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Lateral-Directional Controller Design for an Electric Aircraft

A study on different control concepts and their stability and performance as yaw damper systems

Master's thesis in Systems, Control and Mechatronics

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Abstract

In this project, different controller strategies and methods have been designed and evaluated for a yaw damper to an electric, high-wing regional aircraft. The controller designs tested have ranged from classical proportional and integral controllers to the more modern methods of model-based optimal control. Finally, gain scheduled, and linear parameter varying control has been explored to improve robustness throughout the aircraft's flight envelope. Result analysis shows promising results and suggests that while all controllers can dampen the dutch roll, modern control methods can perform more robustly and with less pilot effort. Flight testing in Heart Aerospace's engineering flight simulator has been carried out using the different control methods to evaluate the aircraft's handling qualities with the Cooper-Harper rating scale. The LPV yaw damper improved aircraft performance with reduced pilot workload, demonstrating real handling quality improvements for the aircraft.

Keywords: control theory, flight controls, aircraft, LPV, yaw damper, flight test, gain scheduling, LQR

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Carl Folkesson
Henric Jorholm Andersson
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List of Acronyms

Below is the list of acronyms that have been used throughout this thesis listed in alphabetical order:

CG	Center of Gravity
HQR	Handling Quality Rating
LPV	Linear Parameter Varying
LQR	Linear Quadratic Regulator
MIMO	Multiple Input Multiple Output
PBH	Popov-Belevitch-Hautus
PI	Proportional-Integral
PID	Proportional-Integral-Derivative
SAS	Stability Augmentation System
SHSS	Steady Heading Sideslip
SISO	Single Input Single Output
SQP	Sequential Quadratic Programming
TAS	True Airspeed

Nomenclature

Below is the nomenclature of indices, sets, parameters, and variables that have been used throughout this thesis.

Variables

α	Angle of attack
β	Sideslip angle
γ	Flight path angle
u_b	Velocity in x-axis in body frame
v_b	Velocity in y-axis in body frame
w_b	Velocity in z-axis in body frame
V	Airspeed
ϕ	Roll angle
θ	Pitch angle
ψ	Yaw angle
p	Angular roll rate
q	Angular pitch rate
r	Angular yaw rate
H	Height above sea level
n_z	Load factor

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1 Introduction

The movement for electrification to achieve sustainability in transportation is rapidly accelerating. Industries are quickly scrambling to develop new solutions to replace fossil-fuel driven vehicles. Heart Aerospace is a start-up company with the ambition to bring electrification to the aviation industry. Heart Aerospace's idea is to develop an electric aeroplane for short-haul, zero-emission regional flights. They are currently working on their first aircraft, called the ES-19. It is a 19 passenger aircraft with four electric propeller motors.

1.1 Background

The development of electrical aircraft has been around for several years. However, due to limitations in viable energy solutions, electrical aircraft has not been able to compete with conventional aircraft since batteries are heavier and have a lower energy density than conventional fuel sources [1]. Battery technologies have significantly improved in recent years due to the pursuit of sustainable fuel sources. Electrification has come a lot further in land-based transportation since vehicles on land are less affected by the increased mass than aircraft since they do not need to lift their weight. Modern batteries have the capability of providing enough energy for shorter flights [2]. To increase the range, the weight of the aircraft needs to be reduced, which can be done in many ways, e.g. by replacing conventional hydraulic systems that are commonly used to control the actuators with lighter wire systems. These modifications yield an aircraft with slightly different characteristics than a conventional one.

This Master's thesis will explore different control algorithms for controlling the lateral-directional stability of an aircraft. This control system is called a yaw damper in aviation, and it controls the rudder deflection of the aircraft. The different control algorithms are then flight-tested and analyzed to see what controller might be most suitable as a yaw damper.

1.2 Objective and Scope

This thesis project aims to design, analyze, and compare a range of control algorithms for a yaw damper system. These control methods include classical proportional-integral control, linear quadratic regulator and linear parameter varying control. The different control methodologies will be evaluated and compared for the application area.

The evaluation of the control methodologies will consist of both quantitative and qualitative metrics. The quantitative evaluation will consist of commonly used metrics in control theory, such as rise time, settling time and overshoots. The qualitative evaluation will consist of pilot-in-the-loop simulations where a flight test plan will be drafted and conducted with a certified test pilot. The flight test will assess metrics such as pilot workload and aircraft handling characteristics for the different control methodologies. Due to human interaction with the final system, both types of evaluations are necessary to capture the full effect of the control system implementations.

1.3 Limitations

Some limitations were made to make the most of the project. First, due to the high complexity and many parameters in the model, some parameters are kept constant. For example, the aircraft's weight and centre of gravity are constant throughout the thesis. The evaluations of the control systems are considered in cruise flight conditions, i.e. no effect of the landing gear or different flap settings will be taken into consideration. All these parameters and others not mentioned need consideration for a complete system evaluation.

The flight testing will be conducted at a single altitude, and velocity since testing on the full range of the aircraft would be very time-consuming. However, a single test point is enough for the relative comparison between control methodologies for the thesis.

1.4 Ethics and Data

During the project, no sensitive individual or personal data is used. The results presented is based from a simulated environment using synthetic data. The methods presented in this thesis can not be used to bias the results in favor of a specific method. Fairness is guaranteed since no advantage can be given to an algorithm to discriminate the results.

The control strategies are designed to aid and not constrain the pilot. The output of the controller is clearly interpreted by the pilot.

To put a control system in an aircraft, extensive testing is needed to ensure the reliability and safety of the system to not cause any harm to the occupants. Although this project is far from realization, it is important to incorporate this mindset from the beginning of the project.

2
Theory

This chapter describes the theory used during the project to provide the reader with the necessary knowledge needed to understand the upcoming chapters.

2.1 Flight dynamics

To create a model of the aircraft, it is essential to understand which forces and moments affect the aircraft, and which inputs are available in the aircraft. There are three primary sources of forces that can influence an aircraft in flight; aerodynamic forces, propulsive forces and gravitational forces. The aerodynamic forces are products of the shape of the aircraft and the surrounding environment and can be divided into two major components; lift and drag. The propulsive forces stem from the propulsion system of the aircraft, and the gravitational forces are based on the mass of the aircraft [3].

2.1.1 Essential aerodynamics

Aerodynamics is the science of airflow. It is vital to understand the aircraft's aerodynamics forces and moments since these are the primary factor in flight dynamics.

The aerodynamic forces and moments on a body moving through the air originate from two primary sources; pressure distribution over the body surface, p, and shear stress distribution over the body surface, τ. p acts normal to the body surface while τ acts tangential to the surface area due to friction between the air and the body surface. p and τ can be integrated over the surface area of the body to obtain a resulting force vector R and moment M, where R is commonly divided into a lift force vector L and drag force vector D [4] for an aircraft body. The total aerodynamic forces and moments subjected to the aircraft have contributions from all airframe parts.

It is possible to model the aerodynamic forces and moments as products of a reference area and dynamic pressure [5]. Dynamic pressure measures how high the pressure of the incoming air is. The freestream is the air stream far ahead and free of airflow disturbances from the aeroplane. The freestream dynamic pressure q_∞ is defined in (2.1), where ρ_∞ is the freestream density and V_∞ is the freestream velocity [4].

$$q_\infty = \frac{1}{2}\rho_\infty V_\infty^2. \tag{2.1}$$

Dimensionless aerodynamic coefficients are defined to model a body's ability to generate force and moments as it moves through the air. These coefficients are obtained by dividing a force by a representative reference area of the body, and a reference length in the case of moment. These coefficients are functions of the angle of attack of the body [5]. The coefficients are based on forces and moments calculated and measured via computational fluid dynamics simulations and wind tunnel tests. These coefficients are then used in the aircraft model to obtain the forces and moments on the aircraft in a given flight condition.

2.1.2 Equations of motion

The equations of motion use the forces and moments that the aircraft is subjected to and translate these into velocity and acceleration.

There are four sets of equations generally referred to as equations of motion; force equations (2.2), kinematic equations (2.3), moment equations (2.4), and navigation equations (2.5). These equations are expressed in body-frame with the earth-frame as an inertial reference frame and the flat-earth assumption. The flat earth assumption introduces some inaccuracies in the determination of position. However, it is widely used to analyze aircraft dynamics and used to derive linear models for further analysis, and design [5].

$$\begin{aligned}
\dot{u}_B &= rv_B - qw_B - g_D \sin\theta + \left(X_A + X_T\right)/\mathrm{m} \\
\dot{v}_B &= -ru_B + pw_B + g_D \sin\phi\cos\theta + \left(Y_A + Y_T\right)/\mathrm{m} \\
\dot{w}_B &= qu_B - pv_B + g_D \cos\phi\cos\theta + \left(Z_A + Z_T\right)/\mathrm{m}
\end{aligned} \tag{2.2}$$

$$\begin{aligned}
\dot{\phi} &= p + \tan\theta(q\sin\phi + r\cos\phi) \\
\dot{\theta} &= q\cos\phi - r\sin\phi \\
\dot{\psi} &= (q\sin\phi + r\cos\phi)/\cos\theta
\end{aligned} \tag{2.3}$$

$$\begin{aligned}
\Gamma\dot{p} =& J_{xz}\left[J_x - J_y + J_z\right]pq - \left[J_z\left(J_z - J_y\right) + J_{xz}^2\right]qr + J_z\ell + J_{xz}n \\
J_y\dot{q} =& \left(J_z - J_x\right)pr - J_{xz}\left(p^2 - r^2\right) + m \\
\Gamma\dot{r} =& \left[\left(J_x - J_y\right)J_x + J_{xz}^2\right]pq - J_{xz}\left[J_x - J_y + J_z\right]qr + J_{xz}\ell + J_x n
\end{aligned} \tag{2.4}$$

$$\begin{aligned}
\dot{x}_E =& u_B \mathrm{c}(\theta)\mathrm{c}(\psi) + v_B(-\mathrm{c}(\phi)\mathrm{s}(\psi) + \mathrm{s}(\phi)\mathrm{s}(\theta)\mathrm{c}(\psi)) + \\
& w_B(\mathrm{s}(\phi)\mathrm{s}(\psi) + \mathrm{c}(\phi)\mathrm{s}(\theta)\mathrm{c}(\psi)) \\
\dot{y}_E =& u_B \mathrm{c}(\theta)\mathrm{s}(\psi) + v_B(\mathrm{c}(\phi)\mathrm{c}(\psi) + \mathrm{s}(\phi)\mathrm{s}(\theta)\mathrm{s}(\psi)) + \\
& w_B(-\mathrm{s}(\phi)\mathrm{c}(\psi) + \mathrm{c}(\phi)\mathrm{s}(\theta)\mathrm{s}(\psi)) \\
\dot{z}_E =& -u_B\ \mathrm{s}(\theta) + v_B\ \mathrm{s}(\phi)\mathrm{c}(\theta) + w_B\mathrm{c}(\phi)\mathrm{c}(\theta)
\end{aligned} \tag{2.5}$$

In equations (2.2), g_D is the down-component of the gravitational force vector and is denoted as such to distinguish that the equations uses the flat earth assumption. $F_A = \begin{bmatrix} X_A & Y_A & Z_A \end{bmatrix}^\top$ and $F_T = \begin{bmatrix} X_T & Y_T & Z_T \end{bmatrix}^\top$ in equation (2.2) are the external aerodynamic- and thrust forces. m in equations (2.2) is the mass of the aircraft. Note that this m is not the same as in equations (2.4). J in equations (2.4) is the inertia matrix of the aircraft and $\Gamma = J_x J_z - J_{xz}^2$ for convenience. $M = \begin{bmatrix} \ell & m & n \end{bmatrix}^\top$ is the external moments applied in their respective axis in the body frame of the aircraft.

The state vector x for the equations of motion in this form is depicted in (2.6) and given a value for x the state derivatives can be evaluated [5].

$$x = \begin{bmatrix} \phi & \theta & \psi & p & q & r & u_b & v_b & w_b & x_E & y_E & z_E \end{bmatrix} \tag{2.6}$$

The full derivation of the equations of motion is outside the scope of this thesis, but can be found in the book by Stevens, Lewis and Johnson [5].

2.1.3 Flight envelope

The flight envelope is a defined range of operating conditions in which the aeroplane is allowed to operate. The operating condition is dependent on several parameters such as airspeed, altitude, and mass. The plane must fly in a controllable and predictable way within the flight envelope. Further in this thesis, only operating conditions within the flight envelope are considered.

2.1.4 Aircraft stability

Aircraft stability is a product of many factors. The placement of the centre of gravity and shape and placement of wings, fuselage, and empennage have significant impacts. There are two kinds of stability concepts related to aircraft, static stability and dynamic stability.

Static stability is related to the instantaneous response of the aircraft when subjected to a disturbance in an equilibrium state, meaning that the forces and moments exerted by the aircraft when disturbed determine the static stability. If a plane shows a tendency to return to equilibrium when disturbed, it is statically stable [6]. Otherwise, it is statically unstable.

Dynamic stability is related to the aircraft's response over time after being subjected to a disturbance. If an aircraft returns to equilibrium after being disturbed, it is dynamically stable. Otherwise, it is dynamically unstable. While static stability covers the instantaneous tendency of the aircraft after being subjected to a disturbance, it does not cover if the aircraft returns to an equilibrium state [6].

From a control systems perspective, the definition of stability is slightly different. A system is said to be input-output stable if the system has a finite gain, i.e. the impulse response is absolutely integrable. This is when (2.7) is fulfilled, where $g(\tau)$ is the impulse response of the system [7].

$$\int_0^\infty |g(\tau)| d\tau < \infty \tag{2.7}$$

2.2 Trimming of non-linear model

In this thesis, trimming an aircraft refers to finding a solution to the equations of motion where the translational and rotational derivatives are either constant or zero in a specific flight condition. [8]. This means that the state derivatives are required to be either zero or constant, depending on the flight scenario that is analyzed [5]. The trimmed condition is either called a trim point or an operating point.

Operating points are essential as they provide the information required for linearizing the non-linear aircraft dynamics [5], which is needed for e.g. the implementation of a range of control methods. Trimming can also indicate whether or not a model is correctly implemented and, e.g. if control surface deflection limits are sufficient for the intended flight envelope since the plane should be able to trim throughout its flight envelop. Perhaps the essential use of operating points is in the stability analysis, where the response of the aircraft around an equilibrium is analyzed after the aircraft is subjected to a force or moment, either as an intended pilot input or from an external disturbance [6].

Trim points are found via numerical optimization tools where algorithms such as sequential quadratic programming (SQP) or Simplex are commonly used [8]. Depending on the flight scenario, specific constraints on the model states need to be fulfilled. For example, straight and level flight requires the roll-, pitch- and yaw rates, along with the yaw and roll angle, to be strictly zero. In appendix A, an example of the trim specifications for level flight is shown.

2.3 Dutch roll

The Dutch roll is a lightly damped, oscillatory motion in yaw, coupled to the roll and sideslip of the aircraft. The Dutch roll motion can arise when the aircraft is disturbed in the yaw or roll axis. Yawing of the aircraft causes the air velocity over the wings to change in an oscillatory manner, which induces a lift and drag difference between the wings. This dynamic results in a coupled oscillation in yaw and roll referred to as Dutch roll [9].

The characteristics of the Dutch roll mode of an aircraft are primarily determined by the vertical fin, where a larger vertical fin is generally better for a more stable Dutch roll mode. However, this contradicts the desired design requirements for the spiral mode. This usually calls for a design compromise where the result often is to have a lightly damped Dutch roll mode in order to have decent characteristics in the spiral mode [9]. Therefore, many aircraft are equipped with an automatic control system to help improve the characteristics of the Dutch roll mode. This control system is called a yaw damper.

2.4 Coordinated turn

A coordinated turn is when the aircraft turns with zero sideslip angle and lateral directional acceleration. A coordinated turn requires precise control of the rudder, aileron and elevator input. The aircraft will slip towards the lower wing if a turn is initiated by purely aileron input. A turn using only rudder input results in the aircraft skidding [3].

2.5 Yaw damper

The yaw damper is a stability augmentation system (SAS). Stability augmentation systems are automatic systems that are present on most modern aircraft in some forms and are meant to enhance the stability and flying characteristics of the aircraft. Common examples of SAS are to counteract undesired motions that can appear naturally in flight due to disturbances, e.g. phugoid motions or Dutch roll motions [5].

The yaw damper is meant to suppress the yaw-rate perturbations that can happen when an aircraft is disturbed from its steady-state mode, e.g. when flying through turbulence. Depending on the aircraft, the system response to such a disturbance can be poorly damped, and the resulting motion, commonly known as Dutch roll, of the aircraft, can cause discomfort to passengers and crew [10].

The most basic formulation of a yaw damper states that the yaw damper uses yaw rate as sensed output and has the authority of the rudder as means to counteract any deviations from the desired mode. In steady, level flight, the desired value for the yaw-rate is zero [10]. More sophisticated definitions of the yaw damper that utilizes modern control techniques, e.g. linear quadratic regulator (LQR), use several states and outputs as inputs to the controller [11]. In steady, level flight, the values for all lateral states will be zero. However, the aircraft will not always be in steady and level flight, which creates a problem with the general formulation of the yaw damper. The yaw damper will always try to regulate the sensed outputs to zero, and thus it will fight the pilot when, e.g. setting up a turn. One solution for this problem is to filter the states fed to the controller with washout filters. The washout filter is essentially a high-pass filter to make sure only frequencies corresponding to non-pilot induced disturbances get regulated [10].

2.6 Control theory

This section will give the theoretical background of the control methods implemented in this thesis. The controllers to be used range from classical single input, single output (SISO) controllers to more modern and sophisticated methods.

2.6.1 Classic control methods

The Proportional-Integral-Derivative (PID) controller is the most widely used control method in the industry. It is a method that is easily understandable and delivers efficient solutions to many applications [12]. Therefore, it is reasonable to use this control method as a baseline to compare against when constructing more modern controller techniques such as LQR control.

A PID control law is defined as in equation (2.8), where K_P is the proportional gain, K_I is the integral gain and K_D is the derivative gain [12].

$$u(t) = K_P e(t) + K_I \int_0^t e(\tau) d\tau + K_D \frac{de(t)}{dt} \tag{2.8}$$

The three degrees of adjustment in a PID controller capture different characteristics of the error signal. The proportional gain compensates for the output by actuating control input proportional to the error. The integral gain is primarily used to reduce low-frequency stationary error. Finally, the derivative term improves high-frequency transient behaviours in errors.

The derivative term is omitted from the classic control methods used in this project since no emphasis has been put on transient response handling. Instead, proportional-integral (PI) control is considered and used as the baseline control method.

2.6.2 Linear quadratic regulator

The classical control methods suffers from a couple of limitations:

- Classic control methods can not handle multiple input, multiple output (MIMO) systems without introducing inner and outer loops.
- Feedback gains require manual tuning, which is not always intuitive or easy to achieve. Adjusting weights on states and inputs is more intuitive to use and understand from an engineering stand point when implementing the controller [5].
- Classic control methods are not optimal. Modern control techniques are optimal to minimize the cost function.

The LQR controller fills these gaps and limitations that the classic control methods leave behind.

A linear time-invariant system can be described using a state-space model (2.9). x is the state vector of the system, u is the input vector, and $\dot{x}$ contains the state derivatives.

$$\begin{aligned} \dot{x} &= Ax + Bu \\ y &= Cx \end{aligned} \tag{2.9}$$

LQR is a full state-feedback control law for which the control (2.10) needs to be decided [13].

$$u = -Kx \tag{2.10}$$

Inserting (2.10) in (2.9) yields the state-space model in equation (2.11).

$$\dot{x} = (A - BK)x \tag{2.11}$$

The difference between LQR and regular state-feedback control is that the feedback gain K is based on the aforementioned cost function (2.12). The matrices $Q \geq 0$ and $R > 0$ are design parameters referred to as weighting matrices. These matrices decide how much cost should be associated with each state and input of the closed-loop system and thus is how the designer can affect the dynamic behaviour of the closed-loop system [5].

$$J = \frac{1}{2}\int_0^\infty (x^\top Qx + u^\top Ru)dt \tag{2.12}$$

The optimal feedback gain K is found as in equation (2.13), where P is the solution to the algebraic Ricatti equation (2.14).

$$K = R^{-1}B^\top P \tag{2.13}$$

$$0 = A_c^\top P + PA_c + Q + PBR^{-1}B^\top P \tag{2.14}$$

2.6.3 Gain scheduling

Up to this point, controller design has only been discussed for a single operating condition. However, the control design must work in the entire flight envelope of the aircraft. For instance, as described in section 2.1.1, the dynamic pressure grows quadratically as a function of the airspeed. Hence, the same amount of low and high-speed control surface deflection will produce vastly different responses.

The valid flight envelope is multi-dimensional, and parameters can have an extensive range of variability. An idea would be to design a controller sufficiently robust to cover the whole flight envelope. The drawback of that approach is that the controller design will be conservative. Another idea that is widely adopted in aviation controller design is the concept of gain scheduling [14], where the controller gains vary with respect to one or several scheduling parameters. The gain scheduling technique captures non-linear dynamics by using several linear controllers and switching between them with the scheduling parameters that have non-linear trajectories.

It is crucial to choose parameters that closely capture the dynamics of the non-linear model. For example, parameters such as velocity and altitude directly impact the dynamic pressure the airframe will be exposed to, affecting the overall dynamics of the airframe. Other parameters that capture both longitudinal and lateral dynamics are also essential to consider to capture the non-linearity of an aeroplane.

However, gain scheduling is an ad hoc method which has some implications. Analysis of controller implementation is complicated since design criteria such as controller performance and robustness are not specified in the design process. Therefore, only performance and robustness can be guaranteed in the exact trim points where the controllers have been designed [15]. Extensive simulations and flight testing is required to tune and establish the stability and safety of the controller design.

2.6.4 Linear parameter varying control

Due to the above mentioned limitations of the gain scheduling methodology, there has been an increased interest of the concept linear parameter-varying (LPV) control [15].

An LPV system can be desribed by the state-space model

$$\begin{aligned} \dot{x}(t) &= A(p(t))x(t) + B(p(t))u(t) \\ y(t) &= C(p(t))x(t) + D(p(t))u(t) \end{aligned} \tag{2.15}$$

where the vector $p(t)$ contains parameters that the matrices A, B, C and D depend on.

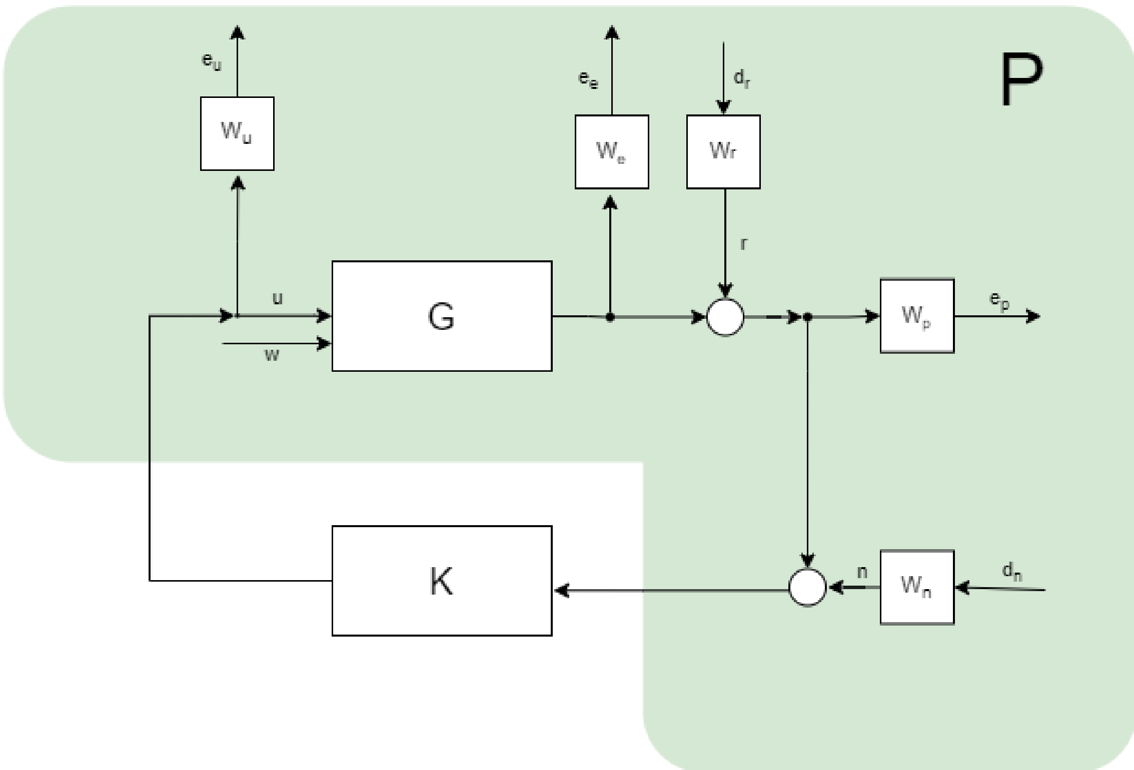

Figure 2.1: The LPV system and controller architecture in a P-K structure.

To design the controller K, it is useful to have a standard problem formulation such as the P-K structure, which any control problem can be manipulated to [16]. The augmented system is shown in figure 2.1. The plant model P can be described as

$$\begin{bmatrix} \dot{x}(t) \\ e(t) \\ y(t) \end{bmatrix} = \begin{bmatrix} A(p(t)) & B_1(p(t)) & B_2(p(t)) \\ C_1(p(t)) & D_{11}(p(t)) & D_{12}(p(t)) \\ C_2(p(t)) & D_{21}(p(t)) & D_{22}(p(t)) \end{bmatrix} \begin{bmatrix} x(t) \\ d(t) \\ u(t) \end{bmatrix}. \tag{2.16}$$

Using the same principle as in LQR controller design, the weight matrices describe how vital and how costly an action or an error is. The weight matrix W_u describes the cost associated with the control input W_p weights the reference tracking property of the controller, and W_e weights the system's output. There are also weighting matrices for inputs, such as the sensor noise W_n. W_r describes the ideal reference system that is wanted for reference tracking [16]. Furthermore, it is also possible to tune W_e and W_p as transfer functions, which provides even more adjustability in tuning since that means the weights are frequency dependent.

A simple but valuable result of the controller analysis is the $\mathcal{L}_2$ gain of the closed-loop system. The $\mathcal{L}_2$ gain describes the input-output stability, meaning that a bounded input to the system would result in a bounded output. Therefore, if the gain of the closed-loop is finite, then the system is input-output stable [7].

Let $S(p)$ denote the LPV system in (2.15), then [17] defines the induced $\mathcal{L}_2$ norm of the system as described in equation (2.17).

$$\|S(\rho)\|_{2\to 2} = \max_{\rho \in \mathcal{P}} \max_{\substack{u \in L_2 \\ \|u\|_2 \neq 0}} \frac{\|S(\rho)u\|_2}{\|u\|_2} \tag{2.17}$$

The LPV toolbox generates a quadratically stabilizing controller over the whole parameter set p while still trying to minimize the $\mathcal{L}_2$ gain [17]. Hence, the closed-loop LPV system is nominally stable. Under certain conditions, nominal and robust performance and robust stability can be claimed. These claims require the norm of the system to be less than 1.

2.7 Cooper-Harper scale

The Cooper-Harper scale is a pilot rating scale used to assess the handling qualities of aircraft [18] and remains the most widespread method for pilot assessment more than 50 years after it was first introduced [19].

Cooper-Harper defines handling qualities of an aircraft as "Handling qualities is defined as those qualities or characteristics of an aircraft that govern the ease and precision with which a pilot can perform the tasks required in support of an aircraft role" [18, p.2]. From the definition, it is clear that assessing the handling qualities of an aircraft is subjective. Despite numerous attempts of trying to capture handling qualities in objective measures, no attempt has ever been successful [19]. The Cooper-Harper scale has thus become the standard rating method for evaluating

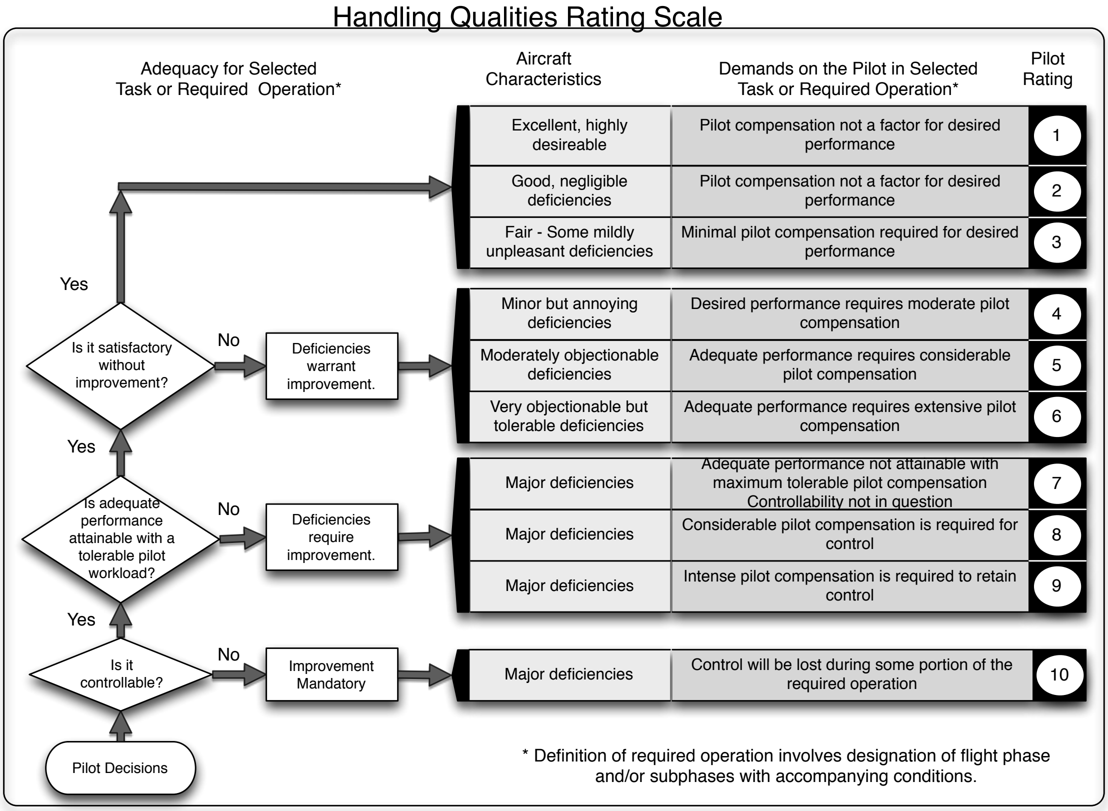

Figure 2.2: "The Cooper-Harper Handling Qualities Rating Scale" by Ccotting is licensed under CC BY-SA 3.0 [20].

aircraft handling qualities. The Cooper-Harper handling qualities rating scale is depicted in Figure 2.2.

The Cooper-Harper rating scale follows a specific procedure. After a finished moment/manoeuvre that is to be rated, the pilot will answer several questions in order. These questions are depicted in the left-most column in Figure 2.2. The questions are answered with a yes or no, which are meant to guide the final rating into subcategories. The best category involves ratings 1-3 and is the desired category, where the aircraft is desirably fulfilling the manoeuvre. The next category involves ratings 4-6 and is the adequate category. In this category, the aircraft performs the task adequately, but improvements can be made. The third category involves ratings 7-9 when the aircraft has major deficiencies which require an improvement. Finally, a rating of 10 indicates that the aircraft is not controllable, and improvements have to be made to make the aircraft flyable. Each of the subcategories then uses a pilot rating of the aircraft characteristics and the pilot workload required to achieve the final handling quality rating (HQR) [18].

A criterion for desired and acceptable performance has to be defined to use the rating scale. The simplest way to do this is in terms of what the result of the manoeuvre will be [18]. For example, suppose the pilot will conduct a turn where the goal is to roll out on a targeted heading. The desired and acceptable ranges could be defined, in degrees, on how much the captured heading is allowed to deviate from the targeted heading. These must be selected to be challenging but reasonable values as they directly affect the final scores.

3
Methods

This chapter covers the methods used to explore the previously stated research question of the project. Methods of modelling and control of the aircraft are discussed in detail to give the reader insight into the design process of the controllers.

3.1 Aircraft modelling

Modelling is a vital part of controller design methodology. Since we are adopting a model-based controller design, the model will directly affect how the controller is going to respond.

3.1.1 Model inputs, states, and outputs

This section gives an overview of the model regarding what inputs, states and outputs are present.

Inputs

The model has six available inputs shown in Table 3.1.

Input	**Min**	**Max**	**Actuator**
RollCmd	-1	1	Ailerons
PitchCmd	-1	1	Elevator
YawCmd	-1	1	Rudder
FlapCmd	0	2	Flaps
ThrottleCmd	0	1	Motors
LdgCmd	0	1	Landing Gear

Table 3.1: Inputs in the model along with their available ranges and which actuator(s) they control.

The roll, pitch, and yaw commands are the sidestick and pedal inputs available to the pilot. These are measured in percentages of their max values, then mapped to the max deflections of their respective control surfaces.

The throttle command is a collective throttle input, i.e. there is no individual control of the four motors.

The flap command has three settings (0,1,2) which map to different flap deflections used for cruise, take-off and landing, respectively. This project only analyses the aircraft in conditions where the flaps are in a cruise setting to reduce complexity.

The landing gear is not yet implemented, so this input is redundant.

States

The underlying dynamics determine the states of the model. Twelve states stem from the equations of motion (2.2)-(2.5), these are presented in Table 3.2. The actuator models introduce two new states per actuator, one position state and one state for the rate. There are four actuators in the model, two for each aileron and one for the rudder and elevator respectively. This means eight new states are introduced by the actuators, which yields a total of 20 states in the aircraft model.

States	**Unit**
Attitude	
ϕ	rad
θ	rad
ψ	rad
Angular rates	
p	rad/s
q	rad/s
r	rad/s
Velocities	
u_B	m/s
v_B	m/s
w_B	m/s
Position	
x_E	m
y_E	m
z_E	m

Table 3.2: List of states that stem from the equations of motion.

Outputs

The outputs are determined by what sensors are present on the aircraft. In this thesis, the modelling of sensors has been omitted, and it is assumed that all state information of the aircraft is available, which makes the choice of outputs completely arbitrary. However, some metrics are combinations of multiple states necessary for analyzing the aircraft, e.g. the angle of attack α or sideslip angle β. The complete list of outputs used in the model can be seen in Table 3.3.

Outputs	**Unit**
α	deg
β	deg
γ	deg
V	m/s
n_z	-
H	m
ϕ	deg
θ	deg
ψ	deg
p	deg/s
q	deg/s
r	deg/s

Table 3.3: List of outputs used in the model with their corresponding units.

3.1.2 Actuators

The dynamics of the actuators are also essential to consider when modelling the plane. The control inputs from the pilot are delayed by the time it takes for the electro-mechanical actuators to move the control surfaces to the correct position. The dynamics of these are essential to consider since this will directly affect the controller design. If the actuating dynamics were omitted, the controller would not be as robust as the controller analysis might suggest since the actuators would delay high-frequency control signals.

The actuators were modelled as second-order transfer functions. For this study, the natural frequency and damping ratio is considered to be to $\omega_n = 31.4$ rad/s and $\zeta = 0.71$ respectively.

$$G_{act}(s) = \frac{\omega_n^2}{s^2 + 2\zeta\omega_n + \omega_n^2} \tag{3.1}$$

Only the actuators of the primary control surfaces, i.e. aileron, elevator and rudder, were considered in the project. That is sufficient since the controller only uses the primary control surfaces to control the aircraft. Otherwise, if the controller would incorporate auto-throttle, the electric motors with the aerodynamic effects of the propellers would also need to be considered in the model.

3.2 Trimming

Trimming of the aircraft is done via MATLAB and the built-in function findop() [21]. Findop() takes the model object and operating point specification, which specifies the constraints on the states of the aircraft as input arguments. The model has four inputs that can influence the trimming procedure; roll-, pitch-, and yaw-command and thrust. In addition, the model has a large number of outputs which can be constrained to specific values depending on the flight condition.

It is possible to run several different optimization algorithms for trimming. After some trial and error with different algorithms, SQP was chosen since it provided the most reliable results.

The level flight condition indicates that the aircraft is flying straight and level relative to the ground, i.e. it is neither descending nor ascending, and the bank- and yaw angles are both zero. Several constraints on the states, inputs, and outputs can be derived from this definition. A complete operating point specification for the straight and level flight condition, for altitude $H = 10000$ ft and airspeed $V = 140$ ktas, can be seen in Table A.1, A.2 and A.3. It is worth noting that not all states, inputs and outputs need to be constrained since they are coupled in the model. The number of constraints is kept to a minimum to avoid potential problems with over-constraining.

3.3 Linearization

After finding the trim point of the aircraft, the model is linearized to obtain a linearized approximation of the non-linear aircraft model in the operating point. Matlab's built-in function linearize() goes block-by-block and finds each block's Jacobian. The overall model linearization is then found by combining all individual block linearizations [22]. However, it is not possible to calculate an analytical linearization for some blocks. Some non-linear blocks lack an analytical Jacobian, and discrete blocks are often linearized to zero, which was both experienced during the project as well as stated in the Matlab documentation [22].

3.4 Yaw damper design

The following sections explore different control concepts of the yaw damper. The classic proportional-integral controller is considered first, followed by the more sophisticated LQR and LPV control methods.

3.4.1 Proportional-integral controller

Although more modern control techniques are available today, classic control design is still the most widely used control method in various applications due to its simplicity. Therefore it makes sense to use the classical methods as a baseline to compare to more modern and advanced methods.

As discussed in Section 2.5, it is necessary to filter the sensed yaw rate with a washout filter before feeding it to the controller to prevent the controller from fighting the pilot. The characteristics of the washout filter are decided by the transfer function of $H_{washout}(s)$ (3.2), which in turn decides how aggressive the yaw damper will be. If τ is small, the yaw damper will be passive, and a large value makes the yaw damper more aggressive and will then fight the pilot in manoeuvres. $\tau \approx 4$ seconds is usually a good value for a balanced yaw damper performance [10].

$$H_{washout}(s) = \frac{\tau s}{\tau s + 1} \tag{3.2}$$

One of the downsides of using a controller relying only on proportional gains is that they are prone to yield stationary errors relative to the desired reference signals. One remedy to this problem is to introduce integral action to the controller.

The non-linear dynamics is linearized at a steady-state flight condition with the altitude of $H = 10000$ ft and a true airspeed (TAS) of $V = 140$ ktas. The linearized dynamics of the aircraft is a MIMO system. However, the PI-controller only uses the yaw rate as the sensed input and only needs the authority of the rudder, which means we can extract the SISO-system relating the rudder input and yaw rate output to use for the controller design. This SISO-system is denoted G^r_{rud}.

Figure 3.1 shows the impulse response of the closed-loop system for a range of values for K_P and K_I. These values are obtained using the built-in function pidtune() in MATLAB and changing the target crossover frequency between 4 and 12 rad/s, which are purely empirical numbers and the result of trial and error.

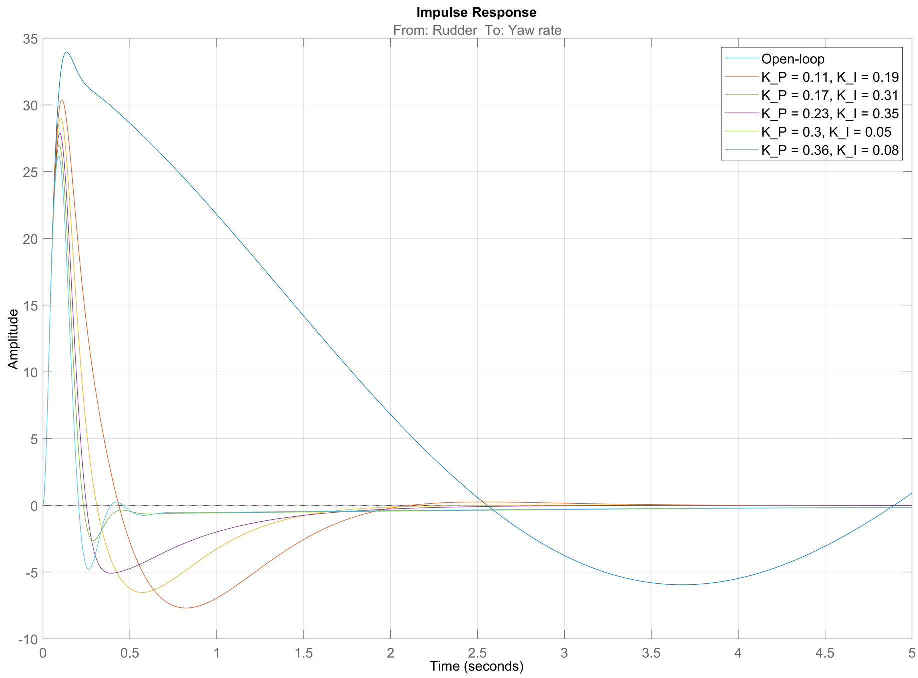

Figure 3.1: Open-loop and closed-loop impulse response from rudder to yaw rate, G^r_{rud}, for a range of values for feedback gains K_P and K_I.

The controller with $K_P = 0.3$ and $K_I = 0.05$ seems like a reasonable choice for a balance between response time and minimal overshoot. The next step is to verify the controller in the non-linear model.

3.4.2 Linear quadratic regulator

In contrast to the classic control methods, the LQR method is no longer constrained to be applied only on SISO systems. Full state-feedback is now considered on the whole MIMO plant. This entails some further analysis on the plant model.

3.4.2.1 Observability and Controllability

The controllability and observability of a system describe how states in the state space are affected by inputs and how they are measured from the outputs. Glad and Ljung defines controllability and observability as:

Definition 1 *The state x^* is said to be controllable if there is an input that in finite time gives the state x^* from the initial state $x(0) = 0$. The system is said to be controllable if all states are controllable.*

Definition 2 *The state $x^* \neq 0$ is said to be unobservable if, when $u(t) = 0$, $t \geq 0$ and $x(0) = x^*$, the output is $y(t) \equiv 0, t \geq 0$. The system is said to be observable if it lacks unobservable states*

[7, p. 45]. Hence, as implied by their names, if a system is not controllable, the regulator will not be able to control all states to their desired setpoint. Furthermore, if a system is not observable, it is impossible to deduce the state values from the measured outputs.

There are a number of methods to check for controllability and observability. Arguably, the easiest method is to use Kalman's test for controllability and observability by checking the null space of the controllability and the observability matrix.

Theorem 1 *A system is observable if $\mathcal{O}$ has full rank and the system is controllable if $\mathcal{C}$ has full rank [7],*

$$\mathcal{O}(A, C) = \begin{bmatrix} C \\ CA \\ \vdots \\ CA^{n-1} \end{bmatrix}, \tag{3.3}$$

$$\mathcal{C}(A, B) = \begin{bmatrix} B & AB & \dots & A^{n-1}B \end{bmatrix}. \tag{3.4}$$

However, a significant drawback of the Kalman tests is that the observability and controllability matrices in (3.3) and (3.4) becomes highly ill-conditioned when applied on large systems with many states. Including actuator dynamics, the plant model of the aircraft has n = 18 states. A more suitable method for high-dimensional systems is the PBH (Popov-Belevitch-Hautus) test. The PBH test does not suffer from numerical issues with regards to the number of states.

Theorem 2 *A system is controllable if and only if $\begin{bmatrix} A - \lambda I & B \end{bmatrix}$ has full rank $\forall \lambda$. It is observable if and only if $\begin{bmatrix} A - \lambda I \\ C \end{bmatrix}$ has full rank $\forall \lambda$ [7].*

3.4.2.2 Kalman decomposition

Controllability and observability are criteria for designing LQR controllers. Since a yaw damper only has authority over the rudder, the system loses controllability over longitudinal states since it is challenging to control pitch and altitude using the rudder. Thus, the system needs to be transformed into a controllable and observable system. The Matlab function minreal() returns the minimal realisation of a system and the Kalman decomposition transformation matrix T. The system can then be transformed as follows,

$$\bar{x} = Tx, \tag{3.5}$$

$$\bar{u} = Tu, \tag{3.6}$$

$$\bar{A} = TAT^{-1}, \tag{3.7}$$

$$\bar{B} = TB, \tag{3.8}$$

$$\bar{C} = CT^{-1}, \tag{3.9}$$

$$\bar{D} = D. \tag{3.10}$$

The reduced system can therefore be described as

$$\begin{aligned} \dot{\bar{x}} &= \bar{A}\bar{x} + \bar{B}\bar{u} \\ y &= \bar{C}\bar{x}. \end{aligned} \tag{3.11}$$

This reduced system is a controllable and observable system approximation of the original system. Hence, it is possible to design LQR controllers for the system 3.11. The robustness of the LQR controller will compensate for the precision lost during the approximation and transformation to the reduced system.

3.4.2.3 Weight matrices

Instead of tuning the feedback gains as previously described in the classic control methods, LQR minimizes the cost function (2.12). The weighting matrices Q and R need to be chosen. Since we use a Kalman decomposition on the system, (3.5) - (3.11), the weight selection of Q has no apparent meaning. However, we can choose the weight matrix Q in the original domain of the system and then obtain $\bar{Q}$ as described in equation (3.12).

$$\bar{Q} = TQT^{-1} \tag{3.12}$$

$R = 1$ is chosen as a starting point. Q is initially selected to have weights on the yaw rate r and the lateral velocity v_B since these are the two most crucial states for a yaw damper. The yaw rate is evident due to the involvement in the Dutch roll mode. The lateral velocity v_B is also penalized since it is comparable to penalizing the sideslip angle β, which is not a state and therefore not accessible. The argument for penalizing v_B is to make the controller regulate the sideslip to zero and help turn coordination. Figure 3.2 shows the impulse response of the LQR-controller with two different tunings.

The response looks reasonable with a quick response time for both tunings. The initial tuning is $R = 1$ and $Q_r = Q_{v_B} = 1$ and the rest of the elements in Q kept at zero. The final tuning was achieved after testing in the flight simulator, where the values $Q_r = 2$ and $Q_{v_B} = 0.5$ were obtained.

When implementing the LQR in the non-linear model, the washout filters are introduced on the four lateral states roll angle ψ, roll rate p, yaw rate r and the lateral velocity v_B to prevent the controller fighting the pilot.

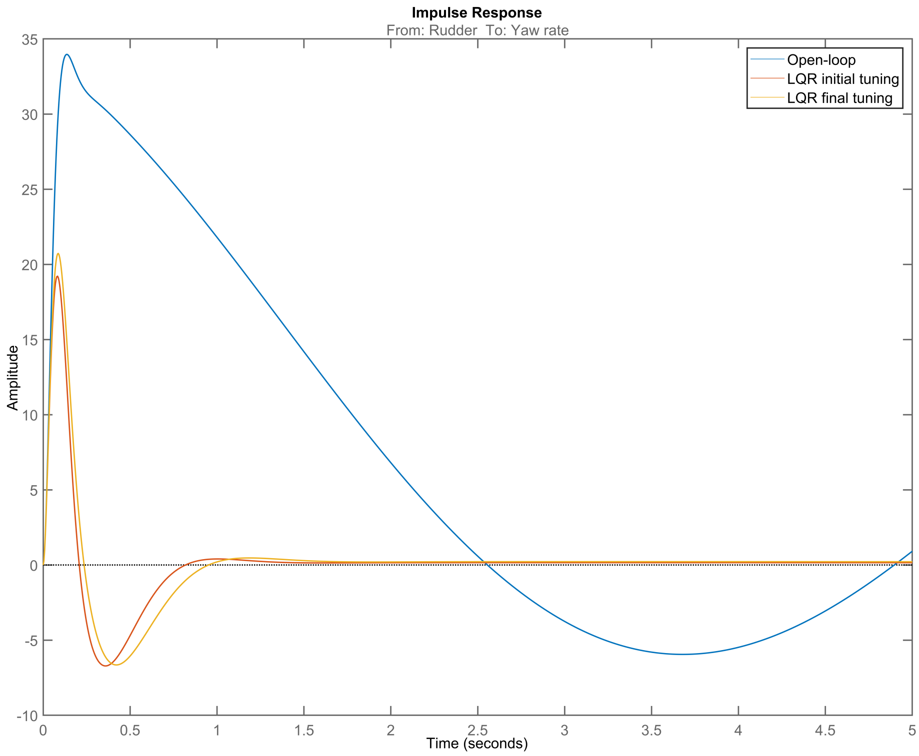

Figure 3.2: Impulse response of the LQR system vs. the open-loop response,

3.4.3 Gain scheduling

The parameters used for scheduling the feedback gains could be either outputs or states. The parameters should be chosen to capture the system's non-linearities to have a good approximation of the system dynamics. Five parameters were selected for scheduling; TAS (V), altitude (H), sideslip angle (β), load factor (n_z) and flight path angle (γ). These choices of scheduling parameters stem from the desire to capture the full range of the flight dynamics in the intended flight envelope. The operating ranges for the scheduling parameters are presented in table 3.4.

Parameter	Min	Max	No. points
H [m]	100	6000	2
V [ktas]	133	190	6
β [deg]	-10	10	3
n_z	0.75	1.5	3
γ [deg]	-5	3	3

Table 3.4: Parameter grid

3.4.4 Linear parameter varying control

In this part, the process of designing a LPV system is explained. The LPVtools toolbox for Matlab is utilised to model, design, and analyse the controller. For further reference about LPVtools, see [17].

3.4.4.1 Linear parameter varying modelling

Similarly, to gain scheduling, parameters need to be selected to capture the non-linear dynamics of the model. By simulating both the LPV and non-linear model for the same disturbances, it is possible to understand how accurately the LPV model describes the non-linearities. The same trim points and parameter selection are used for LPV modelling as with gain scheduling, i.e. the trim points from table 3.4 apply to the LPV model.

Linear interpolation is used for the LPV model between trim points to obtain the interpolated linear system. No extrapolation was used since a linear extrapolation method leads to unstable simulations due to inaccuracies. Therefore, the extrapolation is clipped to the closest trim point in the grid, which is known to be stable for all points outside the defined grid.

As Figure 3.3 demonstrates, the LPV model can resemble the dynamics of the non-linear model closely for the lateral states, which are the most important for the yaw damper implementation.

3.4.4.2 Controller synthesis

During controller synthesis, it must be decided which outputs should be regulated by the LPV controller. Since the plane still should be able to turn, the yaw rate is put as a reference tracking output. As a performance output, the sideslip angle is used. The sideslip angle should be regulated to zero unless a steady sideslip manoeuvre is desired. However, since this is a weighted and not filtered output, the plane can still perform sideslip using this formulation. The weights in equations (3.13)-(3.17) were found to obtain a satisfactory robust controller.

$$W_u = 10, \tag{3.13}$$

$$W_e = 1, \tag{3.14}$$

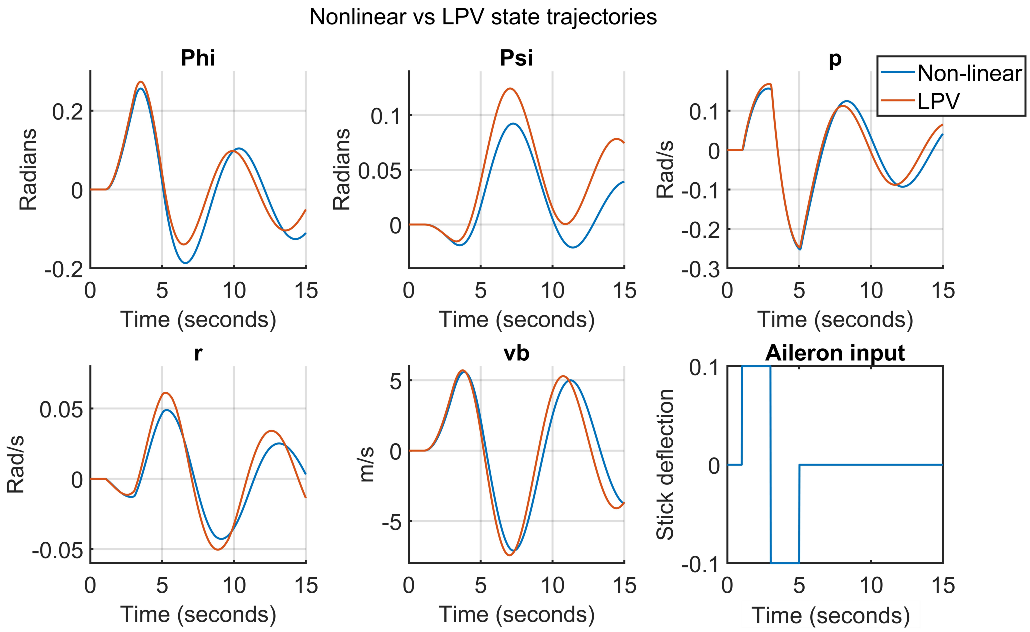

Figure 3.3: Comparison between non-linear model and LPV model response to aileron doublet input.

$$W_p = 10, \tag{3.15}$$

$$W_r = \frac{4}{s+4} \tag{3.16}$$

$$W_n = 0.05 I_2. \tag{3.17}$$

3.4.4.3 Controller analysis

The resulting controller from the weights (3.13)-(3.17) has a $\mathcal{L}_2$ norm of 1.2039. If the weight on performance W_p and weight on reference tracking W_e is lowered, the gain of the system is less than one. Hence the controller would be theoretically robust. However, this assumes that the output disturbance weight matrix W_n accurately describes the disturbances on the sideslip angle and yaw rate in flight. By changing the weights, the $\mathcal{L}_2$ norm will be affected. More time needs to be spent on tuning the weights which might result in reduced induced gain of the system and nominal performance. However, nominal performance comes at the cost of a slower time domain response.

In figure 3.4, the gain of the open-loop and closed-loop for each grid point is compared. For the open-loop system, there are some peaks in magnitude for both the sideslip angle (beta) and the yaw rate at approximately 1 rad/s. This is the resonance frequency for which the rudder has the most effect on the system outputs. Since different grid points have different external conditions affecting the plane, the resonance frequency is shifted slightly between different grid points. These peaks of the gain and the low frequency gain are suppressed for the closed-loop. The open-loop and closed-loop gains converge with heavily damped gains at higher frequencies.

This is reasonable since the high-frequency rudder movement is limited by the actuator's rate limit and cannot disturb the aircraft's inertia. In Figure 3.5 the impulse

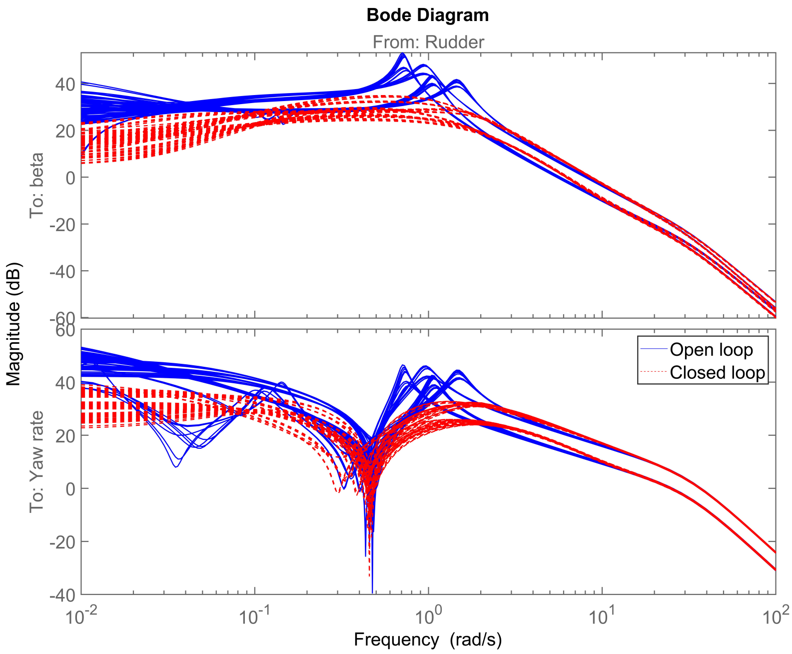

Figure 3.4: Difference between open-loop and closed-loop system gain for each grid point in the LPV model.

response for the open- and closed-loop systems are shown. The closed-loop system can settle the sideslip and the yaw rate significantly earlier than the open-loop system for all parameter combinations. The closed-loop system overshoots twice before settling down. Implementing a more aggressive controller tuning could decrease the overshoot and settling time. However, the controller produced satisfactory results. Since the intended aircraft for this yaw damper is a passenger aircraft and not a fighter jet, some performance can be traded for comfort.

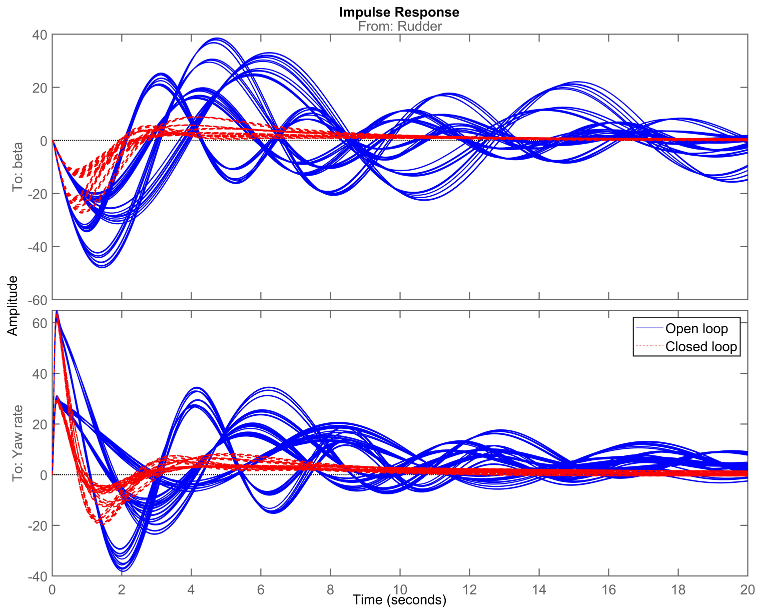

Figure 3.5: Open- and closed-loop impulse response for each grid point in the LPV model.

3.5 Evaluation

One of the primary purposes of a yaw damper system is to ease the potential discomfort for crew and passengers when the aircraft enters a Dutch roll motion. However, developing a control system based on only classical, objective measures can be misleading. A control system that performs excellently on paper may not be excellent in flying comfort and flying qualities. Therefore, it is valuable to weigh subjective opinions on the performance of the resulting closed-loop system, which is done via the widely used Cooper-Harper handling qualities rating scale. The final evaluation of the controller performance will consist of both objective metrics and subjective evaluation.

3.5.1 Dutch roll suppression

The Dutch roll is commonly induced by performing a rudder doublet input. A rudder doublet is when the pilot inputs rudder to one side and then shifts the rudder to the other side with the same amplitude. The input signal of a rudder doublet is depicted in Figure 3.6.

Since the different controllers use slightly different strategies to suppress the Dutch roll motion, it is not easy to quantify precisely how to measure a successful suppression for comparison. One way is to examine the yaw rate since there can be no Dutch roll without it. Although other metrics influence, e.g. the LQR controller regulates more than just the yaw rate. All the lateral states need to be analyzed for

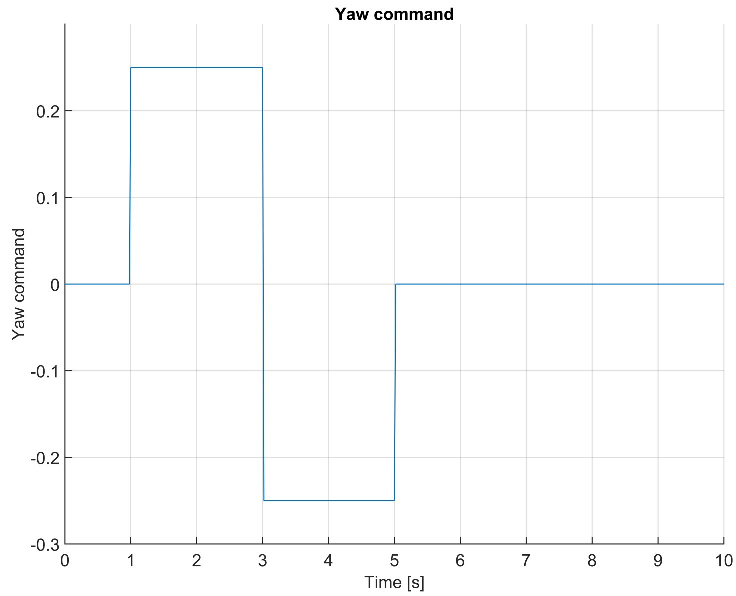

Figure 3.6: Rudder doublet input signal.

a complete evaluation due to the inherent coupling in the aircraft dynamics.

3.5.2 Flight testing in simulator

Although flight testing results are not as precise as simulated fixed responses, they provide more information about the plane's flying characteristics. After all, the aim is to minimize the pilot workload and make the plane fly as safe as possible.

A test program has been set up to test the lateral-directional stability without and with different variations of the yaw damper. This test program can be seen in appendix B.

The weight and center of gravity (CG) of the aircraft were chosen to be light and aft during the flight testing. During testing, it is desired to test the worst-case scenario. The CG has a significant impact on the stability of the plane, and aft CG should provide the least stability to the aeroplane during testing.

The flight testing consisted of three main tasks. The first task was to conduct an aileron only-turn and then roll out on a target heading and capture that heading as precisely as possible. The second task was to establish the plane on a steady heading sideslip with a 15-degree bank angle. Once established, the third task was to get the wings level and capture the heading. These manoeuvres require a pilot's constant feedback and would be difficult to replicate without a pilot-in-the-loop simulation.

The task of rolling out on a heading from a turn is designed to test the ability of the yaw damper to suppress nose sway, i.e. the adverse yaw effect when the pilot is levelling out without any rudder input. It also gives feedback on how much the yaw damper tends to fight the pilot when setting up a turn and shows its ability to maintain a coordinated turn without any rudder input from the pilot. The desired and acceptable ranges for the heading capture are selected as ± 2 and ± 5 degrees, respectively. A narrow but reasonable range was chosen to allow the test to distinguish between better and worse implementations.

Establishing a steady heading sideslip indicates how well the yaw damper cooperates with the pilot when a sideslip is desired. The time it takes the pilot to establish the steady heading sideslip and the effort required to maintain the steady heading sideslip are good metrics for evaluation. Therefore, ten seconds is the desired time to establish a steady heading sideslip in the Cooper-Harper rating scale, while 15 seconds is the acceptable limit.

Capturing a heading from a steady heading sideslip gives insight on roughly the same things as the first task of rolling out on a heading from a turn, but under slightly different premises, e.g. how well the yaw damper compensates for the adverse yaw. The desired and acceptable ranges are the same as in the first task, ± 2 and ± 5 degrees, respectively.

Flight testing will also be conducted with wind turbulence to test the robustness of the controllers. However, only the best controller will be evaluated with wind turbulence due to time constraints. The model used for the wind turbulence is the Dryden wind turbulence model. The parameters used in the model can be seen in the flight test plan in Appendix B.

4

Results

The results of the different yaw damper designs will now be presented. First, the results from the rudder doublet will be presented, followed by the outcome of the flight tests. The most relevant aspects of designing a yaw damper have been tested and analyzed with these results.

4.1 Rudder doublet

In figure 3.6, the input rudder doublet used to excite the Dutch roll is demonstrated. The period of the rudder movement was chosen to be close to the natural frequency of the plane, effectively maximizing the aircraft's response.

The open-loop response for the rudder doublet input is shown in figure 4.1. A large Dutch roll oscillation is excited and lightly damped by the airframe's inherited dynamic stability. There are nine periods of the Dutch roll oscillations until the amplitude of the yaw rate is essentially damped out at around 65 seconds, marking the baseline result.

The PI controller effectively reduces the initial yaw rate of the Dutch roll. As a result, the amplitude is less than 10% of the open-loop response, and the induced yaw rate can be considered to be suppressed after about 10 seconds. However, this is also a problem since it is a sign that the controller will fight the pilot's input. Furthermore, it is evident that although the roll and yaw rates are low, the controller cannot regulate the sideslip closer to zero.

In general, the LQR response is similar to that of the PI controller—the induced yaw rate peaks at about 20 % of the open-loop peak yaw rate. The yaw rate of the LQR controller is higher than the peak value of the PI controller. The yaw rate can be considered to be suppressed at about 10 seconds, which is similar to the PI-controller. However, the vital difference compared to the PI is that the LQR succeeds in reducing the sideslip angle. As the sideslip converges to zero, the yaw rate can also settle down, which entails an output that is more consistent compared to the PI controller.

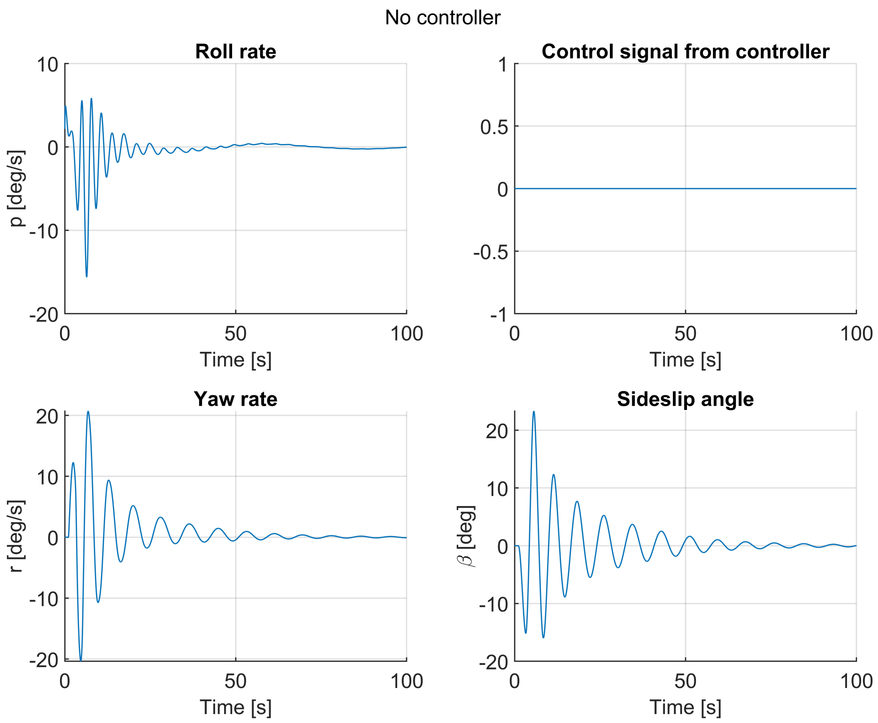

Figure 4.1: Rudder doublet response with no yaw damper.

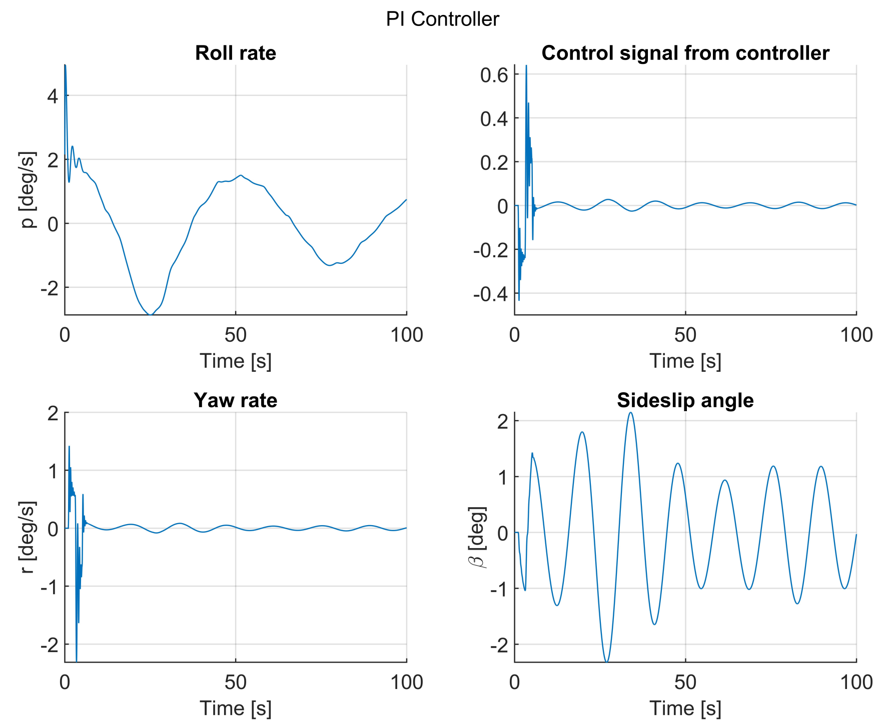

Figure 4.2: Rudder doublet response with PI yaw damper.

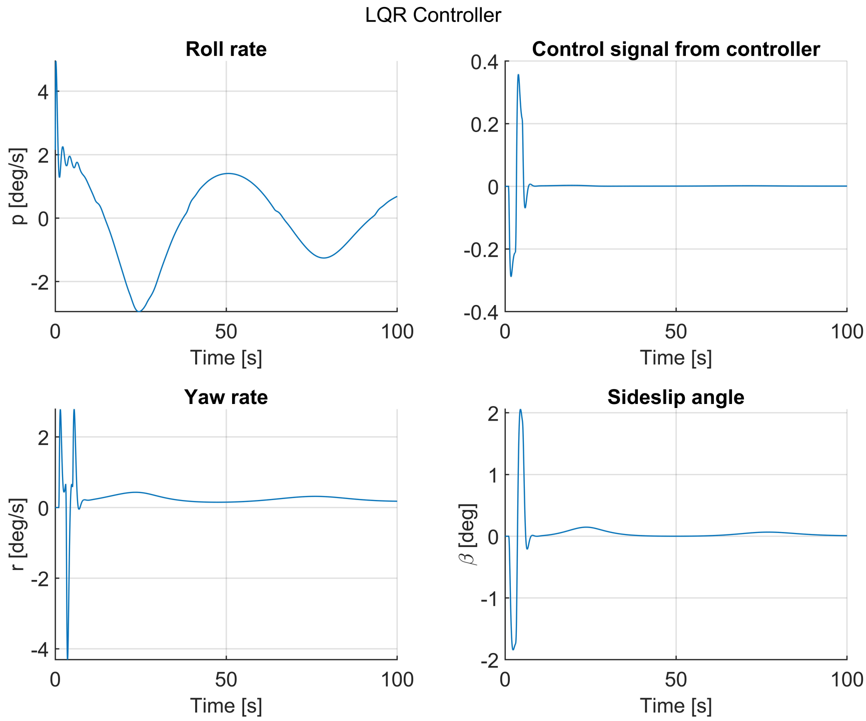

Figure 4.3: Rudder doublet response with LQR yaw damper.

The gain scheduled PI controller in figure 4.4 suffers from the same general problem as the PI regulator in figure 4.2. However, the gain scheduling is a bit more conservative tuned, which causes the sideslip to be higher in the beginning but helps reduce the sideslip during the end of the simulation. As a result, the induced yaw rate peaks at about the same value as the regular PI-controller, but the yaw rate can be considered suppressed after about 50 seconds. The same level of suppression was achieved at about 10 seconds in the regular PI-controller.

The LPV controller in Figure 4.5 has a peak yaw rate of about 25 % of the open-loop value, which is higher than both the PI-controllers and the LQR and manages to suppress the induced yaw rate and sideslip oscillations in about 10 seconds. Although the LPV controller allows for an initially higher yaw rate than other regulators presented, it has a robust response. There are no extra oscillations in yaw rate or sideslip. Note that the x-axis is shortened in 4.5 compared to the other controllers.

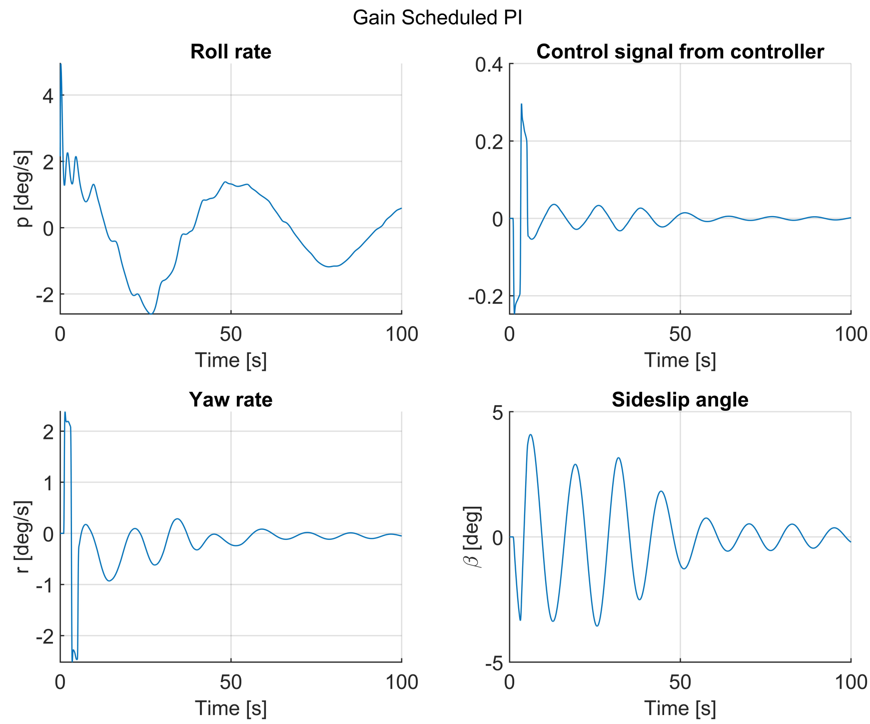

Figure 4.4: Rudder doublet response with gain scheduled PI yaw damper.

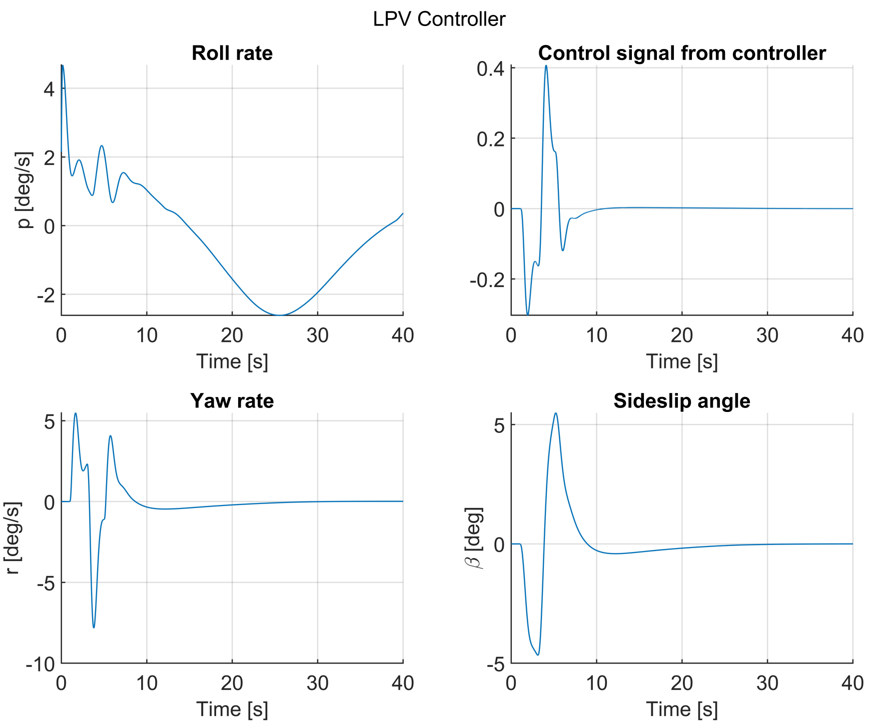

Figure 4.5: Rudder doublet response with LPV yaw damper.

4.2 Flight testing

Below, the flight testing results are presented. For reference to the Cooper-Harper handling quality rating scale, see figure 2.2.

No controller	**Desired**	**Acceptable**	**Obtained**	**HQR**
Roll out on heading	±2	±5	-4deg	6
SHSS - 15 deg bank, left rudder	10s	15s	15s	5
SHSS to level flight, heading capture	±2	±5	6deg	7
SHSS - 15 deg bank, right rudder	10s	15s	13s	5
SHSS to level flight, heading capture	±2	±5	4deg	7

Table 4.1: Handling quality rating for the open-loop response of aircraft.

The handling qualities for the open-loop response are shown in table 4.1. The test pilot stated that it was difficult to steer the aircraft to a specific heading without swaying the nose around. Therefore, the task to roll out on a heading received a rating of 6 since the pilot deemed the performance adequate, but it still required extensive pilot compensation. Due to the pilot workload needed to control the aircraft within the acceptable range, the heading capture task from steady heading sideslip (SHSS) received a rating of 7. However, adequate performance was reached in the steady heading sideslip and ranked with a 5 since a considerable pilot workload was needed.

Gain sheduled PI	**Desired**	**Acceptable**	**Obtained**	**HQR**
Roll out on heading	±2	±5	4deg	8
SHSS - 15 deg bank, left rudder	10s	15s	9s	4
SHSS to level flight, heading capture	±2	±5	-6deg	7
SHSS - 15 deg bank, right rudder	10s	15s	17s	6
SHSS to level flight, heading capture	±2	±5	-2deg	3

Table 4.2: Handling quality rating for gain scheduled PI controller.

The gain scheduled PI controller received poor handling quality ratings due to its strong tendency to fight the pilot's input. Although an acceptable heading capture was obtained on the rollout, the plane fought against the pilot's desired bank angle, forcing the pilot to adjust roll input constantly. That is an unwanted behaviour of the yaw damper, which causes controllability issues, resulting in a higher rating.
The LQR controller demonstrated good turn coordination, and the desired heading angle was achieved due to low nose sway when levelling out on the heading. The time to the establishment of a steady heading sideslip varied a bit from the left and right rudder sideslip. However, the pilot workload was, in both cases, low. A major deficiency was the heading capture from the sideslipping manoeuvre, in which residual yaw from the controller after the manoeuvre pushed the nose far away from the desired heading and forced the test pilot to compensate heavily with roll input. In figure 4.6, the pilot input, controller output, yaw angle and yaw rate can be seen for the moment where the residual yaw effect occurred after recovering from a steady

LQR	Desired	Acceptable	Obtained	HQR
Roll out on heading	±2	±5	-2deg	2
SHSS - 15 deg bank, left rudder	10s	15s	9s	3
SHSS to level flight, heading capture	±2	±5	20s	8
SHSS - 15 deg bank, right rudder	10s	15s	14s	5
SHSS to level flight, heading capture	±2	±5	35s	9

Table 4.3: Handling quality rating for LQR controller.

heading sideslip. From the top right subfigure, showing the pilot's roll input, it is clear that the pilot needs to fight the controller with large roll input pulses. The stick did not even move past the neutral mark to roll out the plane.

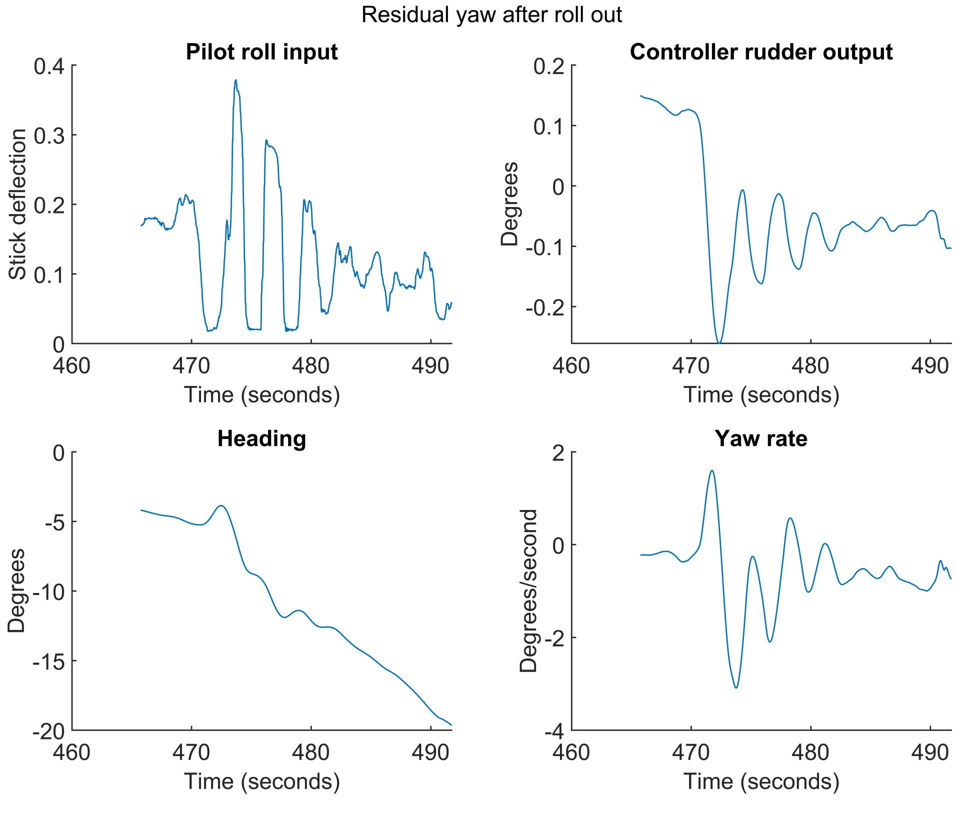

Figure 4.6: Demonstration of the residual yaw effect of the LQR controller.

For the LPV yaw damper, all tasks except the heading capture from the right rudder sideslip produced results that passed the satisfactory without improvement question in the Cooper-Harper scale as seen in Table 4.4. Although the obtained heading angle was 3 degrees of the target, the pilot workload was still minor. Comparing this to the results of the baseline open-loop result in Table 4.1, the LPV controller has made a significant improvement in handling qualities.

LPV	Desired	Acceptable	Obtained	HQR
Roll out on heading	±2	±5	0deg	1
SHSS - 15 deg bank, left rudder	10s	15s	8s	3
SHSS to level flight, heading capture	±2	±5	-1deg	2
SHSS - 15 deg bank, right rudder	10s	15s	9s	3
SHSS to level flight, heading capture	±2	±5	3deg	4

Table 4.4: Handling quality rating for LPV controller.

LPV - with turbulence	Desired	Acceptable	Obtained	HQR
Roll out on heading	±2	±5	2deg	4
SHSS - 15 deg bank, left rudder	10s	15s	12s	5
SHSS to level flight, heading capture	±2	±5	3deg	5
SHSS - 15 deg bank, right rudder	10s	15s	12s	5
SHSS to level flight, heading capture	±2	±5	2deg	4

Table 4.5: Handling quality rating for LPV controller with wind turbulence.

Comparing the data from the LPV controller with turbulence in table 4.5 with the LPV controller without turbulence in table 4.4, it is no surprise that inflation of the handling quality rating has occurred. Interestingly, the test pilot workload was less when flying in turbulence with the LPV yaw damper compared to flying with no yaw damper in completely calm air.

Furthermore, comparing the heading capture from the right rudder sideslip between LPV with and without turbulence, it can be seen that a lower heading error was obtained with turbulence, while also being in the desired range. Due to the turbulence, the pilot workload was elevated. However, this suggests that the heading capture from LPV without turbulence should typically be in the desired range.

In figure 4.7, the difference in heading capture can be seen with and without the yaw damper activated. Several interesting facts can be said from figure 4.7. From the heading graph, it is possible to see the adverse yaw effect in the blue line representing the no yaw damper heading trajectory. As confirmed in the bottom right subfigure, the LPV yaw damper has successfully removed the adverse yaw effect and has a much smoother yaw rate. The yaw rate without the yaw damper overshoots each time there is a change in roll input, i.e. when entering the turn, once established on the bank when levelling off and finally when establishing in level flight. With the yaw damper on, small overshoots are still possible, but the amplitude is significantly lower.

Moreover, the bank angle of the plane is noticeably more steady, both during the turn. Comparing the pilot's input during the turn, it is clear that the pilot has to work less to get the aircraft's desired response. The red curve is very flat and stable compared to the blue curve, where the pilot needs to compensate with much aileron to maintain the bank angle. Additionally, the pilot needs to compensate with less roll input to level off the turn, reducing the workload and allowing for more precise manoeuvring of the plane.

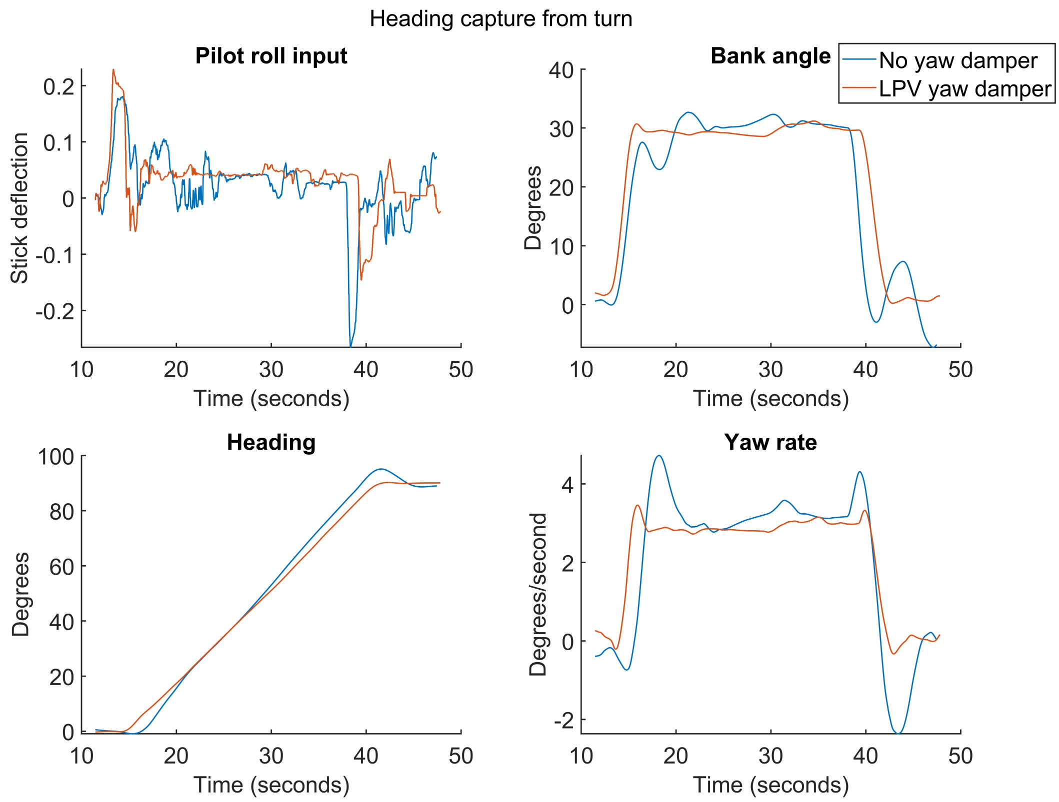

Figure 4.7: A comparison of a heading capture after turn between yaw damper turned off and the LPV yaw damper.

Figure 4.8 shows the sideslip angle, bank angle, yaw rate, and controller output for the aileron-only turn for different controllers during the flight test. It can be seen that the sideslip angle for the LQR and LPV-controller are relatively close to zero during the turn, with a value of about 2 and 4 degrees, respectively. The open-loop response has about the same value, although the value is a lot less steady. The gain scheduled PI controller has a higher value which peaks around 15 degrees. This is not surprising since it does not actively regulate the sideslip angle.

The gain scheduled PI controller is a clear outlier between the different configurations in Figure 4.8. It takes 50 % longer to execute a 90-degree turn because the controller is fighting the pilot. This is because the PI controller attempts to regulate the yaw rate to zero, meaning that this controller will always suppress a turn. However, as mentioned earlier, a washout filter is used to mitigate this effect. Although, it is evident from figure 4.8 that the washout filter needs tweaking.

The LPV-controller has a non-zero sideslip and does not manage to achieve perfect turn coordination due to a too conservative weighting of the sideslip regulation.

The weight associated with the sideslip angle is the performance weighting W_p. If the performance weighting is increased, a better performance would most likely be achieved.

The LQR manages to achieve the lowest sideslip angle during the turn, which is also reflected in the HQR rating 4.3 where the task to roll out on a heading achieved an HQR rating of 2. The big difference between the open-loop, LQR, and LPV responses is that the two regulated responses are much smoother. At the same time, it also requires less pilot effort. This can be seen in the pilot input subplot in Figure 4.8 and the HQR ratings in Tables 4.1, 4.3, and 4.4.

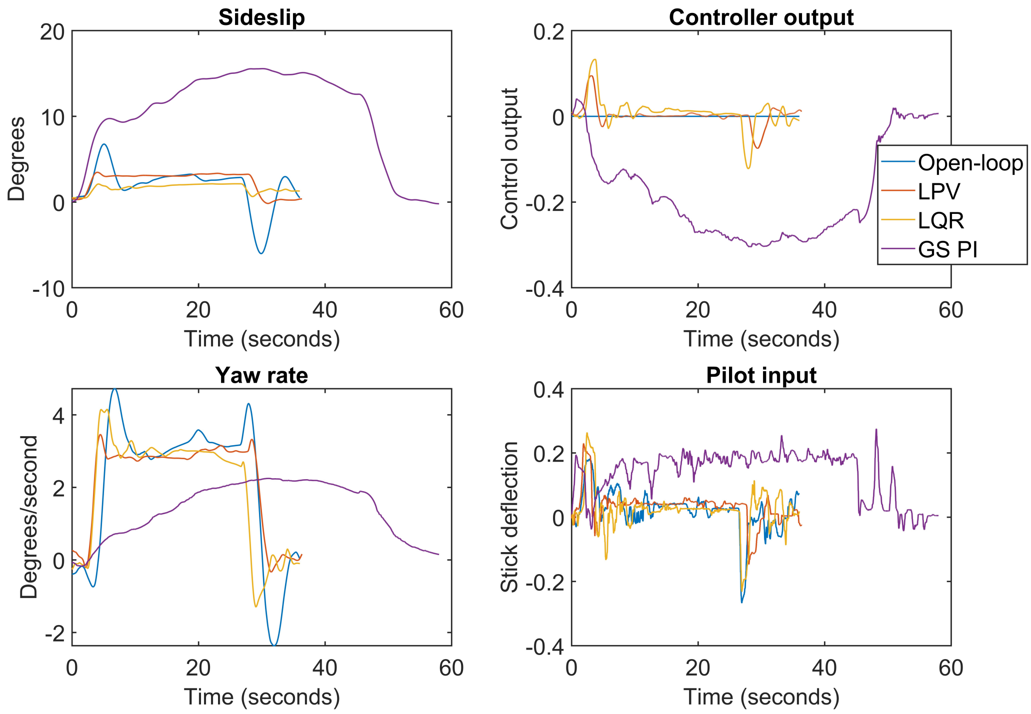

Figure 4.8: The resulting sideslip angle, controller output, yaw rate and pilot input during the aileron-only turn for different controllers.

5
Discussion

The basic definition of a yaw damper is a system used to suppress the Dutch roll by suppressing the yaw rate, as this is the most prominent trait of the Dutch roll motion. Other definitions state that a yaw damper should also aid in turn coordination. The different controllers designed in this thesis project use slightly different strategies to achieve their goals. The LQR and LPV controllers can easily incorporate turn-coordination aid by actively regulating the sideslip angle as it is natural with their respective control structures. This thesis's P and PI controllers do not actively consider turn coordination, although it is possible to construct a cascade control structure that would do that. The P- and PI controllers are mainly used as a baseline for comparison as they are still commonly used within the aerospace industry in various applications.

5.1 Dutch roll suppression

Graphs showing roll rate, yaw rate, sideslip angle and control signal following a rudder doublet input in level-flight for the different controllers are depicted in Figures 4.1 - 4.5. The rudder doublet input is used to induce the Dutch roll motion of the aircraft to showcase the different controller's abilities to suppress it.

5.1.1 PI-controller

In figure 4.2 it shows that the PI-yaw damper manages to suppress the induced yaw rate well compared to the open-loop response. A low amplitude oscillation in yaw rate is lingering, although this oscillation would barely be noticeable by the pilot. However, the lingering oscillation in yaw rate is caused by the sideslip oscillation that, although suppressed compared to the open-loop response, is high enough to cause discomfort to the flight crew and passengers along with increased pilot workload. A secondary control loop for regulating the sideslip angle would probably benefit the overall result. It should also be noted that the sideslip oscillations are slightly worse than that of the open-loop response, which suggests that the controller is slightly too aggressively tuned.

5.1.2 LQR

Figure 4.3 shows the results model response after a rudder doublet input with the LQR controller, and it can be seen that the yaw rate is well suppressed. Compared to the PI controller, the response is slightly less noisy, which can also be seen in the generated control signal of the two controllers. However, the main difference is that the LQR also suppresses the sideslip oscillations in the PI-controlled system, leading to better flight comfort and reduced pilot workload.

5.1.3 Gain scheduled PI

The system's response for the gain scheduled PI-controller is depicted in figure 4.4. As expected, the response is very similar to the PI controller. A rudder doublet does not change the system far away from the initial condition. Thus, the scheduled gains are expected to be similar to the PI controller. However, it can be seen that the generated control signal is less noisy. That is likely due to two main reasons. The first reason is that the controller is scheduled with the sideslip angle, so the gain is tuned to fit the system better when the plane is in this slipping. The second reason is that the gain scheduled PI controller is tuned marginally more conservative, making the control signal more conservative. Combining these two traits makes the gain scheduled PI controller suppress the sideslip oscillations present in the PI controller. However, the suppression is too slow, and a secondary control loop regulating the sideslip angle would benefit the overall system performance.

5.1.4 LPV

Figure 4.5 shows the system's response to the LPV yaw damper after a rudder doublet input. The LPV controller manages to suppress both the induced yaw rate and sideslip as expected. The x-axis is shortened since the LPV-controller regulates the system to a steady state in yaw rate and sideslip rapidly. The response is somewhat comparable to the LQR controller in response time but without any oscillations present in the LQR response, even if these are barely noticeable. The main difference, however, is how smooth the regulated response is. The smoothness indicates that the pilot workload will be kept at a minimum as no extra overshoots or hasty responses will cause the pilot to try to compensate. Overall the response of the LPV controller is very robust and efficient.

5.2 Controller evaluation

Depending on the application, it can be hard to compare the performance of different controllers. The yaw damper is considered a SAS to the aircraft, designed to artificially increase performance in an area where the physical design of the aircraft needs to be a compromise between multiple design goals. A SAS is a complex system to design since it needs to perform the task at hand, in this case, to suppress the Dutch roll motion without being too intrusive to the point where the pilot needs to fight the control system. It is a delicate task where a subjective evaluation of the control system is vital for the outcome. This differs quite a lot from systems without humans in the loop, where performance can be measured solely in control error, overshoot, and settling time. These are all objective metrics.

There are other approaches to aircraft control, e.g. a full closed-loop control system which may utilize the pilot inputs for references, e.g. roll, pitch and yaw, instead of inputting control surface deflections. The control system then has the authority of the control surfaces to drive the aircraft to the references provided by the pilot. This approach circumvents the complexity of trying to design a system that is robust enough while not being too intrusive, which has been a great challenge in this thesis project. However, a full closed-loop control system is not straight-forward to design, and it introduces other complex problems.

5.3 Modelling

When designing a control system, it is essential to have a model that can accurately describe the real system to a degree where the response can be evaluated with some confidence. This thesis project started with a preliminary, high fidelity model developed by Heart Aerospace. However, due to some implementation issues combined with time constraints, the high fidelity model was later changed to a more simplified model to circumvent some issues mainly related to finding trim points in the first model. For example, the simplified model has constant coefficients for the dynamic derivatives, whereas a higher fidelity model would have these vary with some parameters of the aircraft, e.g. angle of attack. Another simplification is that a collective thrust was implemented instead of, as in the case of the ES-19 aircraft, having four individual thrust inputs for each motor. These approximations are impacting the overall model fidelity. However, the model was deemed to perform with a realistic enough response for this project, where a relative evaluation of different control methodologies was to be conducted.

5.4 Flight testing

In contrast to our previous quantitative analysis methods presented earlier, the flight testing evaluation with the Cooper-Harper rating scale is qualitative. It might not always be clear from quantitative analysis how to design a controller that provides good stability and a low piloting workload. Therefore, it is crucial to conduct both types of testing for analyzing control strategies.

A couple of lessons were learned after flight testing. It stood clear that the PI and gain scheduled PI controller were fighting the pilot. The washout filters did poorly extract the reference signal from the yaw rate. The issue might be mitigated by tuning the time constant of the filter.

Interestingly, the third task to level off from the sideslip gave new previously unknown insight from the previous analysis. The LQR controller had residual yaw output when levelling off, which was the cause for the much higher HQR score in table 4.3. The cause could stem from multiple reasons.

Firstly, the LQR controller is a single point controller only designed for one specific condition. During the levelling out manoeuvre, the plane might have been too far from the LQR controller region of validity. The further from the aeroplane's design operating condition, the worse the robustness and performance of the LQR controller get. A gain scheduled controller would help in this case.

Secondly, there is a possibility that the cost function of the LQR controller is too lightly weighted on the actuator input. A higher weight for the control signal implies a less responsive controller, which could reduce the overshoot in the yaw command.

Thirdly, it could also be that the reference signal is not optimally defined. It is not obvious how to define a reference signal since the pilot does not control the reference signal. Washout filters are used to define the reference signal assuming that the low-frequency signals due to pilot input are desired. In contrast, higher frequency signals are considered to be unwanted errors. It might be helpful to decrease the filter's time constant to allow a broader frequency range of the signals not to be regarded as error signals.

The flight test evaluated the relative performance between the controllers and the baseline. To properly test the controllers, the complexity of the testing would increase drastically. It is standard practice to let at least three test pilots execute the same test program to avoid a single pilot's bias. Furthermore, a system like a yaw damper would be tested throughout the entire flight envelope for all airspeeds, altitudes, aircraft weights and centre of gravity and more. Above that, a simplified model was used in the simulator, which means there will be inaccuracies in the aircraft and the environmental models. Although the controllers are far from ready for implementation in an airframe, the project gives insight in to what controllers are worth investigating further in a higher fidelity model.

6
Conclusion & Future Work

In the end, the best yaw damper is the one that prevents the plane from accidents by reducing the pilot workload the most. From flight testing, it is clear that the LPV control strategy is the path going forward. Although the LPV controller did well in the quantitative analysis, so did the rest of the controllers. All controllers drastically increased the performance of the dutch-roll suppression compared to the open-loop response. However, the LPV controller performed outstandingly well during flight testing. A minimal workload from the pilot was required to achieve the desired performance. In contrast, some of the other controllers required more pilot workload to achieve aircraft characteristics worse than what was obtained with the LPV-controller.

A second iteration of flight testing after updating the controller designs would likely yield improved handling quality ratings across all controller designs, even the LPV controller. The flight test gave lots of insight into potential improvements that would lead to higher handling quality rating scores. Another activity for future work would be to implement the controller designs on a higher-fidelity model that should be able to capture the flight dynamics of the aircraft more closely. Additionally, the yaw damper should incorporate and be tested for parameters such as landing gear, flap settings and environmental conditions.

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A

Trim specifications

A.1 Operating point specification for level-flight

Tables A.1-A.3 shows the constraints used on states, inputs and outputs for finding operating point for the flight condition level-flight. These tables shows the specific case for $H = 10000$ ft $= 3048$ m and $V = 140$ ktas $= 72.02$ m/s.

States	**Value**	**Known**	**Steady-state**
ϕ	0	Yes	Yes
θ	-	No	Yes
ψ	0	Yes	Yes
p	-	No	Yes
q	-	No	Yes
r	-	No	Yes
u_B	72.02	No	Yes
v_B	-	No	Yes
w_B	-	No	Yes
x_E	0	Yes	No
y_E	0	Yes	No
z_E	-3048	Yes	No

Table A.1: Constraints used on states for finding operating point for the aircraft in level flight.

States	**Value**	**Known**	**Steady-state**
AilL_{pos}	0	Yes	Yes
AilL_{rate}	0	Yes	Yes
AilR_{pos}	0	Yes	Yes
AilR_{rate}	0	Yes	Yes
Elv_{pos}	-	No	Yes
Elv_{rate}	-	No	Yes
Rud_{pos}	0	Yes	Yes
Rud_{rate}	0	Yes	Yes

Table A.2: Constraints used on actuator states for finding operating point for the aircraft in level flight.

Inputs	**Value**	**Known**	**Min**	**Max**
rollCmd	-	No	-1	1
pitchCmd	-	No	-1	1
yawCmd	-	No	-1	1
flapCmd	0	Yes	0	2
throttleCmd	-	No	0	1
ldgCmd	0	Yes	0	1
Outputs	**Value**	**Known**	**Min**	**Max**
γ	0	Yes	-	-
V_{mps}	72.02	Yes	-	-
H_m	3048	Yes	-	-
ϕ	0	Yes	-90	90
θ	-	No	-90	90
ψ	0	Yes	-90	90

Table A.3: Inputs and output constraints used for flight condition level flight. Note that only the outputs that are constrained are in this table.

B
Flight test plan

FLIGHT TEST PLAN

(FTP)

YAW DAMPER TESTING – DYNAMIC LATERAL DIRECTIONAL STABILITY

	NAME / FUNCTION	DATE	SIGNATURE
Prepared	Henric Jorholm Andersson Flight Science		
	Carl Folkesson Flight Science		
Checked	Given Name SURNAME Department		
	N/A		
Approved	Given Name SURNAME Department		
	N/A		

Table of Contents

Table of Tables

No table of figures entries found.

1 Revision Log

REVISION	DATE	MODIFIED BY	COMMENTS
NC			

2 Distribution

COMPANY / DEPARTMENT	NAME

3 Acronyms and abbreviations

HA Heart Aerospace
N/A Not Applicable

4 Executive Summary

In this project, different controller strategies and methods have been designed for a yaw damper to a high-wing regional aircraft. The controller designs tested have ranged from classical proportional and integral controllers to the more modern methods of model-based optimal control. Also, gain scheduled and linear parameter varying control have been explored to improve robustness and performance through-out the flight-envelope. This test should be used to determine the handling qualities of the aircraft with the controller strategies compared to each other and to the baseline with no controller active.

5 References

Heart Aerospace Documentation

HAD.##	DOCUMENT TYPE	DESCRIPTION	REFERENCE / P/N
HAD.01			
HAD.02			

External Documentation

EXD.##	DOCUMENT TYPE	DESCRIPTION	REFERENCE / P/N
EXD.01			
EXD.02			

6 Introduction

This flight test is part of a master's thesis in stability and control. A couple of different controller strategies for the yaw damper stability augmentation system have been developed. The flight test should provide another perspective of the results and behaviour of the different control designs. Earlier tests have included controller response time, overshoot, and input energy. This test should give some insight into the handling qualities of the aircraft from the pilot's perspective.

7 Requirements / Aim of Test

The aim of the test is to evaluate the controllers from handling qualities point of view. The results could also be used to improve current design by tuning the feedback gains and weights.

8 Prerequisites

There are several prerequisites needed before conducting the testing:

- Model verification in simulator
- Controller design
- Objective evaluation of controller design in non-linear model

These prerequisites are already addressed during earlier parts of the project.

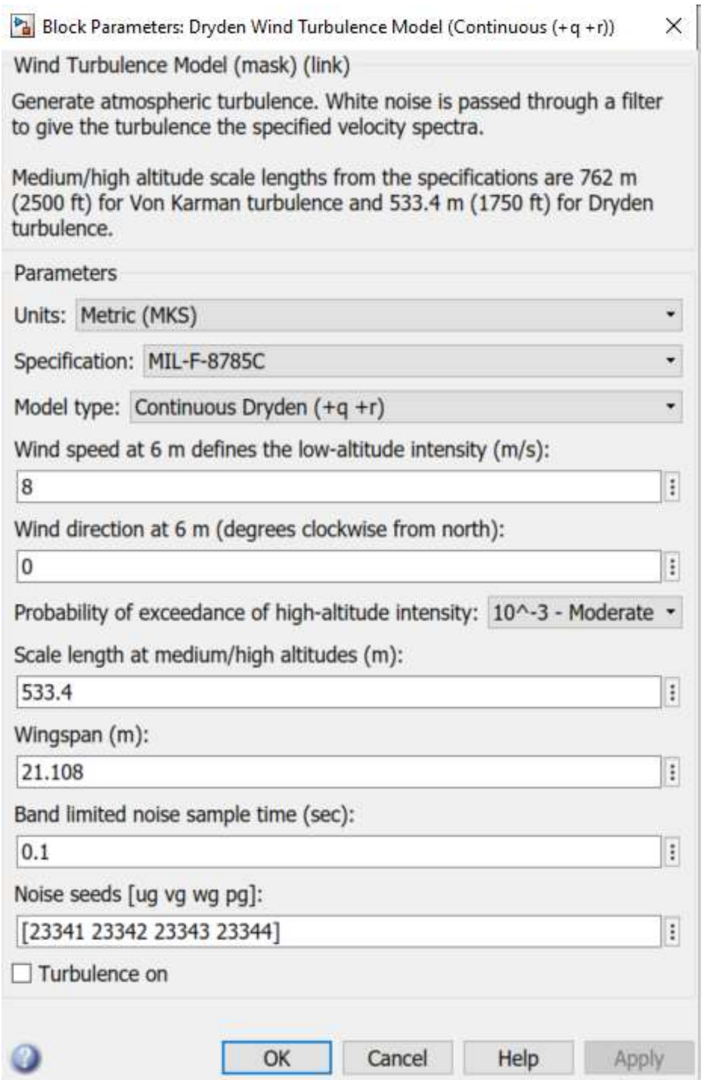
9 Applicability

The tested model is the simplified ES-19 model. The model can be found at git. Commit id 092ab8f3 from branch "stability_control" is used for testing.

The inceptors used are the Brunner sidestick and pedals. Force-feedback, deadband, and output mapping is described in profile **x** and **y**, available in the CLS2Sim software on PC1 in the Heart Aerospace engineering flight simulator.

10 Meteorological Conditions

To have an easier comparison between controller types, the meteorological conditions are held constant with no turbulence. However, it is also important to see how the controllers handle external disturbances. Therefore, moderate turbulence should also be tested. To reduce the complexity of the test, only the best performing yaw damper will be tested for turbulence. A screenshot of how the moderate turbulence was configured is included below.



11 Test Facilities

Engineering flight test simulator at Heart Aerospace.

12 Test Location

Heart Aerospace office, Fredsflottiljens Väg 19.

13 Test Aircraft and Configurations

The aircraft configuration will remain constant throughout the test. The only difference is going to be what type of yaw damper being used. The different configurations are:

- No controller
- PI controller
- LQR controller
- Gain scheduled PI controller

- LPV controller

Specific controller feedback gains and weights can be found in the git repository at the branch “stability_control” at commit id `092ab8f3`.

14 Weight & Balance

The weight and CG in the model will be kept constant throughout the test since it is an electric plane with no fuel burn. The plane is flown in light (9800 kg) configuration. The center of gravity (CG) will be at the most aft position.

15 Test Personnel

15.1 Flight Crew

15.1.1 Test Pilots

- Andrew Larkman

15.2 Ground Crew

15.2.1 Engineering Specialists

- Henric Jorholm Andersson

- Carl Folkesson

16 Data Acquisition, Instrumentation & Real Time Data Presentation

16.1 Instrumentation Required for Test

The picture below demonstrates the basic instrument panel with the real time information presented to the pilot. It contains information about, airspeed, attitude, height, vertical speed, turn coordination and heading.

The most important parameters to log include controller output, pilot input, angular rates, sideslip, and airspeed. This data and much more is logged and saved after each run to a .mat file for later analysis after the flight testing session.

16.2 Pass / Fail Criteria

A controller is considered satisfactory if it can dampen the dutch roll oscillation amplitude to 10% within two oscillations.

16.3 Test Techniques

All tests should be conducted with controls fixed after the dynamic input. Throttle setting should remain constant, and airspeed should be regulated by pitch angle adjustment.

16.4 Test Matrix

Test No.	*Altitude*	*Air speed*	*Gear*	*Flap*	*Power*	*Mass*	*CG*	*Comments*
Lateral Directional Stability								
x.01	*10'000*	*1.4* V_{S1}	*Up*	*Up*	*PFLF*	*Light*	*Aft*	*Roll out on a heading, north to east*
x.02	*10'000*	*1.4* V_{S1}	*Up*	*Up*	*PFLF*	*Light*	*Aft*	*Roll out on a heading, east to north*
x.03	*10'000*	*1.4* V_{S1}	*Up*	*Up*	*PFLF*	*Light*	*Aft*	*SHSS – 15 deg bank, left*
x.04	*10´000*	*1.4* V_{S1}	*Up*	*Up*	*PFLF*	*Light*	*Aft*	*From SHSS, transfer to level flight with heading capture*
x.05	*10´000*	*1.4* V_{S1}	*Up*	*Up*	*PFLF*	*Light*	*Aft*	*SHSS – 15 deg bank, right*
x.06	*10´000*	*1.4* V_{S1}	*Up*	*Up*	*PFLF*	*Light*	*Aft*	*From SHSS, transfer to level flight with heading capture with aileron*

16.5 Weight & Balance

The weight and CG in the model will be kept constant throughout the test since it is an electric plane with no fuel burn. The plane is flown in light (9800 kg) configuration. The center of gravity (CG) will be at the most aft position.

17 Analysis and Reporting

The data logs from the flights will be saved to .mat files directly from Matlab. The data will contain all inputs, outputs, and state variables. The outcome of the tests will be described in the master’s thesis which will be available at the company internal Google drive, and it will also be published at Chalmers University.

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