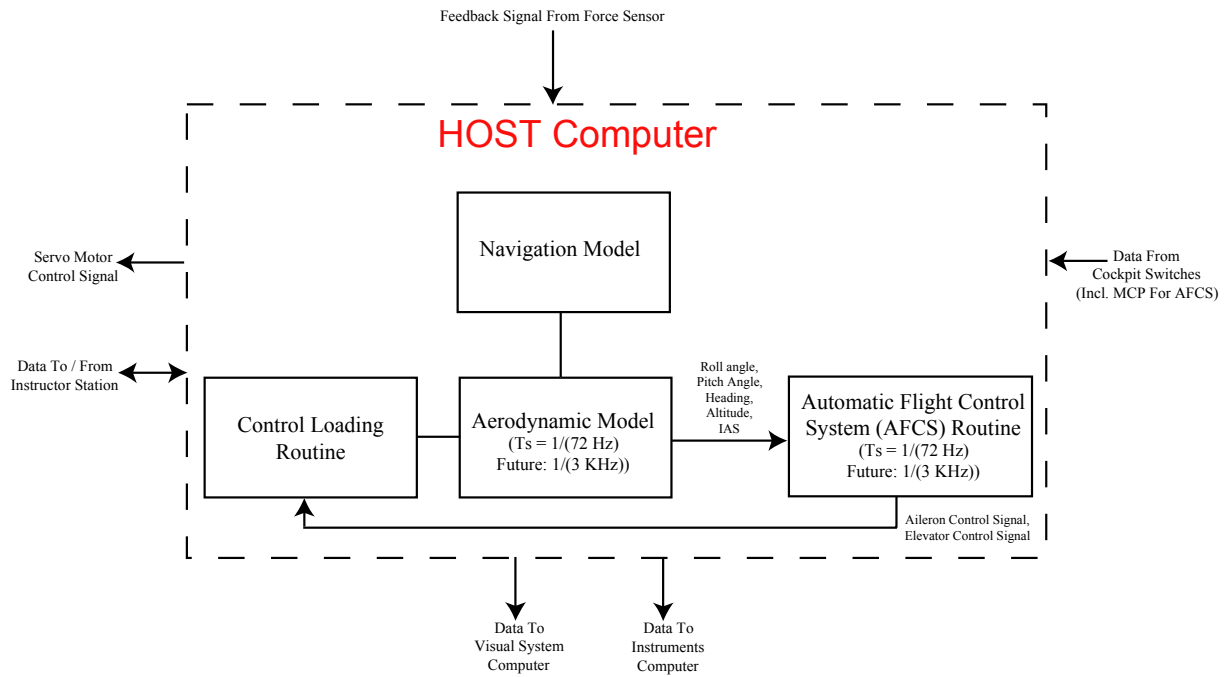


Fig. 1.4: Inside the HOST Computer.

It is illustrated in this figure that the AFCS routine gets the information it needs from the aerodynamic model, the Mode Control Panel or MCP and sometimes from the Horizontal Situation Indicator or HSI. It does the required calculations (depending on which mode is active) and sends an aileron- and / or elevator-output signal to the control loading routine when the AP is active or it sends the command signals for the Primary Flight Display (PFD) to the instruments computer when the FD is active.

For the design of the AFCS, information about only the AFCS Routine block is needed, but the extra information about the place in and the connections to and from the AFCS in the simulator might come in handy. For now, we will concentrate only on the AFCS block diagram as drawn in Fig. 1.5:

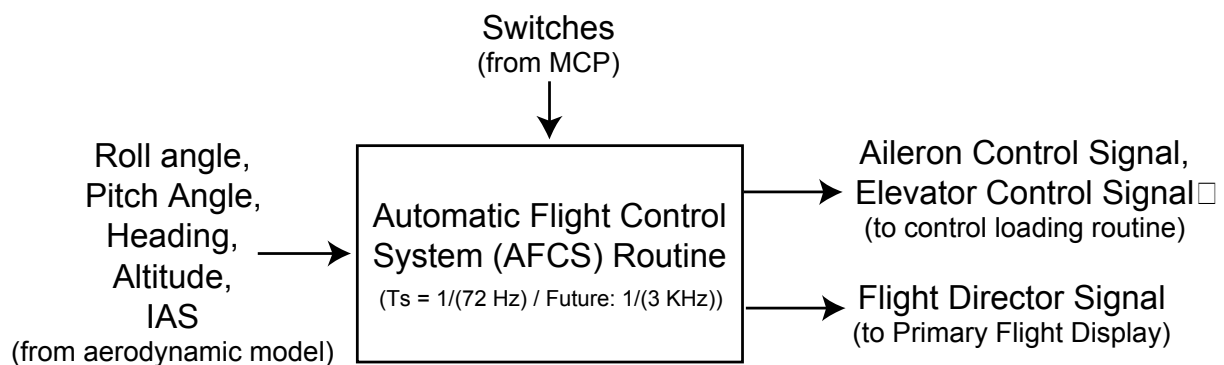


Fig. 1.5: The Automatic Flight Control System Routine.

1.5 Structure Of The Thesis

To familiarize the reader with aviation and certain aspects of controlling aircrafts, chapter 2 will provide some basic aviation theory.

Chapter 3 describes what Automatic Flight Control Systems are, what functions they have and which modes are available. After a short story about what kind of actuators are used in aircrafts, The research of 'real AFCS's', which was done by Saab Aviocomp RIAS, is described. The findings can also be found in appendix B.

The core of this project is the design which is described in chapter 4. Unfortunately, due to time constraint, only basic modes could be described. But these modes are the most important modes of the AFCS design and all to more advanced modes are believed to include these basic inner-loops. The approach which is chosen in this chapter, is that first the design of the longitudinal control of the aircraft is described and the second part describes the design of the lateral control of the aircraft. The two designs are very similar and therefore will the latter description not be as complete as the first one.

Chapter 5 is dedicated to the implementation of the design into a computer program / routine. A generic PID controller implementation is described together with an anti-windup design. The anti-windup design is inevitable for a system which uses an integrator and which has non-linearities.

The last chapter, chapter 6, is called 'Conclusions And Recommendations' and this chapter summarizes and discusses the main results of this project. Recommendations are also given to help others expand this project into a real complete working system for the Vertigo trainer.

For the fanatic reader, addition data can be found in the appendices, but this data is not necessary to understand the main design.

2 BASIC AVIATION THEORY

2.1 The Control Surfaces Of An Aircraft

To understand the contents of this thesis and to familiarize the reader with certain terms, the following short introduction to aircraft control is written.

An aircraft has three axes: the longitudinal axis, the lateral axis and the normal (vertical) axis. All the axes run through the Centre of Gravity (c.g.) of the aircraft and are perpendicular to each other. The basic control surfaces that control an aircraft around these three axes are (see Fig. 2.1):

- The ailerons on the wings for movement around the longitudinal axis (lateral movement). This movement is called 'roll'.
- The elevators on the horizontal stabilizer for movement around the lateral axis (longitudinal movement). This movement is called 'pitch'.
- The rudder on the vertical stabilizer for the movement around the normal axis (lateral movement). This movement is called 'yaw'.

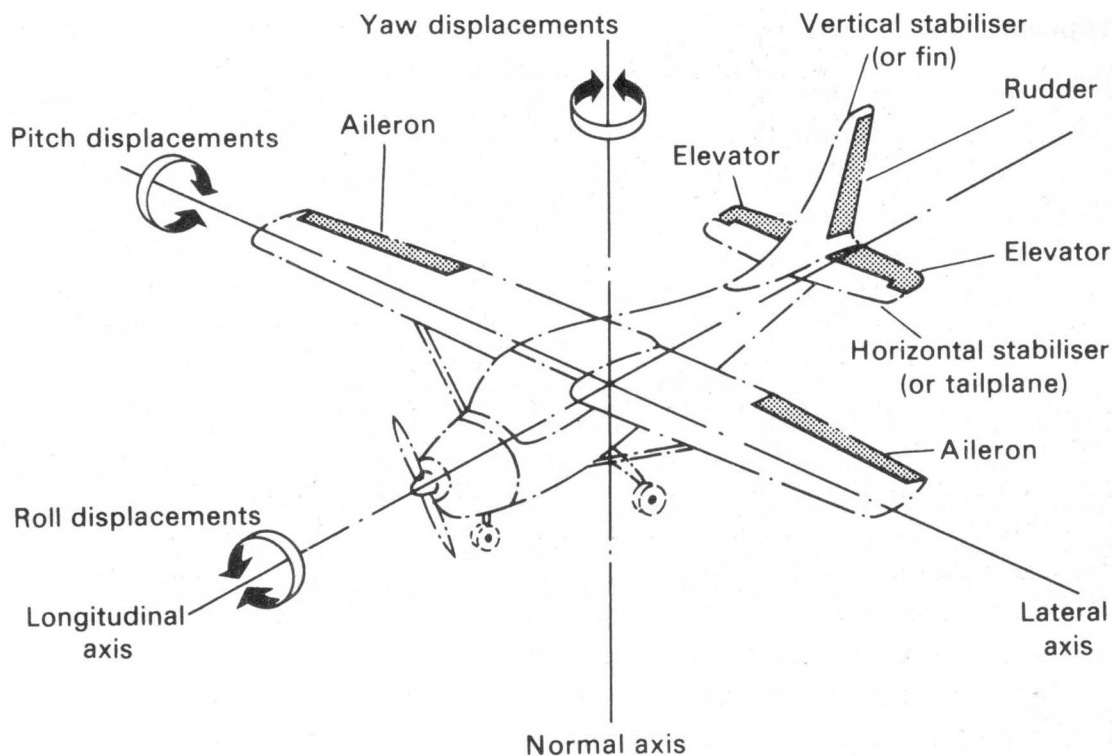


Fig. 2.1: The Basic Control Surfaces Of An Aircraft.

The control device which controls the ailerons and the elevators is called the (control) yoke in conventional aircrafts or joy-stick in some fly-by-wire aircrafts. The ailerons are controlled by a lateral movement of the yoke (left- to right movement) and the elevators are controlled by a longitudinal movement of the yoke (front- to back-movement). To control the rudder, (feet) pedals are used.

2.2 Aircraft Movements

2.2.1 The Roll Movement

The purpose of making the aircraft roll is to change the heading of the aircraft. This action is controlled by the ailerons on the wings of the aircraft and can be divided into 2 parts: the first part consists of initiating the turn and the second part consists of establishing the turn. During the initialization of the turn, the aircraft rolls from wings level (that is normally the default position) to the desired roll angle or bank angle as it also sometimes called. During this manoeuvre, the roll-rate ' p ' is mostly constant (of course not in the beginning and the end). When the desired bank angle is almost reached, the pilot will stop the rolling and if the pilot did his or her job correctly, the roll-rate will be zero at the point when desired bank angle, and as a result the turn, is established. At this point the aircraft will turn with a constant turn-rate.

Because the roll movement is an asymmetric movement, the directional motion have an influence on the lateral motion and vice versa. This effect is called aerodynamic coupling [22].

The ailerons are interconnected in the control system and work simultaneously, but in opposite directions to each other. When a pilot wants to change the roll of an aircraft, he or she will move the yoke to the left (for a left roll movement) or to the right (for a right roll movement). As a result the left aileron moves upwards or downwards and the right aileron moves downwards or upwards. An upward movement of the aileron causes the lift to decrease on that side of the fuselage and a downward movement of the aileron causes the lift to increase on that side of the fuselage. This opposing action allows more lift to be produced by the wing on one side of the fuselage than on the other side. This results in a controlled movement or roll because of the unequal forces on both the wings. In a roll, the angle of attack α of the down-going wing will increase and the angle of attack of the up-going wing will decrease. Because a larger angle of attack will produce more lift and a smaller angle of attack will produce less lift, a rolling moment opposing to the initial roll movement (!) will occur (see Fig. 2.2).

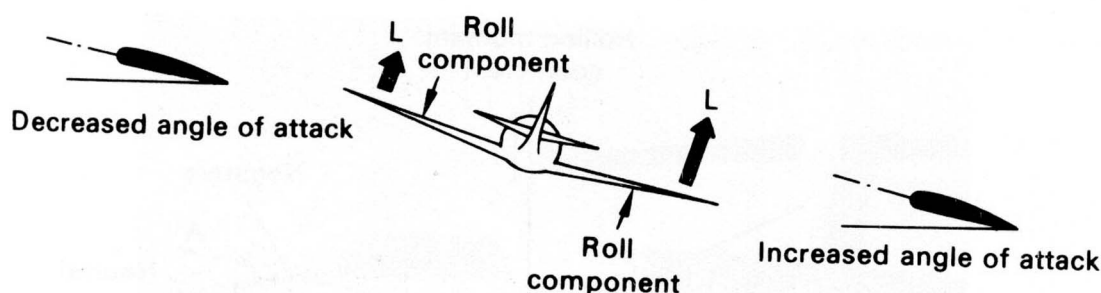


Fig. 2.2: The Roll Movement Of An Aircraft.

The opposing roll movement will damp the initial roll movement with a damping which is proportional to the rate of displacement. When the forces on both wings are equal, a constant roll-rate is established. When the pilot puts the yoke in the centre position, the ailerons will be no longer deflected and this produces again a difference in lift between the down-going and up-going wing. This will cause a rolling moment in the opposite direction until the lift on both wings is equal again and at that moment the roll-rate is zero

and the roll angle remains constant. Therefore, the ailerons control the roll-rate and this is the reason why we call ailerons rate control devices instead of position control devices.

However, the aircraft also experiences a sideslipping motion which is caused by the inclination of the lift vectors at the roll angle (see Fig. 2.3). The moment caused by this motion tends to return the aircraft to a wings-level condition. Therefore a pilot always has to hold the yoke a little bit to the left or to the right when he or she wants to maintain a constant bank angle, but the displacement will be much smaller than the initial displacement. However this differs greatly on aircraft design.

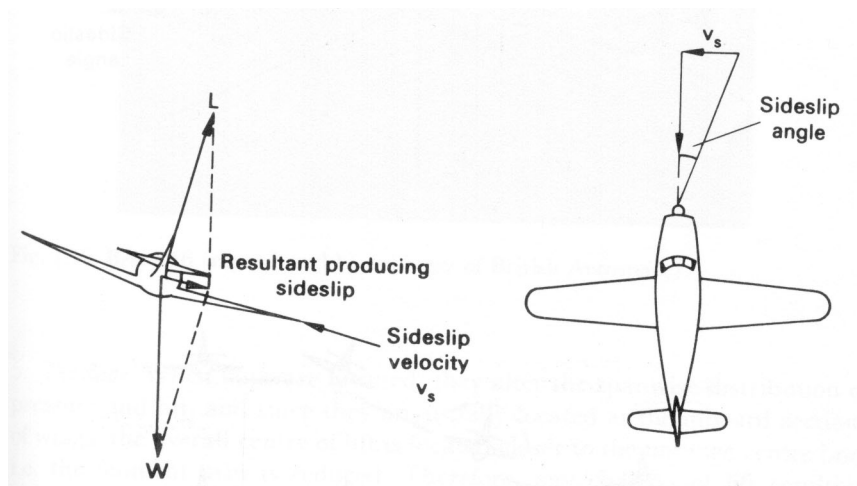


Fig. 2.3: The Sideslipping Motion.

The movement of the ailerons also have a negative side effect: the downward aileron will have a greater drag than the upward aileron and therefore a yawing moment in the direction opposite of being intended is established, resulting into a sideslip. This effect is called 'adverse aileron yaw' (see Fig. 2.4). To counteract for this effect, pilots must often use the rudder to make a coordinated turn (a turn is coordinated when there is no slip along the lateral axis). The aircraft designer can also counteract this effect by letting the ailerons operate differentially: the up-going aileron has a greater deflection than the down-going aileron.

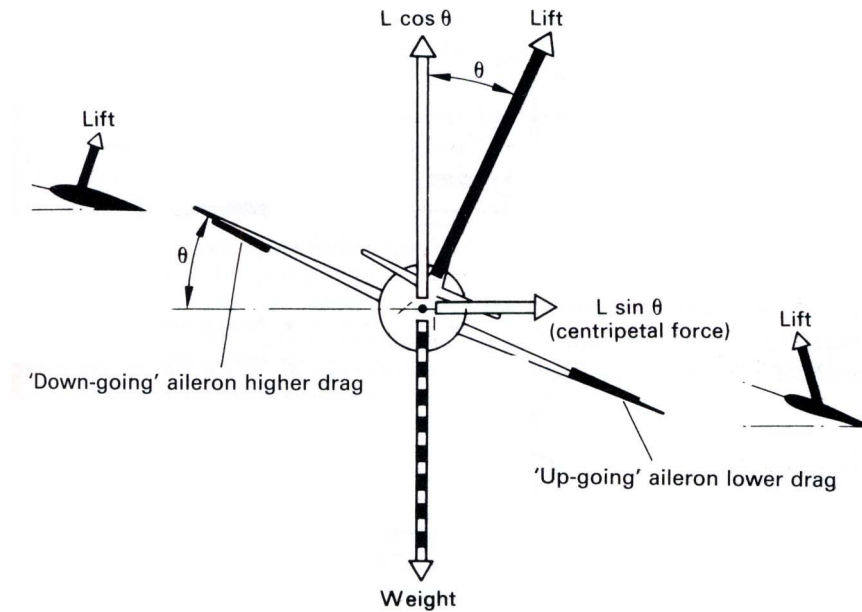


Fig. 2.4: Rolling And Turning.

Another important effect which occurs during a roll is that the wing on the outside of the turn (the up-going wing) has a greater speed than the wing on the inside of the turn (the down-going wing). The faster wing produces more lift than the slower wing ($Lift\ L = C_L \cdot S \cdot \frac{1}{2} \cdot \rho \cdot V^2$) and the result is that the aircraft's roll will be increased. This means that the aircraft has negative lateral static stability. This effect can be counteracted by the aircraft designer by the following methods:

- Dihedral Effect– Placing the wings not completely horizontal, but under a small angle, so the wings will look like a 'V-shape' when watched from the front or back.
- High Wings – Placing the wings on top of the fuselage (above the c.g.).
- Swept Wings – Making the shape of the wings look like a 'V-shape' when watched from the above or below the aircraft.

The roll angle ϕ is defined as the angle between the lateral axis and the horizontal plane as shown in Fig. 2.5:

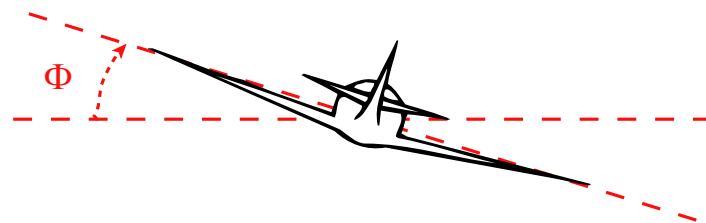


Fig. 2.5: The Roll Angle ϕ .

2.2.2 The Pitch Movement

One purpose of making the aircraft pitch is to change the altitude of the aircraft. When the yoke is moved forward or backward, the elevators on the horizontal stabiliser of the aircraft are moved downwards or upwards. The elevators on both sides of the tail will deflect both in the same way. When the elevators are down-wards deflected, the

(negative) lift on the horizontal stabiliser increases and because the horizontal stabiliser is at the back of the aircraft behind the aircraft's centre of gravity, a pitching moment will occur which reduces the pitch angle of the aircraft. When the elevators are up-wards deflected, a pitching moment will occur which increases the aircraft's pitch angle. When the pitch angle is increased and the aircraft starts climbing, the lift vector of the aircraft will be inclined (see Fig. 2.6). Because of this inclination, the lift vector will be less than the weight vector and therefore a greater thrust vector is needed to let the aircraft gain altitude. A greater thrust vector means of course more power.

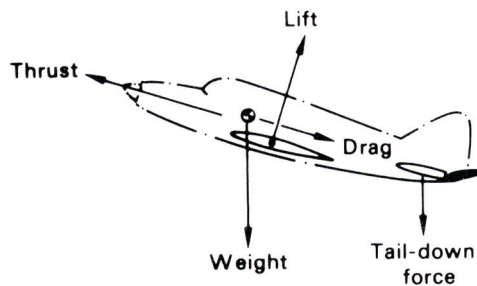


Fig. 2.6: Forces On A Climbing Aircraft.

Elevators are essentially displacement control devices because they control the pitch angle of the aircraft (when the elevators' deflection is fixed, the pitch angle will not change and will also be fixed).

The pitch angle θ is defined as the angle between the horizontal plane and the airframe and it is the summation of the angle of attack α and the flight path angle γ (see Fig. 2.7 and equation (1)),

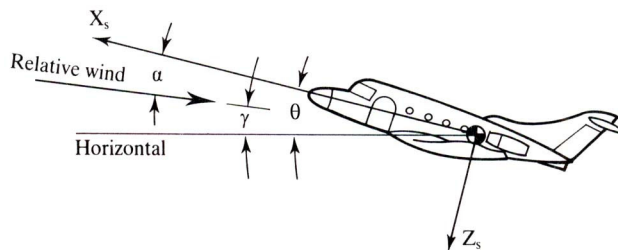


Fig. 2.7: Definition Of Pitch Angle θ , Angle Of Attack α And Flight Path Angle γ .

$$\theta = \gamma + \alpha \quad (1)$$

2.2.3 The Yaw Movement

As explained before, rudder is used to counter effect the 'adverse aileron yaw', but also to counter effect the wind. The rudder is a displacement control device on the vertical stabiliser of the aircraft.

2.3 Interconnection Between The Yoke And The Control Devices

The connection between the yoke and the control devices, described in the previous sections, can be one of four types:

- Completely mechanical control (cables and push-rods).
- Mechanical control with hydraulic assistance (power-assisted; like power-steering in a car).
- Mechanical control with complete hydraulic power (power-operated).
- Fly-by-wire or fly-by-light.

When one of the first two systems is used, the pilot can feel the aerodynamic forces, which work on the control-surfaces, on the yoke because there is some sort of direct mechanical linkage between the yoke and the control-surfaces. With the other two systems, there is only an indirect mechanical linkage between the yoke and the control-surfaces and therefore an artificial feel unit or some sort of spring connected to the yoke is used to simulate the aerodynamic forces on the control-surfaces. This is necessary because without feeling the forces, the pilot will be unable to accurately control the aircraft.

The most important consequence of using a system which is not completely mechanical controlled is a delay between the movement of the yoke and the actual movement of the control-surfaces. Aircraft engineers are always trying to minimize this delay because such a delay is unwanted and it gives the pilot a feeling of not being completely in direct control of the aircraft.

3 AUTOMATIC FLIGHT CONTROL SYSTEMS

3.1 Functions Of An Automatic Flight Control System

An Automatic Flight Control System has two main functions: the first is to control the aircraft without the need for the pilot to fly the aircraft and the second function is to present the pilot with suggestions how he or she should control the aircraft to follow a certain, by the pilot selected, attitude, course, altitude or flight plan / flight path. A block diagram of a basic AFCS is drawn in Fig. 3.1.

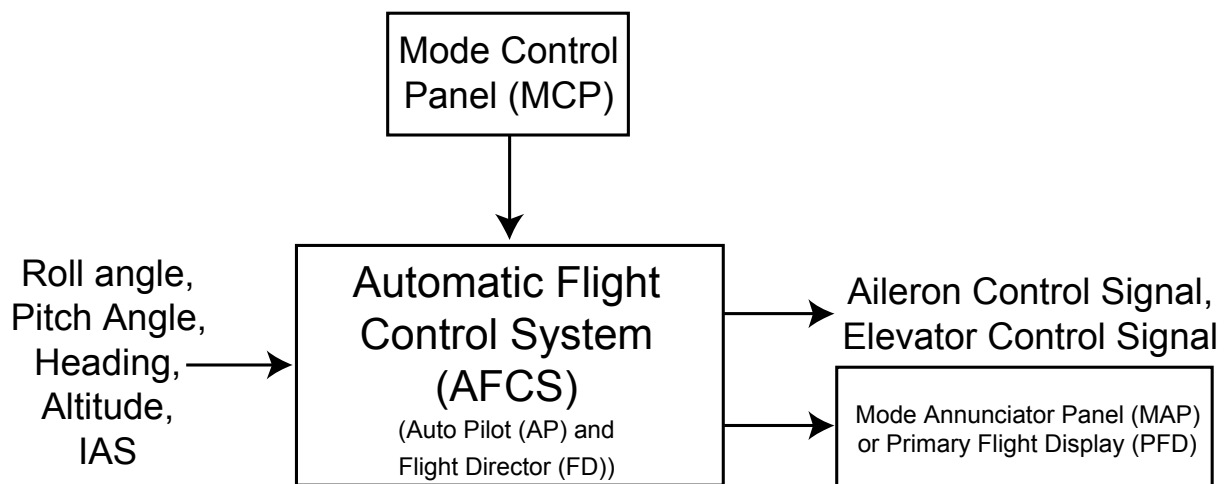


Fig. 3.1: Block Diagram Of An Automatic Flight Control System.

It is seen in this figure that an AFCS has the following inputs:

- ➔ Roll Angle
- ➔ Pitch Angle
- ➔ Heading
- ➔ Altitude
- ➔ Indicated Air Speed (IAS)
- ➔ Inputs From The MCP

It is also possible that more advanced AFCS's have the following additional inputs:

- ➔ Roll-Rate
- ➔ Pitch-Rate
- ➔ Yaw-Rate
- ➔ Vertical Speed
- ➔ Navigational Data (Including GPS & ILS Data)

The outputs of an AFCS are:

- ➔ Aileron Control Signal (to Servo / Actuator (in AP mode) or to AI (in FD mode))
- ➔ Elevator Control Signal (to Servo / Actuator (in AP mode) or to AI (in FD mode))
- ➔ Annunciator Signals (to MAP or PFD)

Advanced AFCS's can have the following additional output signal:

- ➔ Rudder Control Signal (to Servo / Actuator)

The two main functions of an AFCS are:

1. The AutoPilot Function.

An AutoPilot (AP) is a system which automatically controls / flies the aircraft. When the AP is engaged, the pilot can let go of the yoke and the pedals and from that moment on, the aircraft is in the hands of the AP. The pilot's job becomes from then on a monitoring job. When the pilot detects that the AP doesn't work satisfactory, he must disengage the AP and take control of the aircraft himself.

2. The Flight Director Function.

The Flight Director (FD) gives a visual presentation of the AFCS background calculations. The FD can be used with or without the AP engaged and therefore it has two functions: a monitor function and a director function. When the AP is engaged and the FD is switched on, the pilot can monitor which data is computed by the AFCS and consequently he can see if the AP is controlling the aircraft exactly like the AFCS computes. The director function is active when the FD is used without the AP. In this mode, the AFCS is not controlling the aircraft; it only computes how the aircraft should be controlled to satisfy the selected guidance function. The pilot should follow these control suggestions. The computed data is, just like with the monitor function, displayed to the pilot on the Attitude Indicator (AI). The AI can be a stand-alone instrument or it can be integrated into a Primary Flight Display (PFD). Two commonly used presentations for the FD are used:

→ A 'Cross-Hair' Type Flight Director:

The FD is presented as two needles on the AI (one vertical and one horizontal needle, see Fig. 3.2). When the two needles are in the middle, the FD commands are followed correctly. When e.g. the horizontal needle goes upwards, the pilot has to increase the pitch-angle and upon reaching the desired pitch-angle, the horizontal needle will start to move to the centre of the display and the pilot will have to decrease the pitch angle. In other words: the pilot has to 'follow the needles'. In the figure, the commands are: pitch up and roll right.



Fig. 3.2: A 'Cross-Hair' Type Flight Director.

→ An 'Inverted-V' Type Flight Director:

The FD presentation consists here of two elements: the aircraft is presented by a 'triangular-V-shape' symbol; the roll- and pitch commands are presented by an 'inverted-V-shape' symbol (see Fig. 3.3). The pilot has to control the aircraft in

such a way that the aircraft symbol is tucked up into the 'inverted-V-shape' symbol.



Fig. 3.3: A 'Inverted-V' Type Flight Director.

3.2 Control And Annunciator Panels

An AFCS is furthermore always connected to some sort of panel to select the modes and guidance functions. This panel is called a Mode Control Panel (MCP) and is always present in the cockpit. With this panel, the pilot can enable or disable the AP or FD and select the various modes of the AFCS. There are a lot of different types of MCP's used in aircrafts. The MCP is not depended on the aircraft-type, but depended on the AFCS used. The MCP can be very simple with only a few switches, see Fig. 3.4, Fig. 3.5 and Fig. 3.6:

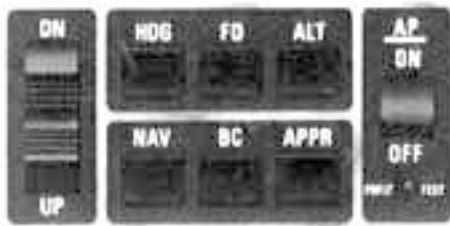


Fig. 3.4: A Mode Control Panel (MCP) From A Bendix / King KFC-200 AFCS (KC-290).

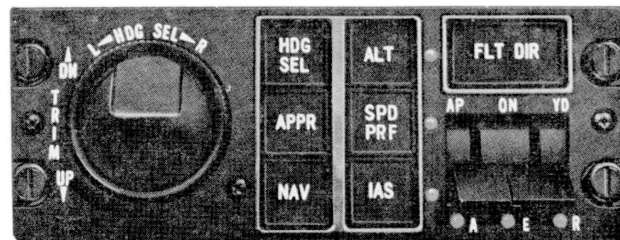


Fig. 3.5: A Mode Control Panel (MCP) From A Bendix / King KFC-300 AFCS (KMC-340).



Fig. 3.6: A Mode Control Panel (MCP) From A Bendix / King KFC-150 AFCS (KC-192).

The MCP can also be very complex when the AFCS consists of many sophisticated modes (e.g. like in Fig. 3.7, which is a Rockwell / Collins MCP used in a Boeing 777).



Fig. 3.7: A Rockwell / Collins Mode Control Panel (MCP) Used In A Boeing 777.

A complex MCP has a button to select the desired heading, but a simple MCP doesn't have such a button and the desired heading in such an aircraft is selected via a turn-knob on a instrument called the Horizontal Situation Indicator (HSI).

The simplest AFCS's also have a separate panel to show the active and selected modes. This panel is called a Mode Annunciator Panel (MAP). An example of such a panel is the Bendix / King KA-285 which is used in the KFC-200 AFCS (Fig. 3.8).



Fig. 3.8: A Mode Annunciator Panel (MAP) From A Bendix / King KFC-200 AFCS (KA-285).

The more complex AFCS's don't have such a panel. With these systems the active and selected modes are shown on the MCP (see Fig. 3.6 & Fig. 3.7) and with some systems they are shown on the PFD.

3.3 Basic Modes

An AFCS will always have a few simple basic modes. AFCS's in small aircraft may only have these basic modes, while AFCS's in large commercial transport aircraft are very sophisticated with a lot of additional modes. The modes of an AFCS can be categorized into lower- and upper-modes. The difference between the two modes is that in a lower-mode, the desired guidance parameter is equal to the current guidance parameter on the moment of engagement of the mode (the parameter is being hold on the specific value). In an upper-mode, the value of the guidance parameter is chosen by the pilot (the parameter is selected).

3.3.1 The basic lower-modes

For now, we will concentrate solely on the following basic lower-modes (their descriptions are self-explanatory):

Roll & Pitch Channel:

→ Attitude Hold Mode (ATT HLD)

Roll Channel:

→ Heading Hold Mode (HDG HLD)

Pitch Channel:

→ Altitude Hold Mode (ALT HLD)

→ Vertical Speed Hold Mode (V/S HLD)

3.3.1.1 Attitude Hold Mode (ATT HLD)

In this mode, no guidance parameter has to be tracked and therefore, this mode is only a control mode.

At the moment this mode is selected, the values of the roll- and pitch angles are stored in memory and these values are used as reference values for the input of the AFCS. Some AFCS's hold the wings of the aircraft level in this mode and consequently only the pitch angle is stored in memory at the moment the mode is active for such systems. When the ATT HLD mode with these systems is selected when the aircraft has a non-zero roll-angle, the AFCS slowly decreases the roll-angle to zero and then holds the wings level.

3.3.1.2 Heading Hold Mode (HDG HLD)

In this mode, the heading Ψ is the guidance parameter and thus is this mode a guidance mode. The reference heading Ψ_{ref} is synchronized to the current heading Ψ_{cur} when the mode is not active. At the moment the mode is selected, a switch opens and the reference heading doesn't change anymore and is held on the last value at the moment of enabling the mode. When the heading of the aircraft changes due to turbulence or due to another circumstance, the AFCS will detect a difference between the reference heading and the current heading and consequently will command a roll-angle so the aircraft will bank in the direction of the desired heading. An inner control loop will notice the new desired roll-angle and as a result the system will move the ailerons so the aircraft starts to bank and changes heading.

3.3.1.3 Vertical Speed Hold Mode (V/S HLD)

The vertical speed 'V/S' of an aircraft can be controlled by either changing the pitch angle or by changing the engine-power. Advanced AFCS's have an additional Auto Throttle System (ATS) which controls the engine power. We will concentrate on simple AFCS's without this ATS. Therefore, the pitch angle is the only parameter which is available to control the vertical speed. This mode is, for example, useful when the pilot wants to climb with a fixed vertical speed after takeoff.

The vertical speed is the guidance parameter and thus is this mode also a guidance mode. The reference vertical speed V/S_{ref} is synchronized to the current vertical speed V/S_{cur} when the mode is not active. At the moment the mode is selected, a switch opens and the reference vertical speed doesn't change anymore and is held on the last value at the moment of enabling the mode. When the vertical speed of the aircraft changes due to turbulence or due to another circumstance, the AFCS will detect a difference between the reference vertical speed and the current vertical speed and consequently will command a pitch-up or a pitch-down command. An inner control loop will notice the new desired pitch-angle and as a result the system will move the elevators so the aircraft starts to change pitch and gains or loses vertical speed.

3.3.1.4 Altitude Hold Mode (ALT HLD)

In this mode, the altitude 'H' is the guidance parameter and thus is this mode also a guidance mode. After reading the description of the V/S HLD mode, the reader may conclude that the ALT HLD mode is simple the V/S HLD mode with the reference vertical speed set to zero. This is however not correct. The reason is that it is possible that the

aircraft loses altitude because of turbulence and that the sudden and short vertical speed change is not detected by the instrument. This is because the instrument is only usable for long-lasting changes in vertical speed. Therefore the altitude is changed, but the system will not detect this change because it believes the vertical speed never changed. It is therefore necessary to use the altitude H as the input for this mode.

The reference altitude H_{ref} is synchronized to the current altitude H_{cur} when the mode is not active. At the moment the mode is selected, a switch opens and the reference altitude doesn't change anymore and is held on the last value at the moment of enabling the mode. When the altitude of the aircraft changes due to turbulence or due to another circumstance, the AFCS will detect a difference between the reference altitude and the current altitude and consequently will command a pitch-up or a pitch-down command. An inner control loop will notice the new desired pitch-angle and as a result the system will move the elevators so the aircraft starts to change pitch and gains or loses altitude.

3.3.2 The basic upper-modes

The basic upper-modes are not very different than the 'hold' modes. The only difference is that instead of holding the input parameter on the value it was at the time of enabling the mode, the reference parameter is chosen by the pilot. The heading, altitude and / or vertical speed are preset on the MCP. In smaller aircrafts, the heading can often be selected by a knob on the HSI.

The following three modes are the basic upper-modes:

Roll Channel:

→ Heading Select Mode (HDG SEL)

Pitch Channel:

→ Altitude Select Mode (ALT SEL)

→ Vertical Speed Select Mode (V/S SEL)

3.4 Actuators Used In AFCS's

For an AFCS to be able to control the control-surfaces, actuators (sometimes called servos, servomotors or servo-actuators) are used. If the aircraft uses a hydraulic power-operated or a fly-by-wire control-system, the units of these systems and the AFCS actuators can be combined into one single unit and therefore eliminating the need for an additional AFCS actuator-unit.

The actuators can be classified into two categories: electro-mechanical and electro-hydraulic. An electro-mechanical actuator is a system which consists of a servomotor, a tacho generator (connected to the servomotor), a synchro system and a clutch and gear system. A tacho generator generates a signal which is proportional to its revolutions per minute. Since the tacho generator is connected to the servomotor, its output signal can be used as a measure for the movement speed of the control-surfaces (this is called rate). An electro-hydraulic actuator uses a Linear Variable Differential Transformer (LVDT) type transducer to provide a position feedback signal to the AFCS.

In the pitch- and roll channel, the servomotor is directly coupled to the steering mechanism. This means the yoke moves with the movement of the servomotor. The actuator is therefore connected in parallel with the primary flight control system. In the

yaw channel (Yaw Damper), the steering-pedals do not move when the servomotor moves and the total deflection of the rudder is an accumulation of the AFCS command and the pilot command. In this case, the actuator is connected in series with the primary flight control system.

3.5 Research Of 'Real' AFCS's

During research, only basic theory of AFCS's was found in literature. Because more info was needed for the design of an AFCS for the Vertigo Trainer, an avionics repair and maintenance company at Schiphol Airport in The Netherlands was found. This company, Saab Aviocomp RIAS (www.aviocomp.nl), was approached and was willing to cooperate to provide the author access to their extensive library with avionics manuals ([5],[6],[7],[8]).

The following systems were researched:

- Bendix / King KFC-150 Flight Control System
- Bendix / King KFC-200 Flight Control System
- Bendix / King KFC-300 Flight Control System

These systems are very basic and pretty old without sophisticated features, but are still widely used in small aircrafts. These systems were chosen on purpose because it is believed that even today's advanced AFCS's are functionally based on the old principles.

Before the research was conducted, a few questions were formulated which needed to be answered during the research:

- How are 'real' AFCS's designed / constructed?
- How are 'real' AFCS's tuned prior and after installation?
- How are the tuning values determined?
- Is a particular AFCS tuned for a range of aircrafts or is each AFCS tuned specifically for one aircraft type?
- Which constraints do 'real' AFCS's use?

3.5.1 The Bendix / King KFC-200

The system first studied was the Bendix / King KFC-200. From the manuals, the author extracted all the relevant in- and output data and the general construction of the system. The result is shown in Fig. 3.9:

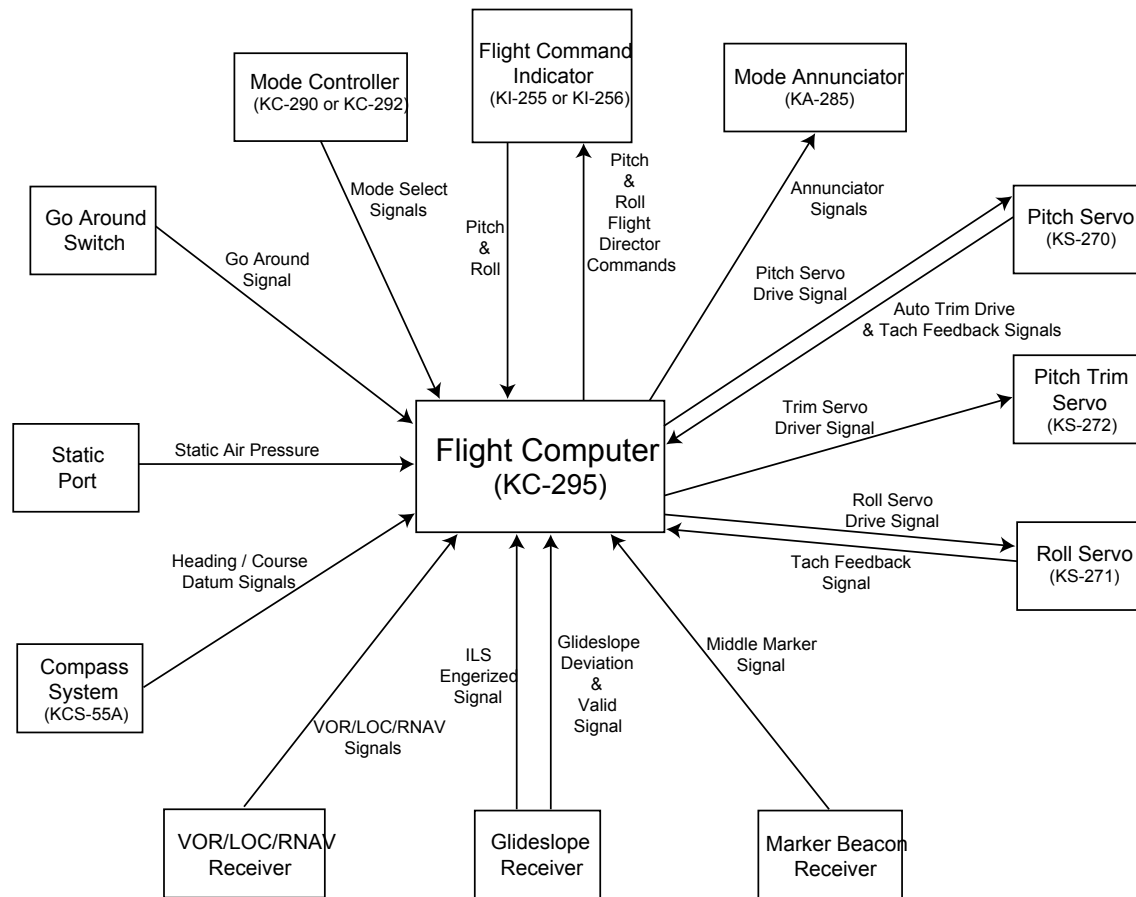


Fig. 3.9: The Structure Of The Bendix / King KFC-200 AFCS.

The AFCS which has to be designed, will be build in the same way as the KFC-200, but the in- and output signals will not be exactly the same. The servo's in the simulator are driven by external servo amplifiers and feedback signals from the servo's are therefore not send back to the AFCS-routine. In the KFC-200, the static air pressure is send to the flight computer. In the flight computer, an air-data computer is present which calculates the altitude. In our AFCS, the altitude is an input signal for the routine, so no static air pressure is necessary.

It was found that the system uses internally the roll- and pitch-rate, but no roll- and pitch-rate input signals from a roll- or pitch-rate gyro were detected. It turned out that the system itself derives the rate's via a differentiator circuit (see Appendix B).

The other systems which were studied (the Bendix / King KFC-150 & KFC-300), were found to be very similar in structure to the Bendix / King KFC-200. In Appendix B, the derived rate schematics of the Bendix / King KFC-150 are given, but they will not be described any further.

3.5.2 Answers To The Formulated Questions

In the previous section, a few questions were formulated. Answers to these questions are now given.

- ➔ How are 'real' AFCS's designed / constructed?

This question is answered in Fig. 3.9.

- ➔ How are 'real' AFCS's tuned prior and after installation?

The tuning-gains are constructed as potentiometers on the circuit-boards of the AFCS. They are constructed in such a way that they are easily reachable so the values can be adjusted and fine-tuned after installation takes place.

- ➔ How are the tuning values determined?

It was nowhere to be found how the initial tuning values are determined. It is believed that the initial tuning values are set very low to begin with to prevent immediate overshoot or instability. When the initial tuning values are implemented into the AFCS, a test flight is made and during this flight, the tuning values are fine tuned to provide a decent aircraft response.

- ➔ Is a particular AFCS tuned for a range of aircrafts or is each AFCS tuned specifically for one aircraft type?

It was found that each AFCS is fitted with an 'adapter-board'. This Printed Circuit Board (PCB) is tuned for a specific type of aircraft and it is this PCB which sets all the tuning values and aircraft-type specific constraints. Therefore, there is a separate installation manual for each aircraft-type. A complete AFCS with fitted aircraft-type specific 'adapter-board' has to be approved by the Federal Aviation Administration (FAA). When the approval is conducted, the system will get a Supplemental Type Certificate (STC) Approval to be installed only in that specific aircraft. A STC Approval offers the highest assurance that all of the airworthiness regulations have been met. The STC's are issued by the FAA Aircraft Certification Office (FAA ACO).

- ➔ Which constraints do 'real' AFCS's use?

In Appendix C, the constraints, limitations and accuracies of the Bendix / King KFC-200 & KFC-400 are given.

4 DESIGN OF THE AUTOMATIC FLIGHT CONTROL SYSTEM

4.1 Classification Of Aircraft-Types

To make the analysis of aircraft characteristics easier, Roskam [25], [26] has classed aircrafts into 10 major classes. If one wants to analysis a specific aircraft, one should first determine in which class the aircraft belongs (based on size / geometry, mass, speed, engine-type etc).

The 10 major classes are:

- Aircraft 'A' – Small, single piston-engine general aviation aircraft
Example: Cessna 182
- Aircraft 'B' – Small, twin piston-engine general aviation aircraft
Example: Cessna 310
- Aircraft 'C' - Small, single jet-engine military training aircraft
Example: SIAI-Marchetti S-211^{*}
- Aircraft 'D' – Small, twin jet-engine military training aircraft
Example: Cessna T-37A^{*}
- Aircraft 'E' – Small, regional, twin-turboprop commuter aircraft
Example: Beech 99
- Aircraft 'F' – Corporate four piston-engine aircraft
Example: Cessna 620
- Aircraft 'G' – Corporate twin jet-engine aircraft
Example: Learjet 24
- Aircraft 'H' – Single jet-engine interceptor fighter aircraft
Example: Lockheed F-104^{*}
- Aircraft 'I' –Twin jet-engine fighter / attack aircraft
Example: McDonnell F-4^{*}
- Aircraft 'J' – Large four jet-engine commercial transport aircraft
Example: Boeing 747-200

^{*} In this thesis, no military aircrafts will be discussed & analysed, therefore Aircraft-types 'C', 'D', 'H' and 'I' are disregarded.

For each aircraft-type, a few flight conditions can be distinguished where the corresponding aircraft behaves differently and therefore are the characteristics for these flight conditions different. The three flight conditions are:

1. Climb
2. Cruise
3. Approach

For aircrafts 'E' and up, the 'Cruise' flight condition can be subdivided into the following conditions:

- 2.1 Cruise Flight At Low Altitude ($\pm 20,000$ feet)
- 2.2 Cruise Flight At High Altitude ($\pm 40,000$ feet)

Each flight condition can be additionally subdivided into a condition where the aircraft has a low weight configuration and where the aircraft has a maximum weight configuration.

In appendix A, various transfer functions are given for aircrafts 'A', 'E', 'G' and 'J'. These data is taken from Roskam [25].

In the next chapters, we will concentrate solely on an aircraft 'A' in cruise flight condition. In section 4.3.7, we will examine what the effect of the different aircraft-types is on the transfer functions and aircraft responses of the control systems and in section 4.3.8, we will examine what the effect of a different flight condition is on the transfer functions and aircraft responses.

4.2 Basic Structure Of An AFCS

Through research of literature it was found that the basic structure of all the AFCS's is more or less identical if we look at the AFCS's in a 'classical control system theory' kind of way. There are basically two loops in the design: one inner-loop which actually controls the aircraft with pitch, roll and sometimes yaw commands (this is the control loop) and one outer-loop which controls the top-level guidance parameters such as heading or altitude (this is called the guidance loop). These loops are illustrated in Fig. 4.1:

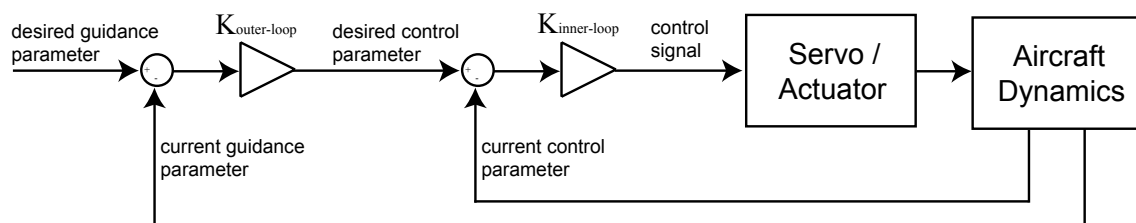


Fig. 4.1: The Inner- And Outer-Loop Design Of An AFCS.

For the design, we will start first with analysis of the inner-loop. When this analysis is completed, the outer-loop together with the inner-loop will be analysed. The inner-loop analysis will be divided into two sections: the first section the longitudinal movements of the aircraft will be evaluated and the second section the lateral movements of the aircraft will be evaluated. It is assumed for now that the longitudinal movements don't influence the lateral movements and vice versa. By this assumption, we can analyse the two movements apart from each other which simplifies the process significantly.

4.3 Longitudinal Control Of An Aircraft

4.3.1 The Longitudinal Open Loop Responses Of An Aircraft (The Phugoid & Short-Period Movements)

We will begin with an analysis of how an aircraft will behave in the pitch when it is controlled solely by the elevator.

In appendix A, the Pitch To Elevator Transfer Function for Aircraft 'A' at a cruise flight of 5,000 feet and with an airspeed of 130.39 kts (= 220.1 ft/sec) is described by:

$$\frac{\theta(s)}{\delta_e(s)} = \frac{-7713.2340 \cdot (s + 1.9982) \cdot (s + 0.0589)}{222.0551 \cdot (s^2 + 8.8994 \cdot s + 27.7798) \cdot (s^2 + 0.0441 \cdot s + 0.0293)} \quad (2)$$

From this equation, we see that this transfer function has two real zeros and two imaginary pole-pairs. It is also seen that there is a minus-sign in the numerator. As a result, the gain in front of the transfer function will be negative to compensate for this convention. In Fig. 4.2, the pole-zero map of this transfer function is plotted.

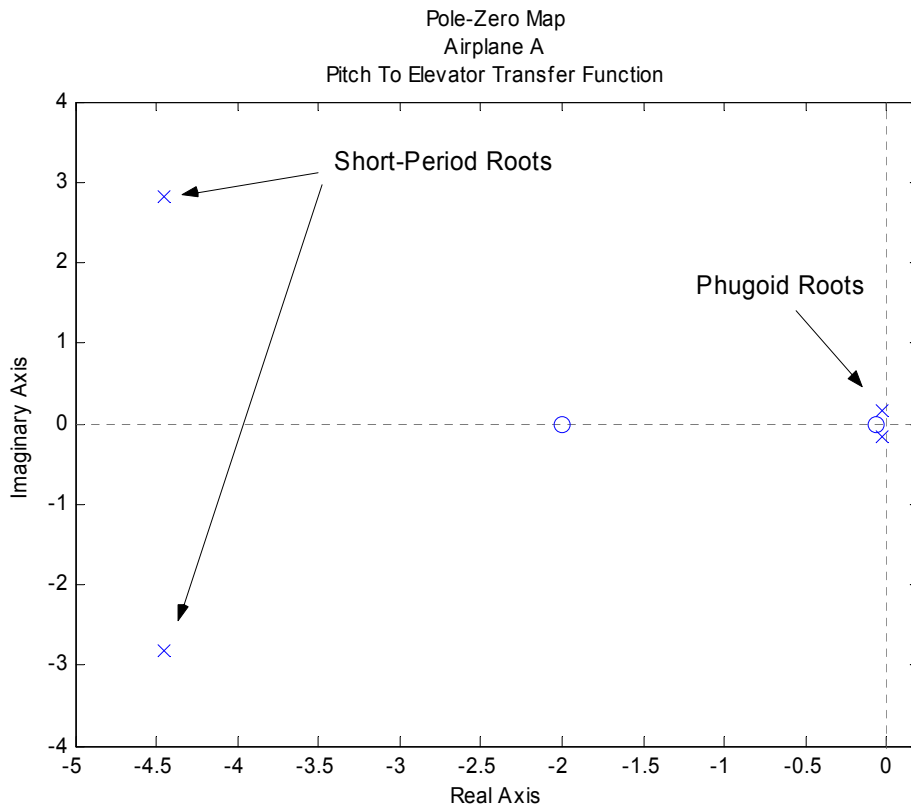


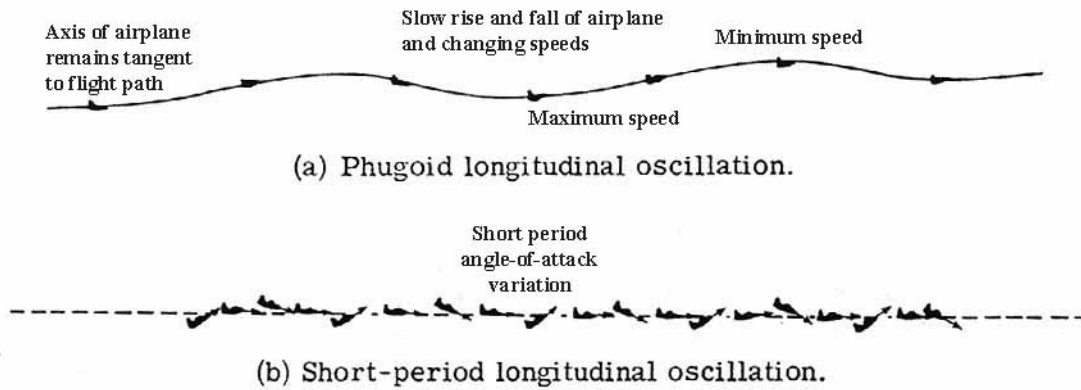
Fig. 4.2: Pitch To Elevator Transfer Function Pole-Zero Map For Aircraft 'A'.

We see in this figure that one pole-pair is very close to the origin and the other one is further away from the origin. The pair closest to the origin are called the Phugoid roots and the pair further away from the origin are called the Short-Period Roots. The damping- and frequency values are given in Table 4.1:

Eigenvalue	Damping	Frequency [rad/sec]	Roots
-0.2221 +/- 0.170i	0.129	0.171	Phugoid Roots
-4.45 +/- 2.82i	0.844	5.27	Short-Period Roots

Table 4.1: Eigenvalue Characteristics For The Pitch To Elevator Transfer Function Of Aircraft 'A'.

It is seen from the values that the Phugoid roots are very lightly damped and they have a very low frequency of $(0.171 / 2 \cdot \pi =) 0.027$ Hz (which corresponds to a period of 36.744 seconds). Because of this long period, the low damping value is not a problem and there are pilots which don't even know this mode exists. A pilot can counteract this effect without any problems. There are aircrafts where the Phugoid roots lie in the right half-plane and therefore have a negative damping. This means that the mode for such an aircraft is unstable and that the oscillation grows with time. But because of the long period, even unstable Phugoid roots can be damped by the pilot. The Phugoid movement involves a trade between kinetic and potential energy: the aircraft climbs, gains altitude and therefore slows down and at a certain moment, it dives, loses altitude and therefore accelerates. During this movement, the aircraft's angle-of-attack (AoA) is assumed to be constant and the pitch and speed vary. The oscillation is basically an oscillation of the aircraft's flight path (see Fig. 4.3-a, credits NASA).

**Fig. 4.3: Phugoid- And Short-Period Longitudinal Oscillations.**

The name 'Phugoid' was improperly given to it by Lanchester in 1908. Since the Greek word 'phug' ($\phi\upsilon\gamma\eta$, pronounced 'fyoo') as well as the Latin word 'fug' both correspond to the English word 'flight', he decided on the name 'Phugoid'. Unfortunately, in both Greek and Latin, this means 'flight' as in 'run away' instead of what birds and aircrafts do.

Lanchester showed that the Phugoid's period is independent of the aircraft's characteristics and aircraft's altitude and that it depends only on the steady state speed U_1 (see Roskam [25], [26]):

$$\omega_{n_{ph}} \approx \frac{g}{U_1} \cdot \sqrt{2} \Rightarrow T_{ph} \approx \frac{2\pi}{g \cdot \sqrt{2}} \cdot U_1 = \frac{\pi \cdot \sqrt{2}}{g} \cdot U_1 \quad (3)$$

He also showed that the Phugoid damping is inversely proportional to the aircraft's lift-to-drag ratio:

$$\zeta_{ph} \approx \frac{1}{\sqrt{2} \cdot \frac{C_{L_1}}{C_{D_1}}} \quad (4)$$

This means that aircrafts with a high lift-to-drag ratio can be expected to have a poor Phugoid damping.

As can be seen from Fig. 4.2 and Table 4.1, the Short-Period roots are strongly damped and have a frequency of 0.839 Hz (which corresponds to a period of 1.192 seconds). The name ‘Short-Period roots’ is evidently chosen because of the short period of the motion. The strong damping causes this response to end before it can even oscillate. During the Short-Period movement, the angle-of-attack as well as the pitch vary, but the speed remains constant (see Fig. 4.3-b).

In Roskam [25], [26], the undamped natural frequency of the Short-Period can be approximated by:

$$\omega_{n_{sp}} \approx \sqrt{\frac{-C_{m_\alpha} \cdot \bar{q}_1 \cdot S \cdot \bar{c}}{I_{yy}}} \quad (5)$$

Where C_{m_α} represents the change in the aircraft aerodynamic pitching moment coefficient due to a change in angle-of-attack, $\bar{q}_1$ is the dynamic pressure in the steady state, S is the area of the wings, $\bar{c}$ is the mean geometric chord of the wings and I_{yy} is the pitching moment of inertia.

From equation (5), it follows that there are three factors that ‘drive’ the magnitude of $\omega_{n_{sp}}$:

1. The centre of gravity (c.g.) location relative to the aircraft’s aerodynamic centre through C_{m_α} .
2. The altitude through $\bar{q}_1$.
3. The pitching moment of inertia. This is influenced by e.g. the mass configuration or mount-position of the engines.

The characteristics of the Pitch To Elevator Transfer Function can be seen in Fig. 4.4, where the open-loop step response of the transfer function is plotted:

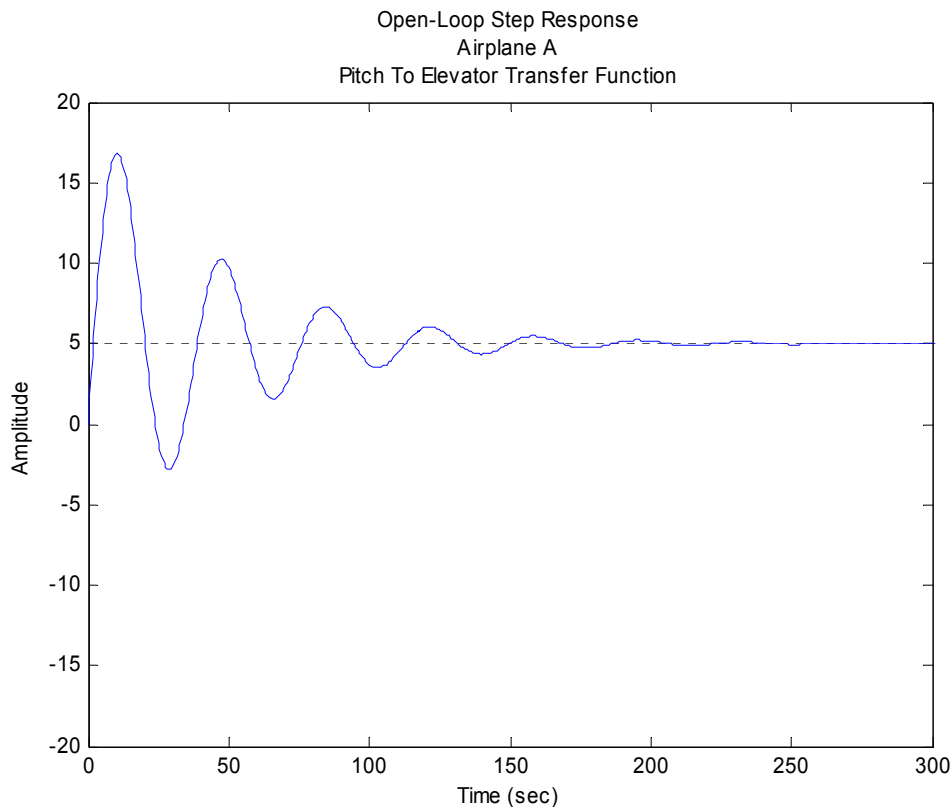


Fig. 4.4: Pitch To Elevator Transfer Function Open-Loop Step Response For Aircraft 'A'.

Because of the strong damping of the Short-Period roots, only the effect of the Phugoid roots can be seen. Even if the time scale is changed, the Short-Period roots effect cannot be seen. The reader may ask him- or herself at this point why these Short-Period roots are described here: it will be shown in the next sections that the effect of these roots can be of influence in the response of a feedback control system.

It was found in many books that often an approximation of the complete transfer function is used. This approximation is called the Short-Period approximation for the Pitch To Elevator transfer function and it what this approximation does, is neglect the Phugoid roots to make the transfer function easier for calculations. In the past where the computers where not that powerful, this made sense because the time to analyze the characteristics could be significantly reduced by using this approximation transfer function instead of the complete transfer function. Nowadays, computers are very powerful and it is no problem to do calculations with the complete transfer functions. Therefore the author believes that at the present time the approximation transfer functions should no longer be used. It is very important to gain insight in all the characteristics and responses of an aircraft to understand its behaviour.

4.3.2 How A Pilot Controls The Pitch Of An Aircraft

It will be instructive to analyze how a pilot controls the pitch of an aircraft. That is important because the inner-loop will control the aircraft in the same way a pilot does. Therefore, if we know how, and with the use of which parameters, a pilot controls the pitch of an aircraft, we basically know how an inner-loop of an AFCS is designed.

First of all, the pitch is controlled by the elevators and that is of course the reason why we analysed the Pitch To Elevator transfer function in this chapter. The elevators are controlled by the pilot by pulling (pitch increases) or pushing (pitch decreases) the flight yoke.

Let us take an example where the pilot wants to change the pitch from 0 to 15 degrees (e.g. when the aircraft needs to climb). The pilot will pull the yoke to increase the pitch of the aircraft. He or she will look at the Attitude Indicator (AI) to know the current pitch. As the yoke is pulled, the elevators will deflect upwards and as a result, the lift of the aircraft-tail will decrease. Because the tail is, of course, at the rear of the aircraft and therefore behind the aircraft's centre of gravity (c.g.), the pitch of the aircraft will increase. The pilot will notice the pitch changing on the AI and he or she will increase or decrease the deflection of the elevators if the pitch change is too slow or too fast. The change of pitch is called the pitch-rate. Upon reaching the desired pitch angle, the pilot will move the yoke to the default position before the desired pitch is reached and consequently prevent overshooting the desired pitch angle. It is important to notice that during the pitch change, when a desired pitch-rate is reached, the pilot will hold the yoke in that position to have a constant pitch-rate (which is the same as a zero pitch acceleration). It is therefore verified that the elevators are position devices (as stated in chapter 2).

From this analysis we know that the pilot looks at two parameters when he want to control the pitch:

1. The pitch.
2. The pitch-rate.

It is therefore obvious that a control system which will perform the same task, will have the same input parameters.

In the next section, a control system which controls the pitch is developed. This control system is the longitudinal inner-loop of the AFCS.

4.3.3 A Pitch Feedback Only Longitudinal Inner-Loop

In the previous section, it was shown that the longitudinal inner-loop will probably have pitch as well as pitch-rate as input parameters. This section we will start with only pitch as input parameter. The result of this analysis should be that no reasonable response is possible with only the pitch as input parameter. In Fig. 4.5, the control system with only the pitch in the feedback path is drawn:

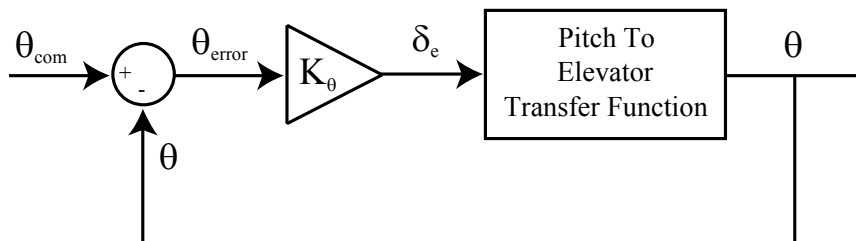


Fig. 4.5: A Pitch Feedback Only Longitudinal Inner-Loop.

We will start the analysis with plotting the root locus of the open loop system; see Fig. 4.6.

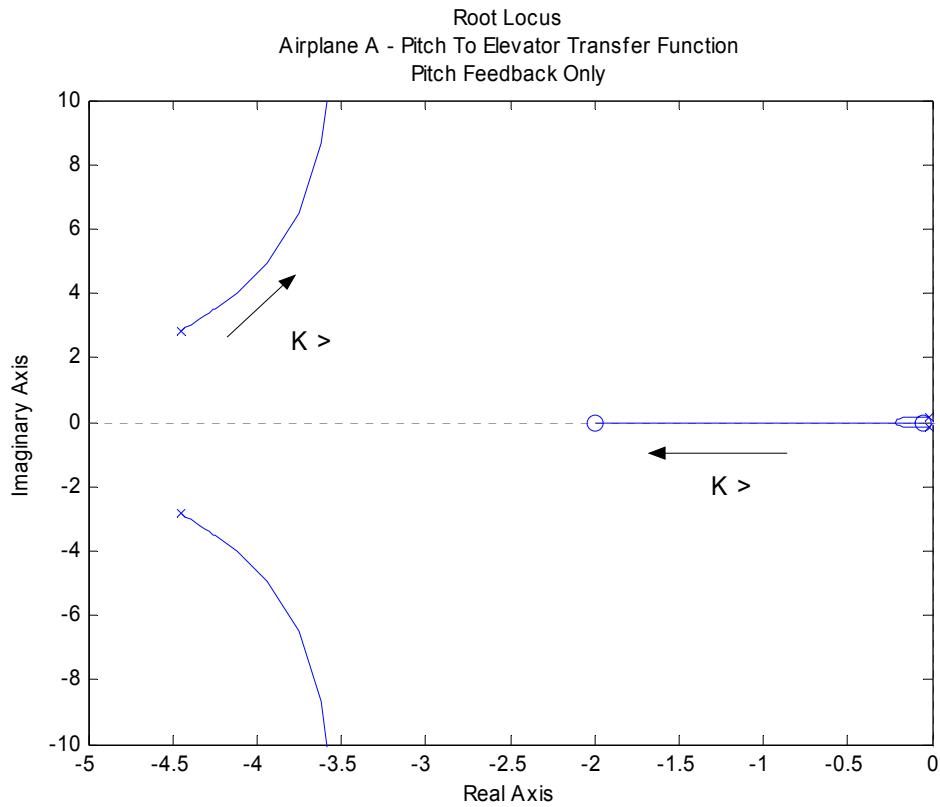


Fig. 4.6: Root Locus Of A Pitch Feedback Only Longitudinal Inner-Loop For Aircraft 'A'.

As one can expect, when the gain K is very small, the roots of the closed loop system will be very close to the roots of the open loop system and hence the response will be almost identical to the response of the open loop system (if we disregard the magnitude). So, the response will be dominated again by the effect of the Phugoid roots. If the gain K is increased, the damping of the Short-Period roots will decrease because the angle between the real axis and the line from the origin to the roots will increase (the damping $\zeta = \cos(\angle)$). To see how the Phugoid roots of the closed loop system behave, we have to zoom in into the origin. This is shown in Fig. 4.7.

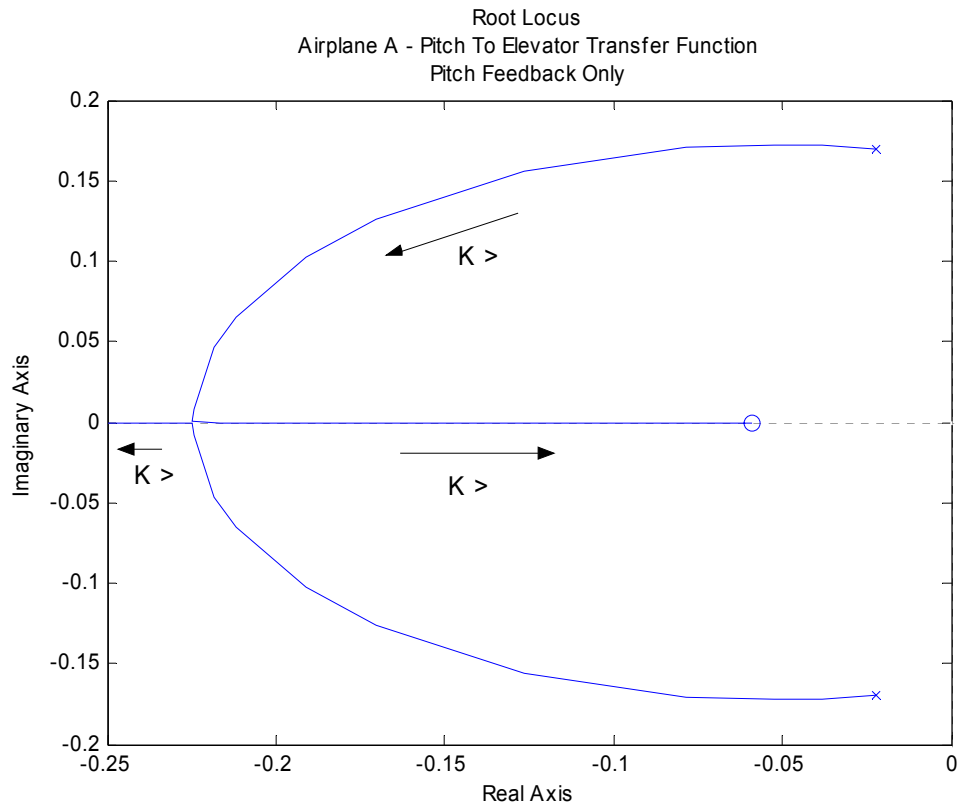


Fig. 4.7: Magnification Near The Origin Of Fig. 4.6.

It is seen that if we increase the gain K , the damping of the Phugoid roots will increase. Therefore, the response of the closed loop system will migrate from a response where the slow Phugoid roots have a dominating effect to a response where the fast Short-Period roots have a dominating effect. We can see this effect in Fig. 4.8, where the step responses are plotted for feedback gain of -0.01, -0.1, -1.0 and -5.0.

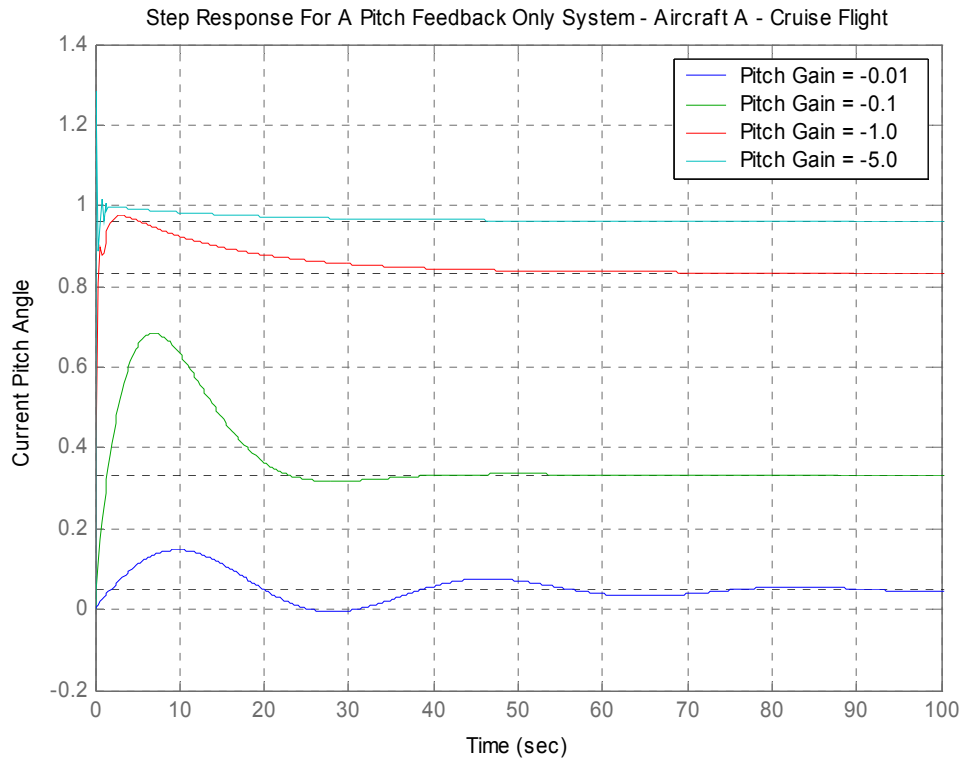


Fig. 4.8: Step Responses Of A Pitch Feedback Only Longitudinal Inner-Loop For Aircraft 'A'.

When the gain is very small (-0.01), the response looks indeed like the open-loop response and when the gain is increased, we see the phugoid response disappear and a high frequency 'ripple' starts to appear at the start of the response. In Fig. 4.9, the scale is adjusted to show the effect of the Short-Period roots.

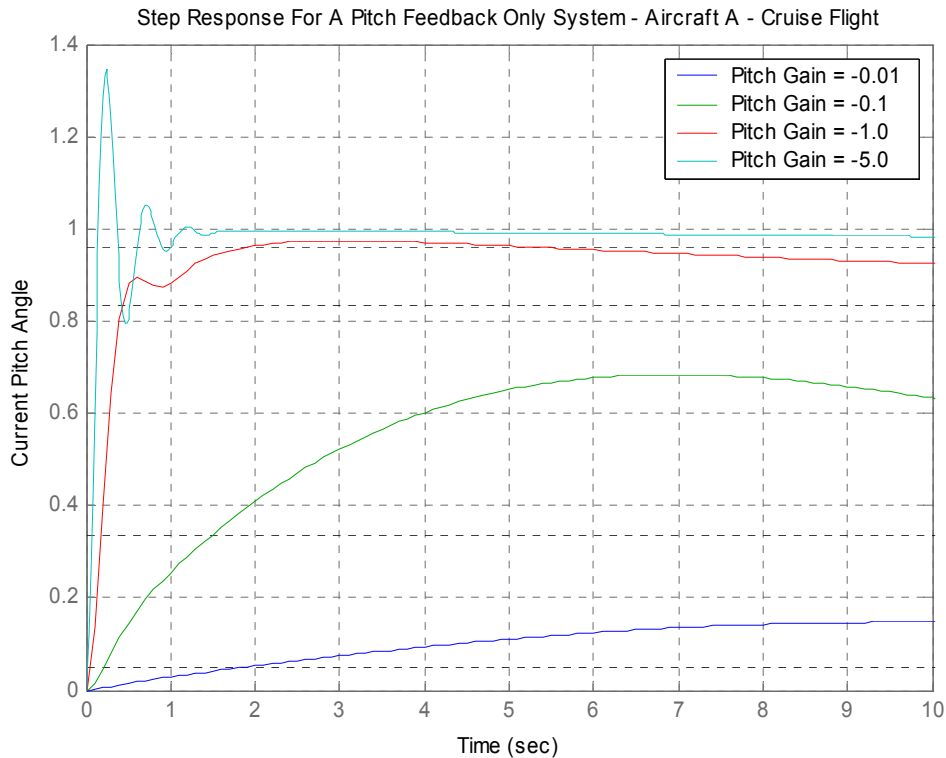


Fig. 4.9: Time-Scale Adjusted Of Fig. 4.8.

If we look at the two previous plots, it is important to notice that there will always be a steady-state error with this system. However, if the gain K is increased, the steady-state error is decreased. But, when the gain is larger than -1 , the Short-Period roots will have a very small damping and the response will start to oscillate (see Fig. 4.9). This oscillation gets worse when the gain is even more increased. It is therefore proved that this inner-loop with only pitch feedback does not work very well. If we want better responses, damping should be added in the form of additional pitch-rate feedback. In the next section this will be done.

4.3.4 Servo- And Cables Models

In the previous section, an ideal, unreal, situation is presumed, because the output of the control system is fed immediate into the aerodynamic model (which for this case is the Pitch To Elevator transfer function). If we really want to simulate systems which are used in real aircrafts, we'll have to simulate the complete aircraft.

In a real aircraft, the signals from the control system, are sent to a servo / actuator. In small aircrafts, mostly electric actuators are used. In large aircraft, electro-hydraulic actuators are used (see McLean [17]). The servo / actuator is connected to the control surfaces (elevators, ailerons & rudder) via cables (in small aircrafts) or via hydraulics (in large aircrafts). A block diagram of the complete system with pitch feedback only, is shown in Fig. 4.10.

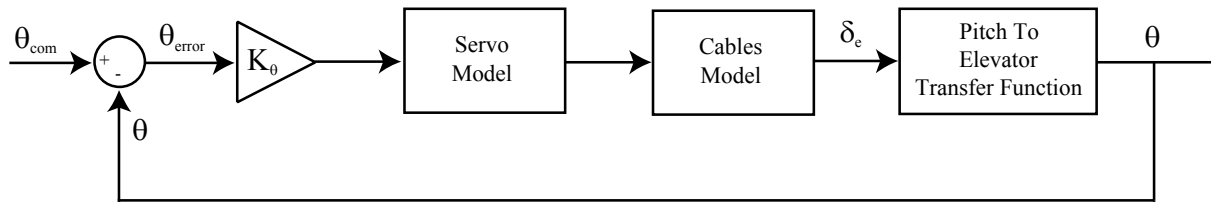


Fig. 4.10: Pitch Feedback Only Inner-Loop With Servo- And Cables-Blocks.

Another characteristic of actuators is that they have a non-linearity behavior. We will incorporate these non-linearities further upwards in this thesis. For now, we will model the servo + cables with one block called 'Servo'. For small aircrafts, this block will be $\frac{10}{s+10}$ and for larger aircrafts, we will use $\frac{20}{s+20}$ (These models were taken from Roskam [25], [26]). The effect of these models will be a (small) delay in the responses. It is a well know fact that delays cause a system to become unstable at some point.

The effect of a $\frac{5}{s+5}$, $\frac{10}{s+10}$ and $\frac{20}{s+20}$ servo on the root locus of a pitch feedback only control system is shown in Fig. 4.11.

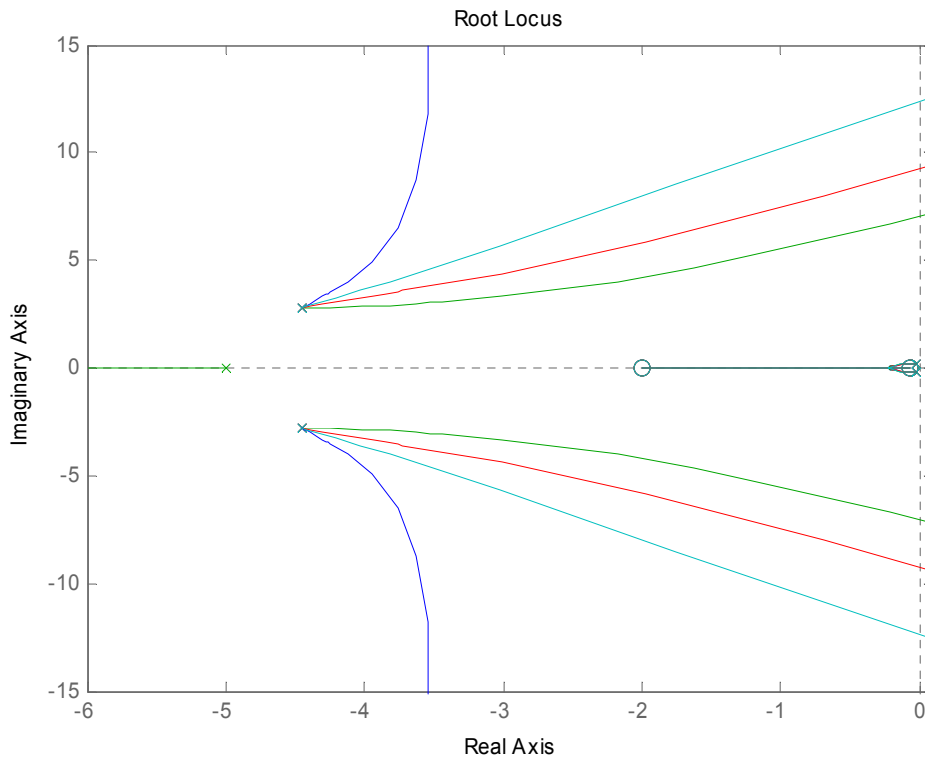


Fig. 4.11: The Effect Of A Servo-Model On The Root-Locus Plot.

This plot shows indeed that the real system will become unstable when the feedback-gain is set too high. The system without the servo-model was always stable and here we see that in reality the system can become unstable. It is therefore useful and important to always incorporate a servo-model in the simulations ! The three servo models shown behave in the same way; there is not very much difference between the slowest (5) and

the fastest (20) servo-model other than delay. It is also seen in this plot that the faster the servo will be, the more the root locus plot will be closer to the root locus plot without a servo-model. We can therefore conclude that the gain margin will be larger when faster servo's are being used.

The effect of adding a servo to the simulation is shown in Fig. 4.12, where the step response without a servo (blue line) and the response with a $\frac{10}{s+10}$ servo (green line) is plotted.

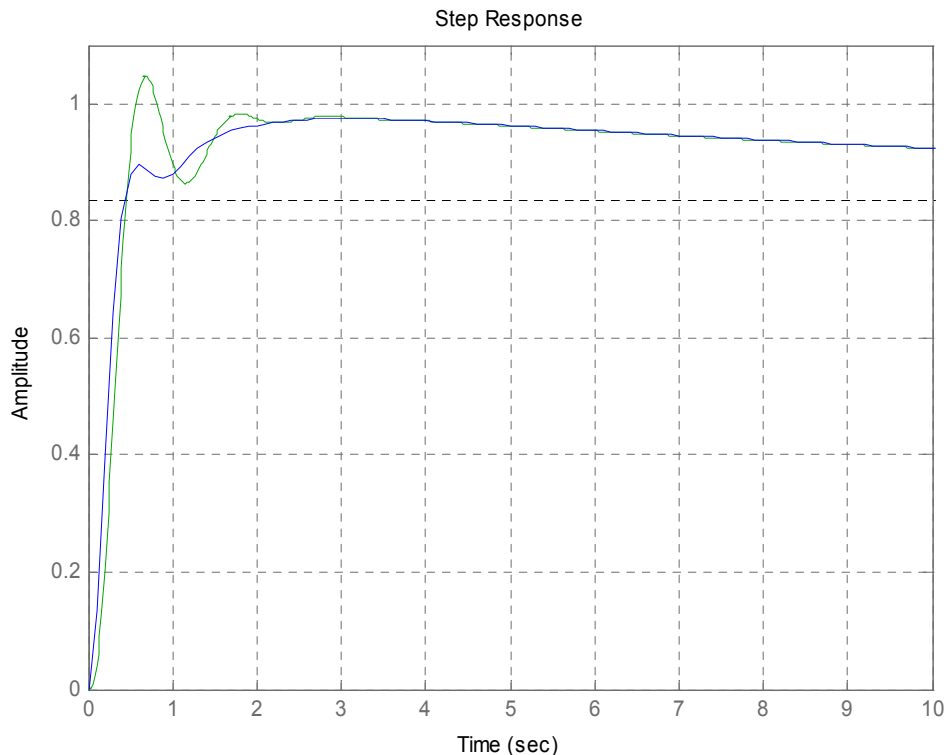


Fig. 4.12: The Effect Of A Servo Model On The Step Response.

As can also be seen from the root locus plot, the damping of the Short-Period roots will be less when a servo is added to the loop and this is also shown in the step response.

4.3.5 Pitch & Pitch-Rate Feedback

It was shown in the previous section that the system which uses pitch feedback only was not sufficient. This was due to the fact that there was a contradiction when the feedback-gain was increased (the damping of Phugoid increases, but damping of Short-Period decreases). Because of this fact, we will investigate how the system behaves when pitch as well as pitch-rate q are being fed back to the control system. The design is shown in Fig. 4.13.

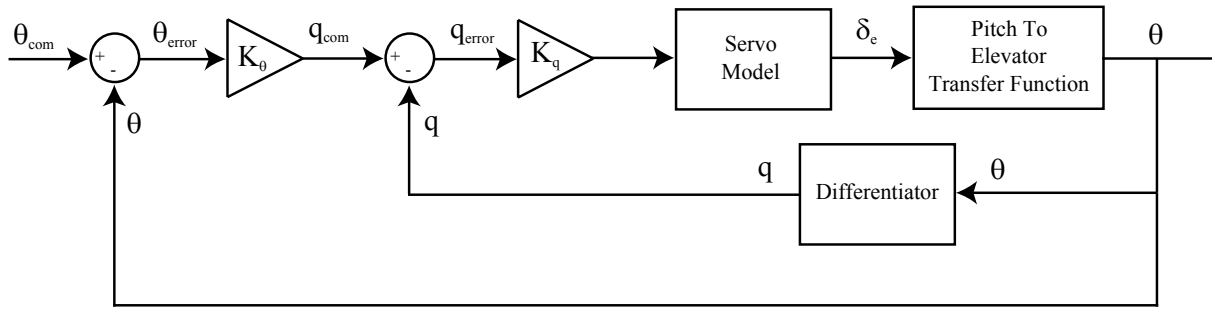


Fig. 4.13: Pitch & Pitch-Rate Feedback Inner-Loop.

It is seen that the inner-loop now consists of an extra loop inside the inner-loop (we could call this loop the innermost-loop of the inner-loop and we could call the other loop, the outermost-loop of the inner-loop, but it is easier to call the innermost-loop the pitch-rate loop and the outermost loop the pitch-loop).

To analyze this system, we will use two steps: in the first step, we will analyze the pitch-rate loop and in the second step, we will analyze the pitch-loop and pitch-rate loop together.

4.3.5.1 The Pitch-Rate Loop

To analyze only the pitch-rate loop, we use the block diagram of Fig. 4.14.

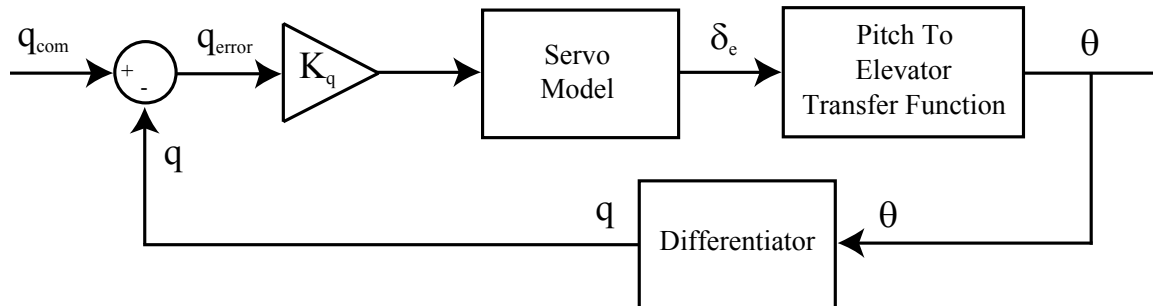


Fig. 4.14: The Pitch-Rate Loop.

The pitch-loop will calculate a commanded pitch-rate and this output is the input of the pitch-rate loop. The differentiator in the feedback-path can of course also be put in the forward-path, but then an extra integrator has to be added to get the pitch for the pitch-loop. The author finds the schematic of figure x easier to interpret. To find an appropriate pitch-rate gain K_q , we will look at the root locus of this system which is plotted in Fig. 4.15.

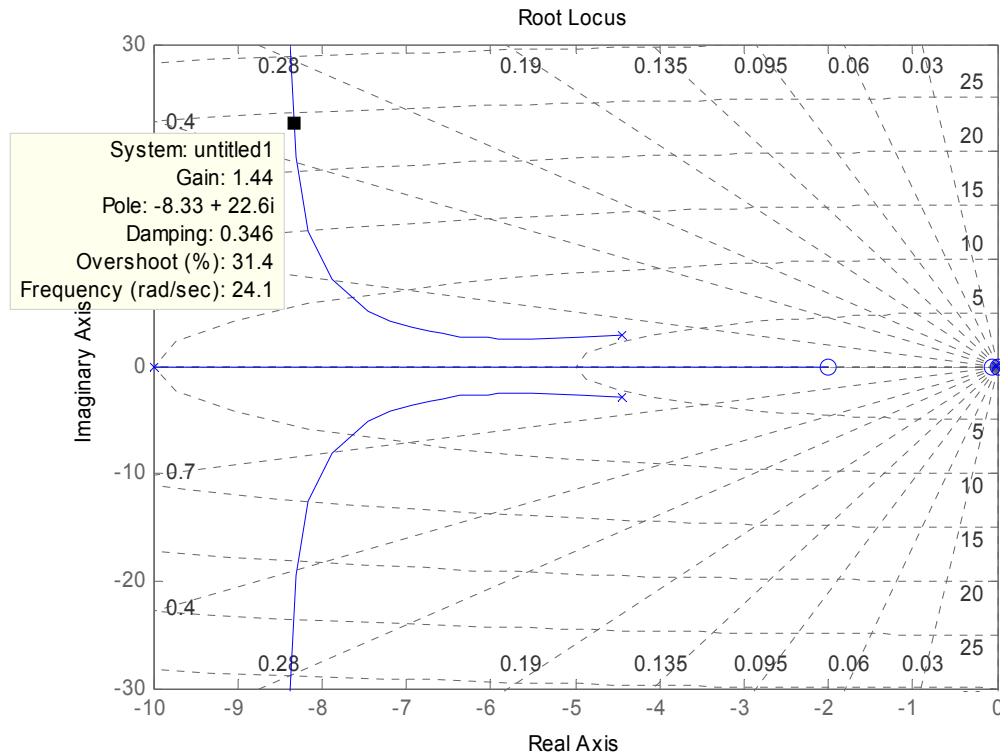


Fig. 4.15: Root Locus Of The Pitch-rate Inner-Loop.

In this plot, we can see that the Short-Period roots are well damped for a small gain, but as the gain increases, the damping decreases rapidly. To see how the root locus near the phugoid roots looks like, we zoom in into the origin (Fig. 4.16).

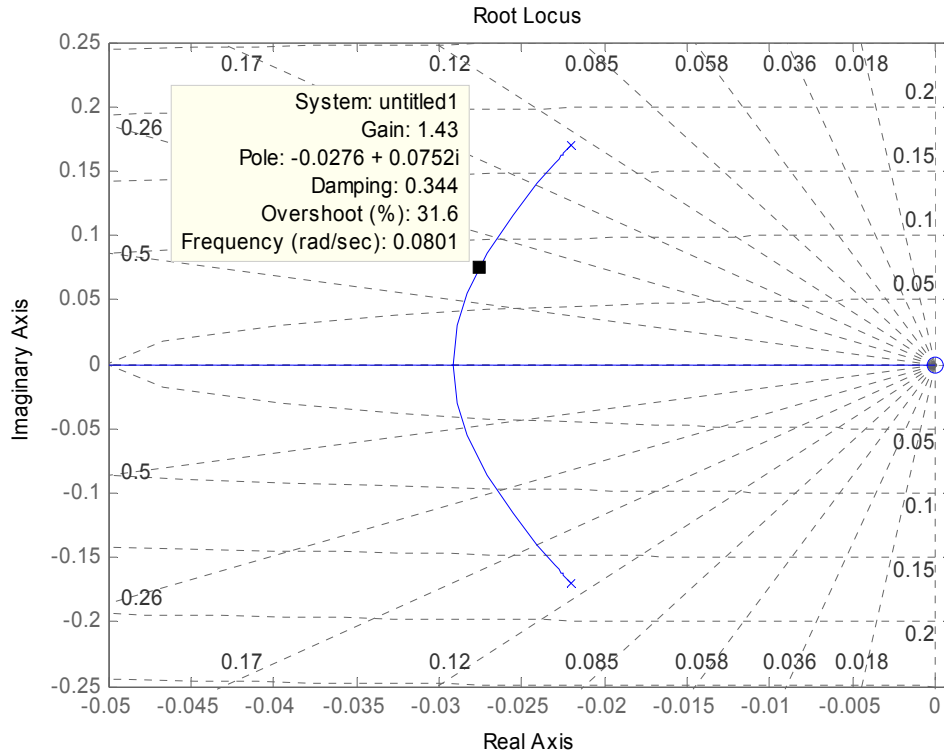


Fig. 4.16: Magnification Near The Origin Of Fig. 4.15.

Just like with the pitch feedback only system, the root locus plot near the Phugoid roots have the opposite direction than the plot near the Short-Period roots. Therefore, if the gain is increased, the damping of the Phugoid roots will also increase and eventually they will be critically damped. To find the best possible pitch-rate gain, a value where the Short-Period as well as the Phugoid roots are damped both satisfactory is wanted. It turns out that if the gain is -1.44, the damping of both roots are the same and equal to ~ 0.35 . If the gain is smaller, the Short-Period roots are better damped and the Phugoid roots are worse damped and if the gain is larger, the Short-Period roots are less damped and the Phugoid roots are more damped. In Table 4.2, the closed-loop values are give for a few different pitch-rate gains K_q (as seen in the root locus plot, the gain margin will always be infinite because the system is always stable):

K_q	ζ_{sp}	$\omega_{n_{sp}}$	ζ_{ph}	$\omega_{n_{ph}}$	Phase Margin [deg]	Frequency [rad/sec]
-0.1	0.779	9.81	0.152	0.153	-100	0.168
-0.5	0.519	15.8	0.227	0.114	-43.6	0.111
-1.0	0.402	20.6	0.296	0.0915	-43.9	0.0857
-1.44	0.346	24.1	0.346	0.0798	44.3	21
-3.0	0.25	33.5	0.484	0.0587	30.3	31.3

Table 4.2: Eigenvalue Characteristics For Different Values Of The Pitch-Rate Gain K_q .

4.3.5.2 The Pitch- And The Pitch-Rate Inner-Loop

With the established pitch-rate gain of -1.44, we can now analyze the complete inner-loop (that is the pitch- and pitch-rate loop together, see Fig. 4.13). The root locus is plotted in Fig. 4.17:

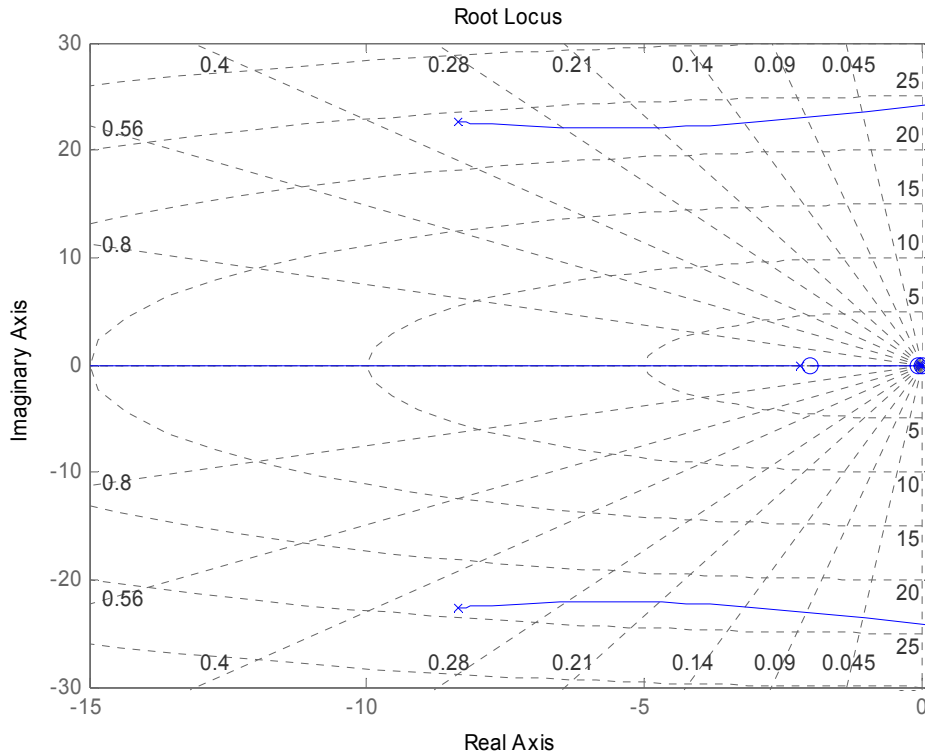


Fig. 4.17: Root Locus Plot Of The Pitch- And Pitch-Rate Inner-Loop.

At a certain pitch gain, the closed-loop poles will enter the right half-plane which causes the closed-loop system to be unstable. The root locus near the Phugoid roots is shown in Fig. 4.18:

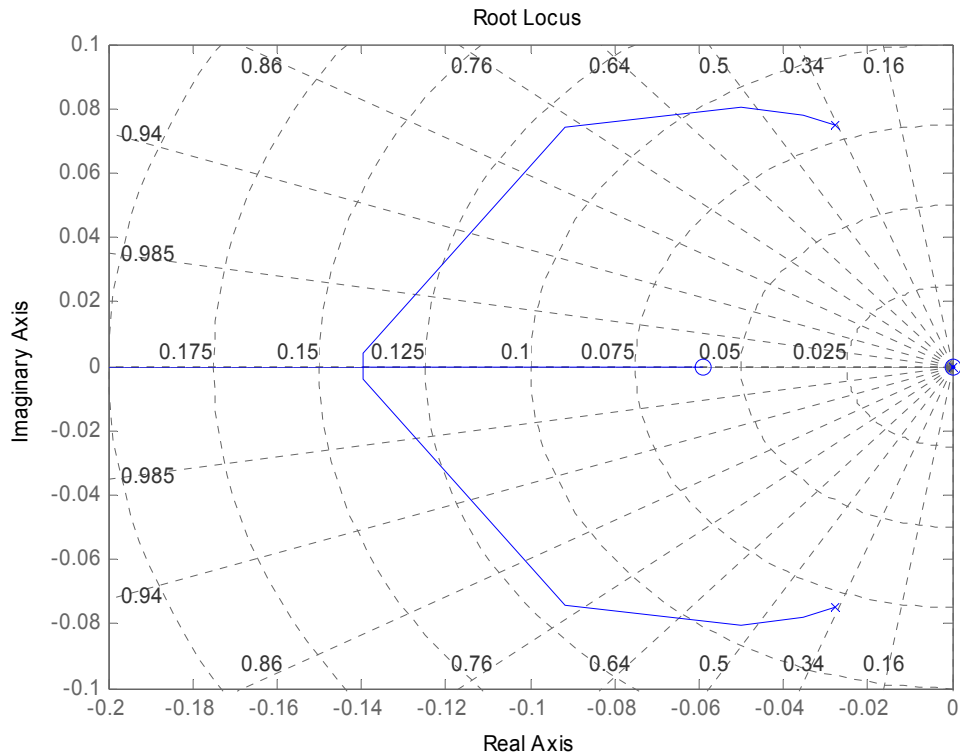


Fig. 4.18: Magnification Near The Origin Of Fig. 4.17.

For a pitch gain of 3, the closed-loop Phugoid roots will lie on the real axis and therefore will be critically damped and the closed-loop Short-Period roots will have a damping of 0.299 at a frequency of 23.3 rad/sec. The step response for this pitch gain is plotted in Fig. 4.19.

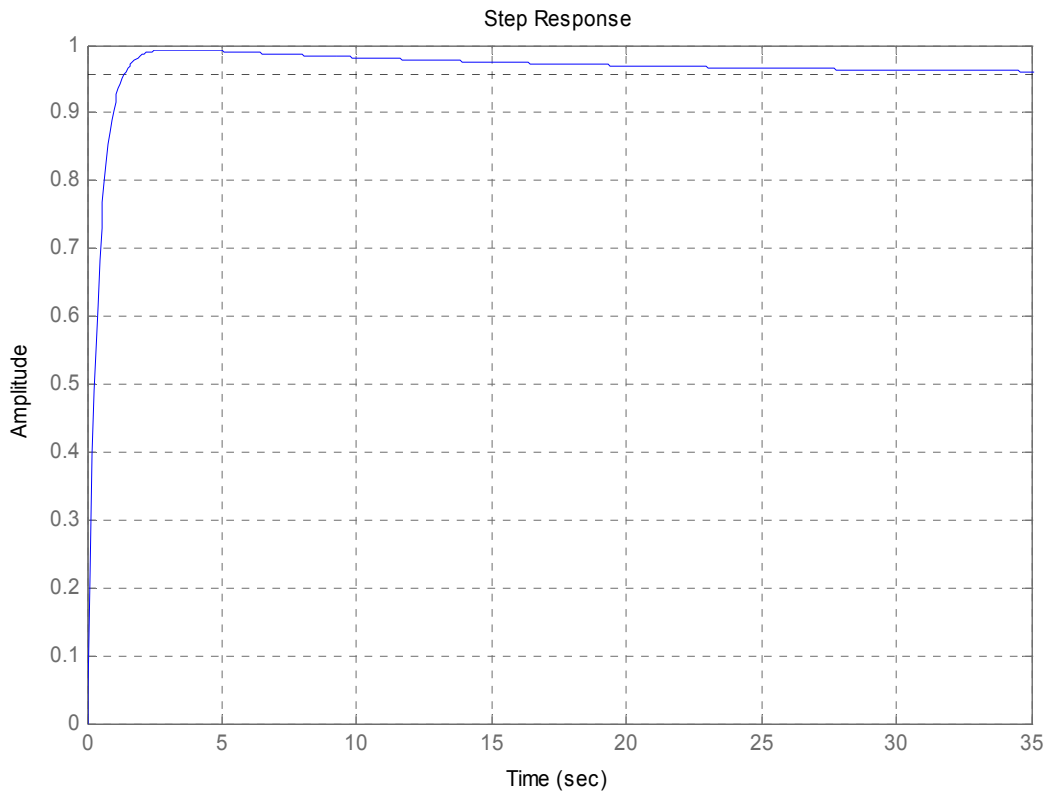


Fig. 4.19: Step Response Of The Pitch & Pitch-Rate Inner-Loop.

This plot shows a pretty nice step response with the only problem being that a steady state error remains. Despite of the low damping of the Short-Period roots, the step response looks acceptable if we disregard the steady state error.

4.3.6 The Elimination Of The Steady-State Error

In the previous sections, all the responses had a steady state error which is obviously not desired. A solution for this behaviour will now be explored.

In the fields of Control Theory, it is well known that a steady state error can be eliminated by adding an integrator to the system. Furthermore, it is believed that the simulated systems in the previous sections outputs not the position, but the speed to the servo. However, the used transfer function requires the elevator position at it's input and therefore an integrator is needed to convert the speed to the position. In the real flight simulator, it depends on what kind of servo is used if this integrator is needed or not.

4.3.6.1 The Pitch-Rate Loop With An Added Integrator

In Fig. 4.20, the block diagram of the pitch-rate loop is drawn with the added integrator.

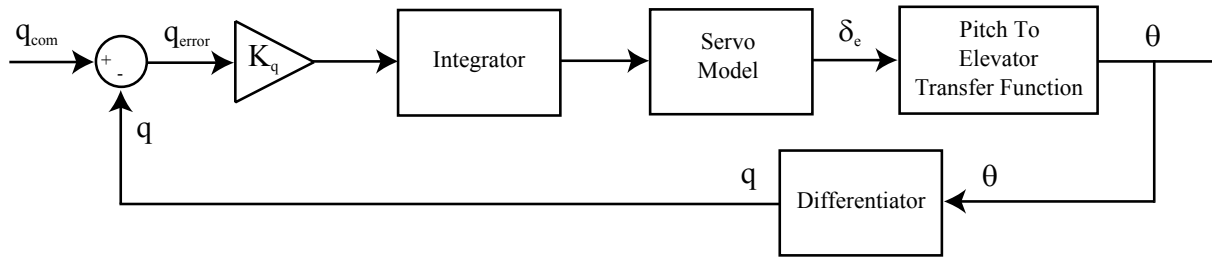


Fig. 4.20: The Pitch-Rate Loop With An Added Integrator.

The root locus of this loop is plotted in Fig. 4.21.

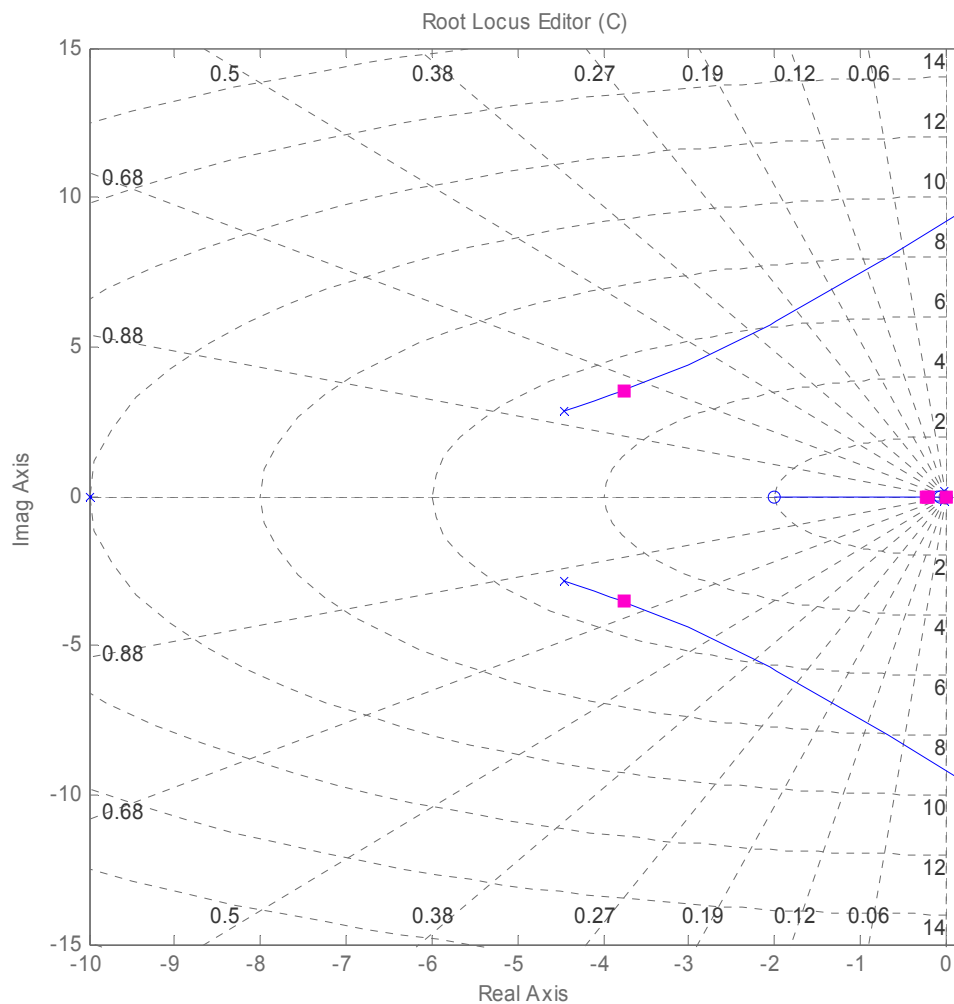


Fig. 4.21: Root Locus Of The Pitch-Rate Loop With An Added Integrator.

If we compare this root locus plot with the root locus plot without the integrator from Fig. 4.15, it is seen in this figure that this plot travels from the Short-Period roots to the direction of the imaginary axis and therefore this innermost loop will become unstable at

some point for a certain gain. We can conclude that the integrator is responsible for pulling the root locus to the right of the imaginary axis. The value of the gain when the loop becomes unstable is found to be -3.84.

The plot near the origin can be seen in Fig. 4.22.

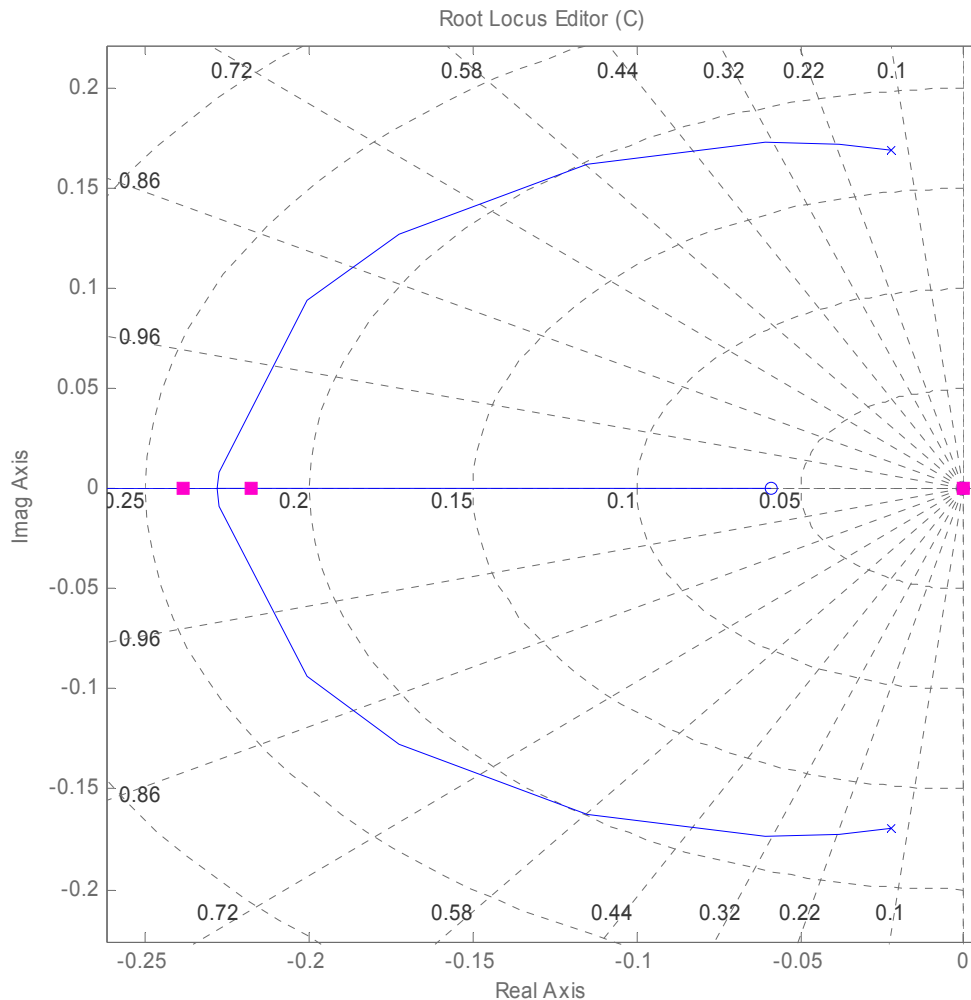


Fig. 4.22: Magnification Near The Origin Of Fig. 4.21.

The root locus near the origin shows that the Phugoid roots will be damped critically for a smaller gain (-0.173 against -13.4 for the system without the integrator).

4.3.6.2 The Pitch – And Pitch-Rate Inner-Loop With An Added Integrator

The system with both the pitch- and pitch-rate loops can be seen in Fig. 4.23.

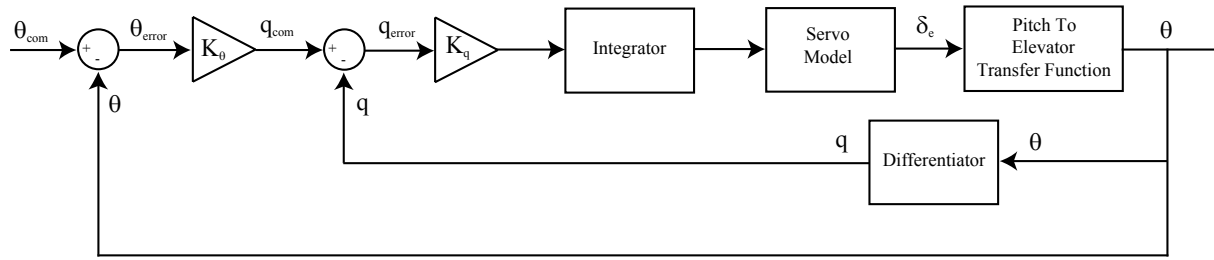


Fig. 4.23: The Pitch- And Pitch-Rate Inner-Loop With An Added Integrator.

If we choose a pitch-rate gain of -0.1, which corresponds to a Short-Period damping of 0.776 and a Phugoid damping of 0.697, the root locus will look like the plot in Fig. 4.24.

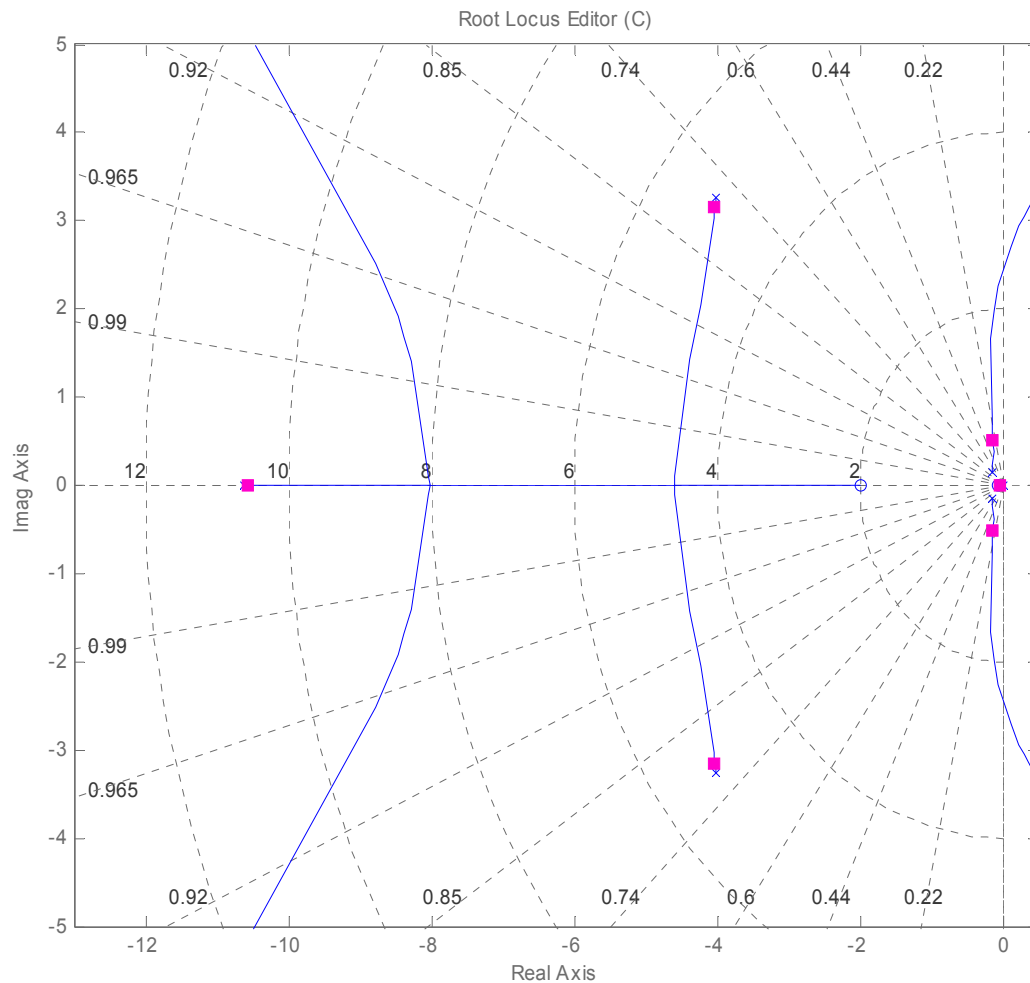


Fig. 4.24: Root Locus Plot Of The Pitch & Pitch-Rate Inner-Loop With An Added Integrator.

This plot should be compared to root locus plot of the system without the integrator in Fig. 4.17. The branch which starts at the Short-Period roots does always have an adequate damping, but if the gain is set too large, the closed-loop poles will be on the leftmost branch and the damping of those roots will rapidly decrease. But before that happens, the closed-loop Phugoid roots will probably be in the right-plane, making the

system unstable. We will therefore look more closely at the root locus near the imaginary axis, which is plotted in Fig. 4.25.

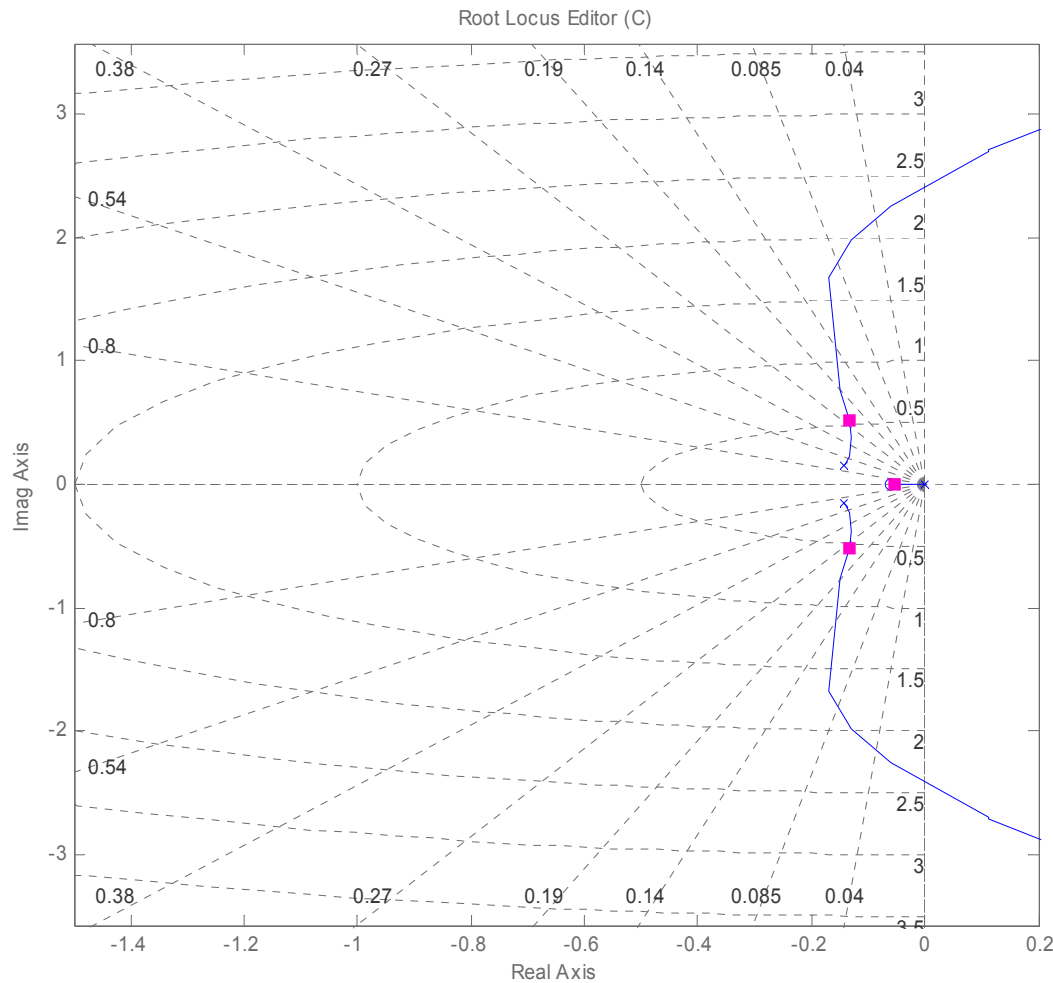


Fig. 4.25: Magnification Near The Origin Of Fig. 4.24.

It is seen here that the closed-loop Phugoid roots indeed enter the right-plane at some point. This pitch-gain value when this happened was determined to be 17. At this gain, the closed-loop Short-Period roots are on the real axis and are indeed not yet arrived at the left-most branch.

We will now look at the step responses of this system for a few pitch- and pitch-rate gains.

For a pitch-rate gain K_q of -0.1, where the closed-loop pitch-rate loop has a damping value ζ_{sp} of 0.78 and a damping value ζ_{ph} of 0.70, four pitch-gains are chosen: $K_\theta = 0.1 / 0.2 / 0.5 / 1$. The step responses are shown in Fig. 4.26.

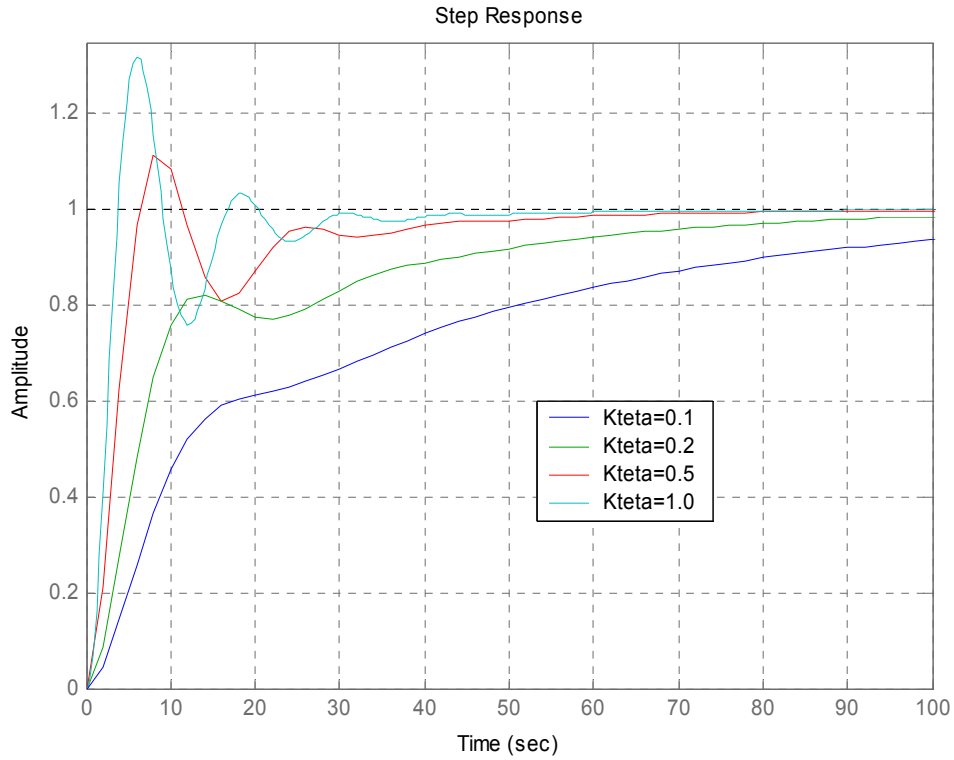


Fig. 4.26: Step Responses With $K_q = -0.1$ And $K_{\theta} = 0.1 / 0.2 / 0.5 / 1$.

To improve these responses, the pitch-rate gain K_q is increased to -0.5 ($\zeta_{sp} = 0.53 / \zeta_{ph} = 1$). The resulting step responses can be seen in Fig. 4.27.

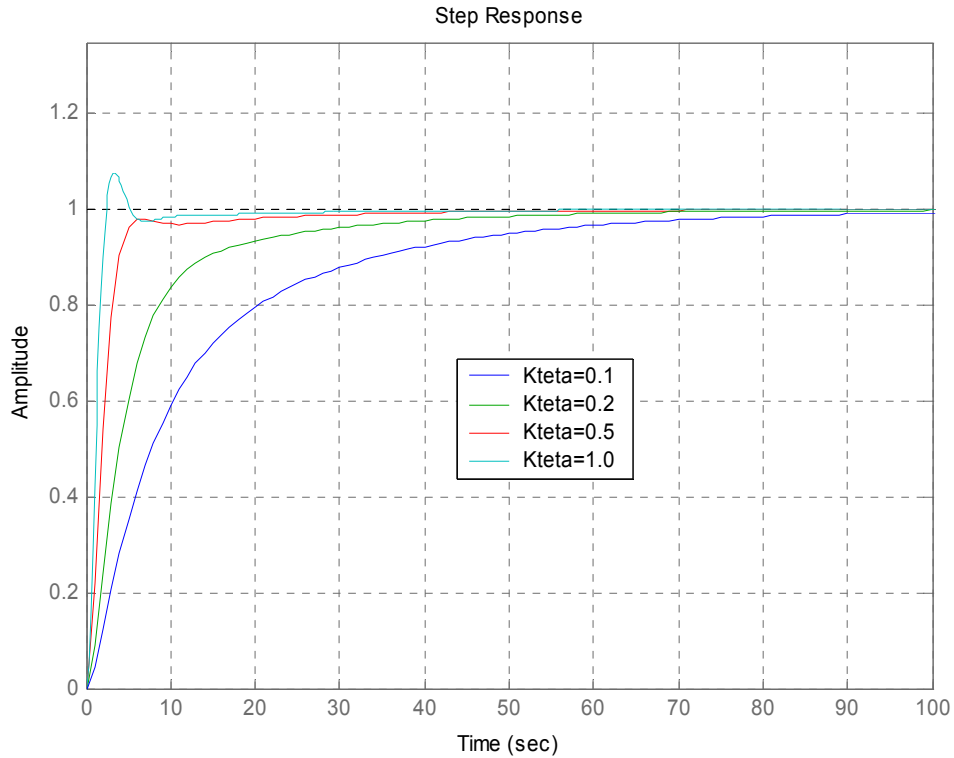


Fig. 4.27: Step Responses With $K_q = -0.5$ And $K_{\theta} = 0.1 / 0.2 / 0.5 / 1$.

This is a noticeable improvement over the previous pitch-rate gain. To try to improve the response even more, the pitch-rate gain K_q is increased to -1.0 ($\zeta_{sp} = 0.34 / \zeta_{ph} = 1$), see Fig. 4.28.

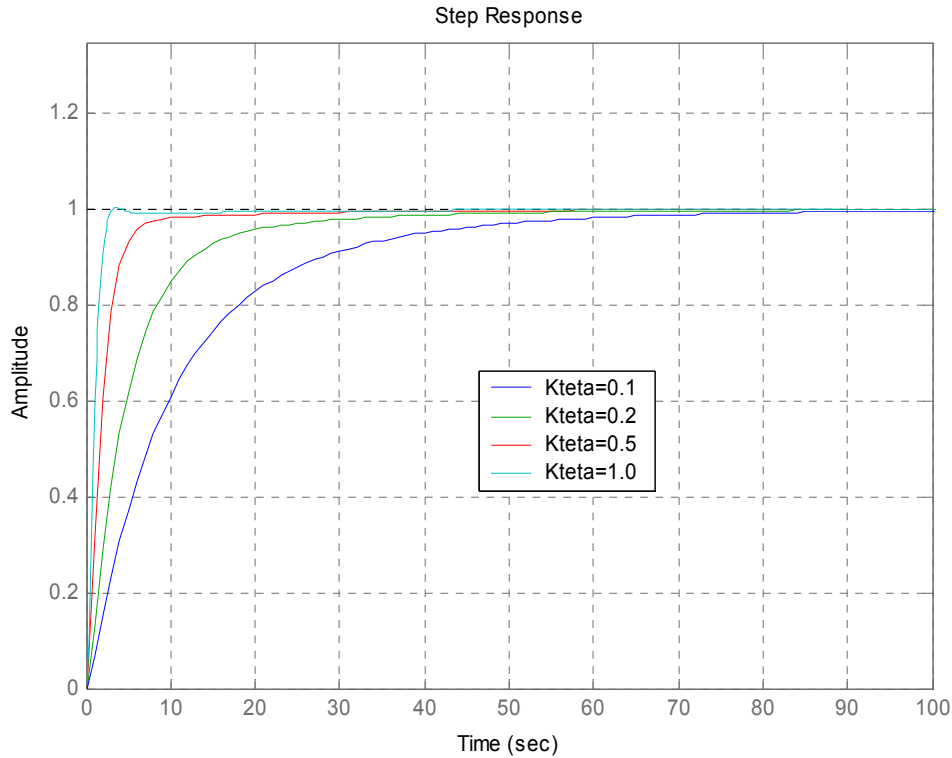


Fig. 4.28: Step Responses With $K_q = -1.0$ And $K_\theta = 0.1 / 0.2 / 0.5 / 1$.

It is seen in the previous figure that a pitch-rate gain K_q of -0.5 & -1.0 give acceptable responses as long as the pitch gain K_θ is not too large. The speed of the response is of course also very important, it is for example not acceptable that it takes 100 seconds to reach a desired pitch angle. With this requirement in mind, we find the following acceptable gain values:

$$K_q = -0.5 / K_\theta = 0.5$$

$$K_q = -1.0 / K_\theta = 0.5$$

$$K_q = -1.0 / K_\theta = 1.0$$

In Table 4.3, relevant data can be found for all the analysed gain values. It is found in this table that the damping of the Short-Period roots for a pitch-rate gain K_θ of -1.0 is very low despite the decent step response in Fig. 4.28. For now, a pitch-rate gain K_q of -0.5 will be preferred.

$K_q = -0.1$	ζ_{SP}	ζ_{ph}	Gain Margin [dB]	Freq. [rad/sec]	Phase Margin [deg]	Freq. [rad/sec]
Inner Loop Only	0.776	0.697	24.6	2.44	31	0.4999
$K_{\theta} = 0.1$	0.777	0.542	44.5	2.44	-180	0
$K_{\theta} = 0.2$	0.778	0.451	38.4	2.44	-180	0
$K_{\theta} = 0.5$	0.781	0.327	30.3	2.44	68.1	0.453
$K_{\theta} = 1.0$	0.787	0.252	24.0	2.44	46.8	0.686
$K_q = -0.5$	ζ_{SP}	ζ_{ph}	Gain Margin [dB]	Freq. [rad/sec]	Phase Margin [deg]	Freq. [rad/sec]
Inner Loop Only	0.534	1.000	16.9	3.94	59.3	0.844
$K_{\theta} = 0.1$	0.535	1.000	36.8	3.94	-180	0
$K_{\theta} = 0.2$	0.536	1.000	30.7	3.94	-180	0
$K_{\theta} = 0.5$	0.540	0.776	22.3	3.94	-180	0
$K_{\theta} = 1.0$	0.546	0.627	15.6	3.93	117	0.876
$K_q = -1.0$	ζ_{SP}	ζ_{ph}	Gain Margin [dB]	Freq. [rad/sec]	Phase Margin [deg]	Freq. [rad/sec]
Inner Loop Only	0.338	1.000	13.7	5.13	71.7	0.94
$K_{\theta} = 0.1$	0.336	1.000	33.5	5.13	-180	0
$K_{\theta} = 0.2$	0.334	1.000	27.3	5.13	-180	0
$K_{\theta} = 0.5$	0.327	0.988	18.7	5.13	-180	0
$K_{\theta} = 1.0$	0.311	0.831	11.7	5.12	-180	0

Table 4.3: Damping, Gain Margin And Phase Margin Values Of The Pitch & Pitch-Rate Inner-Loop With An Added Integrator.

4.3.7 The Influence Of The Aircraft Type On The Longitudinal Simulations

In the simulations in the previous sections, it was assumed that the simulated aircraft was of type 'A' and that the aircraft was in cruise flight. We will now investigate what the influence is of a different aircraft type. In Fig. 4.29, the open-loop root locus' of aircrafts 'A', 'E', 'G' and 'J' are plotted.

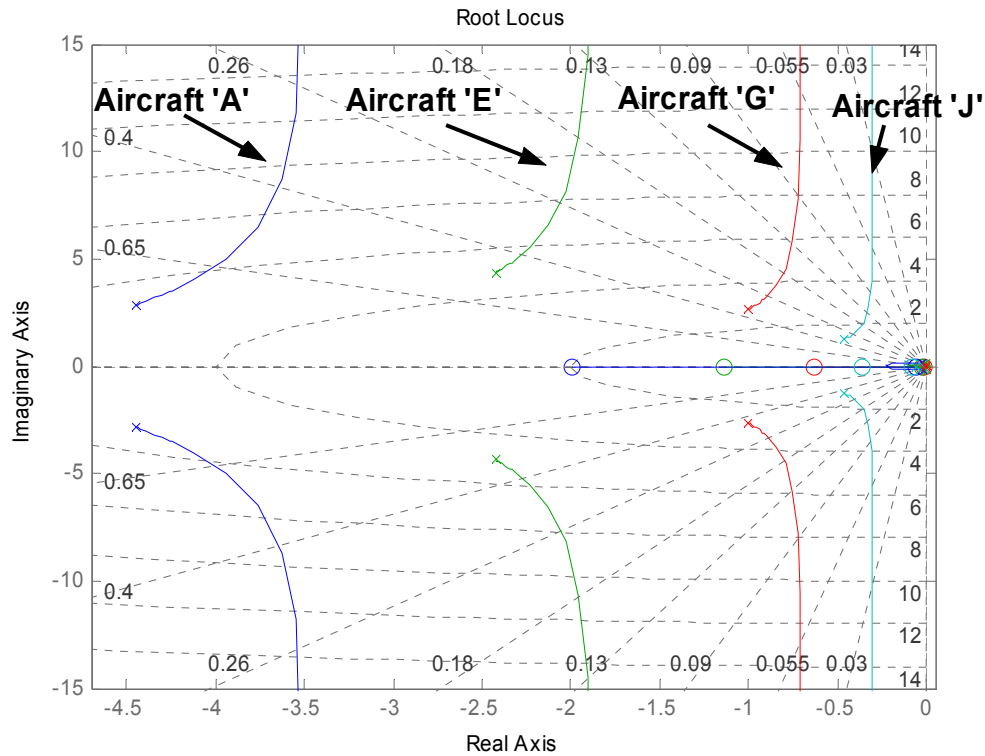


Fig. 4.29: Open Loop Root Locus Of Aircrafts 'A', 'E', 'G' And 'J'.

The root locus near the origin is plotted in Fig. 4.30.

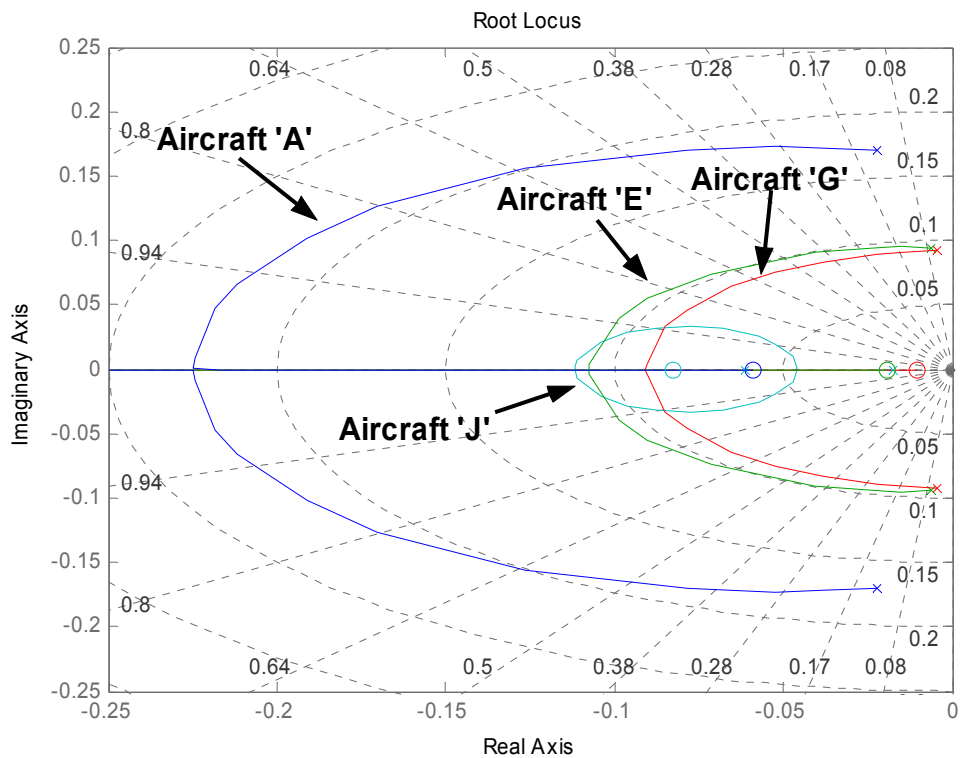


Fig. 4.30: Magnification Near The Origin Of Fig. 4.29.

A few observations are in place when we look at these pictures: The bigger the aircraft, the more the root locus plot will be closer to the imaginary axis and the faster the damping of the Short-Period roots will deteriorate when the gain is increased. Another remarkable fact is that the open-loop Phugoid roots of aircraft 'J' lie on the real axis and are therefore critically damped. As the gain is increased for this aircraft, the damping of these roots is always greater than 0.9.

4.3.8 The Influence Of The Flight Condition On The Longitudinal Simulations

In the previous sections, the transfer functions in 'cruise flight condition' were used. Of course, this condition is valid for most of the flight, but it could be informative to know the influence of the flight condition on the responses. A few different flight conditions were described in section 4.1. In Fig. 4.31, two root locus plots are drawn for 2 different flight conditions: a) Cruise Flight Condition and b) Approach Flight Condition.

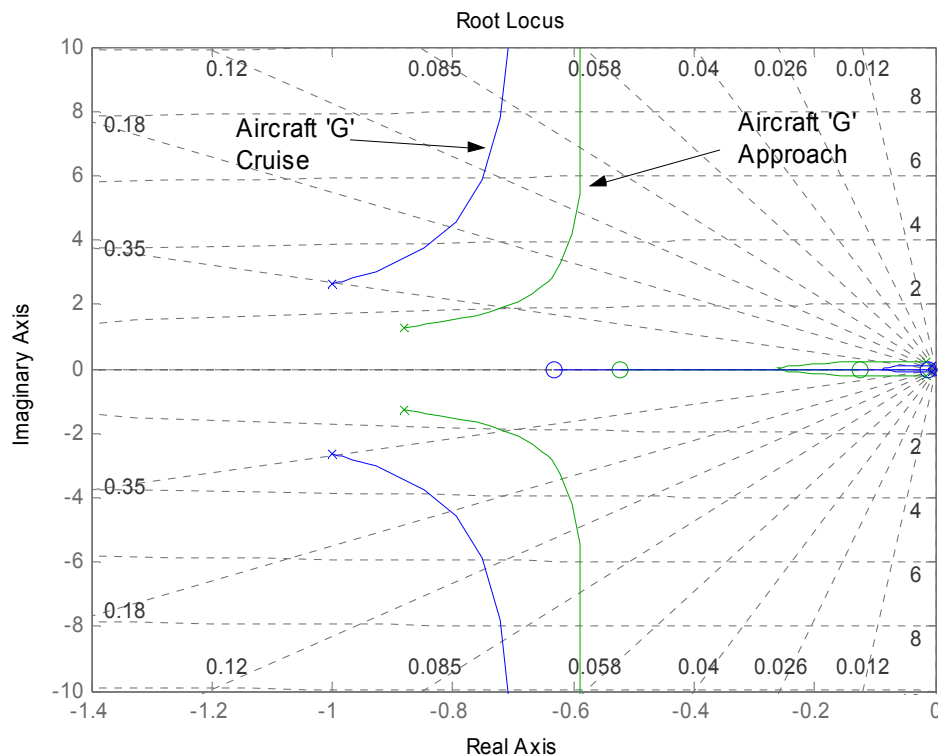


Fig. 4.31: Open Loop Root Locus Of Aircraft 'G' In Cruise And Approach Flight Condition.

The change of root locus plot looks a little bit like the change under influence of a different aircraft type seen in Fig. 4.29. If we compare the 'Cruise Flight Condition' with the 'Approach Flight Condition', we can conclude that the closed-loop Short-Period roots will be a little more damped in the latter condition. The change is however not very significant. In Fig. 4.32, the plot is scaled to look at the effect on the Phugoid roots.

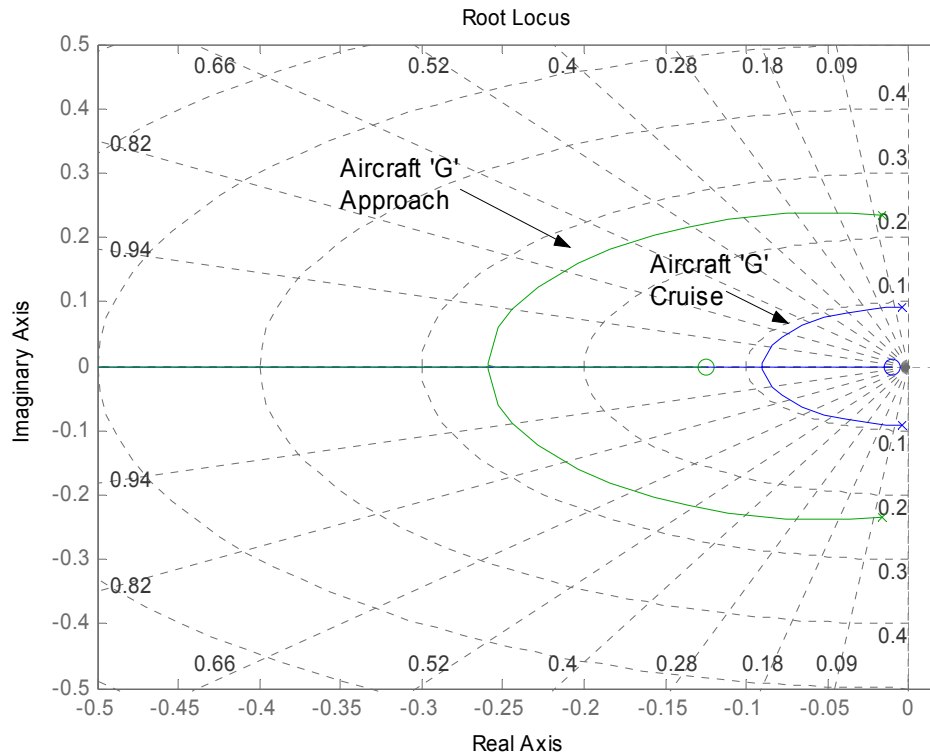


Fig. 4.32: Magnification Near The Origin Of Fig. 4.31.

It is seen that the 'Approach' root locus is further away from the origin, so the period of the Phugoid movement will be a little smaller. The effect of the flight condition is not very significantly, but in the closed-loops AFCS, the flight condition can be of more influence. Because the Phugoid branch is further away from the origin, the root locus of the closed-loop system can be set also further away from the imaginary axis. This, of course, results in a larger gain margin.

4.4 Lateral Control Of An Aircraft

4.4.1 The Lateral Open Loop Responses Of An Aircraft (The Spiral, Dutch Roll & Roll Movements)

In this chapter, the lateral control of an aircraft will be discussed by using the Roll to Aileron Transfer Function for an aircraft 'A' in a cruise flight condition. In Fig. 4.33, the pole-zero map of the Roll to Aileron Transfer Function is plotted.

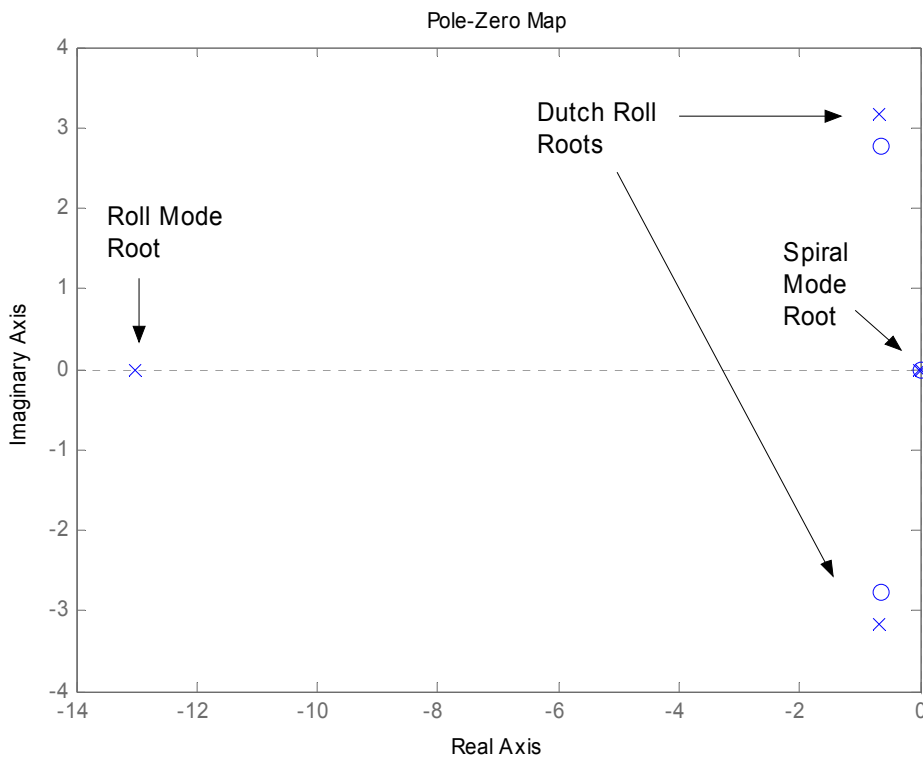


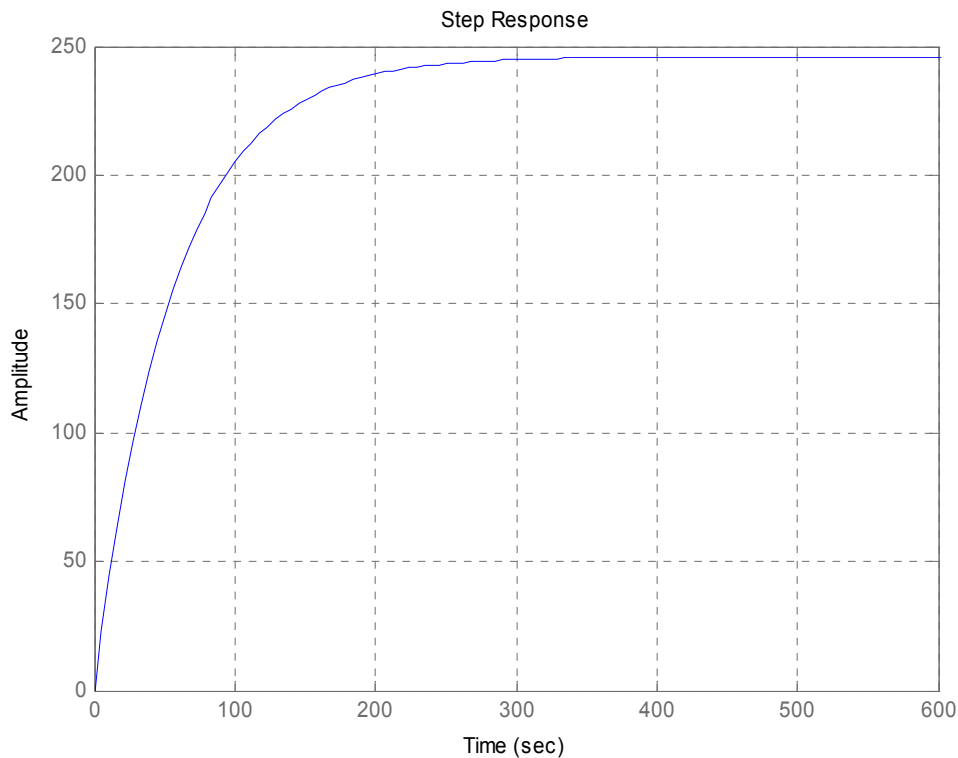
Fig. 4.33: Roll To Aileron Transfer Function Pole-Zero Map For Aircraft 'A'.

Just like with previous discussion of the longitudinal movements, the lateral movements also have a few distinct roots. The pole which is the farthest away from the origin is called the Roll Mode Root. This mode consists of almost pure rolling motion and is generally a non-oscillatory motion because it is critically damped. Therefore, the high frequency is no problem. The root closest to the origin is called the Spiral Mode Root. This mode is normally, just like the Phugoid mode, a very low frequency movement and is therefore not of critical importance. The most important movement is caused by the complex root pair which are called the Dutch Roll Roots. This movement is a coupled roll- and yaw-movement and is normally not very well damped. In large aircrafts, this mode is suppressed by the use of a Yaw-Damper. The damping and frequency data of these modes are shown in Table 4.4 for an aircraft 'A' in cruise flight condition.

Eigenvalue	Damping	Frequency [rad/sec]	Roots
-0.0179	1.00	0.0179	<i>Spiral Mode Root</i>
-0.670 +/- 3.17i	0.207	3.24	<i>Dutch Roll Roots</i>
-13	1.00	13.0	<i>Roll Mode Root</i>

Table 4.4: Eigenvalue Characteristics For The Roll To Aileron Transfer Function Of Aircraft 'A'.

The step response for this open-loop system is given in Fig. 4.34.

**Fig. 4.34: Roll To Aileron Transfer Function Open-Loop Step Response For Aircraft 'A'.**

In normal situations, the aircraft's roll angle will continue to increase when a pilot carries out a step input to the aileron. The change in roll angle, known as the roll-rate ' p ', will be constant very shortly after the beginning of the step input. The roll angle will only return to a steady state when the pilot decreases the input to the aileron.

If we look now at Fig. 4.34, at first sight this plot doesn't look correct if we think of how an aircraft is controlled. But one should notice that the time scale is 10 minutes! In an AFCS, a normal limit for the roll-rate is about 3 degrees / sec and the maximum roll angle is often limited to about 25 degrees. Thus the maximum roll angle is reached in less than 9 seconds! If we plot the step response with a time-scale of 10 seconds, the plot looks almost like a straight line (see Fig. 4.35).

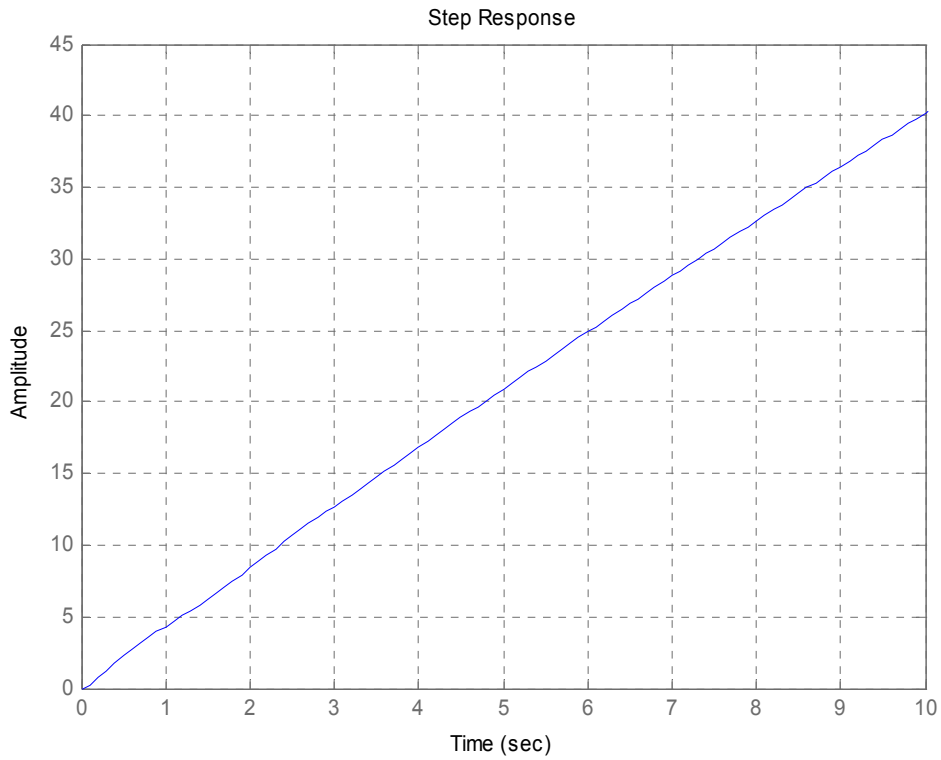


Fig. 4.35: Time Scale Adjusted Of Fig. 4.34.

The derivative of this step response is of course the roll-rate which is plotted in Fig. 4.36.

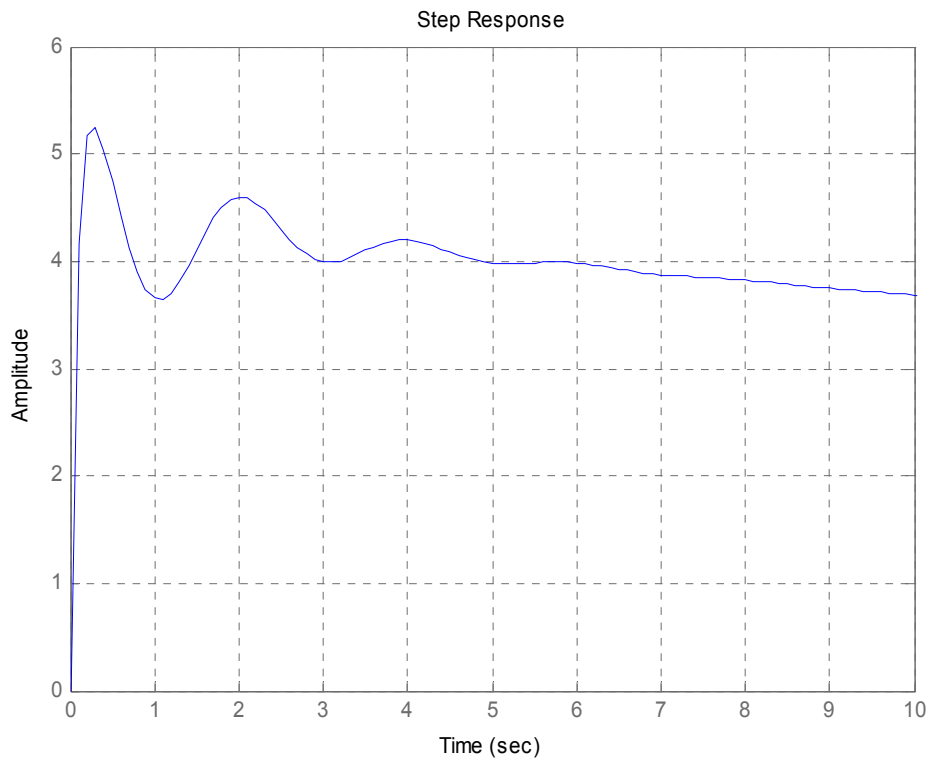


Fig. 4.36: Step Response Of The Roll-Rate.

It is seen in these pictures that the responses are in a way one would expect them to be: the roll-rate is after a very short time almost constant during a aileron step response and therefore the increase in roll angle is almost linear. For this type of aircraft, the maximum roll angle of 25 degrees is reached in about 6 seconds and the constant roll-rate is about 4 degrees / sec.

It is very important to notice that in the roll angle to aileron transfer function, some sort of integrating action must be present because the step response looks very much like a step response from an integrator. If we look at the pole-zero map in Fig. 4.33, we can see that this effect is caused by the spiral-mode root. This root is so close to the origin, that it almost acts like an integrator.

One should observe that this is a great difference with the elevator step responses: therefore the elevators are often called position-devices because they control the 'position' of the pitch angle and ailerons are often called rate-devices because they control the roll-rate of the aircraft.

4.4.2 A Roll Feedback Only Lateral Inner-Loop

We will start the analysis with a feedback system where only the roll angle ϕ is used as a feedback. In Fig. 4.37, this system is shown.

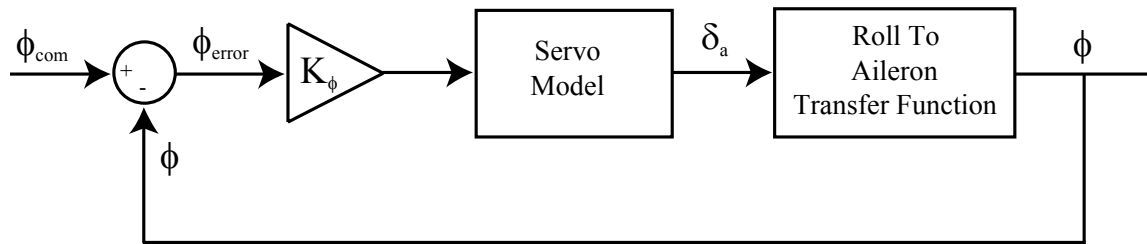


Fig. 4.37: A Roll Feedback Only Lateral Inner-Loop.

The root locus of this system is plotted in figure Fig. 4.38.

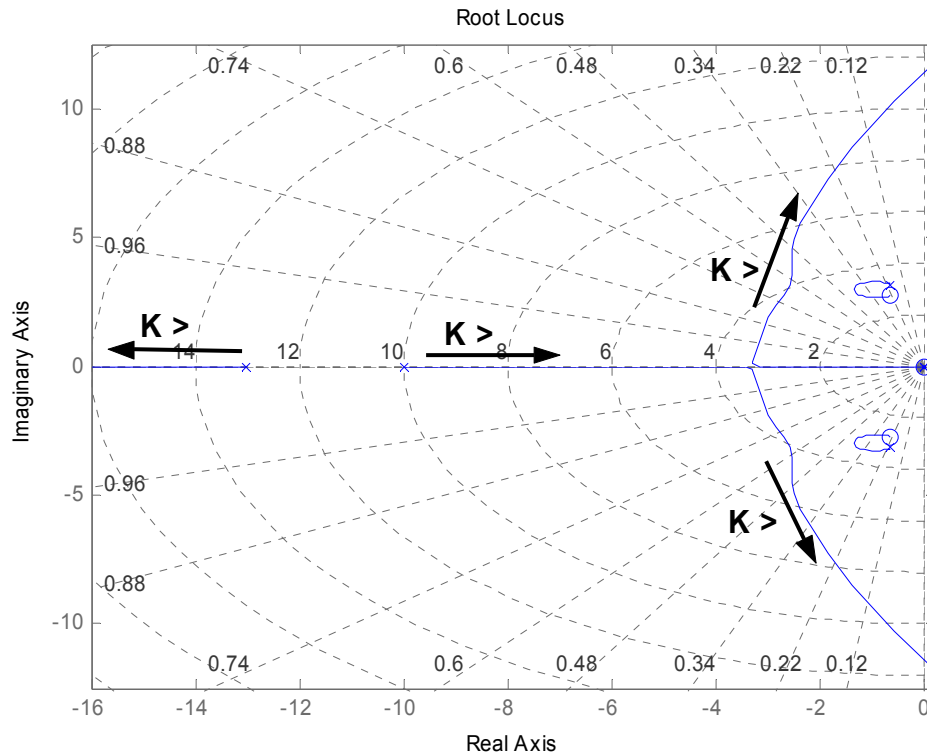


Fig. 4.38: Root Locus Of A Roll Feedback Only Lateral Inner-Loop For Aircraft 'A'.

It can be seen from this picture that the Dutch Roll Roots don't change very much, but as the roll gain is increased, the Spiral Mode Root and the root from the servo-model will form a complex pole-pair at a roll gain of 0.33. This pole-pair causes an oscillation in the step response. If the gain is chosen too large, the system will become unstable. The value of the roll gain where this happens is approximately 3.97. In Fig. 4.39, the branches near the Dutch Roll Roots can be seen more clearly.

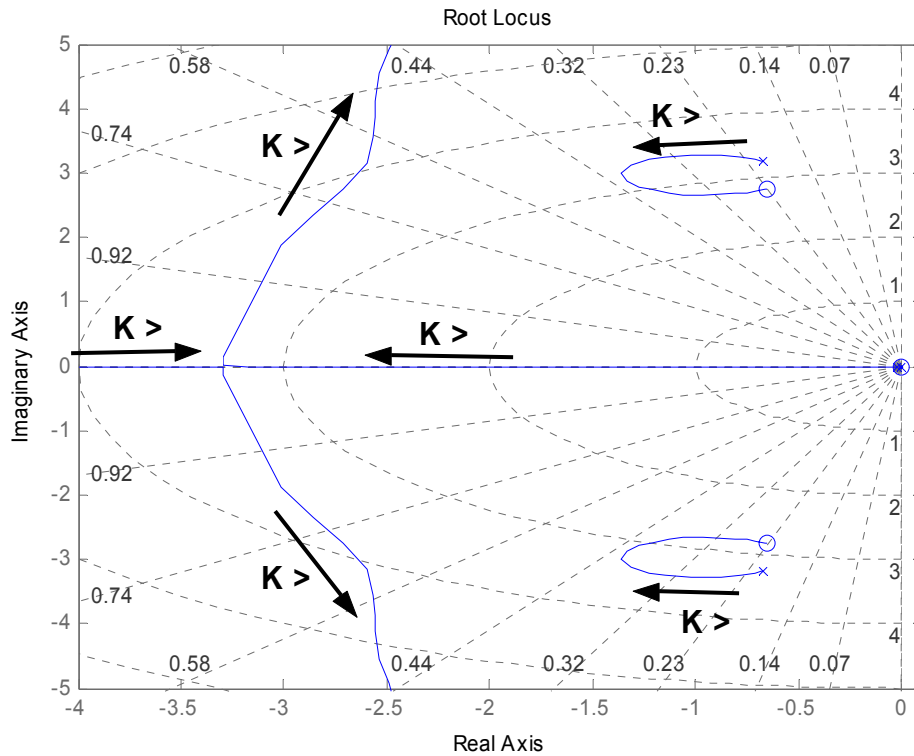


Fig. 4.39: Magnification Near The Origin Of Fig. 4.38.

The step responses for roll gains of 0.1, 0.3 and 1.0 are plotted in Fig. 4.40.

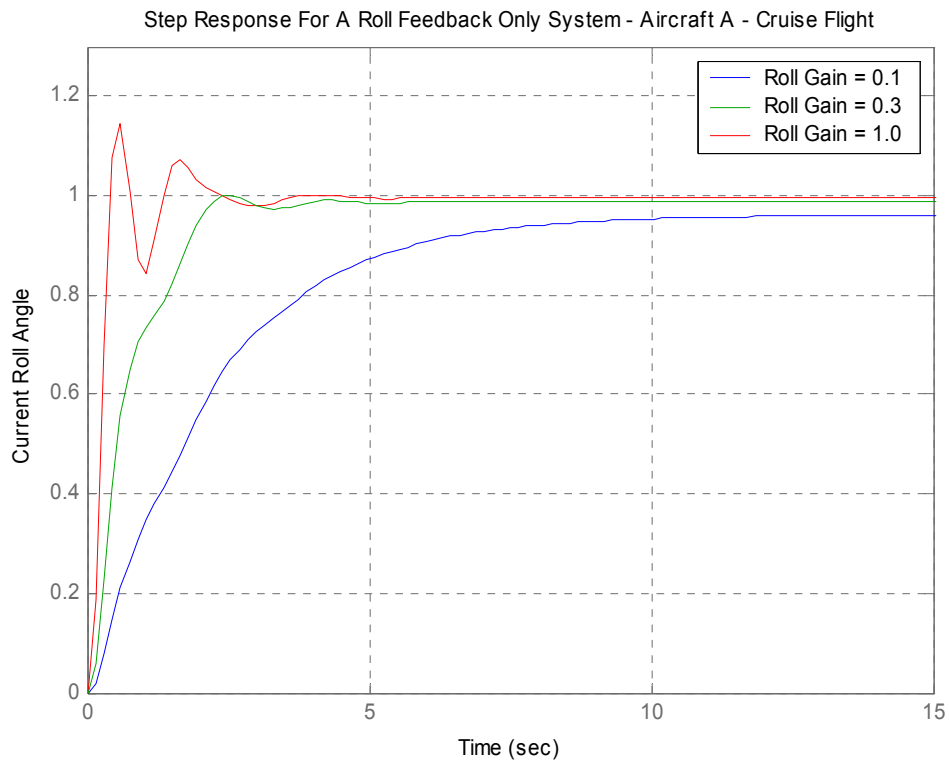


Fig. 4.40: Step Responses Of A Roll Feedback Only Lateral Inner-Loop For Aircraft 'A'.

These step responses shows that the roll gain should be very small (max. 0.3) to have a reasonable system-response. The systems starts oscillating very quickly. Therefore, we will now add roll-rate feedback to improve the damping of the whole system, just like we did with the longitudinal system.

4.4.3 Roll & Roll-Rate Feedback

In Fig. 4.41, the block diagram is showed of the system with both roll and roll-rate feedback. The roll-rate is derived from the roll angle through differentiation.

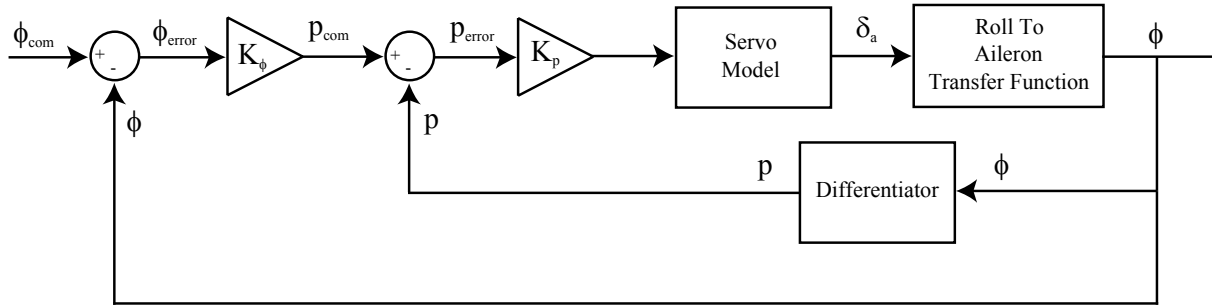


Fig. 4.41: Roll & Roll-Rate Feedback Inner-Loop.

We will analyze this system also in two steps: first we will analyze the innermost roll-rate loop. From the root locus plot, we will determine a suitable roll-rate gain. With this gain, the whole system with roll as well as roll-rate will be analyzed and suitable roll gains will be determined.

4.4.3.1 The Roll-Rate Loop

Fig. 4.42 shows the innermost roll-rate loop which will be analyzed in this section.

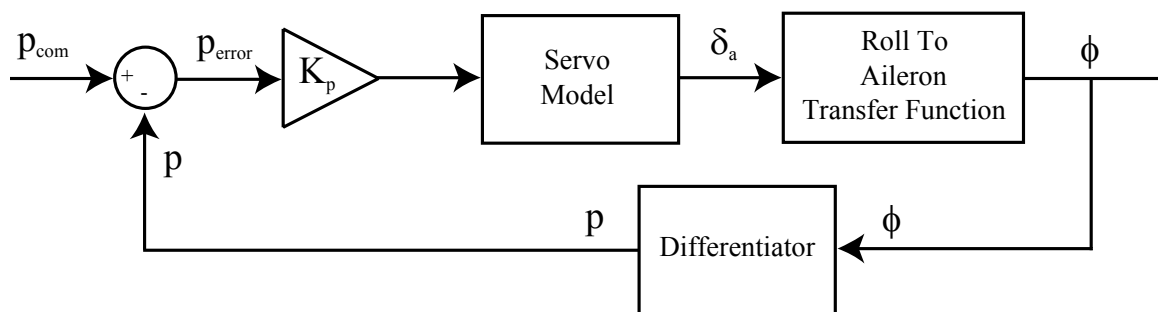


Fig. 4.42: The Roll-Rate Loop.

The root locus of this system is shown in Fig. 4.43.

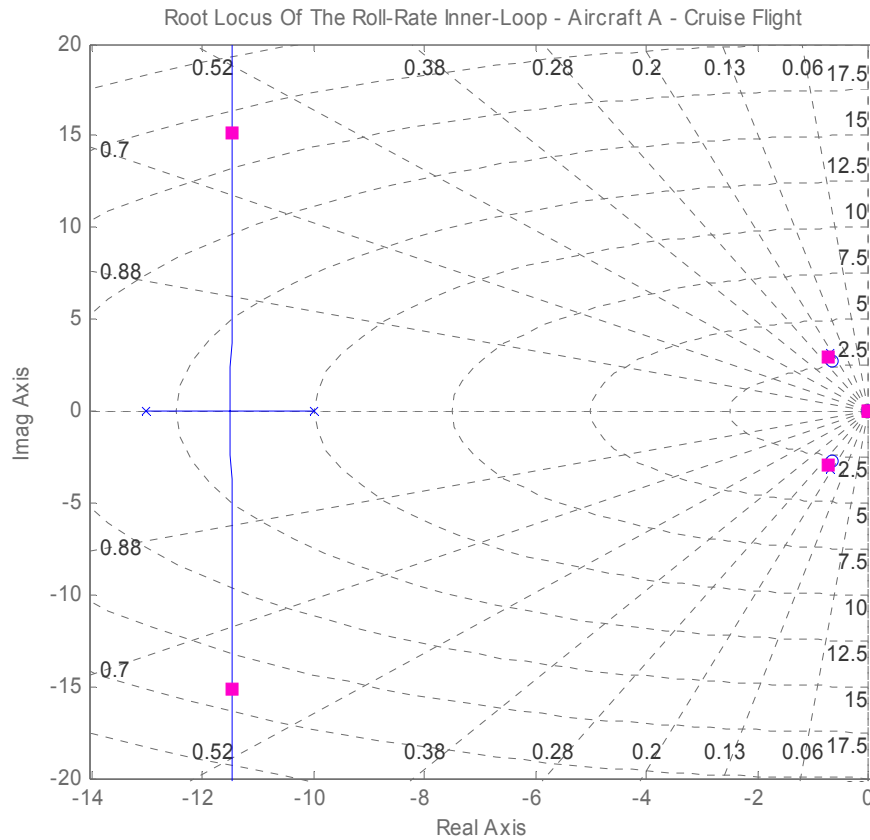


Fig. 4.43: Root Locus Of The Roll-Rate Inner-Loop.

It is concluded from this root locus plot that the roll-rate inner-loop will always be stable and that from a very small roll gain of 0.00307, two complex roots appear in the middle of the Roll Mode Root and the root from the servo. It can be seen in the root locus plot as well as in Table 4.5 that the damping and frequency of the Dutch Roll Roots don't change significantly. For the next analysis, roll-rate gains of 0.5 & 1.0 will be considered.

K_p	ζ_{dr}	$\omega_{n_{dr}}$	ζ_{rm}	$\omega_{n_{rm}}$
0.1	0.230	3.11	0.802	14.3
0.3	0.237	2.99	0.609	18.9
0.5	0.237	2.94	0.51	22.5
1.0	0.235	2.90	0.387	29.7

Table 4.5: Closed-Loop Poles Characteristics For Different Values Of The Roll-Rate Gain K_p .

4.4.3.2 The Roll- And Roll-Rate Inner-Loop

With the found roll-rate gains, we can now plot the root locus of the complete inner-loop with both roll as well as roll-rate feedback. In Fig. 4.44, the root locus for a roll-rate gain of 0.5 is plotted. The root locus for a roll-rate gain of 1.0 looks almost the same, only the two complex branches lie further away from the real axis. We can see in this plot that the further away the complex branch is from the real axis, the less damping the whole inner-loop has. Therefore, we shouldn't make the roll-rate gain too large.

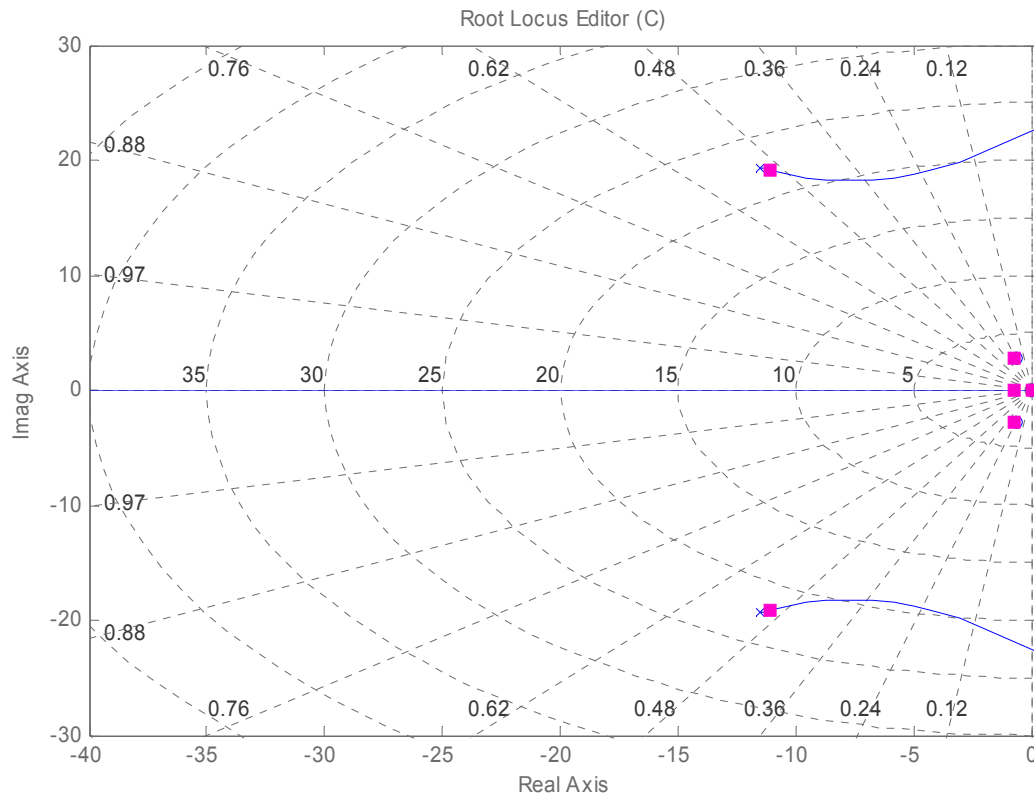


Fig. 4.44: Root Locus Plot Of The Roll- And Roll-Rate Inner-Loop.

For a roll-rate gain of 0.5, the step responses of roll gains of 0.5, 1.0 and 4.0 are plotted in Fig. 4.45

Step Response For A Roll & Roll-Rate Feedback System (Roll-Rate Gain = 0.5) - Aircraft A - Cruise Flight

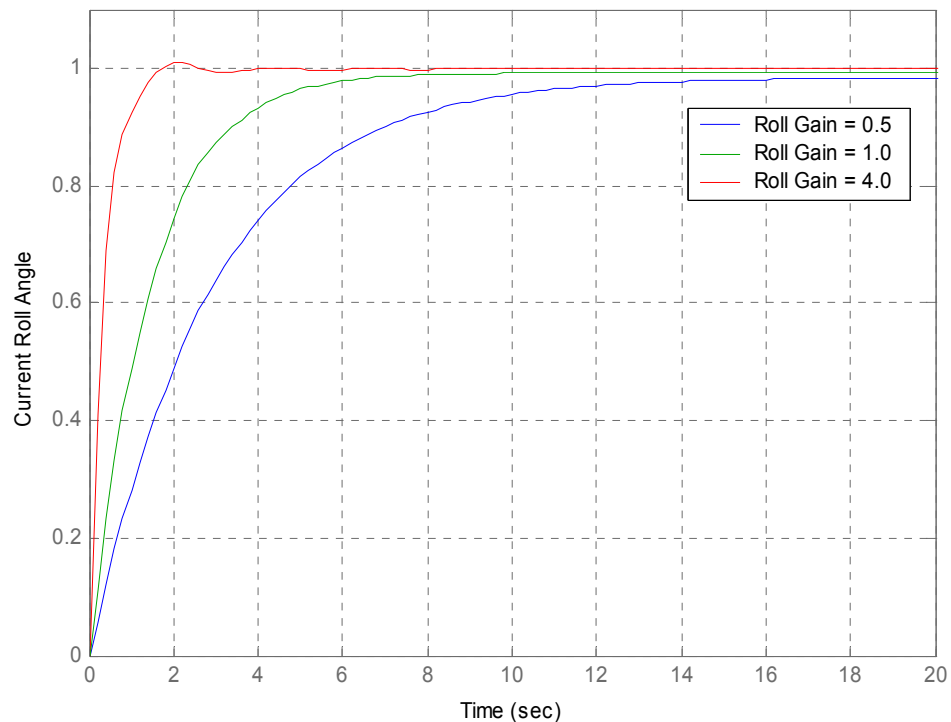


Fig. 4.45: Step Response Of The Roll & Roll-Rate Inner-Loop ($K_p=0.5$).

To see if a roll-rate gain of 1.0 give suitable responses, the step responses for this gain are also plotted (see Fig. 4.46).

Step Response For A Roll & Roll-Rate Feedback System (Roll-Rate Gain = 1.0) - Aircraft A - Cruise Flight

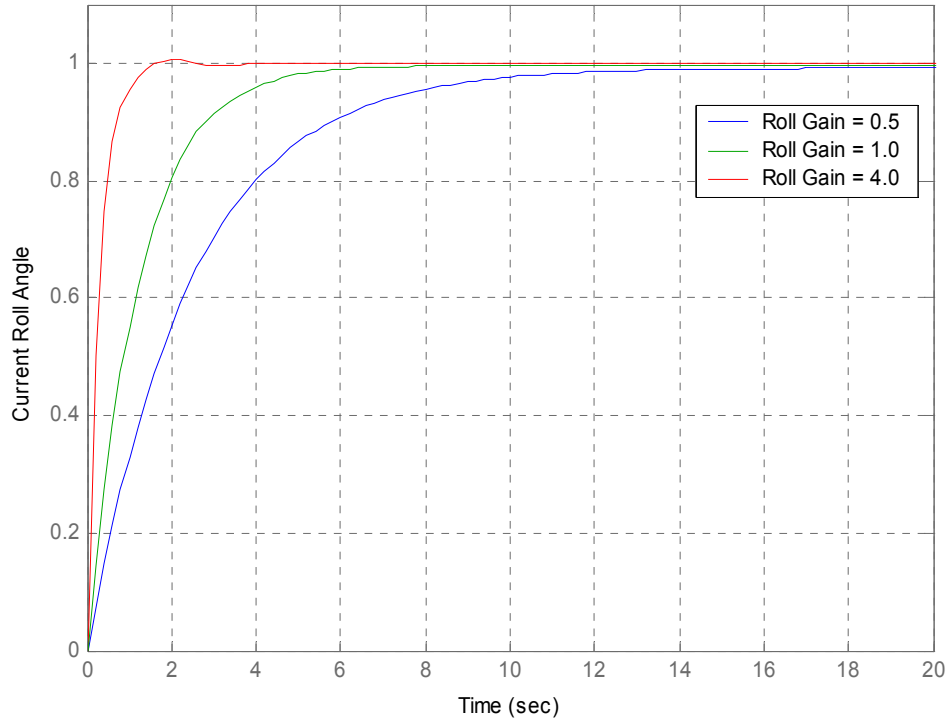


Fig. 4.46: Step Response Of The Roll & Roll-Rate Inner-Loop ($K_p=1.0$).

From these figure, it follows that there is not much difference between the roll-rate gain of 0.5 and 1.0. The latter system is a little fast and the steady-state error is a little bit smaller.

4.4.4 With An Added Integrator

Just like with the pitch inner-loop system, we will now add an integrator to the system to eliminate the steady-state error. An overview of this system can be seen in Fig. 4.47.

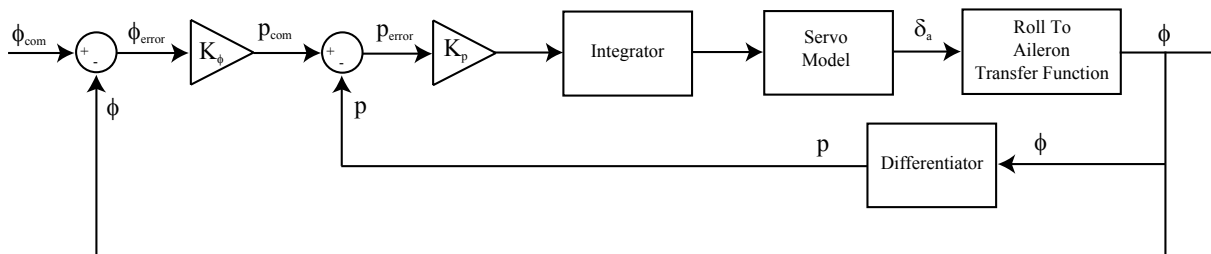


Fig. 4.47: The Roll- And Roll-Rate Inner-Loop With An Added Integrator.

Again, we will first look at the roll-rate inner-loop (see Fig. 4.48) and plot the root locus of this system (see Fig. 4.49).

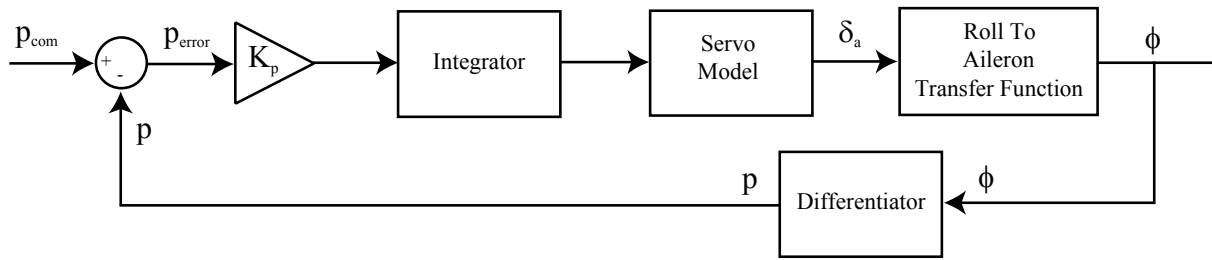


Fig. 4.48: The Roll-Rate Loop With An Added Integrator.

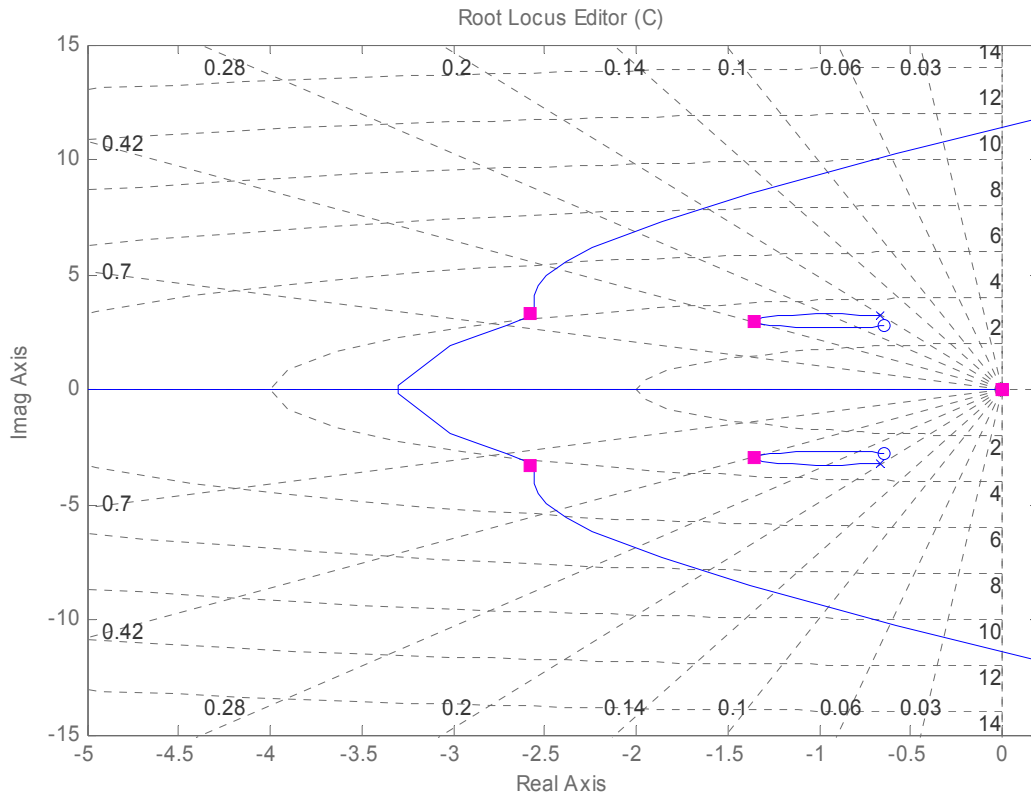


Fig. 4.49: Root Locus Of The Roll-Rate Loop With An Added Integrator.

If we compare this plot with the root locus of Fig. 4.43 where the system didn't have an integrator, we can see that the influence of the integrator is that the main branch is pulled to the right half-plane and therefore making the system unstable at some value of the roll-rate gain (this value appeared to be 3.97). For the further analysis, a roll-rate gain of 0.5 & 1.0 will be considered. The root locus plot for a roll-rate gain of 0.5 is plotted in Fig. 4.50.

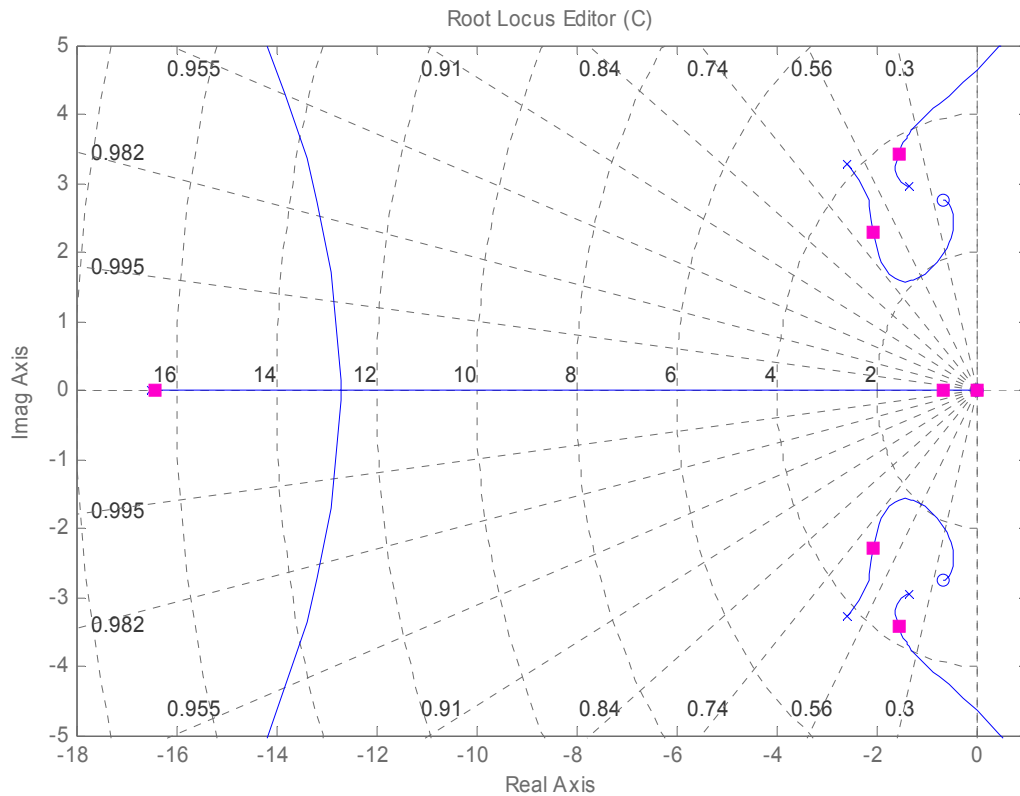


Fig. 4.50: Root Locus Plot Of The Roll & Roll-Rate Inner-Loop With An Added Integrator.

From this plot, the roll gain values of 0.3, 0.5, 1.0 and 2.0 were considered. The step responses for these gains are plotted in Fig. 4.51.

Step Response For A Roll & Roll-Rate Feedback System (Roll-Rate Gain = 0.5) - Aircraft A - Cruise Flight

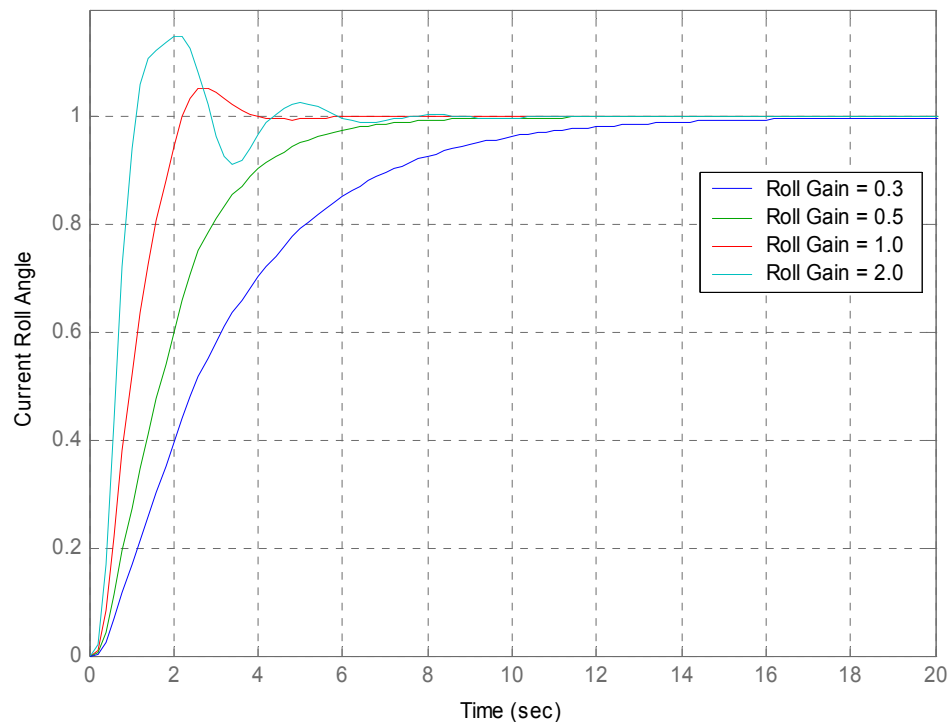


Fig. 4.51: Step Responses With $K_p = 0.5$ And $K_\phi = 0.3 / 0.5 / 1.0 / 2.0$.

It can be concluded from these plots that from a roll gain of 1.0, the overshoot starts. Therefore a roll gain of 1.0 should be the maximum roll gain when a roll-rate gain of 0.5 is used. As expected, the influence of the integrator is that the steady-state error is completely eliminated and therefore the integrator does its job correctly. Step responses for a roll-rate gain of 1.0 are plotted in Fig. 4.52.

Step Response For A Roll & Roll-Rate Feedback System (Roll-Rate Gain = 1.0) - Aircraft A - Cruise Flight

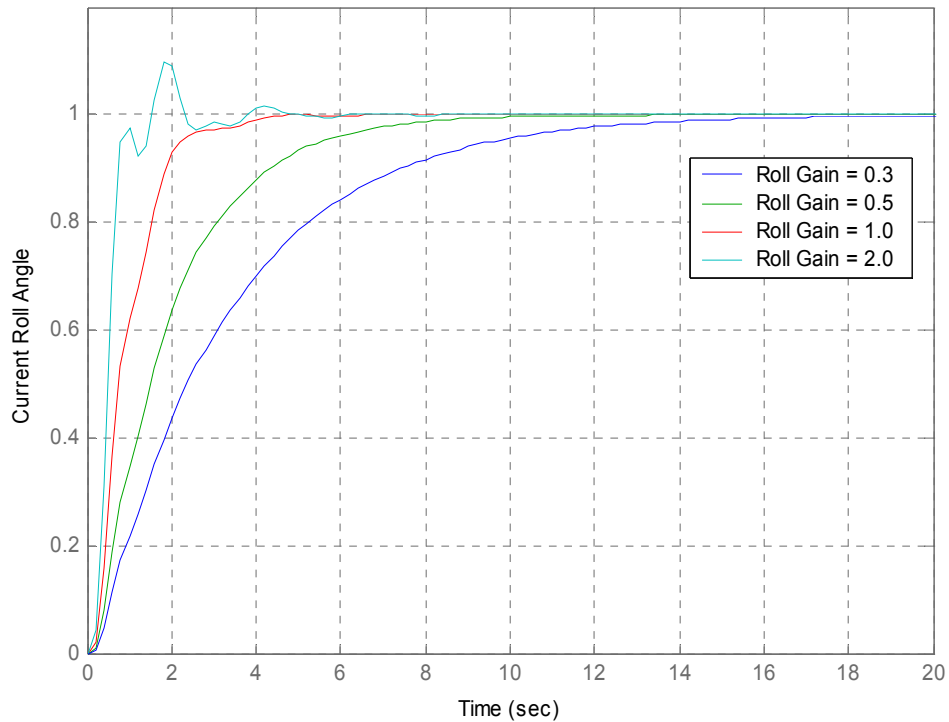


Fig. 4.52: Step Responses With $K_p = 1.0$ And $K_\phi = 0.3 / 0.5 / 1.0 / 2.0$.

If we compare these plots with the plots of a roll-rate gain of 0.5, we see that it takes a little bit longer before overshoot occurs. But when the overshoot occurs at a roll gain of 2.0, the response is very erratic and therefore the gain-margin will be smaller.

The conclusion is that if we use a roll-rate gain of 0.5, roll gains of 0.3 and 0.5 give acceptable responses and if we use a roll-rate gain of 1.0, roll gains of 0.3, 0.5 and 1.0 give acceptable responses.

5 IMPLEMENTATION OF THE AFCS

5.1 The Digital Implementation Of A PID Controller

To implement the designed system into a C++ routine, a short discussion of the implementation of an analogue PID is useful. In Fig. 5.1, a standard analogue PID controller is drawn.

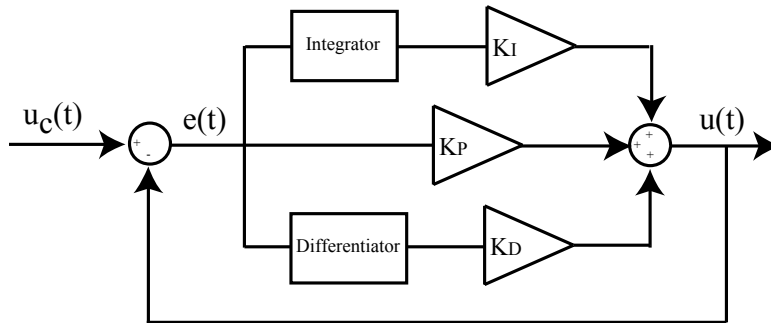


Fig. 5.1: A Standard Analogue PID Controller.

The input signal $e(t)$ is the difference between a command signal $u_c(t)$ and a controller output signal $u(t)$ and therefore is the signal $e(t)$ an error signal. The controller output signal $u(t)$ can be described as:

$$u(t) = K_P \cdot e(t) + K_I \cdot \int_0^t e(t) dt + K_D \cdot \frac{de(t)}{dt} \quad (6)$$

In the past, these analogue PID controllers were used very often and over time it was found that the controllers could be improved in three ways:

1. First, it is better to let only a fraction b of the command signal work on the proportion part. The advantage of this design is that we have control over the position of the zero in the controller transfer function.
2. Second, it was found useful to let the derivative action only act on the output signal and not on the command signal. The advantage of this design is that differentiation of fast changes of the command signal are avoided.
3. Third, to avoid amplification of high-frequency noise, the derivative signal is limited to N at high frequencies.

In Fig. 5.2, a generic block diagram shows the PID controller design with these improvements implemented.

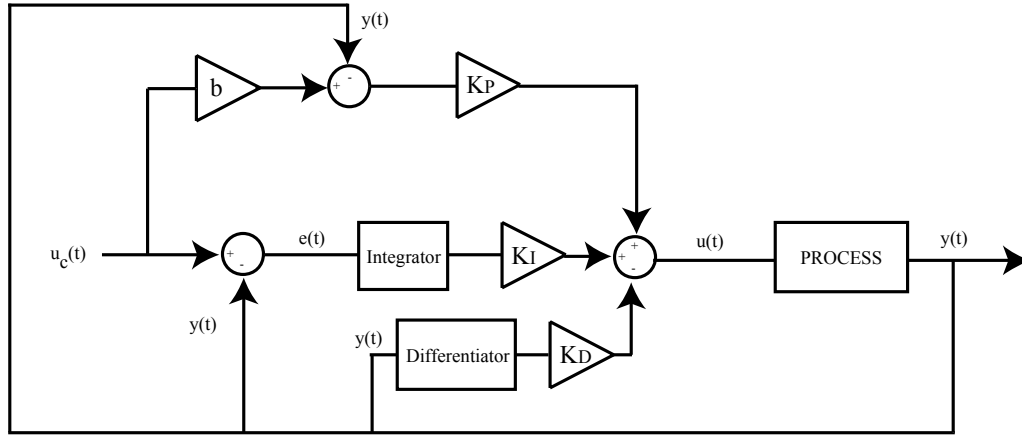


Fig. 5.2: Generic Block Diagram Of An Improved PID Controller.

If we redraw this diagram specifically for our purpose, we get the design in the s-domain (see Fig. 5.3).

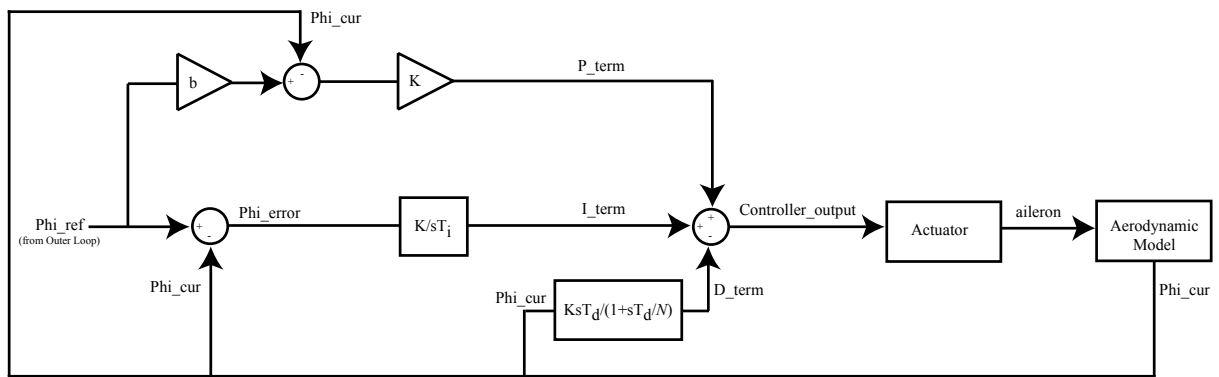


Fig. 5.3: Redrawn Of Fig. 5.2 In The S-Domain.

The overall gain K is present in all the three terms. The integrator is written as: K/sT_i , where T_i is the integrating time constant and the differentiator is written as: $KsT_d/(1+sT_d/N)$, where T_d is the differentiating time constant. By using this form for the differentiator, instead of the standard KsT_d form, the derivative signal is limited to N at high frequencies.

It is important to remember that the actuator is a non-linear actuator. It's output is limited between $aileron_min$ and $aileron_max$.

For the controller output signal, we can write:

$$Controller_output(s) = K \cdot (b\varphi_{ref}(s) - \varphi_{cur}(s)) + \frac{K}{sT_i} (\varphi_{ref}(s) - \varphi_{cur}(s)) - \left(\frac{KsT_d}{1 + sT_d/N} \right) \varphi_{cur}(s) = P(s) + I(s) + D(s) \quad (7)$$

If we want to implement this equation digitally, we must discretize it with common approximation methods.

5.1.1 The Proportional Part

The proportion term is given in the s-domain by:

$$P(s) = K(b\varphi_{ref}(s) - \varphi_{cur}(s)) \quad (8)$$

is a purely static part and therefore requires no approximation. The discrete form used for implementation in C++ is:

$$P(kT_s) = K(b\varphi_{ref}(kT_s) - \varphi_{cur}(kT_s)) \quad (9)$$

Where k is an integer and T_s is the sampling time.

5.1.2 The Integrating Part

The integrating term is given in the s-domain by:

$$I(s) = \frac{K}{sT_i}(\varphi_{ref}(s) - \varphi_{cur}(s)) \quad (10)$$

The term approximated by the rectangular approximation (also often called ‘Euler’s Forward Rectangular Rule’) $s \approx \frac{z-1}{T_s}$:

$$\begin{aligned} I(z) &= \frac{KT_s}{(z-1)T_i} \cdot (\varphi_{ref}(z) - \varphi_{cur}(z)) \Rightarrow (z-1) \cdot I(z) = \frac{KT_s}{T_i} \cdot (\varphi_{ref}(z) - \varphi_{cur}(z)) \Rightarrow \\ &\Rightarrow z \cdot I(z) = I(z) + \frac{KT_s}{T_i} \cdot (\varphi_{ref}(z) - \varphi_{cur}(z)) \end{aligned} \quad (11)$$

For implementation, we re-write this last equation into a discrete-time form by using:

$$z^{-1} \cdot X(z) \simeq q^{-1} \cdot x(kT_s) = x(kT_s - T_s) \text{ or } z \cdot X(z) \simeq q \cdot x(kT_s) = x(kT_s + T_s) \quad (12)$$

which represents a backward or a forward shift operation. This gives:

$$I(kT_s + T_s) = I(kT_s) + \frac{KT_s}{T_i}(\varphi_{ref}(kT_s) - \varphi_{cur}(kT_s)) \quad (13)$$

Note that this equation calculates the I-term in the future at time $kT_s + T_s$! Of course we can write this into the regular form:

$$I(kT_s) = I(kT_s - T_s) + \frac{KT_s}{T_i}(\varphi_{ref}(kT_s - T_s) - \varphi_{cur}(kT_s - T_s)) \quad (14)$$

5.1.3 The Derivative Part

The differentiating term is given in the s-domain by:

$$\begin{aligned}
 D(s) &= -\left(\frac{sKT_d}{1+sT_d/N}\right) \cdot \varphi_{cur}(s) \Rightarrow \left(1+\frac{sT_d}{N}\right) \cdot D(s) = -sKT_d \cdot \varphi_{cur}(s) \Rightarrow \\
 \Rightarrow D(s) &= -\frac{sT_d}{N} \cdot D(s) - sKT_d \cdot \varphi_{cur}(s) \Rightarrow N \cdot D(s) = -sT_d \cdot D(s) - sKT_d N \cdot \varphi_{cur}(s)
 \end{aligned} \tag{15}$$

This equation is approximated by taking backward differences through

$$s \approx \frac{z-1}{zT_s} = \frac{1-z^{-1}}{T_s} :$$

$$\begin{aligned}
 N \cdot D(z) &= -\left(\frac{1-z^{-1}}{T_s}\right) \cdot T_d \cdot D(z) - \left(\frac{1-z^{-1}}{T_s}\right) \cdot KT_d N \cdot \varphi_{cur}(z) \Rightarrow \\
 \Rightarrow NT_s \cdot D(z) &= (z^{-1}-1) \cdot T_d \cdot D(z) - (1-z^{-1}) \cdot KT_d N \cdot \varphi_{cur}(z) \Rightarrow \\
 \Rightarrow (T_d + NT_s) \cdot D(z) &= T_d \cdot z^{-1} \cdot D(z) - KT_d N \cdot (1-z^{-1}) \cdot \varphi_{cur}(z) \Rightarrow \\
 \Rightarrow D(z) &= \left(\frac{T_d}{T_d + NT_s}\right) \cdot z^{-1} \cdot D(z) - \left(\frac{KT_d N}{T_d + NT_s}\right) \cdot (\varphi_{cur}(z) - z^{-1} \cdot \varphi_{cur}(z))
 \end{aligned} \tag{16}$$

For implementation, we re-write this last equation into the discrete-time form:

$$D(kT_s) = \left(\frac{T_d}{T_d + NT_s}\right) \cdot D(kT_s - T_s) - \left(\frac{KT_d N}{T_d + NT_s}\right) \cdot (\varphi_{cur}(kT_s) - \varphi_{cur}(kT_s - T_s)) \tag{17}$$

5.1.4 The Total Output Of The PID Controller

The total output of the PID controller is given by:

$$\begin{aligned}
 Controller_output(kT_s) &= P(kT_s) + I(kT_s) + D(kT_s) = \\
 &= K \left(b\varphi_{ref}(kT_s) - \varphi_{cur}(kT_s) \right) + I(kT_s - T_s) + \frac{KT_s}{T_i} \left(\varphi_{ref}(kT_s - T_s) - \varphi_{cur}(kT_s - T_s) \right) + \\
 &+ \left(\frac{T_d}{T_d + NT_s}\right) \cdot D(kT_s - T_s) - \left(\frac{KT_d N}{T_d + NT_s}\right) \cdot (\varphi_{cur}(kT_s) - \varphi_{cur}(kT_s - T_s))
 \end{aligned} \tag{18}$$

To be able to compute the Controller_output signal, we thus have to save the states of the variables I, D, φ_{ref} and φ_{cur} in order to have these old values available at time $kT_s + T_s$.

5.1.5 Anti-windup design

Because the actuator is a non-linear actuator, its output can become saturated at some time when the output control signal becomes very large or very small. If this happens, the integrator signal is increasing (or decreasing) during the period that the actuator is saturated. If the error signal, after some time, finally decreases (or increases), it takes a considerable amount of time for the integrating signal to decrease (or increase) to a normal value. This effect is called integrator windup. To counteract this effect, an additional feedback signal can be used which measures the difference between the output and the input of the actuator. If this difference is zero, the actuator is not saturated, but if this difference has a value other than zero, the actuator is saturated and the difference signal is fed-back to the integrator through a gain with tracking-time constant T_t as shown in Fig. 5.4.

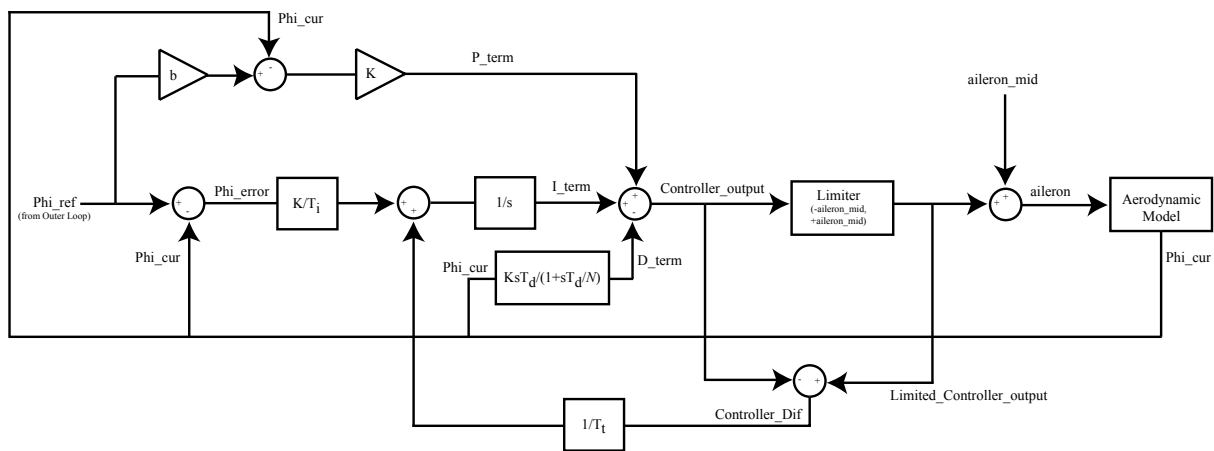


Fig. 5.4: Block Diagram Of The System With An Anti-Windup Scheme.

The integrating term with this anti-windup feedback becomes:

$$I(kT_s) = I(kT_s - T_s) + \frac{KT_s}{T_i} (\varphi_{ref}(kT_s - T_s) - \varphi_{cur}(kT_s - T_s)) - \frac{T_s}{T_i} (\text{aileron}(kT_s - T_s) - \text{Controller_output}(kT_s - T_s)) \quad (19)$$

For tuning, the following applies:

- N is typically in the range of 3-20 (e.g. 10)
- T_t is typically equal to T_i or 0.1-0.5 times T_i

The primary tuning parameters are K , T_i and T_d and the secondary tuning parameters are b , T_t and N .

Note:

It should be noted that the previous discussion was only intended to gain some insight into digital implementation of a (PID) controller. In our AFCS design, not a standard PID controller is used, because the I-term and D-term are not placed in parallel over the P-term. Due to time constraints, the digital implementation of our AFCS-design will not be discussed here.

6 DISCUSSION, CONCLUSIONS AND RECOMMENDATIONS

It should be noted that the author is not an Aerospace Engineer, but a *future* Electrical Engineer with a specialization in Avionics.

At the start of this project, the goal was to design an Automatic Flight Control System which would be, in a way, adaptive. What is meant with the term ‘adaptive’, is that the system could internally adjust itself every time a new or different aerodynamic flight model is loaded in the simulator. This is desired because the P-2 Trainer is build with flexibility in mind and therefore will be adaptable to simulate different types of aircrafts. It was proven to be very difficult at the beginning to find the information needed to design such a system. The design of the system was started from a ‘practical’ point-of-view and various different designs were implemented and tested. The main problem with all this designs was the absence of knowledge how to properly tune the system and provide appropriate output responses.

With the visit to Saab Aviocomp RIAS, a turning point in this project was reached. From that moment on, the approach to the problem was changed to a ‘Classical Control Theory’ point-of-view approach. This approach gives this thesis also the required scientific basis which is of course essential for obtaining the MSc-Degree in Electrical Engineering. For this new approach, The MathWorks Matlab & Simulink Release 13[®] was used to simulate the various systems and to help in the development of the design. For these simulations, various ‘generic’ transfer functions were used which were obtained from Roskam [25]. These ‘generic’ transfer functions instead of the current implemented transfer functions were used, because the author had the ‘adaptive’ aspect always in the back of his mind. Making the design based on ‘generic’ transfer functions, will make the ‘adaptive’ design easier because there are only 10 aircraft classes specified.

Because of the badly chosen approach at the start of this project, a lot of time was lost, but not wasted, because this path proved to be an important learning process. However, the (negative) result was that this approach imposed time constraints on the project. Therefore, the project is not completely finished because the very interesting ‘adaptive’ aspect of this project was omitted due to time constraints. This was inevitable because it is very important to have first a good basic design before one can begin with the ‘adaptive’ design. However, the author has some ideas for making the system ‘adaptive’. These ideas will be described later on in this chapter.

The basic design was based on diagrams the author derived from the maintenance and repair manuals. It turned out that this derived design was basically very consistent with different designs presented in various literature. It was therefore concluded that almost all AFCS’s will have the same basic design with the outer- and inner-loops.

In chapter 4, the development of the design was presented: from the start with a very basic inner-loop to a more complicated inner-loop with addition rate feedback and a steady-state eliminator / integrator. It was concluded from a pilot point-of-view that this additional rate feedback was desired because the pilot unconsciously also uses the rate information (which he or she also derives him- or herself from the changing pitch- and / or roll angle). This conclusion was also confirmed by the derived schematics, where it turned out that the 'real' systems also derive the rate information as shown in appendix B. The basic inner-loops are therefore designed by the diagrams shown in Fig. 4.23 and in Fig. 4.47. As described earlier: these inner-loops control the aircraft and are the same for various modes. The guidance of the aircraft is provided by the outer-loops, which are simply an extra feedback loop on the outside of the inner-loops as shown in Fig. 4.1. Because of the time-constraints, these outer-loops are not described in detail, but a few simulations of the outer-loops with the designed inner-loops, were quickly done for the Heading Select and for the Altitude Select modes. The first results looked promising, but there was a steady-state error present. As shown in chapter 4, this steady-state error can be easily eliminated with the addition of an integrator. Therefore, in future designs of the outer-loops, the design of Fig. 4.1 will have to be modified with the additional integrator after the outer-loop gain and before the input of the inner-loops. During these quick simulations, the signal which is sent to the servo was also examined shortly. It turned out that for some gains which gave acceptable output step responses in chapter 4, the servo control signal showed some oscillations. It is therefore recommended that during all simulations, the servo control signal is also analysed and not only the output response. The oscillations began when the gains were too large.

A consequence of the addition of integrators which can have enormous effects on the response of systems with non-linearities is the 'windup' effect. Our designed systems will also suffer from this effect because our system is non-linear because of various limitations (as is shown in appendix B). A solution to this problem is the use of an 'anti-windup' method as shown in Fig. 5.4. This technique is basically very simple: the output signal of the limiter is subtracted from the input signal of the limiter and this difference is fed back with a gain and is subtracted from the signal which enters the integrator (note that the scheme in Fig. 5.4 uses another convention of the signs, but the working is the same). When the output signal is limited, the difference signal will be subtracted from the input signal of the integrator and thus stopping the integrator to integrate and preventing the output from increasing. It is recommended and in fact essential that these anti-windup schemes are implemented in the future system for proper functioning.

Another aspect which should be implemented is a compensation for the pitch angle in turns. This is necessary to compensate for the loss of lift when an aircraft is in a bank, as explained in chapter 2. Therefore, the lateral and longitudinal system work together in a turn. In chapter 2, it was also described why pilots need to apply rudder to help the aircraft turn. If an AFCS does not have rudder control, the pilot could help the AFCS with applying rudder him- or herself. Advanced AFCS's do have rudder control and they provide the, so called, 'turn-coordination' to help the aircraft turn.

As was proved in chapter 4, the flight condition of the aircraft will have influence on the response of the aircraft. Although, at first, this influence looks not very large, it is very much possible that this influence is significant with other aircrafts and other flight

conditions then the ones examined. The first parameter with which different flight conditions can be determined is the altitude. For example, a 20,000 feet an aircraft is probably in a cruise flight under normal circumstances. At low altitudes, the aircraft can be either in cruise flight, approach or climb flight condition. Therefore it can probably be useful to include also the airspeed for determination of the flight condition. If the flight condition is determined by the system, it can adjust its internal gains to provide good responses for the current condition. This process is called 'gain scheduling' or 'gain adjustment'. For small aircrafts, the simple AFCS's don't offer this feature, but advanced AFCS's always uses this feature. In approach, for instance, the gain is decreased when the aircraft closes in on the runway because at that moment the airspeed is very low and the flight-path would be very erratic and overshoot would occur if the gains were not decreased (it could be of course also very dangerous because the aircraft is so close to the ground during an approach).

Because there was not enough time to design the 'adaptive' aspect of this project, the following short discussion is purely speculative.

Because 'generic' transfer functions were used in the simulations, the 'adaptive' aspect of this project is simplified. The author believes that it could be useful to have a small database with main aerodynamic characteristics of the loaded aerodynamic model. When a new aerodynamic model is loaded into the simulator, only these relevant parameters will have to be entered into the database. From the data in the database, it should be possible for the system to determine to which aircraft type category the new model belongs. This is of course only possible if there is enough relevant data in the database. Research should be conducted to determine which parameters are relevant to determine the aircraft type. There are of course a few parameters which are evident to be of great influence:

- weight
- type of engines
- number of engines
- size
- maximum airspeed

If the category of the new model is determined, the proper gains can be loaded into the model. Therefore, another database should be made before a new model is loaded, which has all the gains determined through simulations. This database however only has to be created once.

However, although it is believed that that the approach of using 'generic' aircraft types is the right one, the gains will of course not be optimal because every aircraft behaves a little bit differently, even when they belong to the same category. A final step can therefore be added to the 'adaptive' process: when the first 'coarse' gains are loaded, an automatic test flight can be conducted. During this test flight, the gains can be fine-tuned into optimal gains for this specific aerodynamic model.

This fine-tuning will be the 'adaptive' aspect of the system. The test-flight can consist of multiple flights / 'test-runs'. During each 'test-run', the AFCS can be programmed to following a fixed flight-path with fixed flight-manoevres. This flight-path and –

manoeuvres will have to be the same for each 'test-run'. During the 'test-run', relevant data should be logged (e.g. roll, pitch, roll-rate, pitch-rate, yaw, yaw-rate, angle-of-attack, altitude, heading, airspeed, aileron-deflection, elevator-deflection, rudder-deflection etc.). After each 'test-run' the logged data can be analysed and from these analysis, the gains could be adjusted to provide better responses. After the adjustments, the next 'test-run' is conducted and the process starts over. For that reason will this process be an iterative process. After some 'test-runs', the responses should be within requirements and the process is ended with storing the fine-tuned gains into a database.

Because it was previously discussed that the flight condition can be of influence on the responses, it will be useful to repeat the process for a few different flight conditions with other fixed flight-paths and –manoeuvres which are relevant for that particular flight condition.

However, the problem which arises from this approach is the transition between flight conditions during normal operation of the flight simulator. When the AFCS is on and the flight conditions changes, for example, from cruise flight to approach flight, the gains will be changed which can give of course an undesired transition-effect. A solution could be to add a fader which changes the gains slowly from one value to the other value.

APPENDIX A: EQUATIONS OF MOTION AND TRANSFER FUNCTIONS OF AIRCRAFTS

Lateral And Longitudinal Equations Of Motion Of An Aircraft

The motion of an aircraft can be described by 8th order equations of motion. Although there is a small amount of coupling of lateral motion into longitudinal motion, this is usually ignored. Therefore, the 8th order equations of motion can be broken up into two decoupled 4th order equations of motion. These two sets represent the perturbations in longitudinal and lateral motion. The longitudinal motion consists of axial (X), vertical (Z) and pitching (pitch angle θ , pitch-rate q) motion and the lateral motion consists of rolling (roll angle ϕ , roll-rate p) and yawing motion (yaw rate r , sideslip angle β (v/U_0)) motion. The longitudinal motion is primarily affected by the elevators and throttle and the lateral motion is primarily affected by the ailerons and rudder.

For a steady state straight, level and constant speed flight condition, the two sets of equations are often linearized under the condition that there is no turning in any axis and the wings will be level so that $\phi = 0$. This gives the linearized equations of motion (obtained from Franklin, [12]).

The linearized longitudinal equations of motion are:

$$\begin{bmatrix} \dot{u} \\ \dot{w} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} = \begin{bmatrix} X_u & X_w & -W_0 & -g_0 \cos(\theta_0) \\ Z_u & Z_w & U_0 & -g_0 \sin(\theta_0) \\ M_u & M_w & M_q & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix} \begin{bmatrix} u \\ w \\ q \\ \theta \end{bmatrix} + \begin{bmatrix} X_{\delta_e} \\ Z_{\delta_e} \\ M_{\delta_e} \\ 0 \end{bmatrix} \cdot \delta_e$$

Where:

u = forward velocity perturbation in the x direction

w = velocity perturbation in the z direction (proportional to the angle of attack, $\alpha = w/U_0$)

q = pitch-rate

θ = pitch angle

X_{u,w,δ_e} = partial derivative of the aerodynamic force in x direction with respect to perturbations in u , w and δ_e

Z_{u,w,δ_e} = partial derivative of the aerodynamic force in z direction with respect to perturbations in u , w and δ_e

M_{u,w,q,δ_e} = partial derivative of the aerodynamic pitching moment with respect to perturbations in u , w , q and δ_e

δ_e = elevator deflection

To determine altitude changes, the following equation can be added to the longitudinal equations of motion:

$$\dot{h} = V_{ref} \cdot \sin(\theta) - w \cdot \cos(\theta)$$

If we linearize this equation, we get the linearized altitude equation:

$$\dot{h} = V_{ref} \cdot \theta - w$$

Substituting this equation in the longitudinal equations of motions yields:

$$\begin{bmatrix} \dot{u} \\ \dot{w} \\ \dot{q} \\ \dot{\theta} \\ \dot{h} \end{bmatrix} = \begin{bmatrix} X_u & X_w & -W_0 & -g_0 \cos(\theta_0) & 0 \\ Z_u & Z_w & U_0 & -g_0 \sin(\theta_0) & 0 \\ M_u & M_w & M_q & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 \\ 0 & -1 & 0 & V_{ref} & 0 \end{bmatrix} \cdot \begin{bmatrix} u \\ w \\ q \\ \theta \\ h \end{bmatrix} + \begin{bmatrix} X_{\delta_e} \\ Z_{\delta_e} \\ M_{\delta_e} \\ 0 \\ 0 \end{bmatrix} \cdot \delta_e$$

The linearized lateral equations of motion are (also obtained from Franklin, [12]):

$$\begin{bmatrix} \dot{\beta} \\ \dot{r} \\ \dot{p} \\ \dot{\phi} \end{bmatrix} = \begin{bmatrix} Y_v & -U_0 & V_0 & g_0 \cos(\theta_0) \\ N_v & N_r & N_p & 0 \\ L_v & L_r & L_p & 0 \\ 0 & \tan(\theta_0) & 1 & 0 \end{bmatrix} \cdot \begin{bmatrix} \beta \\ r \\ p \\ \phi \end{bmatrix} + \begin{bmatrix} Y_{\delta_r} & Y_{\delta_a} \\ N_{\delta_r} & N_{\delta_a} \\ L_{\delta_r} & L_{\delta_a} \\ 0 & 0 \end{bmatrix} \cdot \begin{bmatrix} \delta_r \\ \delta_a \end{bmatrix}$$

Where:

β = side-slip angle ($\beta = v/U_0$)

r = yaw rate

p = roll-rate

ϕ = roll angle

Y_{v,δ_r,δ_a} = partial derivative of the aerodynamic force in y direction with respect to perturbations in β , δ_r and δ_a

$N_{v,r,p,\delta_r,\delta_a}$ = aerodynamic yawing moment stability derivatives

$L_{v,r,p,\delta_r,\delta_a}$ = aerodynamic rolling moment stability derivatives

δ_r = rudder deflection

δ_a = aileron deflection

The transfer functions can be found by entering the appropriate aircraft data into Matlab and using Matlab's SS2TF-function.

Example Transfer Functions Of Aircrafts

The following transfer functions were obtained from Roskam [25].

Aircraft Type 'A' (Small, Single Piston-Engine General Aviation Aircraft) Flight Condition: Cruise (5.000 Feet)

Angle Of Attack α To Elevator δ_e Transfer Function:

$$\frac{\alpha(s)}{\delta_e(s)} = \frac{-44.9854 \cdot s^3 - 7794.8686 \cdot s^2 - 355.6293 \cdot s - 330.5164}{+222.0551 \cdot s^4 + 1985.9525 \cdot s^3 + 6262.2861 \cdot s^2 + 329.8825 \cdot s + 180.5762}$$

Pitch Θ To Elevator δ_e Transfer Function:

$$\frac{\Theta(s)}{\delta_e(s)} = \frac{-7713.2340 \cdot s^2 - 15867.0001 \cdot s - 908.2451}{+222.0551 \cdot s^4 + 1985.9525 \cdot s^3 + 6262.2861 \cdot s^2 + 329.8825 \cdot s + 180.5762}$$

Altitude h To Elevator δ_e Transfer Function:

$$\frac{h(s)}{\delta_e(s)} = \frac{U_1}{s} \left\{ \frac{\Theta(s) - \alpha(s)}{\delta_e(s)} \right\} = \frac{U_1}{s} \left\{ \frac{\Theta(s)}{\delta_e(s)} - \frac{\alpha(s)}{\delta_e(s)} \right\}$$

$$\frac{h(s)}{\delta_e(s)} = \frac{+220.0735 \cdot (+44.9854 \cdot s^3 + 81.6346 \cdot s^2 - 15511.3708 \cdot s - 577.7287)}{+222.0551 \cdot s^5 + 1985.9525 \cdot s^4 + 6262.2861 \cdot s^3 + 329.8825 \cdot s^2 + 180.5762 \cdot s}$$

Roll Φ To Aileron δ_a Transfer Function:

$$\frac{\Phi(s)}{\delta_a(s)} = \frac{+16516.7989 \cdot s^3 + 21473.1354 \cdot s^2 + 132776.7201 \cdot s}{+220.0752 \cdot s^5 + 3162.7190 \cdot s^4 + 6212.5579 \cdot s^3 + 30261.6885 \cdot s^2 + 539.1737 \cdot s}$$

Heading Ψ To Aileron δ_a Transfer Function:

$$\frac{\Psi(s)}{\delta_a(s)} = \frac{-750.8413 \cdot s^3 - 15813.4284 \cdot s^2 - 3308.3966 \cdot s + 19037.9166}{+220.0752 \cdot s^5 + 3162.7190 \cdot s^4 + 6212.5579 \cdot s^3 + 30261.6885 \cdot s^2 + 539.1737 \cdot s}$$

Aircraft Type 'E' (Small, Twin-TurboProp Commuter Aircraft) Flight Condition: Cruise - High Altitude (20.000 feet)

Angle Of Attack α To Elevator δ_e Transfer Function:

$$\frac{\alpha(s)}{\delta_e(s)} = \frac{-62.6410 \cdot s^3 - 10412.5899 \cdot s^2 - 129.8301 \cdot s - 107.8977}{+451.8346 \cdot s^4 + 2197.1573 \cdot s^3 + 11332.6989 \cdot s^2 + 154.0270 \cdot s + 101.9633}$$

Pitch Θ To Elevator δ_e Transfer Function:

$$\frac{\Theta(s)}{\delta_e(s)} = \frac{-10371.2086 \cdot s^2 - 12042.5102 \cdot s - 226.3310}{+451.8346 \cdot s^4 + 2197.1573 \cdot s^3 + 11332.6989 \cdot s^2 + 154.0270 \cdot s + 101.9633}$$

Altitude h To Elevator δ_e Transfer Function:

$$\frac{h(s)}{\delta_e(s)} = \frac{U_1}{s} \left\{ \frac{\Theta(s) - \alpha(s)}{\delta_e(s)} \right\} = \frac{U_1}{s} \left\{ \frac{\Theta(s)}{\delta_e(s)} - \frac{\alpha(s)}{\delta_e(s)} \right\}$$

Roll Φ To Aileron δ_a Transfer Function:

$$\frac{\Phi(s)}{\delta_a(s)} = \frac{+7710.1719 \cdot s^3 + 4778.2179 \cdot s^2 + 29739.2664 \cdot s}{+435.9557 \cdot s^5 + 1495.3368 \cdot s^4 + 1758.4115 \cdot s^3 + 5051.8377 \cdot s^2 + 124.6967 \cdot s}$$

Heading Ψ To Aileron δ_a Transfer Function:

$$\frac{\Psi(s)}{\delta_a(s)} = \frac{+874.8018 \cdot s^3 + 407.6949 \cdot s^2 - 26.7579 \cdot s + 2086.3085}{+435.9557 \cdot s^5 + 1495.3368 \cdot s^4 + 1758.4115 \cdot s^3 + 5051.8377 \cdot s^2 + 124.6967 \cdot s}$$

Aircraft Type J (Large, 4 Jet-Engine Commercial Transport Aircraft) Flight Condition: Cruise - High Altitude (40.000 feet)

Angle Of Attack α To Elevator δ_e Transfer Function:

$$\frac{\alpha(s)}{\delta_e(s)} = \frac{-18.3410 \cdot s^3 - 1055.6956 \cdot s^2 - 84.9709 \cdot s - 1.9952}{+878.5677 \cdot s^4 + 888.9681 \cdot s^3 + 1599.5636 \cdot s^2 + 121.1943 \cdot s + 1.6175}$$

Pitch Θ To Elevator δ_e Transfer Function:

$$\frac{\Theta(s)}{\delta_e(s)} = \frac{-1062.5244 \cdot s^2 - 468.6144 \cdot s - 31.4031}{+878.5677 \cdot s^4 + 888.9681 \cdot s^3 + 1599.5636 \cdot s^2 + 121.1943 \cdot s + 1.6175}$$

Altitude h To Elevator δ_e Transfer Function:

$$\frac{h(s)}{\delta_e(s)} = \frac{U_1}{s} \left\{ \frac{\Theta(s) - \alpha(s)}{\delta_e(s)} \right\} = \frac{U_1}{s} \left\{ \frac{\Theta(s)}{\delta_e(s)} - \frac{\alpha(s)}{\delta_e(s)} \right\}$$

Roll Φ To Aileron δ_a Transfer Function:

$$\frac{\Phi(s)}{\delta_a(s)} = \frac{+149.4192 \cdot s^3 + 33.6557 \cdot s^2 + 133.0699 \cdot s}{+870.7830 \cdot s^5 + 628.6269 \cdot s^4 + 791.3719 \cdot s^3 + 438.0016 \cdot s^2 + 5.4686 \cdot s}$$

Heading Ψ To Aileron δ_a Transfer Function:

$$\frac{\Psi(s)}{\delta_a(s)} = \frac{+6.5121 \cdot s^3 + 2.1241 \cdot s^2 + 0.1082 \cdot s + 4.7950}{+870.7830 \cdot s^5 + 628.6259 \cdot s^4 + 791.3719 \cdot s^3 + 438.0016 \cdot s^2 + 5.4686 \cdot s}$$

APPENDIX B: DERIVED SCHEMATICS OF ‘REAL’ AFCS’S

Complete Derived Block Diagram Of The Bendix / King KFC-200

During the research for the design of ‘real’ AFCS’s by Saab Aviocomp RIAS, the complete schematics of all the circuits of the Bendix / King KFC-200 were studied, analysed and finally worked out into the block diagram of Fig. B.1.

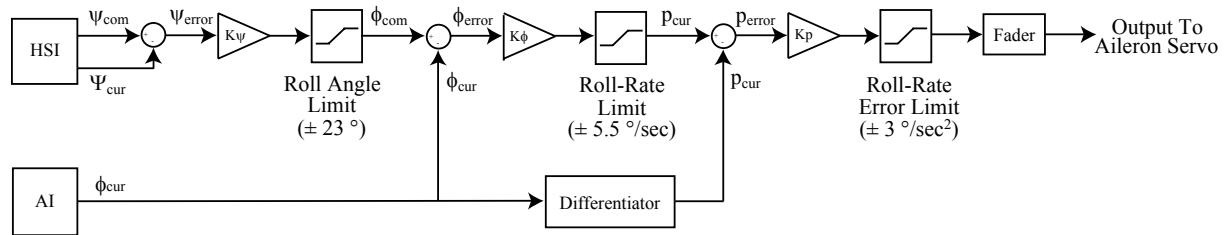


Fig. B.1: Complete Derived Block Diagram Of The Bendix / King KFC-200.

It is seen in this figure that the roll-rate ‘ p_{cur} ’ is obtained through a differentiator. So in this system, no roll-rate input from a roll-rate gyro is used. This differentiator-circuit is also worked out into a schematic in the next section. The system uses a fader at the end to let the ailerons fade in slowly at startup or at large changes. It is of course not desirable that a very fast changing control-signal is sent to the aileron, so that is the reason for this component. Another remarkable element in the diagram is the present of multiple limitations. For safety reasons these limitations are essential, but the values will depend of course on the aircraft and on the type of AFCS.

It can also be seen in the diagram that this system consists of basically three loops:

1. The outermost loop is the heading loop.
2. The middle-loop is the roll angle loop.
3. The innermost-loop is the roll-rate loop.

Our design was based on this diagram, but in addition it was checked if the system could also work satisfactory without the innermost-loop. In our discussion of how a pilot controls an aircraft, the results were also that the pilot uses all the parameters which are also used in this system (reference heading, current heading, current roll angle and current roll-rate).

Derived Roll- And Pitch-Rate Schematics

It was found that the roll- and pitch-rate were used to calculate the control signals for the aileron- and elevator-deflections. Because the AFCS’s did not have an input from a roll- or pitch-rate gyro, it was concluded that the rate’s must have been derived by the system itself. The complete schematics of the systems were analysed and the schematics below were found. Because the roll- and pitch-rates are the derivatives of the roll and pitch respectively, the found schematics will have to be differentiators.

Bendix / King KFC-200 Derived Roll- And Pitch-Rate Schematics

For the Bendix / King KFC-200, the following schematics were found (see Fig. B.2).

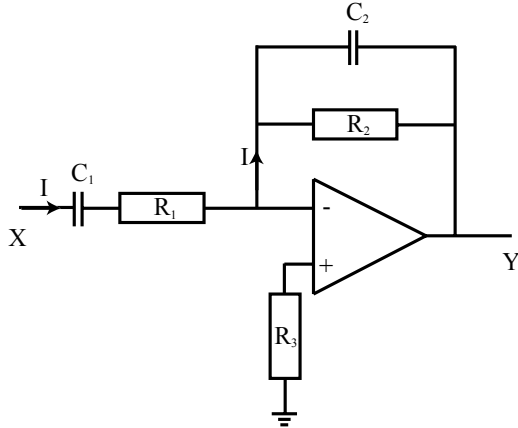


Fig. B.2: Bendix / King KFC-200 Derived Rate Circuit.

The resistor R_3 will have no influence on the transfer function and is chosen so that:

$$R_3 \approx \frac{R_1 R_2}{R_1 + R_2} \text{ (which is the value of } R_1 \text{ and } R_2 \text{ parallel).}$$

The input X is chosen as a variable for the roll- or pitch and the output Y is chosen as a variable for the roll- or pitch-rate.

To derive the transfer function, we look at how the current I travels through the circuit.

We can write for the current I which goes through C_1 and R_1 :

$$I = \frac{X}{\frac{1}{j\omega C_1} + R_1} = \frac{j\omega C_1 X}{1 + j\omega R_1 C_1}$$

And for the current I through R_2 and C_2 :

$$I = \frac{-Y}{\frac{1}{j\omega C_2} + \frac{1}{R_2}} = \frac{-Y}{1 + j\omega R_2 C_2}$$

We state that the input current I does not enter the comparator, but goes entirely through R_2 and C_2 (ideal comparator):

$$\frac{j\omega C_1 X}{1 + j\omega R_1 C_1} = \frac{-Y}{1 + j\omega R_2 C_2} \Rightarrow \frac{Y}{X} = \frac{-j\omega R_2 C_1}{(1 + j\omega R_1 C_1) \cdot (1 + j\omega R_2 C_2)}$$

This transfer function is a 2nd order transfer function with two time constants:

$$\tau_1 = R_1 C_1 \rightarrow \text{This is the low frequency cut-off frequency.}$$

$$\tau_2 = R_2 C_2 \rightarrow \text{This is the high frequency cut-off frequency.}$$

Bendix / King KFC-150 Derived Roll- And Pitch-Rate Schematics

A similar circuit can be found in the Bendix / King KC-192 flight computer schematics which is used in the Bendix / King KFC-150 AFCS (see Fig. B.3):

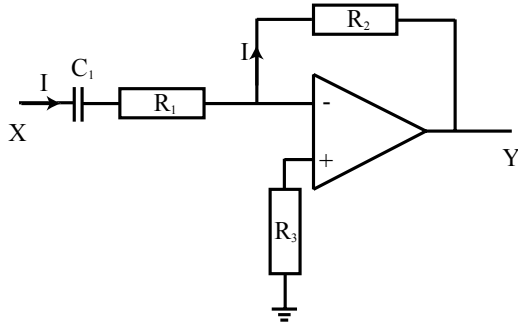


Fig. B.3: Bendix / King KFC-150 Derived Rate Circuit.

Again, the resistor R_3 has no influence on the transfer function and is chosen so that

$$R_3 \approx \frac{R_1 R_2}{R_1 + R_2} \text{ (which is the value of } R_1 \text{ and } R_2 \text{ parallel).}$$

For the input current I which goes through C_1 and R_1 , we can write:

$$I = \frac{X}{\frac{1}{j\omega C_1} + R_1} = \frac{j\omega C_1 X}{1 + j\omega R_1 C_1}$$

For the current I which goes through R_2 , we can write:

$$I = \frac{-Y}{R_2}$$

We state that the input current I does not enter the comparator, but goes entirely through R_2 (ideal comparator):

$$\frac{j\omega C_1 X}{1 + j\omega R_1 C_1} = \frac{-Y}{R_2} \Rightarrow \frac{Y}{X} = \frac{-j\omega R_2 C_1}{1 + j\omega R_1 C_1}$$

This transfer function is a 1st order transfer function and has only a low frequency cut-off and no high frequency cut-off:

$\tau_1 = R_1 C_1 \rightarrow$ This is the low frequency cut-off frequency.

The values used for the components are:

$$R_1 = 49.9K\Omega$$

$$R_2 = 1M\Omega$$

$$R_3 = 51K\Omega$$

$$C_1 = 0.47\mu F$$

This gives a time constant $\tau_1 = R_1 C_1 = 23.5ms$ and a 3-dB frequency of $f_{3dB} = \frac{1}{2\pi\tau_1} = 6.8Hz$.

The gain for high frequencies is $\frac{Y}{X} = \frac{\frac{-R_2}{R_1}}{1 + \frac{1}{j\omega R_1 C_1}} \approx \frac{-R_1}{R_2} = -20$

APPENDIX C: CONSTRAINTS, PERFORMANCE LIMITS & ACCURACIES OF 'REAL' AFCS'S

Bendix / King KFC-200:

ALT HLD / HDG HLD / NAV MODE	LIMITS
Roll Command Limits	$\pm 23^\circ$
Roll-Hold Accuracy (Smooth Air)	$\pm 3^\circ$
Pitch Command Limits	$\pm 20^\circ$
Heading-Hold Accuracy (Smooth Air)	$\pm 2^\circ$
Altitude-Hold Accuracy (Smooth Air)	± 30 ft
ALT HLD / GLIDESLOPE MODE	
Pitch Command Limits	$\pm 25^\circ$

Bendix / King KFC-400:

ALT HLD / HDG HLD / NAV MODE	LIMITS
Pitch Command Limits	$+25^\circ / -15^\circ$
Pitch-Acceleration Command Limits	± 0.05 g
Pitch Hold Accuracy (Smooth Air)	$\pm 0.3^\circ$
Roll Command Limits	$\pm 25^\circ$
Roll-Rate Command Limits	$\pm 3^\circ/\text{sec}$
Roll-Acceleration Command Limits	$\pm 1.6^\circ/\text{sec}$
Roll-Hold Accuracy (Smooth Air)	$\pm 1^\circ$
ALT HLD MODE	LIMITS
V/S Limits	$\pm 2,000$ ft / min
Altitude Accuracy (Smooth Air)	± 20 ft
Pitch-Rate Command Limits	0.15 g
ALT SEL MODE	LIMITS
V/S Limits	$\pm 6,000$ ft / min
APPROACH MODE (CAPTURE)	LIMITS
Roll-Rate Command Limits	$\pm 6^\circ/\text{sec}$
Roll-Acceleration Command Limits	$\pm 4^\circ/\text{sec}$
APPROACH MODE (TRACKING)	LIMITS
Roll Command Limits	$\pm 10^\circ$
GLIDESLOPE MODE	LIMITS
Pitch-Rate Command Limits	$\pm 2^\circ/\text{sec}$ or 0.15 g

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SAMENVATTING (SUMMARY IN DUTCH)

Vertigo Flight Simulation is een in Nederland gevestigd bedrijf dat als doel heeft goedkope vluchtsimulators te ontwerpen, produceren en verkopen voor alle niveaus van de vluchtsimulator opleidingsmarkt. Een van de producten die door Vertigo Flight Simulation wordt ontwikkeld, is de Vertigo P-2 Trainer. De P-2 Trainer is een vaste-basis vluchtsimulator die in staat is om één- en tweemotorige zuigermotoren met een 'Flight & Navigation Procedures Trainer Type II' (FNPT II) goedgekeurd aërodynamisch en motorcertificaat te simuleren.

Een belangrijke element die nog niet beschikbaar is voor de P-2 Trainer, is een 'Automatic Flight Control System' (AFCS) die bestaat uit een AutoPilot (AP) en een 'Flight Director' (FD). Dit systeem is ontwikkeld, wat betekent onderzocht, ontworpen en geïmplementeerd, in dit afstudeerverslag door de schrijver voor zijn Einddoctoraal Afstudeeropdracht om zijn Ingenieursdiploma in Elektrotechniek, specialisatie Avionica, te behalen.

Een 'Automatic Flight Control System' heeft twee hoofdfuncties: de eerste functie is het besturen van het vliegtuig zonder dat de piloot het vliegtuig hoeft te besturen en de tweede functie is het presenteren aan de piloot van suggesties hoe hij of zij het vliegtuig zou moeten besturen om een bepaalde, door de piloot zelf gekozen, stand, koers, hoogte of vluchtplan / vliegroute te volgen.

Gedurende het onderzoek, werd er ontdekt dat 'Automatic Flight Control Systems' fundamenteel allemaal zijn ontworpen en gebaseerd op klassieke regeltechniek theorie: binnen-lussen worden gebruikt om het vliegtuig te besturen (deze lussen worden besturings-lussen genoemd) en buiten-lussen worden gebruikt om het vliegtuig te leiden (deze lussen worden leidings-lussen genoemd). Het probleem dat in eerste instantie werd gevonden in de implementatie van het systeem was het goed instellen van de verschillende binnen- en buiten-lus versterkingen om gewenste vliegtuig-responsies te verkrijgen. Dit probleem werd overwonnen door het probleem te benaderen vanuit het oogpunt van de klassieke regeltechniek theorie.

Omdat de P-2 Trainer is ontwikkeld met flexibiliteit in het achterhoofd, gebruikt het een modulaair ontwerp zodat het mogelijk is om het aërodynamisch vliegtuigmodel en de instrumenten te veranderen om zodoende een nieuw type vliegtuig te simuleren. Door dit ontwerp moet het 'Automatic Flight Control System' ook een flexibel systeem worden. Omdat de methode van simulatie, testen en het instellen van het systeem ontzettend tijdrovend is, zal een methode gepresenteerd worden om het systeem zichzelf te laten aanpassen elke keer als een nieuw aërodynamisch model in de vluchtsimulator wordt geladen.

ACKNOWLEDGEMENTS

It looks like I've finally made it: completing my Master of Science Thesis Project. These last years were the most difficult ones in my life. After finalizing 85% of my academic education in 4½ years, I never imagined how difficult a Thesis Project could be. But I never gave up entirely, although there were quite a few moments when I was very close to 'throw in the towel'. The day I'll graduate will be *the first day of the rest of my life*.

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