AN INTEGRATION OF A MODERN FLIGHT CONTROL SYSTEM DESIGN TECHNIQUE INTO A CONCEPTUAL DESIGN STABILITY AND CONTROLS TOOL, *AEROMECH*

by

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ABSTRACT

AN INTEGRATION OF A MODERN FLIGHT CONTROL SYSTEM DESIGN TECHNIQUE INTO A CONCEPTUAL DESIGN STABILITY AND CONTROLS TOOL, *AEROMECH*

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The aircraft conceptual design (CD) phase is the most abstract, thus challenging phase of the entire aircraft design process. It is the responsibility of the CD engineer to identify and then to explore various aircraft concepts with the goal of arriving at the most promising concept for further evaluation. During this early design phase, the discipline stability and control tends to be underrepresented due to the lack of non-linear aerodynamic and inertia data. The methodology and software *AeroMech* is a vehicle configuration independent aircraft conceptual design stability and control tool, developed to help the conceptual designer to address stability and control for conventional to unconventional design proposals. These include control power assessment for sizing control effectors, trimmed aerodynamics for performance estimation and evaluation of static and dynamic stability for safety verification. This tool has continually been refined over the years from its conception by Dr. Chudoba to its software implementation by Kiran Pippalapalli and Gary Coleman. The ultimate goal of this research undertaking is to increase the capability of *AeroMech* to assess an aircraft design for handling qualities in addition to safe flying qualities

This research is the proposed first step to achieving a capability to shape an aircraft to possess good handling qualities. The objective is to augment the current flight control system design module to include a modern control technique of practical value during the conceptual design phase, which can be utilized to design for desired handling qualities in context with the airframe. This thesis identifies the research problem, the selection of a control technique, the implementation into a FORTRAN source code and the integration of this system into *AeroMech*. A thrust vectoring transport aircraft design example validates and demonstrates the new FCS module.

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NOTATIONS

Abbreviations

AVDS	=	Aerospace Vehicle design Synthesis
CE	=	Control Effector
CD	=	Conceptual Design
CCV	=	Controls Configured Vehicle
CS	=	Configuration Setting
DCFC	=	Design Constraining Flight Condition
DD	=	Detail Design
DBS	=	Data Base System
DiCE	=	Directional Control Effector
DOF	=	Degrees of Freedom
FC	=	Failure Condition
FCS	=	Flight Control System
FT/C/M	=	Flight Test/Certification/Manufacturing
FWC	=	Flying Wing Configuration
I/AI	=	Incident/Accident Investigation
KBS	=	Knowledge Base System
LaCE	=	Lateral Control Effector
LoCE	=	Longitudinal Control Effector
LOTS	=	Linear-Optimum Trim Solution
O	=	Operations
OFWC	=	Oblique Flying Wing Configuration

OWC = Oblique Wing Configuration

QSTORM = Quasi-Steady-State Take-off Rotation

PD = Preliminary Design

PrADO = Preliminary Aerospace Design and Optimization

SAS = Stability Augmentation System

SM = Static Margin

SSSLF = Steady-State Strait line Flight

SSPUPO = Steady-State Pull-up / Push-over

SSRP = Steady-State Roll Performance

SSTF = Steady-State Turning Flight

TAC = Tail Aft Configuration

TFC = Tail First Configuration

TSC = Three Surface Configuration

Symbols

A = Coefficient matrix of state vector

b = Span

B = Coefficient matrix of control vector

C = Coefficient matrix of output vector

$K_{\delta_{LoCE}/\alpha}$ = Angle of attack gain to longitudinal control effector

$K_{\delta_{LoCE}/q}$ = Pitch rate gain to longitudinal control effector

$K_{\delta_{LaCE}/\phi}$ = Bank angle gain to lateral control effector

$K_{\delta_{LaCE}/p}$ = Roll rate gain to lateral control effector

$K_{\delta_{DiCE}/\beta}$ = Sideslip angle gain to directional control effector

$K_{\delta_{DiCE}/r}$ = Yaw rate gain to directional control effector

n_α	=	Load factor gradient
S_{ref}	=	Reference Area
T_i	=	Thrust available per engine
u	=	Output vector
V_T	=	Relative Velocity
x	=	state space vector
y	=	output vector

δ_{DiCE}	=	Directional Control Effector Deflection
δ_{LoCE}	=	Longitudinal Control Effector Deflection
δ_{LaCE}	=	Lateral Control Effector Deflection
δ_{sb}	=	Speed break deflection
δ_t	=	Percentage of available thrust
α	=	Angle of attack
β	=	Side-slip angle
γ	=	Flight path angle
ω_n	=	Natural frequency
ζ	=	Damping Ratio
τ	=	Time constant
T_{double}	=	Time to double amplitude

CHAPTER 1

INTRODUCTION

'Remember, airplanes are not built to demonstrate stability and control, but to carry things from one place to another'; [comment by Otto Koppen after a stability and control lecture]. Perhaps Koppen went too far, because history has shown over and over again that the neglect of stability and control fundamentals has brought otherwise excellent aircraft projects down, sometimes literally.

Abzug and Larrabee

1.1 Research Project Initiation and Motivation

The epigraph espouses the motivation for this research endeavor which is a desire to explore the balance between designing for mission performance and designing for safety performance. On one hand, aircraft designed solely to mission performance can have safety deficiencies which require costly fixes. On the other hand, safety characteristics by themselves are insufficient to define a vehicle which meets all mission requirements. Safety can however be used as design discipline to effect the final shape of the vehicle. It is this authors interest to explore the safety discipline in the early phases of the aircraft design process and use it to improve the overall aircraft product.

The Aerospace Vehicle Design (AVD) Laboratory at the University of Texas at Arlington Mechanical and Aerospace Engineering (UTA-MAE) approaches aircraft design with the aim of improving the overall aircraft product lifecycle from conceptual design to accident/incident investigation (Oza 2008). Within this philosophy, a great emphasis is placed on stability control and safety as shown in (Chudoba 2001). The goal of this research is to increase the capability of the Aerospace Vehicle Design Synthesis (AVDS) process specifically in the area of Flight Control System (FCS) design. This chapter discusses related background information, a description of the specific problem to be addressed and the research approach selected to arrive at a solution.

1.2 Background

The Aerospace Vehicle Design Synthesis (AVDS) methodology and software, developed and applied by the AVD Laboratory at UTA-MAE and its partners, is the aircraft design process at the forefront of this research project. This process is a multi-disciplinary parametric approach to aircraft design which employs carefully crafted tools to simulate the entire life cycle of an aircraft starting from the conceptual design phase. A wealth of information about a particular design or design options is produced, thereby providing both design proficiency and confidence in decision making about the project. For the sake of establishing the basis for this thesis, background information about the aircraft design lifecycle, AVDS and *AeroMech*, the AVDS stability and control tool, are given in the following sections.

1.2.1 Design Lifecycle

The term "design lifecycle" is used in the AVD Lab to describe the life span of a flight vehicle after the mission objectives have been specified. The cycle consists of six continuous phases which are Conceptual Design (CD), Preliminary Design (PD), Detail Design (DD), Flight Test/Certification/Manufacturing (FT/C/M), Operation (O) and Incident/Accident Investigation (I/AI). These terms were coined after talks with industry and academic experts in design (Oza 2008). Figure 1.1 shows the progression of these phases which are described below. It is important to mention that during conceptual design, the AVD Laboratory process simulates all these phases except Detail Design.

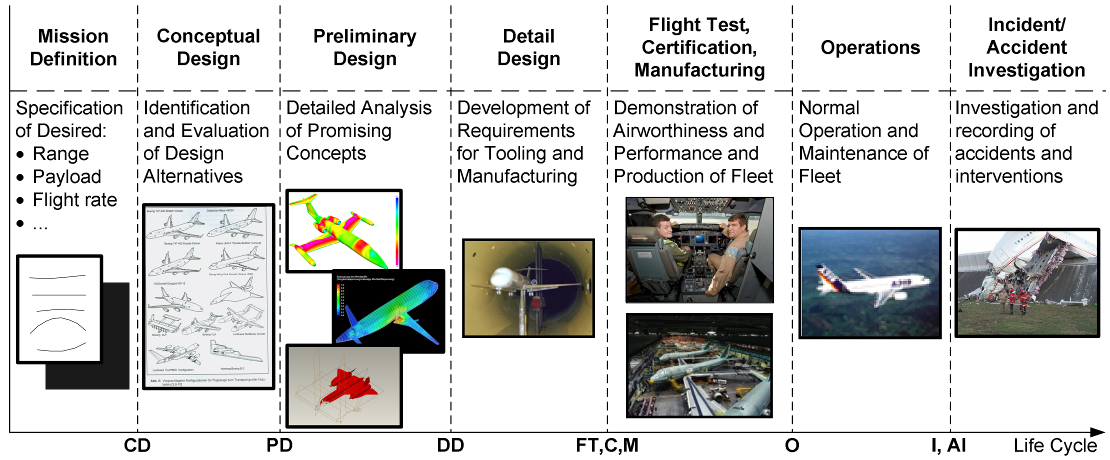


Figure 1.1 Aircraft Lifecycle as described in the AVD Lab

Conceptual Design (CD) refers to the time frame between the initiation of a design project and the selection of the most feasible design concept. The tasks in this phase include the creation of measurable design objectives from mission specifications, exploration of the design solution space where feasible design concepts are located, evaluation of these design alternatives with respect to objectives, and the selection of the most viable concept(s) (McGraw-Hill 2004). The goal is not to create the most accurate but correct design which fulfils the objective design function selected by the design team. In this regard, the tools used in CD do not need to be capable of the most detailed analysis; however, they need to be able to produce show correct trends. The advanced projects departments of the most aerospace companies are active during the CD phase design. Advanced Projects Departments such as Bell's X-Works, Lockheed Martin's Advanced Design Projects (formerly Skunk Works), Boeing's Phantom Works, etc., explore future projects for their respective companies during the conceptual design this phase. The AVD Lab executes with this same mindset via a life-cycle simulation methodology and software.

The Preliminary Design (PD) phase is where the development of baseline specifications for manufacturing is developed. The design concept selected during the conceptual design phase is further evaluated and additional concern is giving to refine the assumptions made during the CD phase to determine if the concept truly meets design objectives (McGraw-Hill 2004). Emphasis is now placed an accuracy of a given correct baseline design. It is at this stage that intense disciplinary studies begin. For example, the aerodynamicists run highest fidelity aerodynamic codes to analyze the aircraft aerodynamics. In the same manner, the structural engineer uses the best methods to determine if the structures provide the desired rigidity, weights and volume. The stability and control engineers develop the flight control laws as well as verify that the aircraft has adequate flying and handling qualities. At some point in PD, there is a design freeze. After this freeze, major changes to the aircraft are not longer allowed only minor modifications are accepted. The end product of preliminary design is a complete aircraft design description including all systems and subsystems.

The Detail Design (DD) phase is where physical components are selected and integrated to form a complete aircraft prototype for flight testing and certification (McGraw-Hill 2004). Attention is given to design the hardware in order to ensure that finial prototype properly represents the initial design concept.

During the Flight Test/Certification/Manufacturing (FT/C/M) phase, the designers prove the viability of the aircraft to be successfully manufactured, manufacture the test vehicle, show airworthiness and demonstrate the performance promised to the customers. There is still room for design changes in this phase however, there is very little flexibility. Towards the end of real time flight testing, production and manufacturing begin then the aircraft are supplied its customers.

The operations (O) phase is where fleets of the manufactured aircraft are flown by the customers. Customers include the military, airliners, business owners, aircraft enthusiast, research organizations and the like. Most customers use aircraft within the specified limits while others push them beyond the flight envelops in unintended ways for the sake of research and pleasure. This design stage generates more design information and validation points especially in the context of off-design conditions. Design changes at this stage come in the form of retrofitting packages or upgrades and cost money.

During the entire lifecycle, it may happen that unforeseen situations, incidents or accidents occur that are attributed to design or operational flaws. These incidents generate valuable lessons learned which can help refine current or future designs. The Incident/Accident Investigation (I/AI) phase represents the time period for all these activities. It could overlap the operations phase; it could be based on post-operations flight tests or it could not eexist at all. It depends on the vehicle and the incidents that may or may not occur.

These phases makeup the entire aircraft product lifecycle in which the AVD Lab attempts to simulate. The lifecycle simulation system and synthesis tools are briefly described next.

1.2.2 Design Lifecycle Simulation and Aerospace Vehicle Design Synthesis

The idea of lifecycle simulation is to emulate all the phases of the aircraft design cycle starting from the CD phase onwards in order to prove concept viability and increase continuity throughout the actual lifecycle. That is, during the CD phase, all relevant design phases are simulated up to incident and accident investigation. The product simulation results are analyzed, and lessons learned can be rapidly implemented by iterating back to the beginning of the design life-cycle. These extra analyses and simulations help augment upfront knowledge generation; accelerate design response time; increase design freedom; and improve correctness and reliability of design decisions (Oza 2008). Figure 1.2 depicts the interplay between product lifecycle and the interactions between knowledge, cost of design change, flight test and freedom. For more information on this process, see (Oza 2008).

The primary ingredients required for lifecycle simulation are a Data Base System (DBS), a Knowledge Based System (KBS), a lifecycle focused methodology and a combination of multidisciplinary design tools which fit into this methodology. The DBS contains data on existing designs, while the KBS contains design methodologies as well as lessons learned (Chudoba

2001). The AVDS methodology, shown in Figure 1.3, is iterative in nature. The multidisciplinary tools include AVD Sizing, a parametric sizing tool for visualizing the solution space (Coleman 2010, 404); *AeroMech*, a generic stability and control tool (Chudoba 2001; Coleman 2007, 283); PrADO, a vehicle synthesis tool for PD level analysis (Osterheld, Heinze, and Horst 2000); and VATES, a flight characteristics modeling and simulation tool for simulating flight tests (Oza 2008; Burdun and Parfentyev 2000, 75-92). This research is focused on increasing the capability of *AeroMech*; hence, the next section will briefly introduce *AeroMech*.

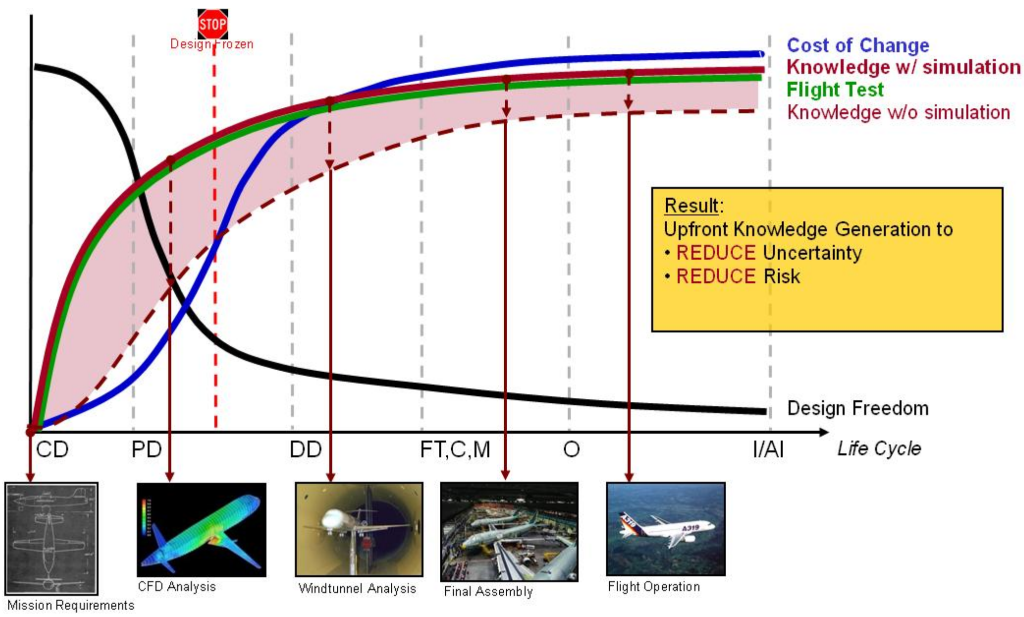


Figure 1.2 Knowledge construction during lifecycle sumulation (Oza 2008)

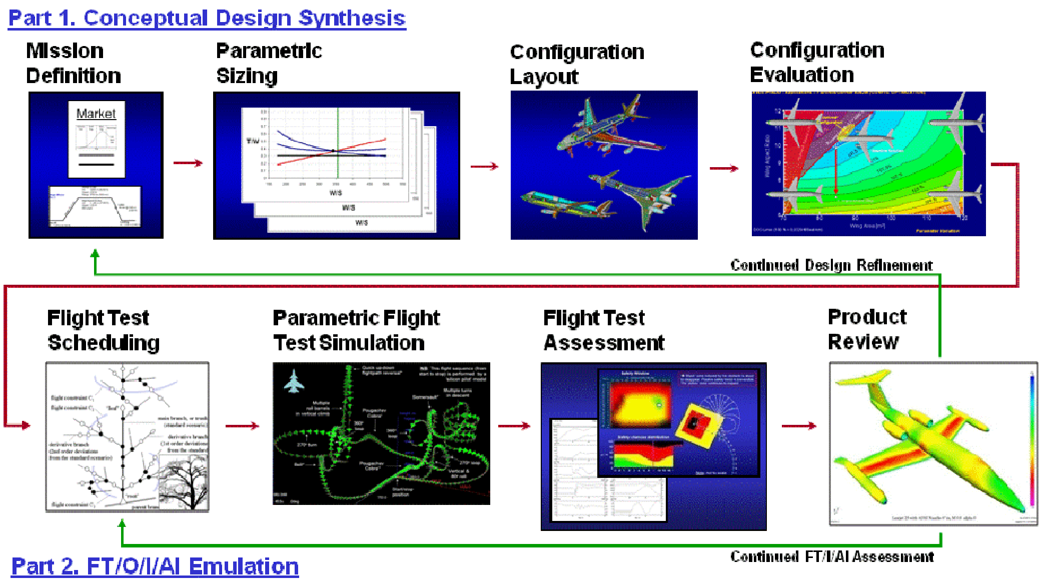


Figure 1.3 Overview of product lifecycle methodology at CD (Oza 2008)

1.2.3 AeroMech

AeroMech is an aircraft configuration independent stability and control design methodology and tool for conceptual design. Noticing that stability and control has been the bane of many aircraft designs, Chudoba did an extensive study on this problem in (Chudoba 2001). It was determined that these failures can be attributed to a deficiency in adequately addressing stability and control design during the CD phase. This inadequacy was expressed through an excerpt of a personal communication with Mr. Blausey former dynamists at Lockheed ADP:

> *"The first steps in conceptual design are fuselage and wing sizing. … Little or no thought is given to the empennage while this portion of the design process takes place. After the wing and fuselage are initially sized, the empennage is sized and added through a separate design effort. Stability and control requirements are considered one-at-a-time and the smallest empennage which meets all the requirements is determined. Wing position on the fuselage and landing gear position are sometimes shifted during the empennage design*

> *process. At some point in the design process, and usually before engineers are ready, management dictates a configuration freeze. After this time design changes are very difficult to make. However, small changes are possible. This is when wing strakes are reshaped; dorsal fins and ventral fins are added; wing and horizontal tail dihedral angles are changed; and wing fences, vortex generators, body strakes, fuselage plugs and wingtip extensions are added. These features usually appear when design deficiencies become evident after configuration freeze. Every bit of control effectiveness is also squeezed out through leading and trailing edge flap deflection optimization. … In the final stages of the design, stability and control takes on the dominant role in the aircraft development process."* (Chudoba 2001)

The *AeroMech* methodology has been proposed as a solution to the stability and control problem in the CD phase. The methodology systematically addresses stability and control concerns for both conventional and unconventional vehicles by presenting a generic means of

1. analyzing control power for Control Effectors (CE) sizing,
2. determining trimmed characteristics for improved performance estimations, and
3. evaluating static and dynamic stability for safety verification.

Control power is a very important stability and control characteristic to quantify during the conceptual design level. It is the ability of the aircraft control effectors to produce sufficient forces and moments to trim, maneuver, and stabilize the aircraft (ROSS and THOMAS 1979). Kay comments that "excessive control power can translate into increased weight and drag, while inadequate control power can result in a failed design" (Kay and others 1993). It is a function of a CE geometric parameter, aerodynamic stability derivatives, and the CE deflection angle (Chudoba 2001) shown in Figure 1.4. These variables are easier to adjust during the CD phase than in later phases. Therefore, it is the responsibility of the CD engineer to ensure that the aircraft has at least sufficient control power at critical non-linear corners of the flight envelope called design constraining flight conditions (DCFC).

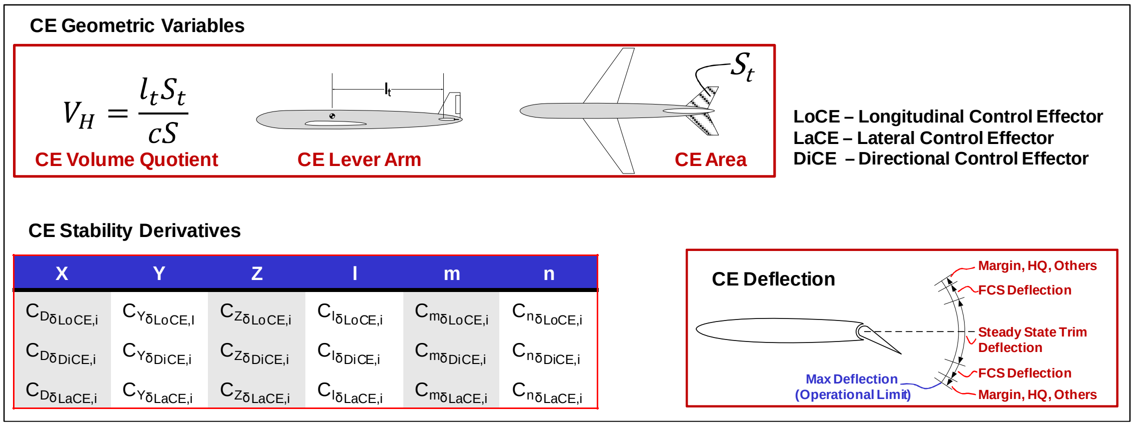


Figure 1.4 Measures of control power

A key component of *AeroMech* is its ‘generic’ nature. The methodology was designed to be capable of analyzing conventional and unconventional aircraft configurations throughout the full speed range. These configurations include Tail-Aft Configuration (TAC), Tail-First Configuration (TFC), Three-Surface Configuration (TSC), Flying Wing Configuration (FWC), Oblique Wing Configuration (OFC) and Oblique Flying Wing Configuration (OBFW) and are shown in Figure 1.5. The ‘generic’ nature is achieved by solving full non-linear 6-DOM trimmed equations during CD as opposed to configuration specific reduced order models. These equations give a good representation of cross coupling and stall effects for modeling all configurations (Chudoba 2001). The equations are also used to produce trimmed aerodynamic data such as trimmed drag polars, lift curve slopes and pitching moment curves around the trim point. This information is useful in evaluating performance characteristics such as climb and decent performance.

AeroMech is also capable of performing static and dynamic analysis about the trim point. Linear derivatives from the aerodynamic input are interpolated around the design point and used to create a linear model from small perturbation equations of motion. In the original conception of *AeroMech*, coupled linear equations were derived for generic stability and control analysis as well as flight control system emulation (Chudoba 2001). However, in the current version of the software, a decoupled small perturbation analysis subroutine called ILOCS from (Abzug 1998)has been integrated to perform stability and control analysis and estimate stability augmentation system gains (Coleman 2007, 283). The evolution of *AeroMech* software is discussed next.

TAC	**T**ail-**A**ft **C**onfiguration	symmetric
TFC	**T**ail-**F**irst **C**onfiguration	symmetric
TSC	**T**hree-**S**urface **C**onfiguration	symmetric
FWC	**F**lying-**W**ing **C**onfiguration	symmetric
OWC	**O**blique-**W**ing **C**onfiguration	asymmetric
OFWC	**O**blique **F**lying-**W**ing **C**onfiguration	asymmetric

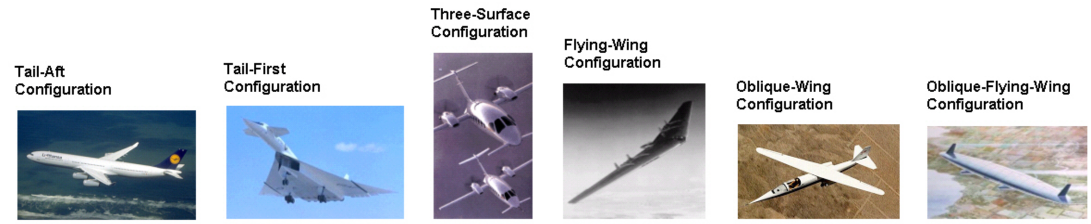


Figure 1.5 Aircraft configurations (Chudoba 2001)

The current version of *AeroMech* has undergone three generations of development. The first generation was the work done by Bernd Chudoba in his Ph. D. dissertation. His contributions are highlighted in (Coleman 2007, 283) as

1. the Identification of the problem with stability and control in CD which required *AeroMech*;
2. the development of the basic methodology for *AeroMech*, able to estimating control power for generic aircraft configurations in CD;
3. the selection of VORSTAB as an appropriate independent aerodynamic prediction tool;

4. the derivation of coupled 6-DOF steady state equations for trim, pull-up and push over, turn, rolling and take-off rotation;
5. the derivation of coupled small perturbation equations of motion for open and closed loop dynamic mode analysis.

The next evolution was by Kiran Pippalapalli, Chudoba's former graduate student, Kiran's Master's Thesis work "*AeorMech* – A Conceptual Design Stability and Control Design Tool" (Pippalapalli 2004). His contributions as described in (Coleman 2007, 283) are

1. the verification of the 6-DOF steady state equations of motion by Chudoba;
2. the creation of *AeroMech* Code Structogram;
3. the integration of VORSTAB and *AeroMech*;
4. the development of *AeroMech* prototype in FORTRAN.

The version of *AeroMech* preceding this current master thesis research was by Gary Coleman, another graduate student of Chudoba. In his Master's research, Gary restructured the *AeroMech* source code (Coleman 2007, 283). He developed a working version and validated it with a case study of the YB-49. The following are his contributions.

1. The Restructure of Kiran's *AeroMech* FORTRAN Code to increase functionality.
2. The integration of a second aerodynamic prediction tool, Digital DATCOM, which is easier to operate than VORSTAB and acceptable for tube and wing aircraft configurations.
3. The implementation of preexisting aircraft dynamics stability analysis package.
4. The developments of output file organization and visualization.
5. The validation of *AeroMech* with the YB-49 case study.

Given this current version of *AeroMech* as a baseline, the goal of the present research is to further improve *AeroMech*'s capability in the area of designing for good handling qualities. The problem is described in the next section.

1.3 Problem Description

It is a reality that aircraft stability and control concerns do not only include flying qualities but they include handling qualities as well. Flying qualities are the inherent flight vehicle characteristics of the airplane, while handling qualities are the characteristics of the pilot interacting with the airplane (US Air Force Test Pilot School 2002). In other words, handling qualities deal with pilot-plane interactions and the difficulty or ease of the pilot to perform required tasks. Flying qualities are typically addressed during the CD phase by developing an inherently safe and controllable aircraft which meets certification requirements (Chudoba 2001; Kay and others 1993; Roskam 2004). During the preliminary design phase, handling quality issues are traditionally ‘fixed’ by using the flight control system to augment the aircraft. The sole reliance on the flight control system to repair handling qualities issues can lead to very complex flight control systems that require heavy high-rate actuators. A unique topic for research is to determine the degree to which handling qualities have an influence on the inherent aircraft design during the conceptual design phase.

The above formulated research topic not at all addressed in any aircraft design nor flight mechanics and flight control system specific texts. Figure 1.6 shows a survey of representative conceptual design texts and a categorization of the stability and control criteria considered. This figure shows that no consideration is given to pilot-in-the-loop shaping of the vehicles[1]. On the other hand, the flight controls engineer does indeed consider handling qualities as a key concern during the preliminary design phase as evident in (US Air Force Test Pilot School 2002; Gibson 1999; Hodgkinson 1999). In actuality, the specifications in (Anonymous1986) are produced relating flying qualities to empirical data on handling qualities. However, these are insufficient for the new age of aircraft with augmented dynamics, hence, dedicated handling qualities design specification are necessary (Gibson 1999).

[1] It is important to note that the terms handling qualities and flying qualities are used interchangeably in some of these texts. However, in the context of this research handling qualities refers specifically to pilot airplane interactions.

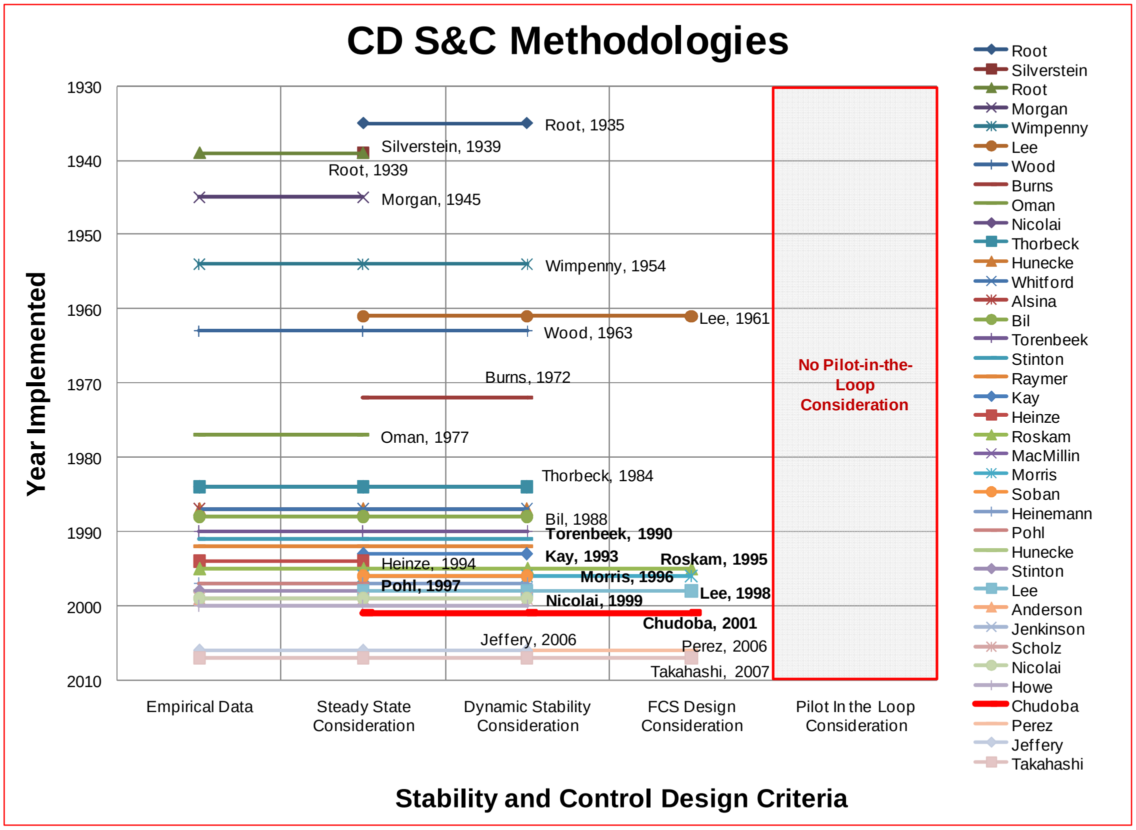


Figure 1.6 Stability and control conceptual design texts surveyed for handling qualities

Consequently, the research objective was to determine if shaping aircraft for good handling qualities during the aircraft conceptual design phase can alleviate flight control system requirements, thereby leading to cheaper and lighter aircraft with similar, identical or even better performance. This involves answering the following questions: Is at all possible to shape handling quality characteristics during the conceptual design phase? Is the resolution of information at this level sufficient enough to model handling qualities? What effects do handling qualities have on the physical shape of the aircraft?

After the review of representative handling qualities texts such as (US Air Force Test Pilot School 2002; Gibson 1999; Hodgkinson 1999; Mooij and Nationaal Lucht- en Ruimtevaartlaboratorium (Netherlands) 1985), it was discovered that the issue of determining handling qualities sensitivities during the conceptual design phase is too lofty for a Master of Science thesis. In addition, a recurrent theme of the dependency of handling qualities on the design of the flight control system was observed. As a result of these finding, the research problem is reduced to the following question: Can an FCS design technique, capable of designing for good HQ, be integrated into a CD S&C design methodology and software, AeroMech? The handling qualities questions are reserved for a later study. The research objective and tasks for addressing this problem are discussed in the next section.

1.4 Research Objective and Tasks

The goal of this research is to build some of the necessary frame work required to perform the study of handling qualities assessment during the conceptual design phase. The goal of this research is on the integration of an advanced flight control system design tool into the *AeroMech* source code, which offers an improved modeling capability of the flight control system (FCS) with view to handling qualities & airframe shaping while being executable during the conceptual design phase. The tasks required to achieve this objective include:

1. To develop an understanding of the effects of flight control system design on aircraft design in general.
2. To outline specifications for a flight control system design tool which is detailed enough to capture preliminary design details that affect handling qualities, yet is simplistic enough to be executed during the conceptual design phase.
3. To survey existing and available flight control system design tools and methodically select one based on the above outlined specifications.
4. To program, implement and validate this flight control system design tool in a stand-alone format compatible with the *AeroMech* source code.
5. To integrate this software tool into the *AeroMech* environment and demonstrate its applicability to conceptual design related case study.

1.5 Master's Organization

This Master's thesis is organized in a logical fashion and takes the approach of problem statement and solution concept while stepping through the research, implementation and validation phases.

Chapter 1 is the introduction of the research topic. It discusses a background to the research work, the desired problem to solve, the research objectives and organization.

Chapter 2 develops the idea of flight control system design for conceptual design. It discusses the need for flight control systems, the effects of flight control systems on design, specifications for a desired flight control systems design technique and the selection process of a design technique.

Chapter 3 documents the implementation of the selected flight controls system design technique. It outlines the implementation from the theoretical development to the source code realization and validation.

Chapter 4 is the integration of the selected FCS design technique into the *AeroMech* baseline methodology and software.

Chapter 5 documents the above build-up in the context of a relevant design case study.

Chapter 6 offers the thesis contribution summary, recommendations for future study and reflection on the research experience.

1.6 Chapter Summary

This chapter gives an introduction to the current research endeavor including background information, problem description, Master of Science (M.S.) approach and objectives. The background information pertaining to the research involves a description of the design lifecycle, the AVD Lab at UTA-MAE lifecycle simulation methodology and synthesis tools. The problem desired to be addressed by this research is to determine the degree of influence the conceptual design phase has on shaping for good handling qualities. This vast subject has been narrowed down to the identification and integration of a flight control design tool which can be used as a stepping stone towards handling qualities research. Finally, the objective and tasks for the M.S. research are concisely presented.

CHAPTER 2

FLIGHT CONTROL SYSTEM DESIGN FOR CONCEPTUAL DESIGN

2.1 Introduction

Abzug *et al* report that the XB-47 was one of the first aircraft with a stability augmentation system (Abzug and Larrabee 2002). The aircraft required a yaw damper because there was a rolling motion at high angles of attack caused by the swept wings' induced dihedral effect. The decision to use a stability augmentation system was uncommon but novel at the time. In fact, there was some opposition to its use during some presentations on the idea in 1949 (Abzug and Larrabee 2002). Abzug et al however explain why stability augmentation is necessary saying:

> "there is a perfectly sound aerodynamic reason why yaw stability augmentation is needed on jet airplanes and it is not an evidence of poor design. Approximately, Dutch roll damping ratio is directly proportional to atmospheric density. An airplane with a satisfactory damping ratio of 0.3 at sea level will have a damping ratio of 0.06 at an altitude of 45,000ft." (Abzug and Larrabee 2002)

In order to promote an appreciation of the importance of flight control systems in conceptual design, a proper introduction to Flight Control Systems (FCS) is necessary. This chapter will explain the need for control systems, describe how they work, make specifications for a flight control system design package suitable for conceptual design and outline the selection process.

2.2 The Need for Flight Control Systems

The current evolution of modern aircraft has resulted in vehicles with exotic propulsion systems, extravagant wing shapes, various body sizes, peculiar control effector types and novel vehicle configurations. These varieties of vehicle concepts, while improving performance, can have adverse effects on stability and controllability. For example, on one hand, swept wings postpone transonic drag rise for better aircraft performance at high speeds (Stevens and Lewis 2003). On the other hand, positive wing sweep makes the dihedral effect more negative which decreases the Dutch-roll damping ratio which is important for stability (Roskam 2001). The performance-stability equation is further complicated by the new trend of aircraft with multiple missions and speed ranges such as the extended range B777 and the multiple versions of the F-35. Expanded envelopes place aircraft in a wide range of dynamic pressures and as shown with the Abzug statement earlier, dynamic stability is sensitive to dynamic pressure changes. The resulting conundrum is a tough decision between sacrificing performance for safety or safety for performance. Roskam comments on this issue by saying “Designing for good inherent stability nearly always results in some performance, weight and balance penalties. The designer must find the appropriate balance between the carious conflicting factors.”(Roskam 2001).

A solution to the problem of arriving at the best compromise between safety and performance is to use control feedback to modify aircraft dynamics for desired stability characteristics while retaining high performance geometry. McRuer *et al* list the following as advantages of feedback control (McRuer, Ashkenas, and Graham 1974):

1. to provide stability;
2. to adjust dynamic response, including
3. reduction of lags,
4. provision of desire or specified command/response relationships, especially as regards the improvement of linearity and reduction of the effect of vehicle cross-coupling forces;
5. to suppress unwanted inputs and disturbances;
6. to suppress the effects of variations and uncertainty in the characteristics of the controlled element (i.e., a stable, indifferent or inherent airframe)

In addition to these advantages, flight control systems can be designed to give an aircraft good stability characteristics throughout its flight envelop without major detriment to performance.

There are three major categories of flight control systems used in aerospace namely

1. Stability Augmentation Systems (SAS) – used to improve aircraft transient response (damping ratios and natural frequencies) to control effector inputs from the pilot.
2. Control Augmentation Systems (CAS) – used give the pilot control of aircraft modes that are not directly or precisely controllable by control effector inputs such as pitch rate.
3. Autopilots – used to reduce pilot workload by automatically performing auxiliary tasks such as speed hold, heading hold and landing.

Some examples of each of these types of FCS are given in Table 2.1 below obtained from (Stevens and Lewis 2003).

Table 2.1 Examples of the different classifications of FCS (Stevens and Lewis 2003)

SAS	**CAS**	**Autopilots**
Roll damper	Roll rate	Pitch attitude hold
Pitch damper	Pitch rate	Altitude hold
Yaw damper	Normal acceleration	Speed/Mach hold
	Lateral/directional	Automatic landing
		Roll-angle hold
		Turn coordination
		Heading hold/VOR hold

CAS and Autopilot structures are generally outer-loops over SAS inner-loops and they are used to provide secondary functions. For these reasons, the discussion of Flight Control Systems in this research is limited to SAS and these terms FCS and SAS are used interchangeably in this context. A description of how a feedback FCS works is given in the next section.

2.3 How Feedback Flight Control System Work

A Feedback control system works by using aircraft output variables to generate signals which are then applied as additional input to the aircraft control effector actuators thereby modifying the dynamic response (Stevens and Lewis 2003). The effect can be seen more clearly by examining the following block diagrams and matrix equations. A block diagram of an aircraft Linear Time Invariant (LMI) state-space model is shown in Figure 2.1 below.

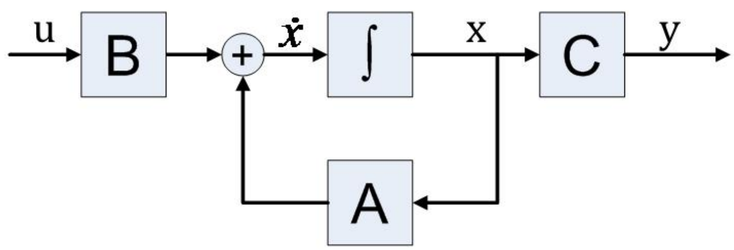


Figure 2.1 Block diagram of aircraft linear time invariant model

The equation of the LMI state-space model given in Figure 2.1 is

$$\dot{x} = Ax + Bu \tag{2.1}$$

Where x and u are the state vector and control input, while A and B are coefficient matrices. Typical aircraft states and control vectors are $x = [v_T \quad \alpha \quad \beta \quad p \quad q \quad r \quad \phi \quad \theta \quad \psi \quad x \quad y \quad z]^T$ and $u = [\delta_T \quad \delta_e \quad \delta_a \quad \delta_r]^T$. The coefficient matrices which are also known as Jacobian matrices are obtained by linearizing the equations of motion. Hence, matrix A indicates the dynamic response of the aircraft about the linearization. The eigenvalues of this state vector coefficient matrix therefore give the aircraft dynamic modes such as short-period, phugoid, spiral modes, etc. It follows that these aircraft dynamic modes can be altered by adjusting the coefficient matrix of the state vector. This is accomplished by using feedback.

Now consider the block diagram of a LTI aircraft model with feedback control shown in Figure 2.2.

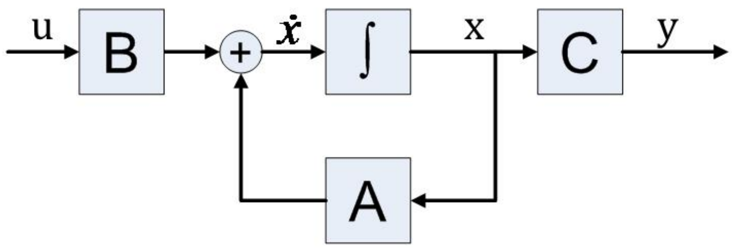


Figure 2.2 Block diagram of aircraft LTI model with feedback control

The output equation and control law in the block diagram can be are written respectively as

$$y = Cx \tag{2.2}$$

$$u = -Ky + v \tag{2.3}$$

Where the output vector y is the measurable part of the state vector x and v is an auxiliary input such as the pilot commands. The matrix C is the transformation matrix from the state to the output and Matrix K is the gain matrix. The resulting equation of motion of a model using feedback can be written as

$$\dot{x} = (A - BKC)x + Bv. \tag{2.4}$$

This equation shows how the coefficient of the state vector is augmented by feedback control. The new state vector coefficient matrix, $(A - BKC)$, is a function of the gain matrix K. As a result, the aircraft modes can be modified to make the aircraft more stable by specifying the appropriate gain matrix. This has to been done with care however, because in the same way the matrix K can stabilize the aircraft, it can also destabilize it (McRuer, Ashkenas, and Graham 1974). There are consequently two questions that need to be addressed in the context of this research.

1. How does feedback control affect aircraft design?
2. How can the feedback gains be adequately determined?

These questions will be answered in the following sections.

2.4 The Effect of Feedback Control on Aircraft Design

The aircraft designer is tasked to provide the customer an aircraft which satisfies the decided mission specifications as well as the airworthiness regulations defined by an applicable regulatory body such as the Federal Aviation Administration (FAA). The mission specifications may include a desired range, design cruise speed, payload capacity, level of maneuverability, minimum acceptable fuel consumption, ride comfort, etc. The regulations include minimum damping ratios, minimum allowable natural frequencies, allowable cockpit forces, minimum stall speeds, service ceiling, etc (Office of the Federal Register (U.S.) 2010). Mission specifications tend to be performance driven while airworthiness regulations are safety oriented. This automatically places the aircraft designer in the safety versus performance debate that makes flight control systems attractive as mentioned in Sec. 2.3 .

Roskam stresses that "the choice between inherent stability, [no SAS], and de-facto stability, [use of SAS], is made by the designer (together with the customer) and not by the regulations." (Roskam 2006). It is therefore important for the designer to understand some of the effects of FCS on an aircraft design. Two immediate primary benefits of implementing an FCS on aircraft design are:

1. enhanced maneuverability of military aircraft by flying them statically unstable (Gibson 1999);
2. Control Configured Vehicle (CCV); allowance of tail size reduction, thus, there is a decrease in overall weight, wetted surface area and drag designer (Roskam 2006).

These benefits come at a price however. There are four major adverse effects on aircraft design resulting from the use of stability augmentation system which are:

1. increase in control power requirement,
2. increase in system complexity,
3. increase of flight control system versus structural coupling,

4. introduction of higher order dynamics which affect handling qualities.

These detriments are discussed as follows.

2.4.1 Increase in control power requirement

Roskam states that "the airplane designer must be aware of the fact that a penalty paid for 'de-facto' stability is that stability augmentation systems use control power to achieve their objective" (Roskam 2006). This is an issue because control power is also required for trimming and maneuvering the aircraft (see Figure 2.3 and Chapter 1). It therefore is of concern to the designer to determine the increase in control power requirement by the FCS. This is because the increase in control power might require changes which negate the benefits that led to the decision to use an FCS in the first place. The reason for the increase in control power requirement is illustrated in Figure 2.4.

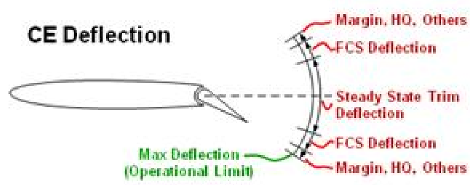


Figure 2.3 Control power representation using control effector deflection

The additional control power amounts from the response of the feedback system to disturbances as shown. When an unaugmented aircraft experiences a disturbance, the control effectors are not directly affected by it because inherent stability acts in a restoring sense. In case there is an additional deflection required to counter this disturbance, however, this control power is usually factored in by sizing the control effector at design critical flight conditions. On the other hand, when an augmented aircraft experiences a disturbance, the control effectors deflect proportionally to this disturbance because of the feedback gains. This could lead to control effector saturation especially applicable to aircraft with full authority flight control systems (i.e. aircraft with no limit on the allowable deflection from the stability augmentation system).

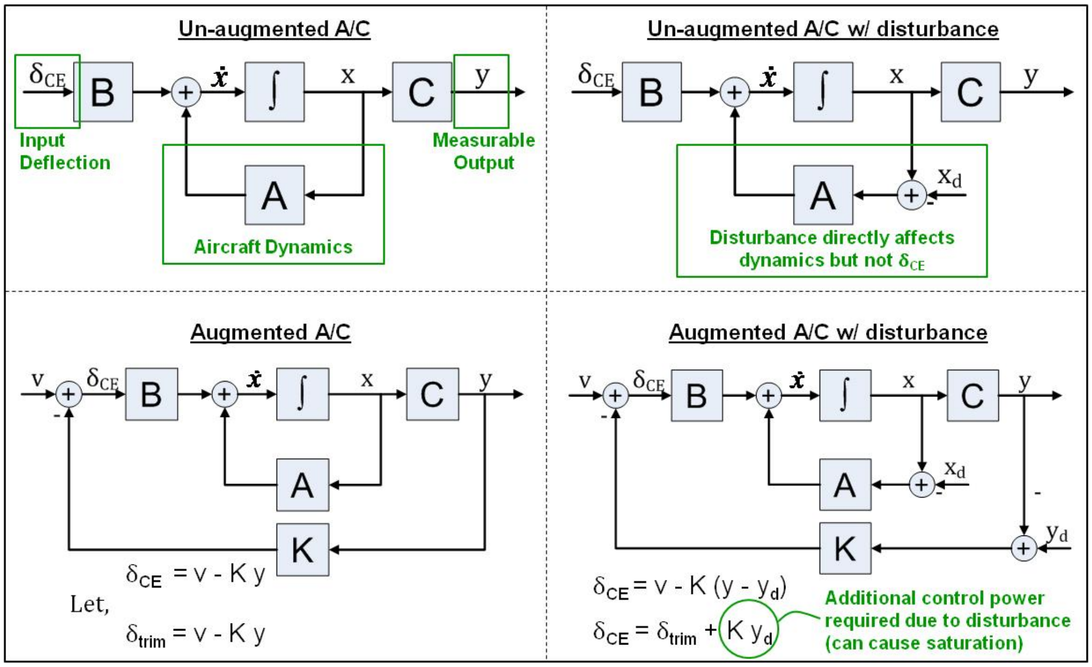


Figure 2.4 block diagrams showing increase in control power requirement

For example, assume an augmented aircraft has an angle-of-attack-to-elevator feed-back gain, K_α, of 15 deg/deg. If a wind gust induces an angle of attack of 2 deg, it would result in an additional elevator deflection of 30 deg. This is unacceptable and would cause control saturation and serious unpredictability for the pilot. If the designer discovers excessive gains such as this during the design process, it indicates that the control power of the aircraft needs to be increased (Roskam 2001). Rules of thumb for acceptable controller gain limits are given in Table 2.2 which is assembled from (Roskam 2006). The expected values in the table represent the magnitudes of expected disturbances and typical values are given in Table 2.3. Table 2.4 shows typical maximum control effector deflections. The next detriment of the use of stability augmentation systems is the increase in system complexity.

Table 2.2 Rules of thumb for controller gain limit assembled from (Roskam 2006)

Gain Criteria	Description	Criteria Limit
$\boldsymbol{K_{u/\delta_{sb}}}$	Velocity feedback to speed break	< 150 deg/unit of u/U_1
$\boldsymbol{K_{\alpha}}$	Angle of attack feedback to elevator	< 5 deg/deg
$\boldsymbol{K_{q}}$	Pitch rate feedback to elevator	< 2 deg/deg/sec
$\boldsymbol{K_{\beta/\delta_r}\,\beta_{expected}}$	Product of sideslip feedback to rudder and expected sideslip angle	< $0.3\delta_{r_{available}}$
$\boldsymbol{K_{r/\delta_r}\,r_{expected}}$	Product of yaw rate feedback to rudder and expected sideslip yaw rate	< $0.3\delta_{r_{available}}$
$\boldsymbol{K_{\beta/\delta_a}\,\beta_{expected}}$	Product of sideslip feedback to aileron and expected sideslip angle	< $0.3\delta_{a_{available}}$

Table 2.3 Typical expected magnitudes of disturbances from (Roskam 2006)

Airplane Type	$\boldsymbol{\beta_{expected}}$	$\boldsymbol{r_{expected}}$
Transports	5 deg	10 deg/sec
Fighters	10 deg	10 deg/sec
Light airplanes	10 deg	20 deg/sec

Table 2.4 Typical maximum control effector deflections from (Roskam 2006)

Control Effector	Description	Typical Max deflection
$\boldsymbol{\delta_e}$	Elevator deflection angle	25 deg
$\boldsymbol{\delta_a}$	Aileron deflection angle	25 deg
$\boldsymbol{\delta_s}$	Speed break angle	60 deg
$\boldsymbol{\delta_{ih}}$	Elevator incidence angle	15 deg
$\boldsymbol{\delta_r}$	Rudder deflection angle	25 deg for single hinge line rudders
		35 deg for single hinge line rudders

2.4.2 Increase in System Complexity

Flight control systems require sensors, filters, compensators, redundant parts and other equipments in order to function properly. These additional systems add a level of complexity to the design. This can lead to increased aircraft cost or reliability and maintainability penalties. Roskam explains that “savings in tail areas and weight are as well as savings in drag! These savings must be traded against greater complexity of the flight control system and its sensors” (Roskam 2006). It is therefore important that complexity be factored into the sizing process to give a proper evaluation of the design. The next demerit to be discussed is the introduction of FCS versus structural coupling.

2.4.3 Flight Control System versus Structural Coupling

When control gains are designed, the aircraft is usually assumed to be a rigid body (Stevens and Lewis 2003); however, aircraft have flexible modes due to aerolasticity. Stevens explains that “these unmodeled high-frequency dynamics can act to destabilize a control system that may have quite suitable behavior in terms only of the rigid-body model” (Stevens and Lewis 2003). Additionally, actuator forces are transmitted to the structure which induces flexural modes detected by the sensors which in turn commands additional control deflection thereby creating a cycle leading to resonance and structural failure (US Air Force Test Pilot School 2002). The instability and resonance issues resulting from flight control systems are key issues which need to be addressed.

One part of the solution is to design controllers with stability robustness (Stevens and Lewis 2003). This is achieved by reducing the loop gain (US Air Force Test Pilot School 2002). Another part of the solution is to filter out the high frequency oscillations introduced by the flexible motion of the aircraft (Pratt 1999). This can be achieved by placing notch filters in the feed-forward and feedback paths of the flight control system. The level of information required for proper notch filter design and placement is usually not available in the early stages of design; hence, they are usually left out until vibration testing (US Air Force Test Pilot School 2002). The designers in the earlier design stages can, however, account for these effects by increasing the cost due to complexity in the sizing process. The final critical effect of flight control systems on aircraft design is the introduction of high order dynamics.

2.4.4 Introduction of Higher Order Dynamics Which Affect Handling Qualities

Gibson explains that stability augmentation *"has sometimes had unforeseen effects on the short term response characteristics as well as the long term ones. Simple modal parameters exactly equivalent to the conventional frequency and damping may be absent because the modes have changed completely or because of a high order control law structure or both"* (Gibson 1995). In other words, the use of stability augmentation introduces higher order modes and/or displaces the regular modes such that they cannot be correlated with conventional regulations (US Air Force Test Pilot School 2002). This means the handling quality of augmented aircraft is difficult to ascertain without the use of additional preliminary design criteria. It is this very effect that motivates this research undertaking. In order to shape aircraft for good handling qualities, it is necessary to include preliminary design considerations which begin with FCS design.

These effects of the flight control system on aircraft design highlight the need of the designer to be able determine the required controller gains in order to properly evaluate the benefits an augmented aircraft concept. This also means that the method used in selecting gains is of key importance. A specification for an appropriate means of calculating control gains for this research is given in the next section.

2.5 Specifications for a Suitable Control Design

There are very many flight control design schemes and philosophies for solving the controls problem. They range from reduced order modeling of control effects as in (Roskam 2001) to the use of tools that model the minutest of control system detail as in (Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997). It is therefore necessary to specify a desired methodology for this research. Since the aim is to develop a tool that bridges the gap between the conceptual designer and the preliminary designer, the following categories of specifications have been defined.

1. Conceptual Design Specification

2. Preliminary-Detail Design Specification
3. *AeroMech* Compatibility Specification

2.5.1 Conceptual Design Specification

As mentioned in Chapter 1, the primary objective in conceptual design is to generate sufficient information to support early design decision-making. Roskam comments that “the objective here is to arrive at a decision about the feasibility of a certain configuration with a minimum amount of engineering work” (Roskam 2004). This translates to the following CD specification:

1. speed of calculation,
2. accepts minimal input,
3. minimum work is required,
4. generic (configuration independent),
5. captures top level design details,
6. results translate directly to design decisions.

The next section identifies PD specifications.

2.5.2 Preliminary-Detail Design Specification

The preliminary and detail designers, in this case flight dynamicists, are tasked to ensure that a selected configuration will meet all stability and control requirements. “The objective here is to arrive at a realistic, reasonable detailed layout of an airplane configuration. The goal now is to ‘fine tune’… that means to determine whether or not the configuration meets …specifications” (Roskam 2004). In this phase, more attention is given to the system structure and high fidelity. McRuer et al. give some qualities of the best control systems in history:

1. simplicity of mechanization,
2. Economy equalizations
3. Commonality of elements and settings for different operational modes
4. Simplicity of gain compensation

5. Versatility across vehicles
6. Lack of response to unwanted inputs
7. Lack of susceptibility of the sensors to unwanted inputs
8. Lack of sensitivity to controller tolerances and airframe configuration changes
9. Lack of sensitivity to controlled element uncertainties and parasitic nonlinearities
10. Inherent reliability and maintainability

These qualities provide criteria for preliminary design specifications. In addition for good handling qualities design, the design technique should allow "the engineer [to] maintain a detailed knowledge of and exercise control over the signal pathways and interconnection" (Gibson 1999). That is the flight control technique must allow the ability to design FCS with desired control structure. These considerations translate to the following preliminary design specifications:

1. ability to select any desired control structure;
2. Visibility of all the command paths
3. Simplicity in the resulting system
4. Lack of response to unwanted inputs
5. Robustness of the control scheme
6. Physical interpretation of design scheme
7. Ability to model different compensator dynamics

2.5.3 AeroMech Compatibility Specification

AeroMech is a generic (configuration independent) conceptual design stability and control tool (Chudoba 2001; Coleman 2007, 283). The source code, written in FORTRAN, is capable of analyzing control power required for maneuver and trim conditions. It also estimates trimmed aerodynamics and evaluates static and dynamic characteristics. It generates linearized state space models for flight control system analysis as well. Currently, it uses Abzug's ILOCUS subroutines for flight control system design ((Coleman 2007, 283; Abzug 1998), but a method of Equivalent Derivatives was proposed in the original methodology (Chudoba 2001). The methodology selected from the present research investigation will be integrated into *AeroMech* resulting in increased FCS modeling capability.

The selected methodology therefore needs to be compatible with *AeroMech*. The following specification ensures compatibility with *AeroMech*.

1. an availability of a source code or programmable algorithm
2. the source code written in FORTRAN Language
3. there is permission and capacity to modify source code if needed
4. there is proper documentation of the source code or algorithm

This gives a total of twenty criteria which would be used in select a suitable system for this research. The systems will graded based on the following color scheme.

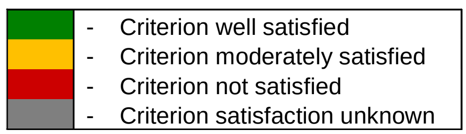


Figure 2.5 Color Scheme for FCS design technique evaluations

The different flight control schemes for consideration are given in the next section.

2.6 Flight Control Design Options and Assessments

A list of the flight control design techniques reviewed, a brief description, key references and available tools are given with Table 2.5. This list of methods has been generated based on the references provided. The assessments of these techniques are based on the specifications and template from Sec. 2.6 as shown in Table 2.6 to Table 2.8. These qualitative charts are used to show some of the characteristics of the methods in order make the selection process more visual.

Table 2.5 Flight Control System design techniques considered

Name	Philosophy	Key References	Tool / Algorithm
Flight Control System Emulation (Equivalent Stability Derivatives)	Assuming no lag in the SAS, the action of the SAS can be thought of as a superposition of the inherent stability derivative of an airplane and the contribution to that derivative by the idealized stability augmentation system.	(Roskam 2001) (Chudoba 2001) (Abzug 1998)	AAA
Classical Control Theory	Gains in the transfer function of a single input and single output feedback system are varied until the system displays a desired performance. The philosophy for multiple loops and multiple input and outputs is successive loop closure (Stevens)	(Roskam 2003) (McRuer, Ashkenas, and Graham 1974) (Stevens and Lewis 2003)	ILOCS(Abzug and Larrabee 2002) CONDUIT(Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997)
Eigenvector Assignment	Matrix operations are used to to locate the poles and zeros of a multiple input and output feedback system so that the system meets performance requirements	(Andry, Shapiro, and Chung 1983, 711-729) (Nieto-Wire and Sobel 2007) (Pratt 1999)	CONDUIT(Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997)
Linear Quadratic Regulator w/ Full state Feedback	Matrix operations are used to close feedback loops on all states simultaneously with the gains selected based on performance criteria	(Stevens and Lewis 2003) (Nelson 1998)	CONDUIT(Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997)
Linear Quadratic Regulator w/ Output Feedback	Matrix operations are used to close feedback loops of available output simultaneously with the gains selected based on performance criteria	(Shapiro, Fredricks, and Rooney 1981, 505) (Stevens and Lewis 2003) (Choi and Sirisena 1977, 134-136)	CONDUIT(Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997)
Linear Quadratic Regulator w/ Explicit Model Following	A regulator is design to make the system behave like an ideal model of desired performance with the model part of the regulator	(Stevens and Lewis 2003)	CONDUIT(Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997)
Linear Quadratic Regulator w/ Implicit Model Following	Performance indices are selected to make the feedback system behave like an ideal model of desired performance without including the ideal model in the controller	(Stevens and Lewis 2003)	CONDUIT(Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997)
Dynamic Inversion	A nonlinear system is linearized using a feedback loop containing the system's dynamics then the linear system can be controlled via an outer tracking loop.	(Stevens and Lewis 2003). (LOCKHEED MARTIN AERONAUTICS CO FORT WORTH TX and others 2001, 70)	CONDUIT(Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997)
Linear Quadratic Gaussian Design	Full state feedback regulator is used in conjunction with an observer for estimating immeasurable states. It is made possible by the separation principle	(Stevens and Lewis 2003) (Anderson and Moore 2007).	CONDUIT(Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997)
H infinity Design	The use of frequency domain techniques to design a robust modern-controller (ie a controller with noise and uncertainty rejection). The resulting system is of higher order and in the case of output feedback; an estimator is used to determine unknown states.	(Pratt 1999; Doyle and others 1989, 831-847)	CONDUIT(Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997)

Table 2.6 Assessment based on conceptual design requirements

	Speed of calculation	Accepts minimal input	Minimum work is required	Generic	Captures top level design details	Results translate directly to design decisions
Equivalent Stability Derivatives						
Classical Control Theory						
Eigenvector Assignment						
Linear Quadratic Regulator w/ Fullstate Feedback						
Linear Quadratic Regulator w/ Output Feedback						
Linear Quadratic Regulator w/ Explicit Model Following						
Linear Quadratic Regulator w/ Implicit Model Following						
Dynamic Inversion						
Linear Quadratic Gausian Design						
H infinity Design						

Table 2.7 Assessment based on preliminary design requirements

	Ability to select any desired control structure	Visibility of all the command paths	Simplicity in the resulting system	Lack of response to unwanted inputs	Robustness of the control scheme	Physical interpretation of design scheme	Ability to model various compensators
Equivalent Stability Derivatives							
Classical Control Theory							
Eigenvector Assignment							
Linear Quadratic Regulator w/ Full state Feedback							
Linear Quadratic Regulator w/ Output Feedback							
Linear Quadratic Regulator w/ Explicit Model Following							
Linear Quadratic Regulator w/ Implicit Model Following							
Dynamic Inversion							
Linear Quadratic Gaussian Design							
H infinity Design							

Table 2.8 Assessment based on *AeroMech* compatibility requirements

	Availability of source code or algorithm	FORTRAN Language Source Code	Permission and capacity to modify code	Proper documentation
Equivalent Stability Derivatives				
Classical Control Theory				
Eigenvector Assignment				
Linear Quadratic Regulator w/ Full state Feedback				
Linear Quadratic Regulator w/ Output Feedback				
Linear Quadratic Regulator w/ Explicit Model Following				
Linear Quadratic Regulator w/ Implicit Model Following				
Dynamic Inversion				
Linear Quadratic Gaussian Design				
H infinity Design				

2.7 Selection

The flight controls design technique ultimately chosen as the most suitable for this research undertaking is the linear quadratic output feedback design technique. There was some contemplation that led to this final selection and they are discussed in the next sections.

2.7.1 FCS Emulation vs. Automatic Control Theory

Flight control system emulation by equivalent stability derivatives is a technique for handling flight control system in conceptual design proposed in (Chudoba 2001; Roskam 2001; Roskam 2006). This method assumes a no lag situation in which all the mechanisms of control, such as actuators and sensors, are infinitely fast and their dynamics can be neglected (Roskam 2001). Therefore the equivalent stability derivatives can be written as

$$C_{x_{y_{SAS}}} = C_{x_{y_{Airframe}}} + C_{x_\delta} K_y y, \tag{2.5}$$

where $C_{x_{y_{SAS}}}$, $C_{x_{y_{Airframe}}}$, C_{x_δ}, K_y and y are the Equivalent stability derivative, inherent stability derivative, control derivative, control gain and control variable respectively. This allows the control gain to be estimated as

$$K_y = \frac{C_{x_{y_{SAS}}} - C_{x_{y_{Airframe}}}}{C_{x_\delta} y} \tag{2.6}$$

This equation gives a framework for quickly estimating feedback gains for design. The equivalent derivative approach presents the following advantages:

1. It meets almost all the conceptual design specifications in Sec. 2.5.1
2. It does not require extensive knowledge of automatic control theory (Roskam 2001)
3. It can be used to estimate required actuator performance (Chudoba 2001)

The disadvantages of this methodology include:

1. It does not satisfy preliminary design specifications in Sec. 2.5.2
2. It neglects important dynamics which are critical to handling qualities
3. It is not entirely generic (Chudoba 2001)

The disadvantages of the equivalent stability derivative approach are typical for any reduced order mindset. In contrast, automatic control theory involves the analysis and synthesis of control systems using established methodologies which capture all (or selective) dynamics of the aircraft mechanisms of control. Therefore, it has been decided that the present research endeavor will concentrate on an augmenting automatic control theory methodology. Clearly, the automatic control theory methodology approach will strike a better progression between conceptual design and preliminary design.

2.7.2 Classical Control Theory vs. Modern Control Theory

Automatic control theory has two classifications, classic control theory and modern control theory. Classical control theory, as the name implies, is the oldest form of the two classifications originating as far back as 1877 with the development of Routh's Criteria (McRuer, Ashkenas, and Graham 1974). Classical theory involves synthesizing control systems by analyzing single loop transfer functions and adjusting feedback gains to provide desired performance. Most of the techniques used for analysis are in the frequency domain (Stevens and Lewis 2003). They include root locus, Bode plot, Nichols chart and Nyquist diagram methods (Roskam 2003). Processes for using classical theory are given, for example in (McRuer, Ashkenas, and Graham 1974) and via FORTRAN subroutines in (Abzug 1998). The benefits of classical theory include that

1. it has proven standardized processes for controls design,
2. It has a wealth of knowledge as this is the oldest controls design technique
3. the control structures have physical connection to the real world
4. the techniques meet most of the preliminary design specifications in Sec. 2.5.2
5. it can handle generic systems
6. Abzug's Fortran source code meets *AeroMech* compatibility requirements in Sec. 2.5.3

The demerits of classical control theory include that

1. it requires a great deal of intuition and experience,
2. gains for multi loop system are not selected based on parametric indices and require a great deal of trial-and-error
3. it becomes more tedious and unreliable as the number of control loops increase
4. it requires a lot of designer involvement and trial and error
5. it does not meet most of the Conceptual design requirements in Sec. 2.5.1

The problems with classical control theory stem from the restriction to successive loop closure and lack of a mathematically relationship between performance objectives and multi loop gains (Stevens and Lewis 2003). Modern control theory addresses these very issues by using matrix operations to determine control gains based on precise performance criteria. If the control problem is properly phrased, it can reduce design time and effort significantly. For these reasons, modern control theory was selected over classical theory for this research. There have, however, been many criticisms of modern control theory and its applicability to aircraft design, see (Abzug and Larrabee 2002)(Abzug 1998). These criticisms are backed by the unsatisfactory performance of modern control theory in the design the X-29A and some other aircraft as discussed in (Abzug and Larrabee 2002). These poor performances are not because of the use of modern control but miss use of it. Stevens and Lewis defend modern theory explaining that

> "The traditional modern design techniques based on state variable feedback that are available in current texts are not suitable for aircraft controls. This is due to several things, one of which is their dependence on selecting large numbers of design parameters – namely, the performance index weighting matrices. Any design method for aircraft controls should eliminate the need for this trial and error selection" (Stevens and Lewis 2003).

It is, therefore, necessary to choose a modern control technique suitable for aircraft controls in the present research context. This has resulted in comparing the merits of the linear quadratic regulator with full state feedback and the linear quadratic regulator with output feedback, see the discussion presented in the next sub-chapter.

2.7.3 LQR Full-State Feedback vs. LQR Output Feedback

Linear Quadratic Regulator design is one of the modern control methodologies that, if applied properly, will design an aircraft with good stability characteristics (Stevens and Lewis 2003). It is also probably the simplest to implement. It involves the estimation of control gains that minimize a quadratic cost function of the form

$$J = \frac{1}{2}\int_0^\infty (x^T Q x + u^T R u)\, dt, \tag{2.7}$$

where Q and R are symmetric positive semidefinite weighting matrices (i.e. they have all positive eigenvalues). The idea here is that since the control vector, u, is a function of the gains, K, K can be calculated which will drive a weighted function of the state vector, x, to zero. This in turn guarantees a stable system with the performance dependent on the selection of the weighting matrices Q and R. A parametric methodology such as this is very suitable for conceptual design. There is, however, an issue with selecting the control law (i.e. the relationship between u and x). The two control laws are full state feedback and output feedback which are respectively written as

$$u = -Kx, \tag{2.8}$$

$$u = -Ky = -KCx. \tag{2.9}$$

The difference between these two laws is, that in the case of state feedback, all the states are used as inputs to the controls. While in output feedback, only select states or a linear function of the states are fed back for control via matrix C. The advantages of state feedback include that:

1. gains are selected based on parametric indices,
2. the calculation of the gain is fast
3. the system is guaranteed to be stable with proper selection of Q and R (Lewis)

4. gains are analytically computed hence do not need numerical initialization
5. tools are available which meet *AeroMech* compatibility requirements in Sec. 4
6. the gains calculated are the global optimum for the system
7. it has good robustness characteristics (i.e. system performs well in the presence of uncertainties)

The demerits include that:

1. it does not meet key preliminary design specifications in Sec. 2.5.2 ,
2. the gain matrix is populous because all the states are feedback and this is inefficient and costly
3. all states are seldom measurable in real world applications
4. the control law cannot be designed to have structure
5. the gain structure introduces cross coupling
6. it loses touch with the real world

Although full state feedback has many merits, its demerits make this approach undesirable for aircraft control. The biggest issues are that not all aircraft states are measurable and the control laws have no structure. One solution to the immeasurable states problem is the use of a dynamic observer to estimate the unknown state. This fix, however, dramatically increases system complexity without solving the structure problem (Stevens and Lewis 2003).

Stevens and Lewis suggest that using output feedback with a more general than usual performance criteria, “it is straight forward to design controllers that have sensible structure from the point of view of the experience within the aircraft industry, without the trial-and-error selection of a large number of design parameters”. To demonstrate such a control structure using output feedback, consider the following yaw damper block diagram in Figure 2.6,

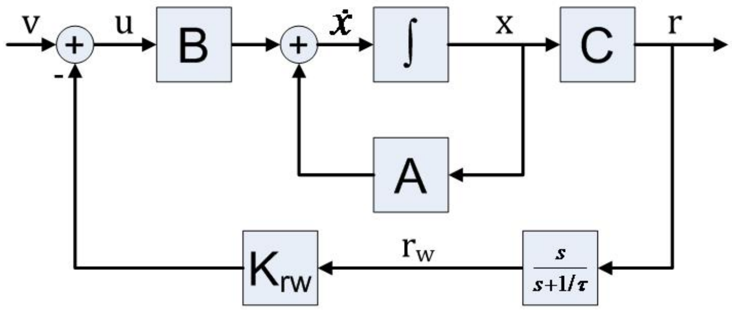


Figure 2.6 Block diagram of a yaw damper with a washout filter.

where $x = [\beta \quad \phi \quad p \quad r \quad \delta_r]^T$ is desired for the control gain to be applied to the washed-out yaw rate, r_w only in order to prevent the pitch damper from fighting the pilot during a bank. This problem is difficult to model with state feedback but it is simple with output feedback. First a washout state, x_w, is introduced with the following equations

$$x_w = r - r_w, \tag{2.10}$$

$$\dot{x}_w = \frac{1}{\tau}(r - x_w), \tag{2.11}$$

With a state vector $x = [\beta \quad \phi \quad p \quad r \quad \delta_r \quad x_w]^T$, the output feedback regulator can be implemented as

$$u = -KCx + v = -K_{u_r/r_w}[0 \quad 0 \quad 1 \quad 0 \quad 0 \quad -1]x + v. \tag{2.12}$$

The ability to provide control structure was a key factor in the selection of the linear quadratic output feedback regulator in the context of the present research undertaking. The merits of output feedback include that

1. it allows the design of automatic controllers with structure,
2. classical control structures can be implemented, thereby allowing the application of a wealth of industry experience,
3. it meets conceptual design requirements in Sec. 2.5.1 with proper formulation of control problem

4. it meets preliminary design requirements in Sec. 2.5.2 with proper formulation of control problem
5. tools are available which meet *AeroMech* compatibility requirements in Sec. 4

The demerits are that

1. gains are selected based on parametric indices,
2. it requires a numerical solution and hence it need an initial stabilizing gain
3. the calculated gains are sub-optimal because they represent different local minimums depending on the initial stabilizing gain
4. stability is not guaranteed unless problem is properly structured
5. it is computationally intensive and can take longer than other modern control techniques
6. robustness characteristics are not guaranteed

In light of the merits, linear quadratic output feedback seems to be the most suitable technique since it strikes a proper balance between the specifications given in Sec. 2.5 . In addition, there are ways to curb some of the demerits as will be shown in the chapter on implementation. It is important to mention that other modern control techniques have been considered but not selected because the complexity exceeded that of the LQR approach. For more information about these techniques, consult the references in Table 2.5. An additional note is that the design tool CONDUIT (Tischler, Ames Research Center., and United States Army Aviation and Troop Command. Aeroflightdynamics Directorate. 1997) seems to be capable of synthesizing most available control design techniques. However, since the source code is unavailable at the time of this research, CONDUIT had to be ruled out for implementation.

2.8 Chapter Summary

This chapter discusses the importance of flight control systems, its effect on aircraft design and the selection process for a flight control system design technique suitable for conceptual design. The key argument for flight control systems is that it answers the need to maximize both performance and safety. The drawback however is that it increases control power requirement, complexity and susceptibility to structural noise. The solution is that the designer needs to identify these issues during the early conceptual design phases. This requires an FCS modeling technique capable of estimating the control gains without hindering the conceptual designer. LQ Output feedback design was chosen as a suitable technique.

CHAPTER 3

IMPLEMENTATION OF LQR OUTPUT FEEDBACK DESIGN

3.1 Introduction

In the previous chapter, the rationale behind the selection of linear quadratic output feedback was given. The LQ controller does strike a proper balance between conceptual design and preliminary design objectives while providing a parametric process for control design via performance indices. Additionally, LQ allows the modeling of control structures with different levels of detail via proper definition of coefficient matrices. This makes this approach suitable for lower level preliminary and conceptual design phase modeling. This chapter will discuss the implementation of the linear quadratic output feedback into conceptual design. This chapter covers theoretical development, algorithm proposal, FORTRAN implementation and validation. The theories and algorithms are compilations of desirable output feedback elements from various available sources. While the source code implementations are written specifically for this current research undertaking.

3.2 Theoretical Development of LQ Output Feedback Control Design

In this section, the theory of an LQR output feedback algorithm is be developed and tailored for the specifications discussed in Chapter 2. The following LQR derivation is given in (Stevens and Lewis 2003). Consider the following linear time-invariant system

$$\dot{x} = Ax + Bu \tag{3.1}$$

$$y = Cx \tag{3.2}$$

where $x(t) \in \mathbb{R}^n$, $u(t) \in \mathbb{R}^m$, and $y(t) \in \mathbb{R}^p$ are the state, control input and the measured output vectors. It is to be controlled by output feedback of the form

$$u = -Ky \tag{3.3}$$

Where the gain matrix K is an $m \times p$ matrix of constant coefficients to be determined. The resulting closed loop system can be written as

$$\dot{x} = (A - BKC)x \equiv A_c x \tag{3.4}$$

Since it is the desire for the system to be stable, hence, the control input u, via K, must be selected to guarantee stability by forcing the states to zero. This can be achieved by using u to minimize the quadratic cost

$$J = \frac{1}{2}\int_0^\infty (x^T Q x + u^T R u)\, dt, \tag{3.5}$$

where Q and R are positive semi-definite weighting matrices. Substituting equations 0 and 0 gives

$$J = \frac{1}{2}\int_0^\infty (x^T Q x + C^T K^T R K C)\, dt, \tag{3.6}$$

Suppose there is a constant symmetric positive semi-definite matrix P which satisfies the equation

$$\begin{gathered} \frac{d}{dt}(x^T P x) = -x^T (Q + C^T K^T R K C) x \\ \dot{x}^T P x + x^T P \dot{x} = -x^T (Q + C^T K^T R K C) x \end{gathered} \tag{3.7}$$

substituting 0 gives

$$g \equiv A_c^T P + P A_C + Q + C^T K^T R K C = 0 \tag{3.8}$$

Then the quadratic cost is written as

$$J = \frac{1}{2} x^T(0) P x(0) - \frac{1}{2}\lim_{t\to\infty} x^T(t) P x(t) \tag{3.9}$$

If the system eventually stabilizes

$$J = \frac{1}{2} x^T(0) P x(0) \tag{3.10}$$

which can be written as

$$J = \frac{1}{2} tr(PX) \tag{3.11}$$

where the trace, $tr(\cdot)$, of a matrix is the sum of its diagonals and the matrix X is an $n \times n$ matrix defined as

$$X = x(0)x^T(0) \tag{3.12}$$

Note, it is common to select $X = I_n$ (where I_n is an $n \times n$ identity matrix). This is a good assumption for the regulator problem but not for tracking (Stevens and Lewis 2003).

It is shown in (Stevens and Lewis 2003) that equation 0 must satisfy the following

$$(A - BKC)^T P + P(A - BKC) + Q + C^T K^T RKC = 0 \tag{3.13}$$

$$\frac{\partial J}{\partial K} = 2(RKCLC^T - B^T PLC^T) = 0 \tag{3.14}$$

where L is an $n \times n$ positive semidefinite matrix solution to

$$(A - BKC)L + L(A - BKC)^T + X = 0 \tag{3.15}$$

Equations 0 and 0 are special equations called Lyapunov equations. Lyapunov equations are linear symmetric matrix equations which are identical to their transpose. They can be solved using the ATXPXA (ARMSTRONG 1978) or SB03MD (Benner and others 1999, 499-539) subroutines.

The solution to the output feedback gain problem requires the minimization of equation 0 using K. This needs to be solved numerically because 0-0 are coupled and nonlinear (Stevens and Lewis 2003). One numerical solution technique is to use the Davidon-Fletcher-Powell (Press 2007) gradient-based subroutine. Using this subroutine, at each step, the value of K and 0 are used to solve for the cost, while 0 and 0 are used to update the direction of K. Note: It is important that each new value of K stabilizes the system. Hence, the update algorithm needs to be subject to a stability check (Choi and Sirisena 1974, 257-258). The following conditions are necessary for convergence (Stevens and Lewis 2003):

1. The existence of a gain K such that A_c is stable (i.e. the system is output stabilizable).
2. The output matrix C has a full row rank p.
3. The control weighing matrix R is positive defininte (i.e. all the inputs are weighed).
4. The state weight matrix Q is positive semidefinite and $\left(\sqrt{Q},\ A\right)$ is detectable.

There are two major problems with linear quadratic output feedback design as discussed:

1. The numerical solution techniques require an initial stabilizing gain K_0.
2. The weighing matrices R and Q need to be carefully selected.

There are some ways around these problems and they will be discussed progressively in the next sections.

3.2.1 Case 1: LQ Full State Feedback

In the case of LQR full state feedback, all the states are assumed to be measurable and feedback to the control. That is $C = I_n$ and

$$u = -\widehat{K}\, x. \tag{3.16}$$

Therefore 0 and 0 become

$$\left(A - B\widehat{K}\right)^T P + P\left(A - B\widehat{K}\right) + Q + K^T R\widehat{K} = 0 \tag{3.17}$$

$$\frac{\partial J}{\partial \widehat{K}} = 2\left(R\widehat{K}L - B^T PL\right) = 0 \tag{3.18}$$

$$\left(A - B\widehat{K}\right)L + L\left(A - B\widehat{K}\right)^T + X = 0 \tag{3.19}$$

From 0

$$\widehat{K} = R^{-1}B^T PLL^{-1} = R^{-1}B^T P \tag{3.20}$$

substituting in 0 gives

$$A^T P + PA + Q - PBR^{-1}B^T P = 0. \tag{3.21}$$

This is an Algebraic Riccati Equation which can be solved using the RICTNWT (ARMSTRONG 1978) or the SB02MD (Benner and others 1999, 499-539) subroutines. Notice that 0 is not a function of $\widehat{K}$. This means that there is a direct solution of P, J and $\widehat{K}$ from 0, 0 and 0 respectively. This is a major benefit of full state feedback; there is no need for an initializing stabilizing gain or numerical solution technique. The drawback however is, that the assumption $C = I_n$ eliminates the ability to design a controller with desired structure (see Chapter 2). Nevertheless, the result of the full state feedback case is important to the LQR output feedback solution, as will be shown next.

3.2.2 Case 2: Constrained LQ Output Feedback

One of the problems with LQ output feedback is the need for an initial stabilizing gain in order to minimize the performance index. This, however, is not a problem in the case of full state feedback as shown in Sec. 3.2.1 . Constrained output feedback is another variation of the output feedback in which some of the constants in the gain matrix are forced to satisfy linear constraints during the performance index minimization. The linear constraint can vary from 'zeroing' specific gains to 'forcing relationships' between other gains. The benefits of constraints include:

1. the removal of gains that have little effect on performance for the sake of reduction in complexity and the number of gains to be scheduled;
2. the elimination of gains that couple unwanted outputs to the inputs. For example, $K_{\delta_{a/r_w}}$, which couples the yaw rate to the ailerons can be eliminated see 0.
3. the ability to truly specify any desired control structure including those which are used in classical control theory;
4. the provision of effective means to perform trade-offs between various control structures.

$$u = Ky \tag{3.22}$$

$$\begin{bmatrix} \delta_a \\ \delta_r \end{bmatrix} = \begin{bmatrix} K_{\delta_{a/r_w}} & K_{\delta_{a/p}} & K_{\delta_{a/\beta}} & K_{\delta_{a/\phi}} \\ K_{\delta_{r/r_w}} & K_{\delta_{r/p}} & K_{\delta_{r/\beta}} & K_{\delta_{r/\phi}} \end{bmatrix} \begin{bmatrix} r_w \\ p \\ \beta \\ \phi \end{bmatrix} \tag{3.23}$$

In addition to theses merits, constrained LQ output feedback gives a means of solving the initial gain problem of output feedback. The gain obtained from the full state feedback case can be used as an initial stabilizing gain and then gains corresponding to inaccessible states are zeroed Algorithms for constraint output feedback are given in (Stevens and Lewis 2003), (Choi and Sirisena 1977, 134-136) and (Shapiro, Fredricks, and Rooney 1981, 505). The method in (Shapiro, Fredricks, and Rooney 1981, 505) was chosen for this research work and will be repeated here for completeness.

Given the system defined in 0 and 0, assume C is a full rank matrix of the form

$$C = \left[I_p \vdots 0_{p \times (n-p)} \right] \tag{3.24}$$

Define

$$\bar{C} = \left[0_{(n-p) \times p} \vdots I_{n-p} \right] \tag{3.25}$$

and an augmented matrix

$$\hat{C} = \begin{bmatrix} C \\ \cdots \\ \bar{C} \end{bmatrix} = I_n \tag{3.26}$$

Then, the following control law can is chosen

$$u = -\widehat{K} C x = -\widehat{K} x \tag{3.27}$$

This is similar to the full state feedback control law in 0, therefore $\widehat{K}$ is the solution to 0 and 0 as described in Sec. 3.2.1 . The objective in this variation of constraint feedback is to use this $\widehat{K}$ as an initializing gain, $\widehat{K}_0$, for a numerical algorithm that minimizes the cost function 0 while

1. zeroing gains corresponding to inaccessible states thereby forming K in 0 (output feedback),
2. eliminating any unwanted gains in K, and
3. forcing a linear relationship between some desired gains.

In order to do this, a careful formulation of the constraints is necessary this process is as follows.

Define a vector

$$f = S(\widehat{K}) \tag{3.28}$$

where $St(\cdot)$ is a column stacking operator which converts a matrix into a vector composed of its columns stack one after the other. Since $\widehat{K}$ is a $m \times n$ matrix, vector f has dimensions, $1 \times mn$.

$$f = \left(\hat{k}_{11}, \dots, \hat{k}_{m1}, \hat{k}_{12}, \dots, \hat{k}_{m2}, \hat{k}_{13}, \dots, \hat{k}_{m3}, \dots, \dots, \hat{k}_{1n}, \dots, \hat{k}_{mn}\right)^T \tag{3.29}$$

This will allow access to the individual gains of $\widehat{K}$.

Now consider three matrices W_1, W_2 and W_3 dimensioned $q_1 \times mn$, $q_2 \times mn$, $q_3 \times mn$ respectively. A matrix W can be formed $q \times mn$ and partitioned as

$$W = \begin{bmatrix} W_1 \\ \cdots \\ W_2 \\ \cdots \\ W_3 \end{bmatrix} \tag{3.30}$$

where

$$q = q_1 + q_2 + q_3 \tag{3.31}$$

The following constraint is applied to the performance index minimization

$$Wf = d \tag{3.32}$$

with $d \in \mathbb{R}^q$. If w_i^T is the i-th row of W, then the i-th equation is

$$w_i^T f = d_i \tag{3.33}$$

Then, to set a certain gain f_{ij} to zero, only the corresponding value in $w_i^T$is set to '1' (e.g. $w_i^T = [0 \quad 0 \quad 0 \quad \dots \quad 0 \quad 1 \quad 0 \quad \dots \quad 0]$) and $d_i = 0$. As a result, W_1 is used to apply the output feedback constraint as by setting

$$W_1 = \left[0_{q_1 \times (mn - q_1)} \vdots I_{q_1}\right] \text{ and}$$
$$d_i = 0, \quad 1 \le i \le q_1 \tag{3.34}$$

where $q_1 = (n - p)m$. Similarly, W_2 is used to eliminate unwanted gains corresponding to accessible outputs by using the form

$$w_i^T = [0 \quad 0 \quad 0 \quad \dots \quad 0 \quad 1 \quad 0 \quad \dots \quad 0] \text{ and} \tag{3.35}$$

$$d_i = 0, \quad q_1 + 1 \le i \le q_1 + q_2$$

where q_2 = number of accessible gains to eliminate. Finally, W_3 is used to form linear relationships between gains and has no particular structure.

To illustrate the constraint definition, consider a system order $n = 4$, with number of inputs $m = 2$ and number of outputs $p = 3$. If the desired structure of the output gain is

$$K = \begin{bmatrix} 3k_{23} & 0 & k_{13} \\ k_{21} & k_{22} & k_{23} \end{bmatrix} \tag{3.36}$$

This implies that $k_{11} - k_{23} = 0$ and W is selected as

$$W = \begin{bmatrix} W_1 \\ \cdots \\ W_2 \\ \cdots \\ W_3 \end{bmatrix} = \begin{bmatrix} 0 & 0 & 0 & 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 \\ \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots \\ 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 \\ \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots & \cdots \\ 1 & 0 & 0 & 0 & 0 & -3 & 0 & 0 \end{bmatrix} \tag{3.37}$$

And vector d is

$$d = 0_{4\times 1} \tag{3.38}$$

With an understanding of the constraint definition, the constrained minimization method is presented. The constrained minimization for constrained LQ output feedback is written as

$$\min_{\widehat{K}} J(\widehat{K}) \tag{3.39}$$

subject to constraint

$$\xi = Wf - d = 0 \quad \text{or} \quad \|\xi\|^2 = \xi^T \xi = 0 \tag{3.40}$$

(Shapiro, Fredricks, and Rooney 1981, 505) show that this constrained minimization can be written as an equivalent dual function unconstrained minimization with a new cost,

$$H(\widehat{K}) = J(\widehat{K}) + \lambda^T \xi + \frac{1}{2}\gamma \xi^T \xi \tag{3.41}$$

where λ is a large constant and λ is a $q \times 1$ vector Lagrange multiplier. The gradient is given as

$$\frac{\partial}{\partial \widehat{K}}\left[H(\widehat{K})\right] = \frac{\partial}{\partial \widehat{K}}\left[J(\widehat{K})\right] + \sum_{j=1}^{q} \lambda_j S_m^{-1}(w_j) + \gamma S_m^{-1}(W^T W f - W^T d) \tag{3.42}$$

where $S_m^{-1}(\cdot)$ un-stacks vectors into matrices columns row m. It is good to note that

$$\left(A - B\hat{K}\right)^T P + P\left(A - B\hat{K}\right) + Q + \hat{K}^T R \hat{K} = 0 \tag{3.43}$$

$$J\left(\hat{K}\right) = \frac{1}{2} tr(PX) \tag{3.44}$$

$$\left(A - B\hat{K}\right)L + L\left(A - B\hat{K}\right)^T + X = 0 \tag{3.45}$$

$$\frac{\partial J}{\partial \hat{K}} = 2\left(R\hat{K}L - B^T PL\right) = 0 \tag{3.46}$$

and the initial gain for this minimization, $\hat{K}_0$, is the solution to the LQ full state feedback case shown in Sec. 3.2.1 .

The constrained output feedback formulation shown here solves the initial gain problem of the linear quadratic output feedback design. In addition to this, it allows the designer to give the gain matrix, K, any desired structure by eliminating gains or forcing a relationship between them. There is, however, one restriction to design a desired structure. This restriction is imposed by the assumption that $C = \left[I_p \vdots 0_{p\times(n-p)}\right]$. A method of working around this problem using similarity transformation is discussed next.

3.2.3 Case 3: Constrained LQ Output Feedback of a Similar System

In the case of constrained LQ Output Feedback, the assumption was made that the coefficient matrix, C, is a full rank matrix of rank r and in the form

$$C = \left[I_p \vdots 0_{p\times(n-p)}\right] \tag{3.47}$$

This is undesirable because it puts a restriction on the control structure that can be implemented. One example is the design of a yaw damper by applying a feedback gain to the washed-out yaw rate $r_w = \frac{s}{s+1/\tau} r$. As discussed in (Chapter 2,), the desired control law is of the form

$$u = -KCx = -K_{u_r/r_w}[0 \quad 0 \quad 1 \quad 0 \quad 0 \quad -1]x \tag{3.48}$$

where $x = [\beta \quad \phi \quad p \quad r \quad \delta_r \quad x_w]^T$ and x_w is the washout state.

The constrained LQ output feedback formulation in Sec. 3.2.2 can be expanded to any full rank C matrix by using a similarity transformation. A similarity transformation changes the coordinates, basis and eigenvectors of the system while retaining the same eigenvalues (Smith 2007). Therefore a system, 0 and 0, with any full rank matrix C, is stabilizable by a stabilizing gain for a similar system with

$$\tilde{C} = \left[I_p \vdots 0_{p\times(n-p)}\right] \tag{3.49}$$

To implement this, define the following change of coordinates using a matrix T

$$x = T^{-1}\tilde{x} \tag{3.50}$$

Substituting this into 0 and 0 and multiplying 0 by T gives

$$\dot{\tilde{x}} = TAT^{-1}\tilde{x} + TBu \tag{3.51}$$

$$y = CT^{-1}\tilde{x} \tag{3.52}$$

Defining coefficient matrices and initial condition as

$$\tilde{A} = TAT^{-1} \tag{3.53}$$

$$\tilde{B} = TB \tag{3.54}$$

$$\tilde{C} = CT^{-1} \tag{3.55}$$

Therefore, the similar system is

$$\dot{\tilde{x}} = \tilde{A}\tilde{x} + \tilde{B}u \tag{3.56}$$

$$y = \tilde{C}\tilde{x} \tag{3.57}$$

Note that since the value of the output y is not affected by the transformation, the control laws for both systems are equivalent. That means

$$u = -Ky = -KCx \equiv -K\tilde{C}\tilde{x} \tag{3.58}$$

Since the goal is for $\tilde{C} = \left[I_p \vdots 0_{p\times(n-p)}\right]$, 0 gives

$$CT^{-1} = \left[I_p \vdots 0_{p\times(n-p)}\right] \tag{3.59}$$

There is a non-unique solution

$$T = \begin{bmatrix} C \\ \cdots \\ E \end{bmatrix} \tag{3.60}$$

where E is any matrix that makes $rank[T] = n$. If this is satisfied, the constrained output feed-back equations from Sec. 3.2.2 can be used. Note that the cost function for this formulation is

$$J = \frac{1}{2}\int_0^\infty (\tilde{x}^T \tilde{Q}\tilde{x} + u^T R u)\,dt, \qquad \tilde{X} = \tilde{x}_0 \tilde{x}_0^T \tag{3.61}$$

However since the desired cost function is of the form

$$J = \frac{1}{2}\int_0^\infty (x^T Q x + u^T R u)\,dt, \qquad X_0 = x_0 x_0^T \tag{3.62}$$

select

$$\tilde{Q} = (T^{-1})^T Q T^{-1} \tag{3.63}$$

$$\tilde{X}_0 = T x_0 x_0^T T^T = T X_0 T^T \tag{3.64}$$

Therefore we obtain

$$J = \frac{1}{2}\int_0^\infty (\tilde{x}^T (T^{-1})^T \tilde{Q} T^{-1}\tilde{x} + u^T R u)\,dt = \frac{1}{2}\int_0^\infty (x^T Q x + u^T R u)\,dt, \tag{3.65}$$

This formulation of the constrained LQ output feedback problem solves the problem with determining the initial gain and shall be used in the proposed algorithm as shown later. There is still the issue of selection the weighting matrices R and Q. A method for selection of weighting matrices suitable for conceptual design is discussed next.

3.2.4 Selecting a Suitable Weighting Function for Conceptual Design

For a performance index such as 0, the weighting matrices Q and R are the means of posing the controls problem. The entries of theses matrices place penalties on the different states on their corresponding vectors. This means that the selection of the values in the matrix define the minimization problem and thus the performance of the system.

$$J = \frac{1}{2}\int_0^\infty (x^T \tilde{Q} x + u^T R u)\,dt, \tag{3.66}$$

Using the entries of the Q matrix, one can penalize the yaw rate over the bank angle, for example. Using values of the R matrix, one can favor rudder response over aileron response, for example. Since the performance index is additive, it is the relative magnitudes that determine the penalties. As discussed at the beginning of Sec. 3.2 , there are restrictions on the values of the weighing matrices. These restrictions are:

1. the control weighing matrix R is positive definite (i.e. all the inputs are weighed);

2. the state weight matrix Q is positive semi-definite and $\left(\sqrt{Q},\ A\right)$ is detectable.

This means that a great deal of engineering judgment has to go into the selection of these matrices. In fact, the selection of these matrices could take a great deal of trial and error or lead to choices that have no physical significance (Stevens and Lewis 2003). This will defeat the benefits for choosing this method for conceptual design. The idea, therefore, is to use these indices to pose a problem with physical significance and requires a little amount of trial and error.

There are various methods available for selecting weighting matrices and some are given in (Stevens and Lewis 2003). One of them that is suitable for this research undertaking since it eliminates the need for observability in the Q matrix, thereby providing more freedom. This method is the time-dependent weighting and it is discussed next.

3.2.5 Time-Dependent Weighting

In time-dependent weighting, the matrix Q is multiplied by an extra term, t^k, in the performance index. This gives the form

$$J = \frac{1}{2}\int_0^\infty \left(t^k x^T Q x + u^T R u\right) dt. \tag{3.67}$$

Skipping the derivation, the results which are shown in (Stevens, Lewis, and Al-Sunni 1992, 238-Feb.) give the equations for the performance index as

$$\begin{gathered} 0 = g_0 \equiv A_c^T P_0 + P_0 A_c + Q \\ 0 = g_1 \equiv A_c^T P_1 + P_1 A_c + P_0 \\ \vdots \\ 0 = g_{k-1} \equiv A_c^T P_{k-1} + P_{k-1} A_c + P_{k-2} \\ 0 = g_k \equiv A_c^T P_k + P_k A_c + k!\, P_{k-1} + C^T K^T R K C \end{gathered} \tag{3.68}$$

$$\begin{gathered} 0 = A_c L_k + L_k A_c^T + X \\ 0 = A_c L_{k-1} + L_{k-1} A_c^T + k!\, L_k \\ 0 = A_c L_{k-2} + L_{k-2} A_c^T + L_{k-1} \\ \vdots \end{gathered} \tag{3.69}$$

$$0 = A_c L_0 + L_0 A_c^T + L_1$$

with the performance index and gradients given with

$$J = \frac{1}{2} x^T P_k x = \frac{1}{2} tr(P_k X) \tag{3.70}$$

$$\frac{\partial J}{\partial \hat{K}} = 2[RKCL_k C^T - B^T(P_0 L_0 + \cdots + P_k L_k)C^T] = 0 \tag{3.71}$$

Note that these equations are for the standard LQR output feedback problem and not for the full-state feedback.

The time-dependent weighting method eliminates the observability restriction on Q as long as $k > 1$ (Stevens, Lewis, and Al-Sunni 1992, 238-Feb.; Boukas and Liu 2002, 49-65) , thus allowing more freedom in the selection of the weighting matrices Q and R.

3.2.6 Selected Structure of Q and R Matrices

The structure of the performance index chosen from the options listed in (Stevens and Lewis 2003) is

$$J = \tfrac{1}{2}\int_0^\infty (t^2 y^T y + \rho u^T u)\, dt, \tag{3.72}$$

where y is the desired output and ρ is a constant. This implies

$$Q = C^T C \tag{3.73}$$

$$R = \rho I_m \tag{3.74}$$

Two primary considerations justify this selection. Firstly, the goal of the Stability Augmentation System is to minimize the final states of the output. Therefore, only the output states need to be included in the cost function. Secondly, from a conceptual design perspective, the smaller the number of variables required to tune performance, the better. In this selection, only ρ is required for performance tuning while the observability issues, that might occur in this sort of formulation, is removed by the time-dependent weighting (with $k = 2$). This approach posses physical in-sight, it is efficient and examples in (Stevens and Lewis 2003) show it produces good results. For additional tuning flexibility, a weighting can be selected as

$$Q = C^T \hat{Q} C \tag{3.75}$$

$$R = \rho I_m \tag{3.76}$$

and k can be varied. This gives more design flexibility where the ratio of the weighting on each state can be varied individually. This formulation might not be necessary but it is available. With all the issues of the LQ output feedback resolved, an algorithm of a flight control system design tool can be proposed.

3.3 Algorithm for LQ Output Feedback Control Design Suitable for Conceptual Design

The algorithm proposed is based on (Shapiro, Fredricks, and Rooney 1981, 505) but it incorporates all the elements discussed in this chapter which are not in that text.

Step 1:

Input the matrices $A(n \times n)$, $B(n \times m)$, $C(p \times n)$, $X(n \times n)$, $W(p \times n)$; vector $d(q \times 1)$; integer k and scalars, ρ, β, ϵ, M. (Note, β, ϵ, M are used for the iteration of the constraint)

Step 2:

Form weighting matrices

$$Q = C^T C \tag{3.77}$$

$$R = \rho I_m \tag{3.78}$$

Step 3:

Determine a matrix $E\big((n-p) \times n\big)$, such that $rank[T] = n$. And Form

$$T = \begin{bmatrix} C \\ \cdots \\ E \end{bmatrix} \tag{3.79}$$

Step 4:

Perform the change of coordinates

$$\tilde{A} = TAT^{-1} \tag{3.80}$$

$$\tilde{B} = TB \tag{3.81}$$

$$\tilde{C} = CT^{-1} \tag{3.82}$$

$$\tilde{Q} = (T^{-1})^T Q T^{-1} \tag{3.83}$$

$$\tilde{X} = T X_0 T^T \tag{3.84}$$

Step 5:

Initialize counter $i = 0$. Set the Lagrange multiplier vector $\lambda(0) = 0_{q\times 1}$ and select $\lambda(0)$ to a large positive scalar.

Step 6:

Use a Riccati solver to solve for P_k from

$$\tilde{A}^T P + P\tilde{A} + \tilde{Q} - P\tilde{B}R^{-1}\tilde{B}^T P = 0. \tag{3.85}$$

and solve for the initial gain $\hat{K}(0)$ from

$$\hat{K} = R^{-1}\tilde{B}^T P \tag{3.86}$$

Step 7:

Form

$$\xi(i) = Wf(i) - d \tag{3.87}$$

If $\xi^T(i)\xi(i) < \epsilon$ go to Step12. Else go on.

Step 8:

Perform the inner loop minimization with respect to K using Davidon-Fletcher-Powell Algorithm (remember to limit step such that K stabilizes), the cost function is given by

$$H\left[\hat{K}(i)\right] = J\left[\hat{K}(i)\right] + \lambda^T\xi(i) + \frac{1}{2}\gamma\xi^T(i)\xi(i) \tag{3.88}$$

where

$$J\left[\hat{K}(i)\right] = \frac{1}{2}tr[P_k(i)\tilde{X}] \tag{3.89}$$

and $P_k(i)$ is solved from

$$\begin{aligned} 0 = g_0 &\equiv \left[\tilde{A} - \tilde{B}\hat{K}(i)\right]^T P_0 + P_0\left[\tilde{A} - \tilde{B}\hat{K}(i)\right] + \tilde{Q} \\ 0 = g_1 &\equiv \left[\tilde{A} - \tilde{B}\hat{K}(i)\right]^T P_1 + P_1\left[\tilde{A} - \tilde{B}\hat{K}(i)\right] + P_0 \\ &\vdots \end{aligned} \tag{3.90}$$

$$0 = g_{k-1} \equiv \left[\tilde{A} - \tilde{B}\hat{K}(i)\right]^T P_{k-1} + P_{k-1}\left[\tilde{A} - \tilde{B}\hat{K}(i)\right] + P_{k-2}$$

$$0 = g_k \equiv \left[\tilde{A} - \tilde{B}\hat{K}(i)\right]^T P_k + P_k\left[\tilde{A} - \tilde{B}\hat{K}(i)\right] + k!\,P_{k-1} + \hat{K}^T(i)R\hat{K}(i)$$

The gradient of $H\left[\hat{K}(i)\right]$ with respect to $\hat{K}(i)$ is given by

$$\frac{\partial}{\partial \hat{K}}\left\{H\left[\hat{K}(i)\right]\right\} = \frac{\partial}{\partial \hat{K}}\left\{J\left[\hat{K}(i)\right]\right\} + \sum_{j=1}^{q} \lambda_j(i) S_m^{-1}\left[w_j\right] + \gamma(i) S_m^{-1}\left[W^T \xi(i)\right] \tag{3.91}$$

and

$$\frac{\partial}{\partial \hat{K}}\left\{J\left[\hat{K}(i)\right]\right\} = 2\left[R\hat{K}(i)L_k - B^T(P_0 L_0 + \cdots + P_k L_k)\right] = 0 \tag{3.92}$$

where

$$\begin{aligned}
0 &= \left[\tilde{A} - \tilde{B}\hat{K}(i)\right]L_k + L_k\left[\tilde{A} - \tilde{B}\hat{K}(i)\right]^T + \tilde{X} \\
0 &= \left[\tilde{A} - \tilde{B}\hat{K}(i)\right]L_{k-1} + L_{k-1}\left[\tilde{A} - \tilde{B}\hat{K}(i)\right]^T + k!\,L_k \\
0 &= \left[\tilde{A} - \tilde{B}\hat{K}(i)\right]L_{k-2} + L_{k-2}\left[\tilde{A} - \tilde{B}\hat{K}(i)\right]^T + L_{k-1} \\
&\;\;\vdots \\
0 &= \left[\tilde{A} - \tilde{B}\hat{K}(i)\right]L_0 + L_0\left[\tilde{A} - \tilde{B}\hat{K}(i)\right]^T + L_1
\end{aligned} \tag{3.93}$$

Increment the counter for i by setting $i \to i + 1$, and denote the solution to the inner loop minimization by $\hat{K}(i)$.

Step 9:

Update the Lagrange multiplier vector according to

$$\lambda(i) = \lambda(i-1) + \gamma(i-1)\xi(i-1) \tag{3.94}$$

Step 11:

If

$$\frac{\|\xi(i-1)\|^2}{\|\xi(i)\|^2} > M \tag{3.95}$$

Then

$$\gamma(i) = \gamma(i-1) \tag{3.96}$$

Else

$$\gamma(i) = \beta\gamma(i-1) \tag{3.97}$$

Step 11:

Go to Step 7

Step 12:

$$K = \widehat{K}(i) \tag{3.98}$$

A summary of this algorithm is shown in Nassi-Schneiderman diagram format in Figure 3.1. Nassi-Schneiderman plots are a clear and concise way to display linear programming; see description in (Coleman 2007, 283). In the next section, the implementation and validation of standalone codes written for each of the different cases are discussed.

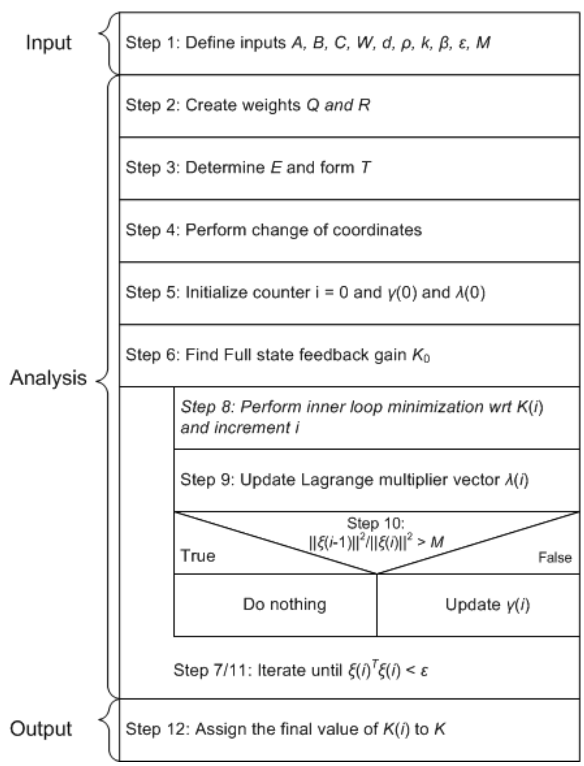


Figure 3.1 Summary of algorithm for constrained output feedback of similar system

3.4 Validation of Cases

The different cases discussed in the previous sections have been programmed in FORTRAN for this research endeavor. The SB02MD and SB03MD subroutines from (Benner and others 1999, 499-539) are used to solve the Riccati and Lyapunov equations respectively. While, the Dfpmin and lnsrch subroutines from (Press 2007) are both used for gradient minimization. Additionally, Dfpmin is modified such that each update of $\hat{K}(i)$ stabilizes the system. In order to validate the subroutines individually, similar examples from available texts are used for comparison. The algorithm is programmed progressively with features of each new case added at a time. The results are discussed in like manner.

3.4.1 Validation of Case 1: Full state feedback

The full state feedback FORTRAN subroutine written is called "STATE_FEED". The required inputs are, integers n and m, plus matrices $A(n \times n)$, $B(n \times m)$, $Q(n \times n)$, $R(m \times m)$; and resulting in the matrix $K(m \times n)$ as its output. The validation example chosen for this subroutine is from (Stevens and Lewis 2003). The system given is

$$\dot{x} = Ax + Bu = \begin{bmatrix} 0 & 1 \\ 0 & 0 \end{bmatrix} x + \begin{bmatrix} 0 \\ 1 \end{bmatrix} u \tag{3.99}$$

with a performance index

$$J = \frac{1}{2} \int_0^\infty (x^T Q x + u^2)\, dx. \tag{3.100}$$

The weights are

$$Q = \begin{bmatrix} q_d^2 & 0 \\ 0 & q_v \end{bmatrix} \tag{3.101}$$

$$R = 1 \tag{3.102}$$

An algebraic weighting was selected for this special case because an analytical solution is possible. The resulting optimal gain given in the text is

$$K = \left[q_d \quad \sqrt{2}\sqrt{q_d + \frac{q_v}{2}}\right]. \tag{3.103}$$

The STATE_FEED subroutine however, only computes numerical values. Therefore the numerical values are selected for q_d and q_v. The results are compared in Table 3.1.

Table 3.1 Validation results for "STATE_FEED" subroutine

$\boldsymbol{q_d}$	$\boldsymbol{q_v}$	$\boldsymbol{K = \left[q_d \quad \sqrt{2}\sqrt{q_d + \frac{q_v}{2}}\right]}$	**STATE_FEED**
1	1	$[1 \quad 1.7321]$	$[1 \quad 1.7321]$
1	2	$[1 \quad 2]$	$[1 \quad 2]$
2	1	$[2 \quad 2.2361]$	$[2 \quad 2.2361]$
10	1	$[10 \quad 4.5826]$	$[10 \quad 4.5826]$
1	10	$[1 \quad 3.4641]$	$[1 \quad 3.4641]$

The “STATE_FEED” subroutine results are equal to the results of the validation example. It can be concluded that the algorithm syntax is programmed correctly. The next program is the pure output feedback case described in Sec. 3.2 . It is the next logical progression, but since it has not been given a case number, it will be called case 1.5.

3.4.2 Validation Case 1.5: Pure Output Feedback

The output feedback subroutine written is called “OUT_FEED”. The required input are the integers n, m, and p, plus the matrices $A(n \times n)$, $B(n \times m)$, $C(p \times n)$, $Q(n \times n)$, $R(m \times m)$, $X(n \times n)$, $K_0(m \times p)$. The algorithm produces the matrix $K(m \times p)$ and the cost J as its output. The example chosen for validation is given in (Choi and Sirisena 1974, 257-258). The system is

$$\dot{x} = \begin{bmatrix} -0.037 & 0.0123 & 0.00055 & -1.0 \\ 0 & 0 & 1.0 & 0 \\ -637 & 0 & -0.23 & 0.0618 \\ 1.25 & 0 & 0.016 & -0.0457 \end{bmatrix} x + \begin{bmatrix} 0.00084 & 0.000236 \\ 0 & 0 \\ 0.08 & 0.804 \\ -0.0862 & -0.0665 \end{bmatrix} u \tag{3.104}$$

$$y = \begin{bmatrix} 0 & 1 & 0 & 0 \\ 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 \end{bmatrix} x \tag{3.105}$$

with weighting matrices, initial condition and initial gain as

$$Q = I_4, \qquad R = I_4, \qquad X_0 = I_4, \qquad K_0 = 0_{2\times 3} \tag{3.106}$$

The results are compared in Table 3.2

Table 3.2 Validation results for “OUT_FEED” subroutine

	(Choi and Sirisena 1974, 257-258)	**OUT_FEED**
$\boldsymbol{J_0}$	15568	15567.57
$\boldsymbol{J_{final}}$	79.56	79.53
$\boldsymbol{K_{final}}$	$\begin{bmatrix} -0.36 & -1.53 & -7.61 \\ 1.27 & 3.54 & 5.06 \end{bmatrix}$	$\begin{bmatrix} -0.398 & -1.59 & -7.852 \\ 1.257 & 3.48 & 5.004 \end{bmatrix}$

The results from the example and the subroutine correlate well with an acceptable maximum error of about 10%. This error can be attributed to using different computers, precision settings and tolerances. The subroutine has, however, demonstrated a validated syntax. With the following example, the constrained output feedback case is validated.

3.4.3 Validation Case 2: Constrained Output Feedback

The constrained output feedback subroutine written is called “CON_FEED”. Input are the integers n, m, qi, plus the matrices $A(n \times n)$, $B(n \times m)$, , $Q(n \times n)$, $R(m \times m)$, $X(n \times n)$, $W(qi \times mn)$, $d(qi \times 1)$; it produces the matrix $\widehat{K}(m \times n)$ and the cost H on output. The structure of matrix $\widehat{K}(m \times n)$ depends on the constraints imposed by the W and D matrices (see Sec.3.2.2). The example chosen for its validation is given in (Choi and Sirisena 1977, 134-136). The system is

$$\dot{x} = \begin{bmatrix} -0.154 & 0.004 & -0.990 & 0.178 & 0.075 \\ -1.250 & -2.850 & 1.430 & 0 & -0.727 \\ 0.568 & -0.277 & -0.284 & 0 & -2.050 \\ 0 & 1.0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & -10.0 \end{bmatrix} x + \begin{bmatrix} 0.075 \\ -0.727 \\ -2.050 \\ 0 \\ -10.0 \end{bmatrix} u \tag{3.107}$$

$$y = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 \end{bmatrix} x \tag{3.108}$$

with weighting matrices and initial condition as

$$Q = I_5, \qquad R = I_1, \qquad X_0 = I_5 \tag{3.109}$$

There is also an initial stabilizing gain given as

$$\widehat{K}_0 = [0.976 \quad 0.054 \quad -0.848 \quad -0.175 \quad 0]. \tag{3.110}$$

Although the formulation shown in (Sec.3.2.2) does not require an initial stabilizing gain; the availability of an initial gain in the example gives an extra data point for validation. The desire control structure is

$$\widehat{K} = [k_{11} \quad k_{12} \quad k_{13} \quad 0 \quad 0]. \tag{3.111}$$

Therefore, the constraint is

$$Wf - d = \begin{bmatrix} 0 & 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 \end{bmatrix} f - 0 = 0 \tag{3.112}$$

The results are compared in Table 3.3.

Table 3.3 Validation results for “CON_FEED” subroutine

	(Choi and Sirisena 1977, 134-136)	**CON_FEED using $\widehat{K}_0$**	**CON_FEED w/o $\widehat{K}_0$**
$\boldsymbol{H_0}$	7.37	7.3701	2.852
$\boldsymbol{H_{final}}$	5.6869	5.6869	5.6869
$\boldsymbol{K_{final}}$	$[-0.127 \quad -0.788 \quad 1.215]$	$[-0.127 \quad -0.790 \quad -1.215 \quad 0$	$[-0.127 \quad -0.790 \quad -1.214 \quad 0$

These results of both tests correlate well with the example results including the test not using the initial gain from the example. This validates the CON_FEED subroutine. The next case for validation is that of the constrained feedback of a similar system.

3.4.4 Validation Case 3: Constrained Output Feedback of a Similar System

The similar system constrained output feedback subroutine written is called “SIM-CON_FEED”. Input are the integers n, m, p, , qi, plus the matrices $A(n \times n)$, $B(n \times m)$, $C(p \times n)$, $E\big((n-p) \times n\big)$, $Q(n \times n)$, $R(m \times m)$, $X(n \times n)$, $W(qi \times mn)$, $d(qi \times 1)$; it produces the matrix $\widehat{K}(m \times n)$ and the cost H as its output. The structure of matrix $\widehat{K}(m \times n)$ depends on the constraints imposed by W and D matrices (see Sec.3.2.2). The change of coordinates does not affect the structure of $\widehat{K}$, recall that

$$u = -Ky = -KCx \equiv -K\tilde{C}\tilde{x} \tag{3.113}$$

Hence, if the constraint and change of coordinates are applied properly, then the gains obtained should be adequate.

The example for the validation the SIMCON_FEED subroutine is given from (Stevens and Lewis 2003). The objective of this problem is to design a lateral regulator. The system is

$$A = \begin{bmatrix} -0.3220 & 0.0640 & 0.0364 & -0.9917 & 0.0003 & 0.0008 & 0.0000 \\ 0.0000 & 0.0000 & 1.0000 & 0.0037 & 0.0000 & 0.0000 & 0.0000 \\ -30.6492 & 0.0000 & -3.6784 & 0.6646 & -0.7333 & 0.1315 & 0.0000 \\ 8.5396 & 0.0000 & -0.0254 & -0.4764 & -0.0319 & -0.0620 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 & -20.2000 & 0.0000 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 0.0000 & 0.0000 & -20.2000 & 0.0000 \\ 0.0000 & 0.0000 & 0.0000 & 57.2958 & 0.0000 & 0.0000 & -1.0000 \end{bmatrix} \tag{3.114}$$

$$B = \begin{bmatrix} 0 & 0 \\ 0 & 0 \\ 0 & 0 \\ 0 & 0 \\ 20.2 & 0 \\ 0 & 20.2 \\ 0 & 0 \end{bmatrix}$$

where $x = [\beta \quad \phi \quad p \quad r \quad \delta_a \quad \delta_r \quad x_e]^T$ and $u = [u_a \quad u_r]$. It is desired for the feedback variables to be $y = [r_w \quad p \quad \beta \quad \phi]^T$. Therefore,

$$C = \begin{bmatrix} 0 & 0 & 0 & 57.2958 & 0 & 0 & -1 \\ 0 & 0 & 57.2958 & 0 & 0 & 0 & 0 \\ 57.2958 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 57.2958 & 0 & 0 & 0 & 0 & 0 \end{bmatrix} \tag{3.115}$$

The weights are

$$Q = diag\{50, 100, 100, 50, 0, 0, 1\} \tag{3.116}$$

$$R = 0.1I_2, \tag{3.117}$$

One of the variables used for feedback is the washed-out yaw rate, r_w. As discussed previously, this structure is difficult to model with full state feedback (Sec. 3.2.1) or with the regular formulation of the constrained output feedback (Sec. 3.2.2). However, it is possible by performing a transformation to a similar system with

$$E = \begin{bmatrix} 0 & 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 1 \end{bmatrix}. \tag{3.118}$$

There are two control structures in the text, hence there are two validation points. The structures are (note, only the output gains are shown, the rest will be constrained to zero)

$$K_1 = \begin{bmatrix} k_{11} & k_{12} & k_{13} & k_{14} \\ k_{21} & k_{22} & k_{23} & k_{24} \end{bmatrix} \tag{3.119}$$

$$K_2 = \begin{bmatrix} 0 & k_{12} & 0 & k_{14} \\ k_{21} & 0 & k_{23} & 0 \end{bmatrix} \tag{3.120}$$

The reasoning behind the second structure is to reduce the number of gains to schedule and to eliminate the aileron response to sideslip and yaw rate, in addition to the rudder response to bank and roll rate. This ability to eliminate cross coupling from the gains is one of the reasons output feedback was selected as the most favorable design technique (see. Chapter 2).

The constraints chosen are

$$W = [0_{6\times8} \vdots I_6], \qquad d = 0_{6\times1} \tag{3.121}$$

$$W = \begin{bmatrix} 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 \\ 1 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & 0 \end{bmatrix}, \qquad d = 0_{10\times1} \tag{3.122}$$

The results of the first test are given in Table 3.4.

Table 3.4 Validation results for “SIMCON_FEED” subroutine on gain structure 1

	(Stevens and Lewis 2003)	**SIMCON_FEED**
$\boldsymbol{K_{final}}$	$\begin{bmatrix} -0.56 & -0.44 & 0.11 & -0.35 \\ -1.19 & -0.21 & -0.44 & 0.26 \end{bmatrix}$	$\begin{bmatrix} -0.59 & -0.41 & 0.24 & -0.28 \\ -0.87 & 0.10 & -0.21 & 0.46 \end{bmatrix}$

The gains do no match well, most of them are of similar magnitude but some of them are of different signs. However, consider the plot, shown in Figure 3.1, of the responses of both systems (i.e. $\dot{x} = (A - BKC)x$) to an initial condition of $x(0) = [1 \quad 0 \quad 0 \quad 0 \quad 0 \quad 0 \quad 0]^T$. The responses are identical. This is shows that different combinations of the feedback gains can create the same system response.

The results for the second control structure using 0 are shown in Table 3.5. The gains in this case are very close and they all have the same signs. This is a better correlation probably because there is a fewer number of gains selected, hence there are a fewer number of combinations possible to produce similar responses. The system response plots to a 1 degree sideslip initial condition are shown in Figure 3.3. As in the previous case, the two systems are identical. These new responses are, however, different from the previous case. With the second structure, the bank and roll rate have smaller peaks and the oscillations settle quicker compared to the first structure. The sideslip and yaw rates have larger peaks compared to the first structure while settling quicker. Two possible reasons for these differences are:

1. The elimination of the cross coupling gains reduce the lateral responses to sideslip while increasing the relaxing of the directional disturbance.
2. The effects of weighting ϕ and r by 100 and β and p by 50 is more visible in the structure two than in one.

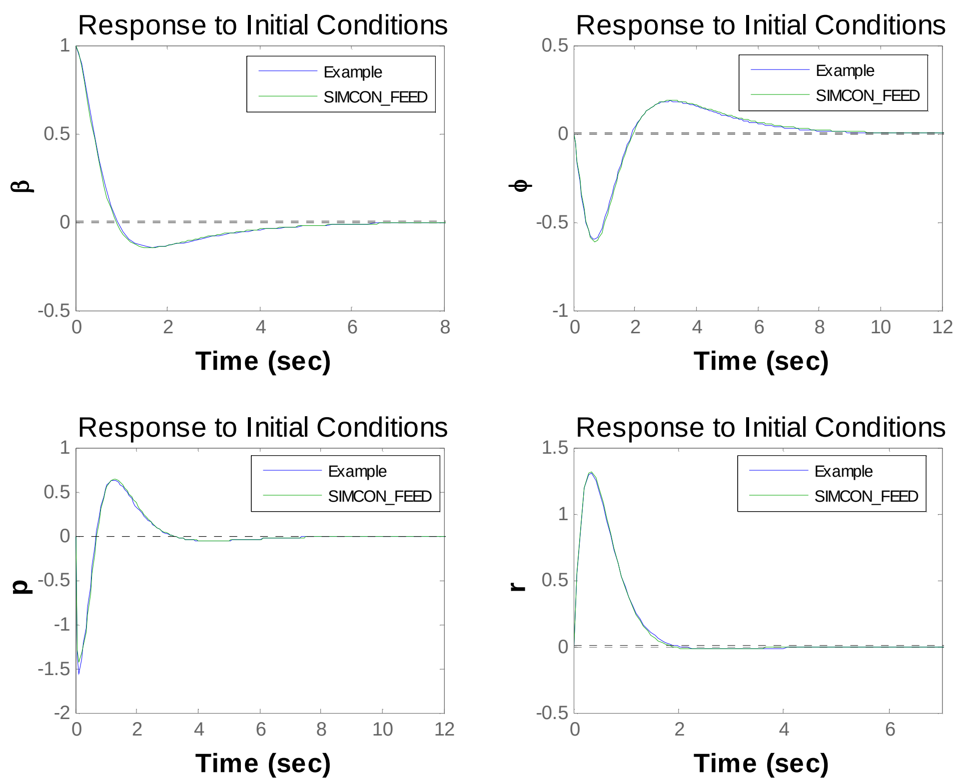


Figure 3.2 Responses of example and SIMCON_FEED systems to the same initial condition

Table 3.5 Validation results for “SIMCON_FEED” subroutine on gain structure 2

	(Stevens and Lewis 2003)	**SIMCON_FEED**
$\boldsymbol{K_{final}}$	$\begin{bmatrix} 0 & -0.55 & 0 & -0.49 \\ -1.14 & 0 & 0.05 & 0 \end{bmatrix}$	$\begin{bmatrix} 0 & -0.56 & 0 & -0.50 \\ -1.18 & 0 & 0.11 & 0 \end{bmatrix}$

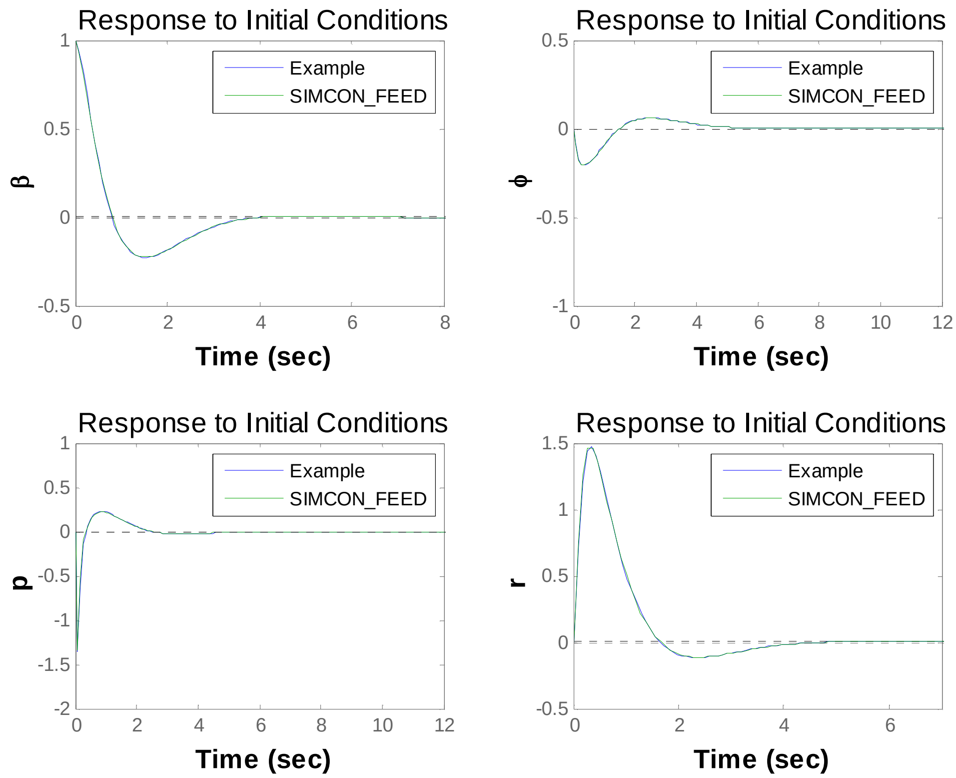


Figure 3.3 Responses of example and SIMCON_FEED systems to the same initial condition

The results of this example validate the "SIMCON_FEED" subroutine. This example also shows that controller design is not trivial, and that the quality of the results depend greatly on how well the problem is posed.. In the next section, an example is given on the suggested performance index which helps to reduce the ambiguity in modern control design.

3.4.5 Validation Case 4: Full Algorithm

In all validation examples up to this point, the weighting matrices have been given. As discussed in Sec. 3.2.4 , the selection of the weighting matrices defines the controls design problem. The major problem with modern controls is the ambiguity in selecting the weighting matrices as discussed in (Abzug and Larrabee 2002; Stevens and Lewis 2003; McRuer, Ashkenas, and Graham 1974; Roskam 2003). The authors of the examples in this chapter chose weighting matrices as identity matrices except for Case 3. In that text, Lewis *et al* explains that the “deficiency is that it was necessary to juggle the entries of Q to obtain a good solution”, but time weighting is suggested as a solution to this problem.

The algorithm in Sec. 3.3 includes this time weighting scheme. The subroutine written with this algorithm is called “TIME_SIMCON_FEED”. The weighting matrices are

$$Q = C^T C \tag{3.123}$$

$$R = \rho I_m \tag{3.124}$$

With this formulation, the only design variable is ρ and the selection of Q is reasonable because the goal of the regulator is to minimize the output states giving the performance index as

$$J = \frac{1}{2}\int_0^\infty (t^2 y^T y + \rho u^T u)\, dt \tag{3.125}$$

There is no similar example in all texts reviewed; therefore the previous example (Sec. 3.4.4) is used for comparison. After comparing the results generated with various ρ, $\rho = 2$ was chosen. The corresponding gain is shown in Table 3.6. Figure 3.4 and Figure 3.5 compare the time response of a system stabilized by this gain to those designed with a similar structure in Sec. 3.4.4 . The initial conditions used are $x(0) = [1 \quad 0 \quad 0 \quad 0 \quad 0 \quad 0 \quad 0]^T$ and $x(0) = [0 \quad 1 \quad 0 \quad 0 \quad 0 \quad 0 \quad 0]^T$ that is $\beta_0 = 1$ and $\phi_0 = 1$ respectively.

Table 3.6 results for “TIME_SIMCON_FEED”

$\boldsymbol{\rho}$	$\boldsymbol{K}$
2	$\begin{bmatrix} 0 & -0.57 & 0 & -1.58 \\ -0.10 & 0 & 1.00 & 0 \end{bmatrix}$

The new system has a quicker response, both to the initial sideslip and to the initial bank angle while maintaining a similar pick response. In addition, there is minimal work required to obtain a good design since only one variable has to be tuned. This is the advantage of time weighting; it gives the ability to specify the design problem and produces decent results. The drawback, however, is that the gains are suboptimal and there are more than likely gain-combinations which could produce better performance. Although, this has to be a problem reserved for preliminary design, where the goal is to optimize vehicle performance. In contrast, the conceptual design phase does strive not for accuracy but correctness. The algorithm implemented can be used to quickly determine a feasible system and to establish a baseline for preliminary design, hence, bridging the gap between PD and CD.

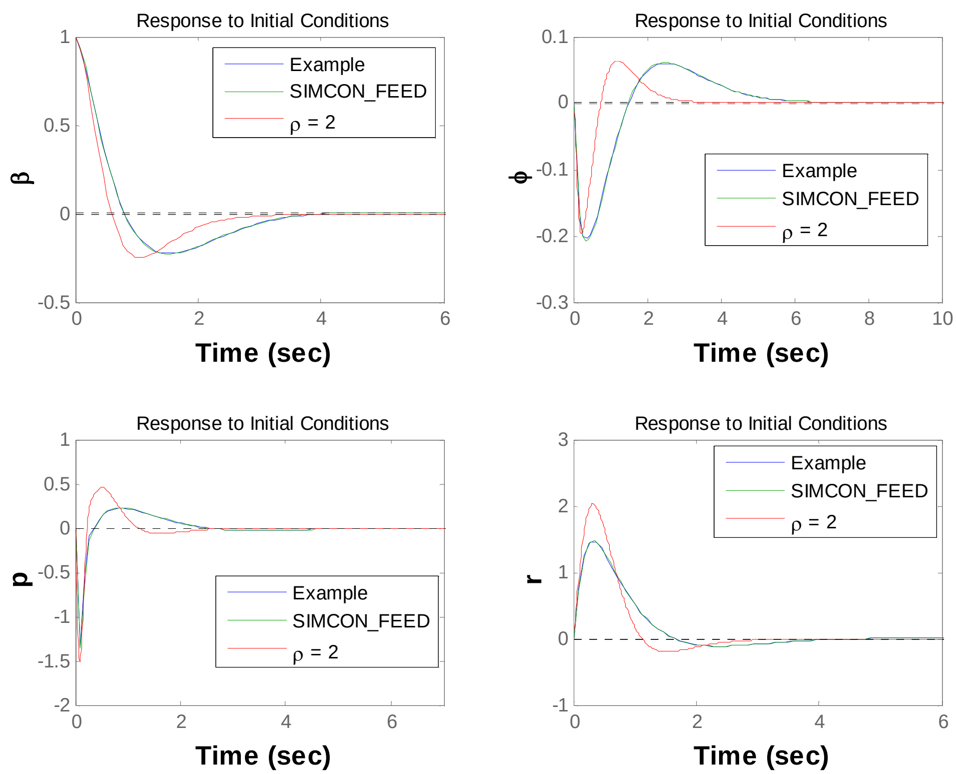


Figure 3.4 Response of system design with proposed algorithm to initial sideslip

A final statement is necessary about what is dimmed as “a good response”. As discussed in chapter 1, the introduction of control feedback increases the order of the system thereby invalidating the use of conventional regulations such as (McGraw-Hill 2004). That is, the regular phugoid, short period, dutch-roll, roll and spiral modes are skewed with the controller dynamics thereby making them difficult to cross reference with regulations. As a result of this various handling qualities criteria (e.g. (Gibson 1995)) have been developed to evaluate these higher order systems. The controller gain design technique presented here lays the foundation for implementing some of these handling qualities criteria at the Conceptual Design level.

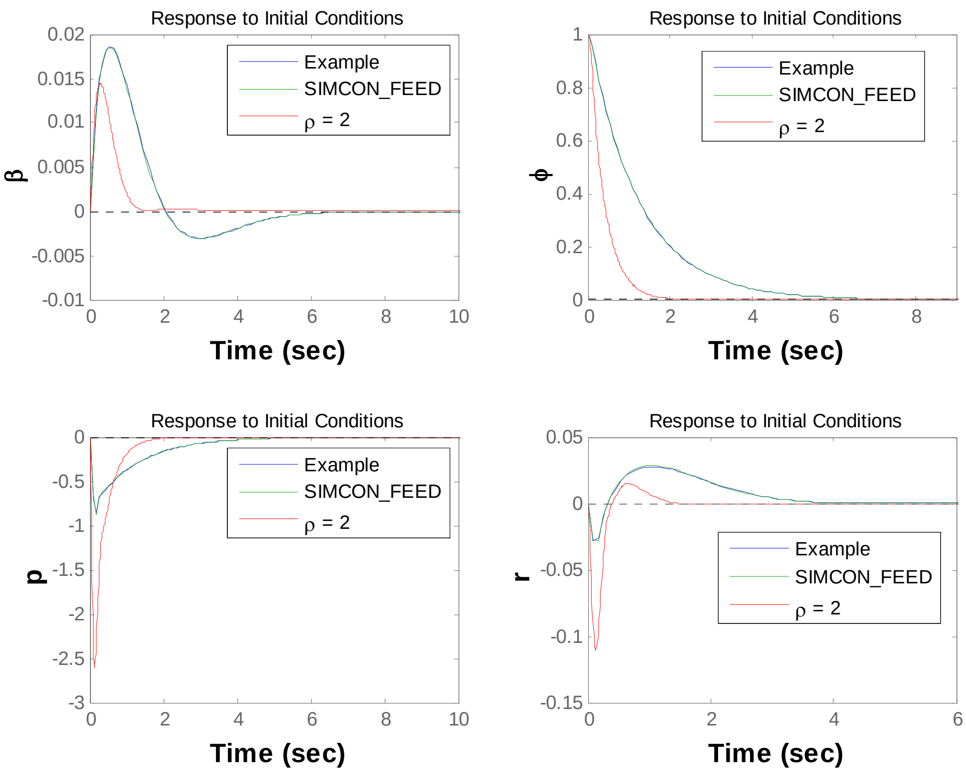


Figure 3.5 Response of system design with proposed algorithm to initial bank angle

3.5 Chapter Summary

The goal of this chapter has been to establish the theory behind output feedback and propose a CD-practical algorithm for a linear quadratic output feedback design. The theory has been established by outlining the overall aim and the deficiencies of alternative technique. Then, different fixes to the deficiencies are given until the final design process is in compliance with the specification and implemented via an algorithm. This algorithm is programmed in FORTRAN and validated with test-examples available with each addition of a new element. The final results show how this system can be used to systematically solve a non trivial controls problem. It can be concluded that this methodology might not produce the optimum design required in a preliminary design setting, but it is a sufficient pointer from a conceptual design standpoint.

CHAPTER 4

INTEGRATION OF FLIGHT CONTROL SYSTEM MODULE INTO AEROMECH

The objective of this research undertaking is to augment the current flight control system design module available in *AeroMech* by integrating a modern control technique of practical value during the conceptual design phase. In previous chapters, constrained output feedback with time weighting has been selected, programmed and validated as a standalone module for this purpose. In order to examine its practicality during the conceptual design phase, this module is to be integrated into the *AeroMech* environment and tested. This chapter gives an overview of the *AeroMech* methodology, source code and integration approach selected into *AeroMech*.

4.1 *AeroMech* Methodology and Source Code Overview

As stated in Chapter 1, *AeroMech* is both a methodology and software for stability and control analysis during the conceptual design phase. The goal of the system has been from the outset to provide a means for adequately sizing control effectors of flight vehicle design alternatives. The initiator, Chudoba, envisioned a tool that is vehicle configuration independent and consistent throughout the speed range (Chudoba and others 2008, 293). The objectives of *AeroMech* are to

1. assess control power at design constraining flight conditions (DCFCs) identified throughout the flight envelop for adequately sizing control effectors,
2. determine trimmed aerodynamics for performance estimations at any desired flight condition,
3. evaluate static and dynamic stability for the verification of safety requirements.

The methodology has been developed, followed by its implementation into an executable software application that is continually being refined. The methodology and present version of the source code are discussed next.

4.1.1 AeroMech Methodology Description

The *AeroMech* methodology is discussed in detail in (Coleman 2007, 283)(Chudoba 2001; Chudoba and others 2008, 293). Figure 4.1 shows an outline of the methodology modules. The software consists of six modules: (1) input definition; (2) aerodynamic prediction; (3) steady state control analysis; (4) trimmed aerodynamics estimation; (5) static and dynamic stability analysis; (6) output presentation. These modules are described in the following sections as well as additions to the static and dynamic analysis modules and output organization.

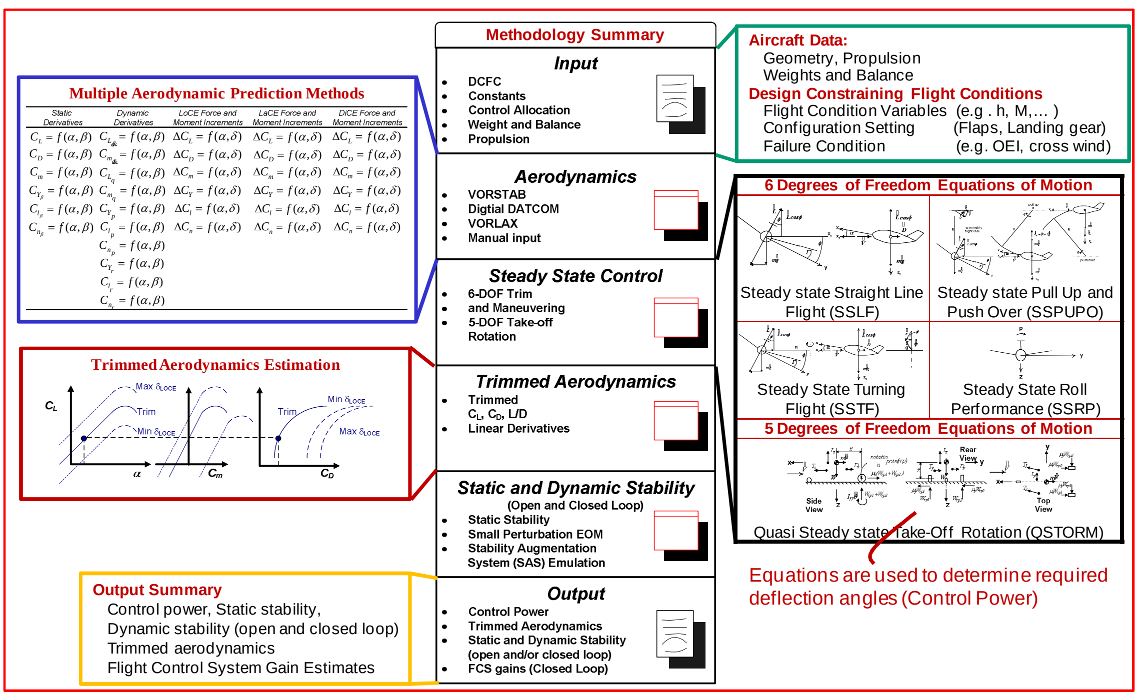


Figure 4.1 *AeroMech* Methodology Overview (Coleman 2007, 283)

The input required for *AeroMech* includes a description of the vehicle, the definition of the design constraining flight conditions, the control allocation schedule and the model setup for aerodynamic prediction. The vehicle description required is in the form of aircraft data such as:

1. geometry variables such as b_{ref}, c_{ref}, s_{ref}, etc,
2. propulsion variables such as the thrust available, thrust location and thrust direction,
3. weight and balance variables such as weight, inertias and cg locations.

The design constraining flight conditions (DCFC) define the testing circumstances to evaluate the aircraft under. These conditions as described in (Chudoba and others 2008, 293) are represented by:

1. mission segment flight condition variables such as altitude, speed, etc.,
2. configuration settings such as flap setting, landing gear setting, etc.,
3. failure conditions such as one engine inoperable (OEI), maximum crosswind, etc.,
4. test cases for evaluation such as steady state straight line flight, steady state turning flight, etc.,
5. vehicle design certification requirements.

It is important to use these variables to define the most critical corners of the flight envelop as shown in Figure 4.2. The most critical design constraining flight condition will size the control effectors (CEs). For example, landing approach in 50ft/s crosswind might size the longitudinal control effector.

Control allocation is a term which refers to the scheme by which redundant control effectors are used. For example, a control scheme needs to be defined for aircraft which have elevators as well as pitch thrust vector control such as the F-22, thereby representing an over-estimated system in pitch. There are two methods presented for dealing with this issue. The first is an ‘ad-hoc’ method where the allocation is manually scheduled based on experience. The second method is to allocate the control effectors to provide minimum trim drag using a Linear Optimum Trim Solution (LOTS), as presented in (Goodrich, Sliwa, and Lallman 1989).

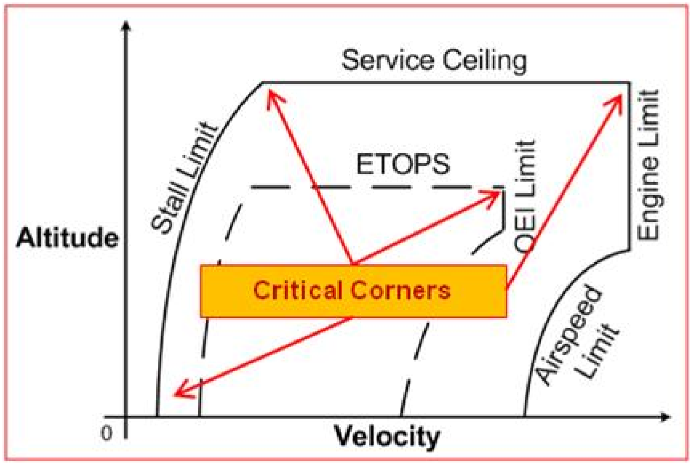


Figure 4.2 A typical flight envelope and the some critical corners

The model set up for aerodynamic prediction depends on the aerodynamics method being utilized. There are four methods of aerodynamic prediction for use with *AeroMech*. These methods are Digital DATCOM, a semi-empirical handbook method; VORLAX, a linear vortex lattice method; VORSTAB, a non-linear vortex lattice method; and manual aerodynamic data input.

The quality of the aerodynamic data available is vital to stability and control analysis since the control effector tend to be sized for aerodynamically non-linear DCFCs. The aerodynamic prediction step involves production of the aerodynamic data required for *AeroMech* analysis. The primary aerodynamic prediction method for selected for use with *AeroMech* is VORSTAB because of its non-linear aerodynamic modeling capability. However as discussed in (Coleman 2007, 283), VORSTAB has deficiencies in modeling unsteady aerodynamic derivatives. Therefore, VORSTAB results are used in conjunction with Digital DATCOM and VORLAX results to produce an initially untrimmed aerodynamic map required by *AeroMech*. It is important to note that *AeroMech* can function with any combination of the aerodynamic prediction methods as long as the output is properly organized. It is up to the engineer to generate the most appropriate aerodynamic input for the desired analysis. A proper combination of these methods can be used to used to produce aerodynamic data for any generic flight vehicle configuration (Chudoba and others 2008, 293).

The steady state control analysis module calculates control power required for trim and maneuvering at defined design constraining flight conditions. It involves the solution of steady state 6-DOF equations of motion for the control effector deflections required of the aircraft to perform basic maneuvers at this flight condition. These maneuvers include:

1. Steady State Straight Line Flight (SSSLF) – This represents all non-accelerating, non-rotating, constant direction flight. It is defined by a flight path angle and a sideslip angle. It can be used to evaluate control power requirements for cruise, climb descent, one-engine inoperative, crosswind landing and other such conditions (Chudoba and others 2008, 293).
2. Steady State Turning Flight (SSTF) - this represents all constant bank turning motions under a prescribed load factor. It can be used to evaluate control power requirements for horizontal turn coordination at a desired turn radius, etc (Chudoba and others 2008, 293).
3. Steady State Roll Performance (SSRP) – this represents the rolling motion about the stability axis commanded by the lateral control effector. It can be used to evaluate the time to bank for a prescribed LaCE deflection, the control power requirement to overcome adverse yaw coupling, *etc.* (Chudoba and others 2008, 293).
4. Steady State Pull-Up/Push-Over maneuver (SSPPO) – this represents all constant longitudinal pitch motion maneuvers under a prescribed load factor and bank angle. It can be used to evaluate control power requirements for speed recovery, load factor maneuvering capability, *etc.* (Chudoba and others 2008, 293)

5. Quasi Steady State Take-Off Rotation – this represents the instantaneous pitch rotation of the aircraft about the main gear induced by the longitudinal control effector during takeoff. The quasi steady state term is used because it models the instant before the wheels leave the ground with no lateral sliding. At this instant, there are horizontal and pitch accelerations but no pitch velocity and the side forces can be neglected (Chudoba 2001).

Trimmed aerodynamics is important in generating data for performance calculations, comparison metrics (such as L/D), static stability analysis and dynamic stability evaluation at a specific design constraining flight condition (DCFC). The steps for producing this data as outlined in (Coleman 2007, 283) are:

1. Solve the steady state straight line flight for attitude and CE deflection at a DCFC.
2. Interpolate trimmed aerodynamic data (e.g. trimmed lift curve slope) from untrimmed data using the CE deflections calculated.
3. Calculate linear derivatives about this trimmed point using a center difference method.

The static and dynamic stability module is the target module integral of the flight control system analysis module. Static stability information is obtained from the trimmed aerodynamic data generated in the trimmed aerodynamics module. From this data, static and maneuver margins are calculated and static stability curves such as pitching moment vs. angle of attack, are produced. These quantities can be examined to ensure that the aircraft meets static stability requirements.

Dynamic analysis is also possible because of the linear derivatives obtained in the trimmed aerodynamics module. The analysis is performed via the small perturbation equations of motion using that data. There are three options for dynamic analysis in the *AeroMech* methodology. These options are outlined in (Chudoba and others 2008, 293) as:

1. Open Loop – this option is for the evaluation of the dynamic characteristics of vehicles that are designed to meet safety requirements without stability augmentation (inherent airframe).
2. Closed Loop Damping Restoration – this option is for the evaluation of the dynamic characteristics of vehicles which are inherently stable with relaxed stability. Stability augmentation with yaw rate, pitch rate and roll rate feedback is used to produce desired damping characteristics. In this case, it is important to determine the additional control power required because of feedback.
3. Closed Loop Stiffness and Damping Restoration – this option is for the evaluation of statically unstable and statically indifferent aircraft. A stability augmentation system with angle of attack and sideslip feedback for stiffness restoration and rate feedback for damping restoration is used. The additional control power requirement required for feedback is also be determined here.

As aforementioned, the dynamics calculations are performed via the small perturbation equations of motion. In (Chudoba 2001), coupled 6-DOF small perturbation equations of motion are derived in order to analyze the dynamic behavior of symmetric and asymmetric aircraft and flight conditions. In the current implementation of the software source code, traditional decoupled lateral and longitudinal equations of motion are incorporated for reasons specified in (Coleman 2007, 283). One reason is that these equations have to be compatible with the stability augmentation design subroutines ILOCS from (Abzug 1998) which have been integrated into *AeroMech*.

For the present research undertaking, the implementation of these small perturbation equations of motion has been revisited. It has been discovered that the ILOCS implantation is not generic enough to handle thrust vector controlled aircraft, which is one of the application case studies selected for this research. The problems stems from the fact that the ILOCS implantation assumes standard aerodynamic control effectors in the model. The ILOCS subroutines are effective modeling conventional aircraft but not unconventional ones. Two options have been considered as solutions to this problem:

1. Implement the generic coupled 6-DOF equations of motion derived in (Chudoba 2001)(Chudoba 2001; Chudoba and others 2008, 293)
2. use a numerical linearization subroutine to obtain the small perturbation 6-DOF equations directly from the coupled 6-DOF non linear equations of motion.

The numerical linearization option has been selected because the subroutine is compatible with the structure of the *AeroMech* software as the equations of motion are used in the steady state control analysis module. The numerical linearization involves calculating the partial derivatives with respect to the state and control vectors about the trim point. The partial derivative with respect to the states gives the coefficient matrix A, while the derivatives with respect to control give the coefficient matrix B. The numerical linearization subroutine chosen for this purpose is the JACOB subroutine from (Stevens and Lewis 2003). This subroutine has been made available courtesy Frank Lewis. The implementation of this code is discussed later in this chapter.

AeroMech is structured in an "Input-Analysis-Output" format. All the required input is prepared upfront before the analysis and all the results of each analysis modules are gathered at the end. The collected output includes the trimmed aerodynamic data, the control power assessment and stability results. This output is in numerical form and (Coleman 2007, 283) suggests various visualizations as part of the stability and control delivery map. During this research, additional visualizations have been developed to present the stability and control analysis results. One of the visuals is the control power assessment chart shown in Figure 4.3.

The control power assessment chart is using MS Excel for visualizing the *AeroMech* output data. It gives a summary of the control power information for each maneuver for a specified design-constraining flight condition (DCFC). It shows parameters characterizing the design constraining flight condition (DCFC) such as the flight condition variables and failure conditions. It also shows the input parameters specified for the maneuvers such as sideslip angle for

SSSLF. The control power measure, such as CE deflections and required trust settings, are also given. These measures are compared with the maximum allowable values, and the results are color coded presenting a data bar for quick interpretation. The other visuals include an input card, trimmed aerodynamics and time response plots which are self-explanatory, thus do not require further discussion.

This information completes the description of the *AeroMech* methodology. In the next section describes the source code implementing this methodology.

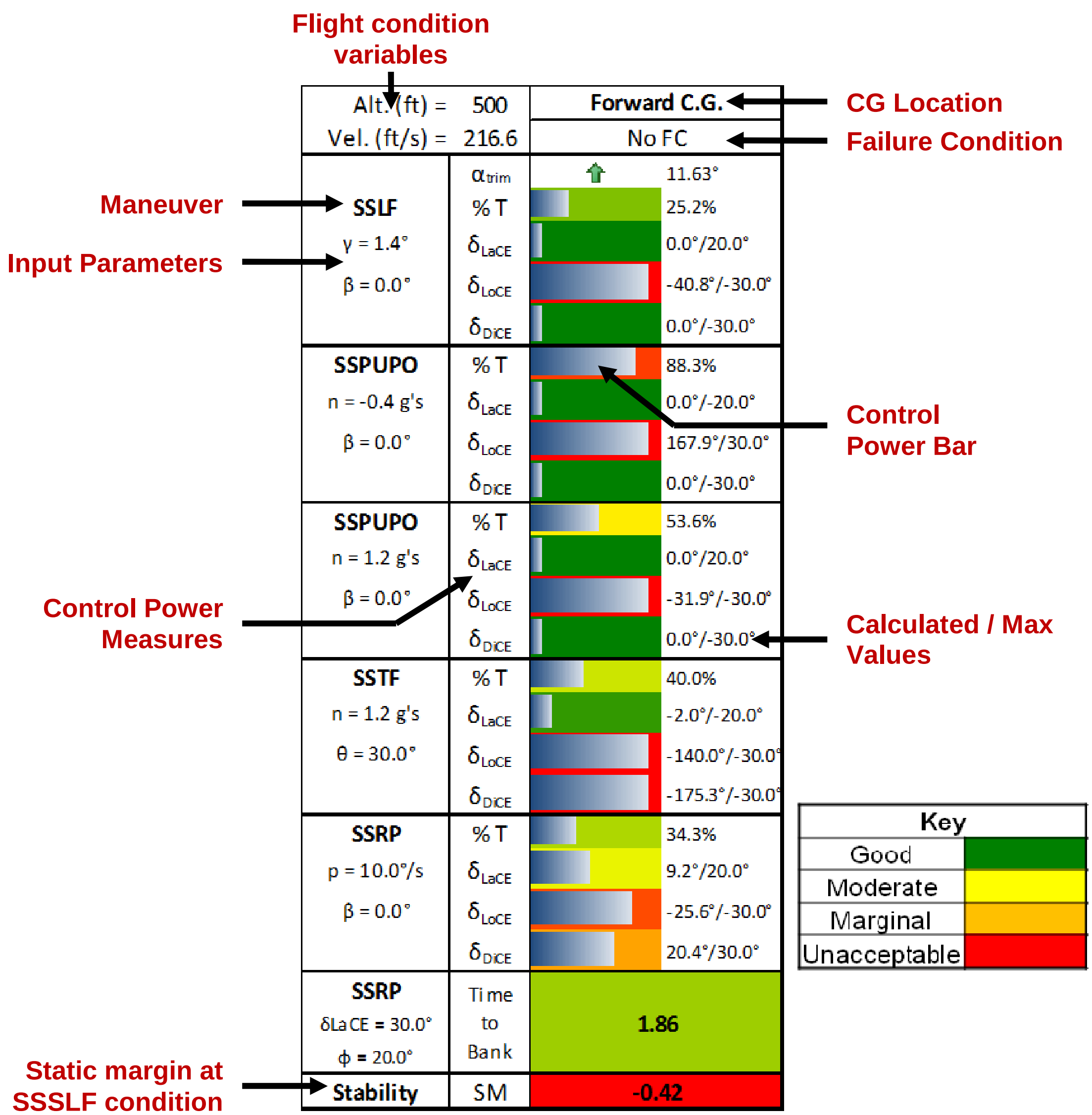


Figure 4.3 Control power assessment chart

4.1.2 AeroMech Source Code Overview

The prototype *AeroMech* software has been developed by Kiran Pippalapalli while a fully functioning version has been implemented by Gary Coleman as described in (Coleman 2007, 283). Figure 4.4 shows the Nassi-Schneiderman diagram outlining the *AeroMech* source code structure. The code follows the same input-analysis-output structure of the *AeroMech* methodology. A summary of the major *AeroMech* subroutines is shown in Table 4.1.

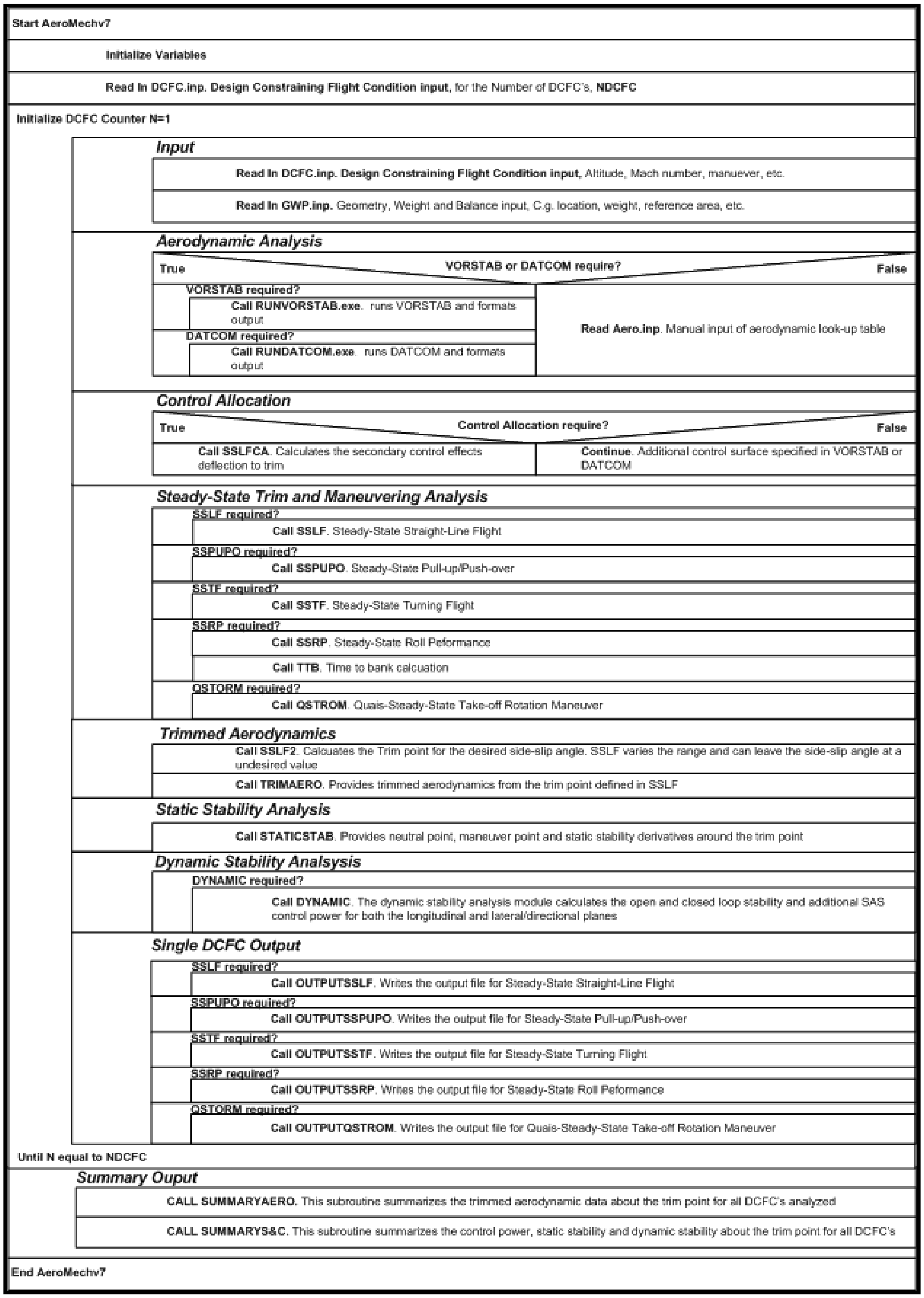


Figure 4.4 Final *AeroMech* driver structogram (Coleman 2007, 283)

Table 4.1 Summary of major *AeroMech* subroutines (Coleman 2007, 283)

Subroutine	***Description***
RUNDATCOM	iterates Digital DATCOM to produce the untrimmed aerodynamic lookup table; stand alone executable
RUNVORSTAB	iterates VORSTAB to produce the untrimmed aerodynamic lookup table; stand alone executable
SSLF	calculates attitude and control variables to trim to 1-g flight
SSLFCA	calculates the secondary control effector deflection require for 1-g trim control allocation
SSPUPO	calculates attitude and control variables to perform a pull-up or push-over maneuver
SSTF	calculates attitude and control variables to perform a horizontal turn
SSRP	calculates attitude and control variables to perform a rolling maneuver
TTB	calculates the time to bank to a predefined bank angle
QSTORM	calculates the rotational pitch velocity given a predefined pitch acceleration and LoCE deflection
SSLF2	calculates attitude and control variables to define the trim point for later calculations
LINAERO	calculates the linear aerodynamic derivatives around the trim point from the aerodynamic lookup table
TRIMAERO	calculates the trimmed aerodynamic properties around the trim point
STATSTAB	calculates the static stability properties around the trim point
DYNAMIC	calculates the open and closed loop dynamic stability around the trim point for both the longitudinal and lateral/directional planes; additional control power required for the SAS function is also calculated

The programming strategy of this code is to maintain simplicity by using a modular approach to integrating the subroutines and to collect variables in a single location (Coleman 2007, 283). This same philosophy is utilized in the integration of the flight control system module developed in the previous chapters.

The subroutines of interest in this version of *AeroMech* are the SSLF and DYNAMIC subroutines. The SSLF subroutine calculates the trim values required for initializing the trim numerical linearization subroutine selected for this research. The DYNAMIC subroutine is the driver for all dynamic and stability analysis in the code. Ideally, this should be the point of integration, however, because the subroutine is only suitable for modeling conventional aircraft, it has been decided to integrate the FCS subroutine directly into the *AeroMech* main structure. The flight control system module driver is called FCS subroutine and it is discussed in the next section.

4.2 FCS Module

The purpose of the FCS subroutine is to create an interface for integrating the linear quadratic regulator with output feedback control design technique into *AeroMech*. This interface is shown in Figure 4.5. It takes in as input the steady state level flight calculations and produces on output open and closed loop Eigenvalues and simulations. The closed loop gains are calculated using the LQR with output feedback, as described in Chapter 3. This allows the design of stability augmentation systems of any desired structure. The subroutine written for this technique is TLQR_OUTFEED. It requires as input the state space matrices A and B; and the control structure matrices C, E and W. On output, the subroutine produces the stabilizing gain matrix K. Based on this subroutine, there are five steps to be accomplished with the FCS module:

1. generate a state space model;
2. create control structure;
3. calculate feedback gains;
4. compute Eigenvalues of the open and closed loop systems;
5. simulate the time responses of the open and closed loop systems;
6. collect results.

The subroutines created for each of these tasks and they are described in the following section.

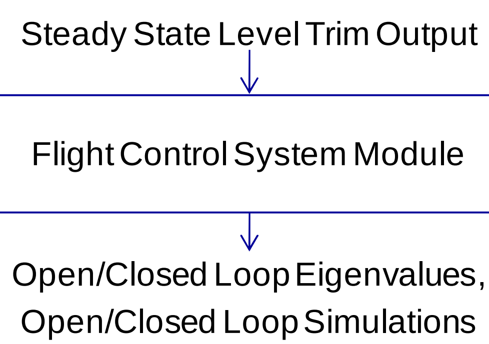


Figure 4.5 FCS Module interface

4.2.1 State Space Linearization

As previously discussed, a numerical linearization technique has been selected to generate the state space model for analysis. JACOB is the linearization subroutine chosen for this task. It computes the partial derivatives with respect to the states and the control variables about the trim point. It requires an equation of motion subroutine called EOM which is based on the coupled equations of motion. A driver subroutine, LINEARIZE, has been written to run JACOB. It is collects the trimmed information and uses it to initial the JACOB subroutine. It also collects the resulting state space matrices and sorts them into longitudinal and lateral matrices. Note that JACOB computes a fully coupled state space matrix. However, since the test case for this project is symmetric, the matrix is partitioned into longitudinal and lateral matrices. The LINEARIZE subroutine assumes this partitioning. The logic needs to be expanded for asymmetric vehicles in a later study. In addition, LINEARIZE augments the state and control coefficient matrices to include yaw washout and angle of attack filters including the actuator dynamics.

4.2.2 Control Structure Creation

The control structure subroutine TLQR_OUTFEED allows the user to implement any control structure of the form

$$u = -Ky \tag{3.126}$$

where $y(t) \in \mathbb{R}^p$ is the output vector defined by

$$y = Cx \tag{3.127}$$

In addition, elements of the gain matrix K can eliminated to give it any structure by specifying the constraint matrices W and d as discussed in Chapter 3.

Two subroutines, LON_CON_STRUC and LAT_CON_STRC have been written to automatically generate C, E and W matrices for typical longitudinal and lateral control structures respectively. There are fifteen longitudinal control structures programmed which are combinations of the longitudinal states v_T, α_F, θ and q (i.e. true airspeed, filtered angle of attack and yaw rate respectively). Additionally, fifteen lateral control structures are programmed for combinations of the lateral states r_w, p, β and ϕ (i.e. washed-out yaw rate, roll rate, sideslip and bank angle respectively). Table 4.2 shows the primary functions of typical feedback relations used from (McRuer, Ashkenas, and Graham 1974).

Table 4.2 Primary functions of typical feedback

Feedback	Primary Functions
$\boldsymbol{v_T} \rightarrow \boldsymbol{\delta_T}$	Stabilize tuck mode
$\boldsymbol{\alpha_F} \rightarrow \boldsymbol{\delta_e}$	Increase short period damping and frequency
$\boldsymbol{q} \rightarrow \boldsymbol{\delta_e}$	Increase short period damping
$\boldsymbol{\theta} \rightarrow \boldsymbol{\delta_e}$	Increase short period damping and frequency Increase phugoid damping
$\boldsymbol{r_w} \rightarrow \boldsymbol{\delta_r}$	Increase directional stability Increase dutch roll damping Reduce inertial cross coupling Improve turn coordination
$\boldsymbol{p} \rightarrow \boldsymbol{\delta_a}$	Improve roll response Reduce ω_r/ω_d
$\boldsymbol{\beta} \rightarrow \boldsymbol{\delta_r}$	Increase directional stability Increase Dutch roll damping Reduce inertial cross coupling Improve turn coordination

4.2.3 Calculating Feedback Gains

The feedback gains are calculated using the TLQR_OUTFEED subroutine. The process is discussed in Chapter 3.

4.2.4 Computing Eigenvalues

The Eigenvalue subroutine written is called EIGENVAL. It uses BALANC, ELMHES, HQR and PIKSR2 from (Press 2007) to compute and organize the Eigenvalues in descending order of the real parts.

4.2.5 Simulating the Time Responses

As discussed in Chapter 3, the introduction of controller dynamics in the closed loop system makes it difficult to correlate their Eigenvalues with regulations such as (Anonymous1986). Therefore, in order to judge the effects of the gains in this project, the time responses of the closed and open systems are compared. The subroutine generated to simulate the responses is called FCS_SIM. It uses fourth order Runge-Kutta integration to do so.

4.2.6 Collecting and Output of Results

All the output of the FCS module are stored in a common location. In the same input-analysis-output philosophy of *AeroMech*, a subroutine, FCSOUT is developed to collect all FCS output in the output files. A summary of the major subroutines is shown in Table 4.3 and the details of the FCS module are shown in Figure 4.6.

Table 4.3 Summary of the major FCS module subroutines

Subroutine	***Description***
LINEARIZE	Calls JACOB to create a state space model and sorts results into longitudinal and lateral matrices
JACOB	Numerical linearization subroutine which computes the partial derivative for creating the state space models
EOM	Contains the coupled nonlinear equations of motion, required by JACOB which
LAT_CON_STRUC	Creates lateral control structure matrices
LON_CON_STRUC	Creates longitudinal control structure matrices
TLQR_OUTFEED	Computes controller feedback gains
EIGENVAL	Calculates Eigenvalues and arranges them in descending order of their real parts
DAMP	Computes damping ratios and natural frequencies from Eigenvalues
FCS_SIM	Performs a time simulation of the system
FCS	Drives all FCS module subroutines
FCSOUT	Writes FCS subroutine outputs to output files

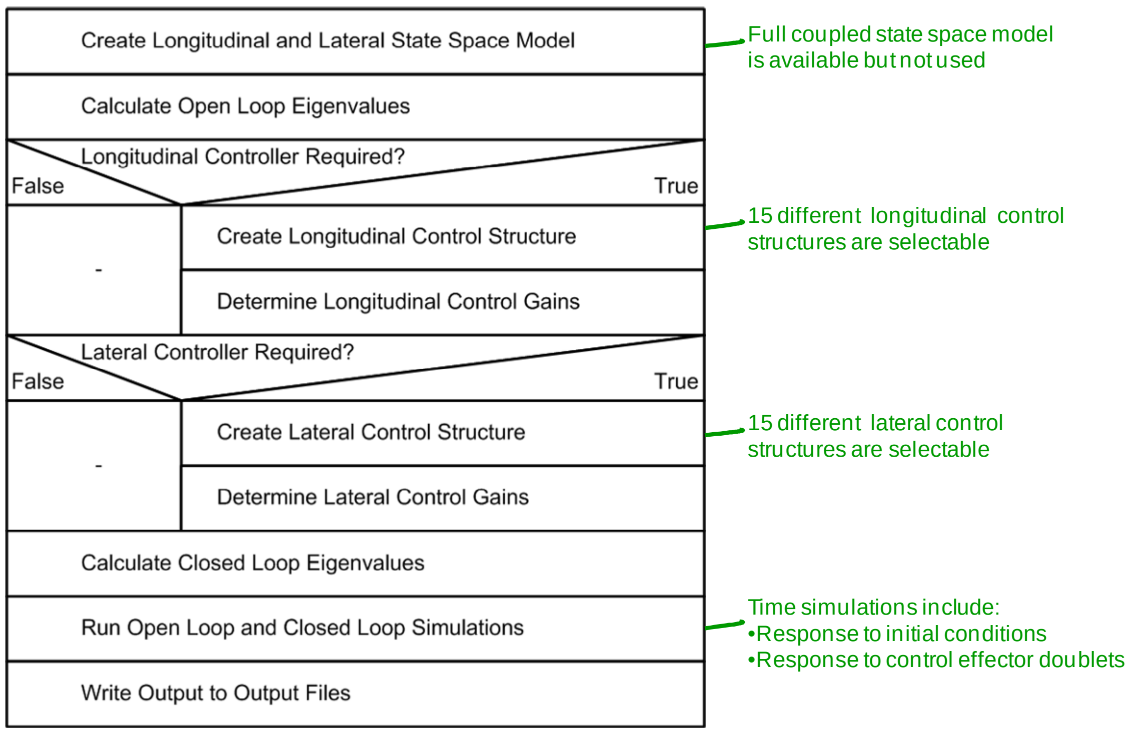


Figure 4.6 Details of the FCS module

4.3 Chapter Summary

In conclusion, the flight control system subroutines, theoretically structured in Chapter 3, has been integrated into the *AeroMech* source code in this chapter to allow testing its viability for conceptual design applications in the following chapter. In this chapter, the *AeroMech* methodology has been summarized and its static and dynamic stability module was identified as the point of integration for the FCS module. The FCS module subroutines have been described. These subroutines are used to accomplish the tasks of state space linearization, control structure creation, control gain calculation, Eigenvalue computation and time response simulation.

CHAPTER 5

APPLICATION OF AEROMECH AND FLIGHT CONTROL SYSTEM MODULE

Clearly, to enable future efficient aircraft design, a truly informed approach is mandatory when addressing the complex issue of aircraft configuration selection as coupled with stability and control, certification issues, and other design disciplines.

Bernd Chudoba

5.1 Introduction

As discussed in Chapter 1, the conceptual designer has the task of providing necessary information to steer the design process towards the best concepts for desired mission objectives. The ability of the designer to do so depends on the knowledge available at this early time in the design process and the capabilities of the tools present. The simulation tools have to be capable of determining the benefits and risks associated with different candidate designs. In addition, these tools aid in performing trade studies in order to understand the sensitivities of the vehicles to various design variables. In the context of this research, *AeroMech* provides the ability to assess stability and control issues of both conventional and unconventional vehicles during the conceptual design phase. Furthermore, various design variables of these vehicles can be perturbed in order to understand how these parameters affect vehicle's stability and control characteristics.

The integration of an advanced flight control system module into *AeroMech*, through this research, gives an extra dimension of variables in order to perform handling quality trades during the conceptual design phase. That is, studies can be done in which both the flight vehicle hardware and the FCS variables are adjusted such to shape the vehicle for good handling qualities whilst minding FCS complexity. Note that this research does not address handling quality issues as discussed although it introduces the FCS variables, ρ and k. A design case study has been selected to demonstrate how these variables can be used in a conceptual design environment.

This chapter highlights the stability and control study conducted on a Thrust Vector Control (TVC) commercial transport concept using *AeroMech*. This study compares the steady state control power assessment using the original *AeroMech* source code with the dynamic stability assessment generated with the new FCS module integrated into *AeroMech*. The goals of this presentation are to

1. demonstrate an understanding of the *AeroMech* methodology;
2. show the effects of the Flight Control System response characteristics of an aircraft concept;
3. illustrate the idea of trading the flight vehicle hardware and FCS variables.

The motivation for a TVC transport study is given in the next section.

5.2 Motivation for a Thrust Vector Control Commercial Transport Study

The Thrust Vector Control (TVC) commercial transport is a concept that the AVD Laboratory at UTA-MAE has been presenting at the 2009 NASA/NIA Truss Braced Wing (TBW) Synergistic Efficiency Technologies Workshop (Chudoba and Coleman 2009). The presentation is a preliminary assessment of the feasibility and synergistic potential of a TVC commercial transport. It has two major analytical steps. The first is a parametric sizing analysis to determine the gross performance benefits of the TVC over a conventional Tail Aft Configuration (TAC) transport. The second is a steady state control power assessment of the TVC transport to examine safety issues concerning this concept. The author of this thesis was responsible for the stability control analysis. This analysis was performed using the original version of *AeroMech*. The results of the sizing study are given in (Coleman 2010, 404) as part of the validation cases for AVDsizing. While the steady-state control power assessment results are show in this present document, because it is part of the familiarization process with the *AeroMech* methodology performed for this research undertaking.

5.3 Steady State Control Power Assessment of a Thrust Vector Control Commercial Transport

The control power required and available of the TVC transport has been assessed with the *AeroMech* methodology and software tool. The methodology follows, as discussed in Chapter 4, the following steps: (1) input definition; (2) aerodynamic prediction; (3) steady state control analysis; (4) trimmed aerodynamics estimation; (5) static and dynamic stability analysis; (6) output presentation. A modified summary diagram of this methodology is shown with Figure 5.1. This diagram shows the intermediate iteration for steady state control power which is necessary when designing an untested vehicle such as a TVC transport. These iterations are necessary because it is important to ensure that the vehicle is capable of performing the basic maneuvers (pitch, yaw, roll) throughout the flight envelope before considering its static and dynamic stability characteristics. If the design is not capable performing these maneuvers, it is a failed concept and there is no need for further analysis. The control power assessment is described in the following sections by stepping through this methodology.

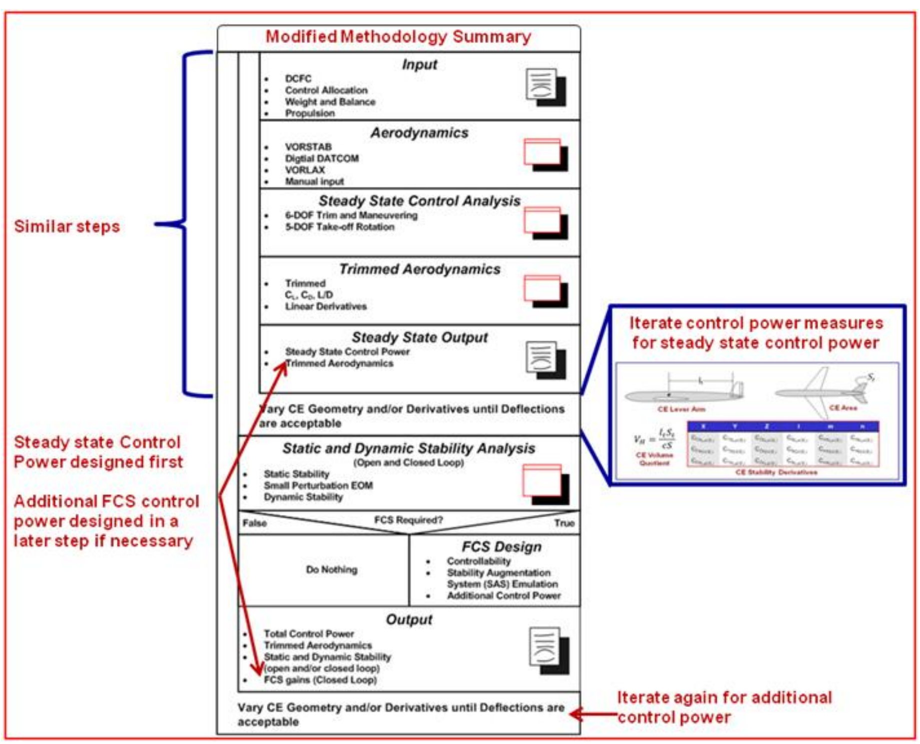


Figure 5.1 *AeroMech* methodology showing iteration steps for control power

5.3.1 Input Definition

As previously mentioned, a parametric sizing study has been performed for the TVC commercial transport before the stability and control assessment can begin. The TVC commercial transport in this study is sized for the B777-300ER mission (Boeing Commercial Airplanes December 2007). The sizing activity provides estimates of weight, geometry and other vehicle parameters. This information is sufficient aircraft data to provide the input required to execute *AeroMech*.

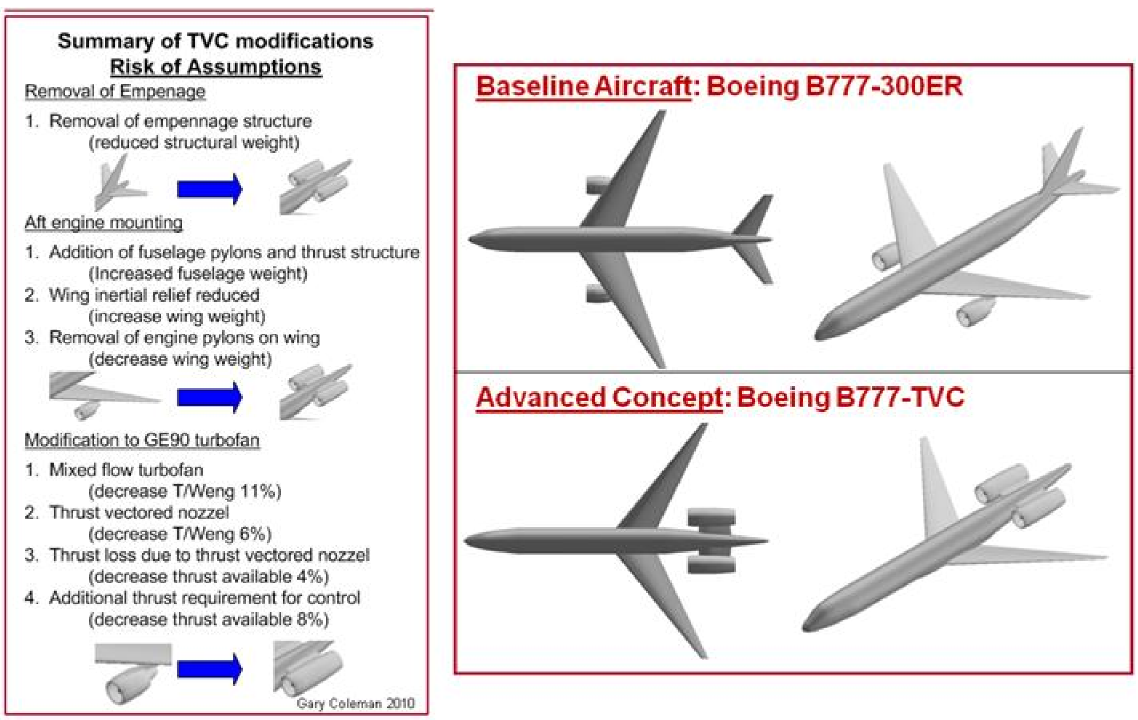


Figure 5.2 Modifications to the B777-300ER for Thrust Vector Control (Coleman 2010, 404)

The design constraining flight conditions are selected to test the steady state control power at the most challenging corners of the envelope for a TVC aircraft. Since the aircraft has a traditional wing as its primary lift supply, the typical stall conditions are critical for sizing the control effectors (CEs). However, since the vehicle is controlled by engine thrust, it is required to consider as well the high altitude and high speed conditions, which typically do not size aerodynamic control effectors. This is because engine thrust limits are reached at these conditions. Figure 5.3 shows the critical corners of the flight envelope for consideration in assessing the control power of a thrust vector controlled (TVC) aircraft. In addition to these flight conditions, the One Engine Inoperable (OEI) case is significant because the vehicle losses half of its control power with the loss of an engine. Considering these factors along with DCFCs summarized in (Chudoba 2001), a DCFC test-matrix is formulated. This test matrix is shown in

Table 5.1.

Mission Segments

1. Take-Off (TO)
2. Initial Climb (IC)
3. Low speed Maneuver (LM)
4. Approach (A)
5. Flare (F)
6. Ceiling (C)
7. High Speed (HS)
8. High Speed – Extended range Twin-engine Operation Performance (HS-ETOPS)
9. Low speed Maneuver– Extended range Twin-engine Operation Performance (LM-ETOPS)

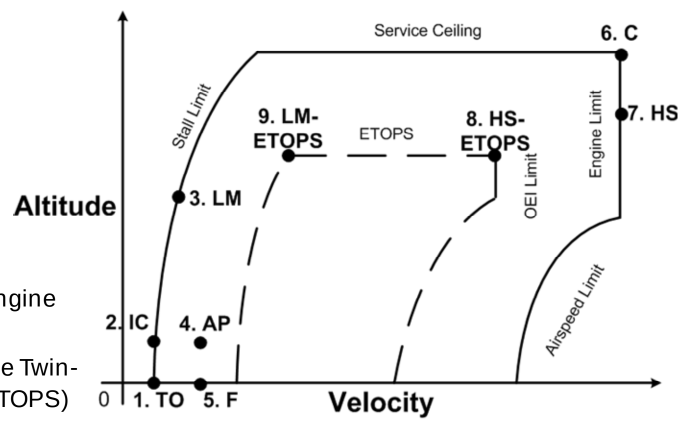


Figure 5.3 Critical corners of the flight envelope for control power assessment of a TVC transport

Table 5.1 Control power assessment test matrix

Design Constraining Flight Conditions (DCFC)																	
Mission Segment		Flight Condition Variables		Configuration Settings		Failure Conditions		Maneuvers									
		Air Speed	Altitude	Flap Setting	Landing Gear	Engine	Cross Wind	SSLF	SSPUPO		SSTF		SSRP	SSRP (TTB)		QSTORM
								γ	n		n	φ	p	δ_{LaCE}	φ	θ''
		(ft/s)	(ft)	(-)	(-)	(-)	(ft/s)	(deg)	(g's)		(g's)	(deg)	(deg/s)	(deg)	(deg)	(deg/s^2)
1	T-O	216.6	0	5	Down	AEO / OEI	42.2	0.0	-0.45	1.2	1.2	30.0	0.0	0.0	0.0	6.0
2	IC	216.6	500	5	Down	AEO / OEI	-	1.4	-0.4	1.2	1.2	30.0	10.0	30.0	20.0	-
3	LM	321.8	10,000	1	Up	AEO / OEI	-	0.0	-0.7	1.0	1.2	30.0	10.0	30.0	20.0	-
4	App	234.7	500	5	Down	AEO / OEI	42.2	-3.0	-0.6	1.7	1.2	30.0	10.0	30.0	20.0	-
5	F	234.7	0	5	Down	AEO / OEI	42.2	0.0	-0.6	1.7	1.2	30.0	10.0	30.0	20.0	-
6	Cel	813.3	42,000	1	Up	AEO	-	0.0	-1.0	2.2	1.2	30.0	10.0	30.0	20.0	-
7	HS	830.5	35,957	1	Up	AEO	-	0.0	-1.0	2.5	1.2	30.0	10.0	30.0	20.0	-
8	HS-ETOPS	688.0	32,000	1	Up	OEI	-	0.0	-1.0	2.3	1.2	30.0	10.0	30.0	20.0	-
9	LM-ETOPS	450.7	32,000	1	Up	OEI	-	0.0	-0.7	1.0	1.2	30.0	10.0	30.0	20.0	-

5.3.2 Aerodynamic Prediction

The aerodynamic prediction method chosen for this study is a modified version of Digital DATCOM. The modifications have been implemented by the AVD Laboratory and include rudder and landing gear aerodynamics prediction methods. In addition, the RUNDATCOM subroutine described in (Coleman 2007, 283) is used to perform sweeps to create the aerodynamic database for *AeroMech*. DATCOM is selected because it is designed for classical wing-body aircraft configurations such as the TVC transport. An isometric view of the TVC transport DATCOM model is shown in Figure 5.4.

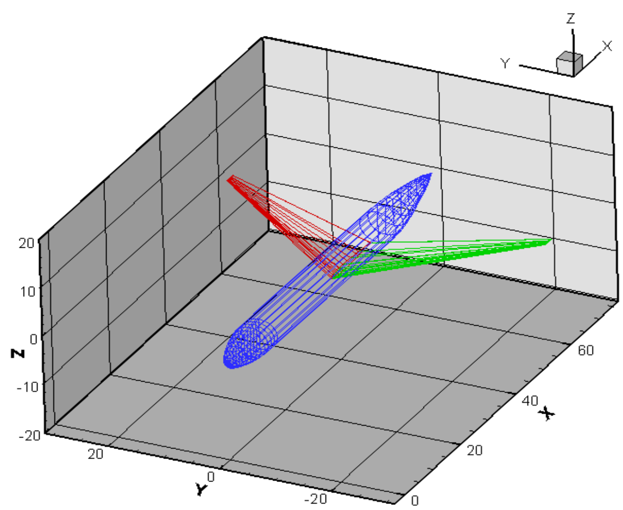


Figure 5.4 the Digital DATCOM model of the TVC aircraft

5.3.3 Steady State Control Analysis and Trimmed Aerodynamics

As shown in the test matrix given with Table 5.1, all the maneuvers are performed at the critical corners of the flight envelope. The inputs for these maneuvers are developed from the regulations in (Office of the Federal Register (U.S.) 2010). For example, the SSLF flight path angle for the initial climb segment corresponds to the 2.4% climb gradient requirements in FAR 25-121. In addition, at each of these points, trimmed aerodynamics is estimated by *AeroMech* for the steady state straight line flight maneuver.

5.3.4 Steady State Output

The output for the analyses as well as results from trades performed are discussed in this section. The results are presented using the control power charts introduced in Chapter 4.

The high speed condition is considered as the design point because this is the point in the mission profile where the vehicle will spend most time. Using the information from the sizing study, the cg location is kept parametric in an MS excel spread sheet. This is because the cg location varies with respect to mission and fuel burnt to get to altitude. For this study, the cg locations chosen are forward and aft cg locations corresponding to a max payload mission and a ferrying mission (all fuel no payload) respectively.

Alt. (ft) =	35,957	Forward C.G.		Aft C.G.	
Vel. (ft/s) =	830.5	No FC	OEI	No FC	OEI
SSLF	α_{trim}	3.65°	3.65°	2.39°	2.39°
	% T	67.8%	140.3%	-217.0%	-434.1%
γ = 0.0°	δ_{LaCE}	0.0°/20.0°	0.9°/20.0°	0.0°/20.0°	-6.3°/-20.0°
β = 0.0°	δ_{LoCE}	23.0°/30.0°	22.2°/30.0°	103.3°/30.0°	-616.7°/-30.0°
	δ_{DiCE}	0.0°/30.0°	16.3°/30.0°	0.0°/30.0°	4.5°/30.0°
SSPUPO	% T	-41.9%	-84.0%	188.0%	376.1%
n = -1.0 g	δ_{LaCE}	0.0°/20.0°	-1.2°/-20.0°	0.0°/20.0°	5.5°/20.0°
β = 0.0°	δ_{LoCE}	457.9°/30.0°	98.8°/30.0°	1164.1°/30.0°	2244.1°/30.0°
	δ_{DiCE}	0.0°/30.0°	25.4°/30.0°	0.0°/30.0°	4.6°/30.0°
SSPUPO	% T	249.8%	511.5%	-568.0%	1135.9%
n = 2.5 g's	δ_{LaCE}	0.0°/20.0°	0.3°/20.0°	0.0°/20.0°	-15.7°/-20.0°
β = 0.0°	δ_{LoCE}	2.0°/30.0°	537.7°/30.0°	-612.7°/-30.0°	-467.3°/-30.0°
	δ_{DiCE}	0.0°/30.0°	15315.4°/30.0	0.0°/30.0°	-181.9°/-30.0°
SSTF	% T	77.2%	233.7%	-247.9%	497.5%
n = 1.2 g's	δ_{LaCE}	0.2°/20.0°	-4.4°/-20.0°	0.0°/20.0°	-7.5°/-20.0°
θ = 30.0°	δ_{LoCE}	23.3°/30.0°	16.0°/30.0°	102.9°/30.0°	-1183.7°/-30.0
	δ_{DiCE}	4.0°/30.0°	-13.5°/-30.0°	4.3°/30.0°	-898.9°/-30.0°
SSRP	% T	69.3%	149.4%	-216.8%	-434.1%
p = 10.0°/s	δ_{LaCE}	4.8°/20.0°	5.6°/20.0°	4.9°/20.0°	-1.5°/-20.0°
β = 0.0°	δ_{LoCE}	23.6°/30.0°	21.8°/30.0°	103.8°/30.0°	-615.9°/-30.0°
	δ_{DiCE}	10.7°/30.0°	25.8°/30.0°	14.6°/30.0°	18.8°/30.0°
SSRP δLa CE = 30.0° ϕ = 20.0°	Time to Bank	0.86	0.86	1.19	1.19
Stability	SM	0.06	0.06	-0.53	-0.53

Figure 5.5 Control power assessment chart for the HS statically stable condition

A statically stable TVC configuration, in which the cg is located ahead of the neutral point, is considered first. The control power result of this test is shown in Figure 5.5. The observations are as follows:

1. There is a large Static Margin (SM) travel between forward and aft cg locations. This is because of the absence of empennages; the wings need to be positioned further aft on the fuselage to gain a typical 5% positive static margin. Since the neutral point moves aft with the wings and the payload cg is located towards the middle of the fuselage, the static margin travels considerable between the max payload and ferrying missions.
2. There is insufficient control power for almost all the maneuvers. This is because by locating the wing so far aft, the moment arm of the thrust vector is significantly decreased.

Some recommendations based on these observations are:

1. Use only the forward cg location for further analysis. This is possible if a fuel transfer system is available to transfer fuel to the fuselage tanks and maintain any desired cg location.
2. Run wing location trades to determine if there is a location where sufficient control power can be identified for maneuvering.

The system complexity of a fuel transfer system and fuselage fuel tanks have to be factored into a later sizing study.

The wing location trades demonstrate the iteration for control power shown in Figure 5.1. The measures of control power are varied in order to gain the desired control power. In the case of thrust vector control, the control power measures are the thrust moment arm, thrust setting and thrust vector deflection angles from the flight path. Since the thrust setting and deflection angles are constrained by the aircraft dynamics, the only design-variable measuring control power is the thrust moment arm. This design variable can be varied by moving the wings.

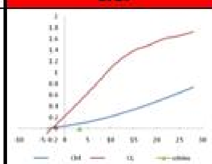
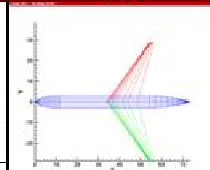
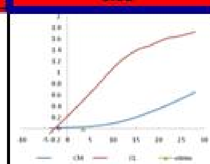
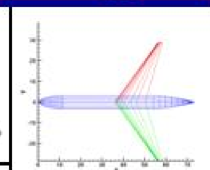
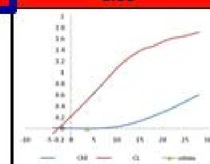
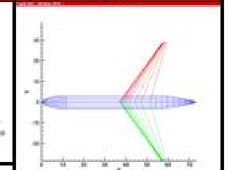
Alt. (ft) = 35,957		Forward C.G.		Forward C.G.		Forward C.G.	
Vel. (ft/s) = 830.5		No FC	OEI	No FC	OEI	No FC	OEI
	α_{trim}	3.19°	3.19°	3.41°	3.41°	3.52°	3.52°
SSLF	% T	83.3%	168.1%	62.8%	129.0%	61.0%	126.4%
γ = 0.0°	δ_{LaCE}	0.0°/20.0°	-1.8°/-20.0°	0.0°/20.0°	-0.5°/-20.0°	0.0°/20.0°	0.1°/20.0°
β = 0.0°	δ_{LoCE}	-46.4°/-30.0°	-45.8°/-30.0°	-18.1°/-30.0°	-17.6°/-30.0°	2.6°/30.0°	2.5°/30.0°
	δ_{DiCE}	0.0°/30.0°	11.4°/30.0°	0.0°/30.0°	13.7°/30.0°	0.0°/30.0°	15.0°/30.0°
SSPUPO	% T	45.4%	-91.0%	9.6%	19.6%	-19.5%	39.3%
n = -1.0 g	δ_{LaCE}	0.0°/20.0°	1.3°/20.0°	0.0°/20.0°	0.1°/20.0°	0.0°/20.0°	-0.5°/-20.0°
β = 0.0°	δ_{LoCE}	-284.1°/-30.0°	-75.8°/-30.0°	24.5°/30.0°	23.8°/30.0°	112.1°/30.0°	293.1°/30.0°
	δ_{DiCE}	0.0°/30.0°	-171.3°/-30.0	0.0°/30.0°	13.8°/30.0°	0.0°/30.0°	-341.7°/-30.0°
SSPUPO	% T	275.1%	-553.7%	247.9%	502.9%	244.5%	498.6%
n = 2.5 g's	δ_{LaCE}	0.0°/20.0°	-4.4°/-20.0°	0.0°/20.0°	-2.1°/-20.0°	0.0°/20.0°	-0.9°/-20.0°
β = 0.0°	δ_{LoCE}	-36.4°/-30.0°	35.9°/30.0°	-18.8°/-30.0°	-18.3°/-30.0°	-8.5°/-30.0°	-8.1°/-30.0°
	δ_{DiCE}	0.0°/30.0°	-169.6°/-30.0	0.0°/30.0°	12.8°/30.0°	0.0°/30.0°	14.1°/30.0°
SSTF	% T	93.9%	224.5%	70.7%	204.4%	69.1%	212.1%
n = 1.2 g's	δ_{LaCE}	0.2°/20.0°	-5.2°/-20.0°	0.2°/20.0°	-4.8°/-20.0°	0.2°/20.0°	-4.6°/-20.0°
θ = 30.0°	δ_{LoCE}	-47.0°/-30.0°	-37.0°/-30.0°	-18.1°/-30.0°	-11.7°/-30.0°	2.8°/30.0°	2.7°/30.0°
	δ_{DiCE}	4.7°/30.0°	-9.3°/-30.0°	4.4°/30.0°	-11.5°/-30.0°	4.2°/30.0°	-12.5°/-30.0°
SSRP	% T	83.1%	170.5%	63.5%	135.0%	62.2%	134.3%
p = 10.0°/s	δ_{LaCE}	4.7°/20.0°	3.0°/20.0°	4.8°/20.0°	4.3°/20.0°	4.8°/20.0°	4.9°/20.0°
β = 0.0°	δ_{LoCE}	-45.4°/-30.0°	-44.0°/-30.0°	-16.8°/-30.0°	-15.8°/-30.0°	3.7°/30.0°	3.4°/30.0°
	δ_{DiCE}	9.6°/30.0°	20.4°/30.0°	10.2°/30.0°	23.0°/30.0°	10.5°/30.0°	24.4°/30.0°
SSRP δLa CE - 30.0° ϕ - 20.0°	Time to Bank	0.86	0.86	0.86	0.86	0.86	0.86
Stability	SM	-0.17	-0.17	-0.05	-0.05	0.00	0.00
Diagrams							
		WAPEX = 111.74 ft		WAPEX = 118.31 ft		WAPEX = 121.59 ft	

Figure 5.6 Control power assessment chart for wing location trades

The results for the wing location trades are show in Figure 5.6. The key observations of this trade are:

1. there are control power issues at extremely negative static margins,
2. the most control power is available at a wing apex location of 118.3 ft, which corresponds to an SM of -5%,
3. at this wing location, there is insufficient thrust-control available for some of the maneuvers.

Some recommendations for further studies are:

4. increase the thrust requirement for cruise in a later sizing study,
5. use this particular wing location for control power assessments at the other corners of the flight envelope.

The results at the other corners of the envelope are shown in the next.

The mission segments near stall conditions include the Initial Climb (IC) and Low speed Maneuvering (LM) as shown in Figure 5.3. These conditions are critical for sizing the Longitudinal Control Effectors (LoCE). The control power assessment results for these conditions are show in Figure 5.7 . The plots of the aerodynamic lift curve slope and pitching moment curves are also shown with the trimmed angle of attack. This additional information helps in the interpretation of the control power results. The key observation is that

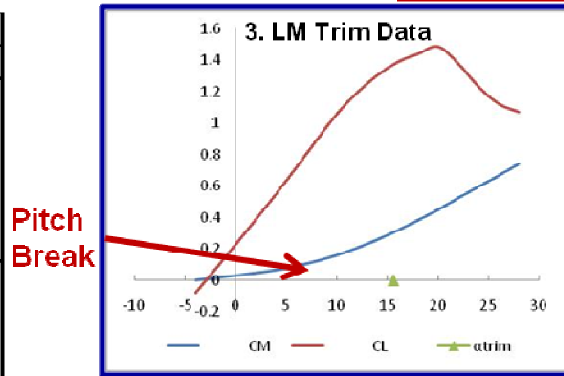
1. the LoCEs are saturated for most of the maneuvers in both conditions. The reason is that at stall the aircraft trims at high angles of attack; since this is an unstable configuration, these angles of attack occur after the pitch break. Thus, large thrust vector deflections are required to overcome the high moments present.

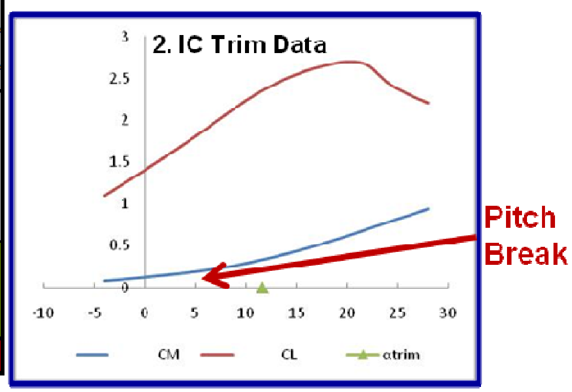
The following are recommendations for further studies:

1. redesign the wing using strakes and other devices to delay the pitch break,
2. use an angle of attack (AoA) limiter to limit the angle of attack to safe pitching moment regions.

DCFC 2: Initial Climb (IC)

Alt. (ft) = 500 Vel. (ft/s) = 216.6		Forward C.G. No FC
SSLF γ = 1.4° β = 0.0°	α_{trim}	11.63°
	% T	25.2%
	δ_{LaCE}	0.0°/20.0°
	δ_{LoCE}	-40.8°/-30.0°
	δ_{DiCE}	0.0°/-30.0°
SSPUPO n = -0.4 g's β = 0.0°	% T	88.3%
	δ_{LaCE}	0.0°/-20.0°
	δ_{LoCE}	167.9°/30.0°
	δ_{DiCE}	0.0°/-30.0°
SSPUPO n = 1.2 g's β – 0.0°	% T	53.6%
	δ_{LaCE}	0.0°/20.0°
	δ_{LoCE}	-31.9°/-30.0°
	δ_{DiCE}	0.0°/-30.0°
SSTF n = 1.2 g's θ = 30.0°	% T	40.0%
	δ_{LaCE}	-2.0°/-20.0°
	δ_{LoCE}	-140.0°/-30.0°
	δ_{DiCE}	-175.3°/-30.0°
SSRP p = 10.0°/s β = 0.0°	% T	34.3%
	δ_{LaCE}	9.2°/20.0°
	δ_{LoCE}	-25.5°/-30.0°
	δ_{DiCE}	20.4°/30.0°
SSRP δLaCE = 30.0° ϕ = 20.0°	Time to Bank	**1.86**
Stability	SM	**-0.42**





DCFC 3: Low speed Maneuvering (LM)

Alt. (ft) = 10,000 Vel. (ft/s) = 321.8		Forward C.G. No FC
SSLF γ = 0.0° β = 0.0°	α_{trim}	15.60°
	% T	65.6%
	δ_{LaCE}	0.0°/20.0°
	δ_{LoCE}	-33.6°/-30.0°
	δ_{DiCE}	0.0°/-30.0°
SSPUPO n = -0.7 g's β = 0.0°	% T	38.4%
	δ_{LaCE}	0.0°/-20.0°
	δ_{LoCE}	-198.4°/-30.0°
	δ_{DiCE}	0.0°/-30.0°
SSPUPO n = 1.0 g β – 0.0°	% T	65.6%
	δ_{LaCE}	0.0°/20.0°
	δ_{LoCE}	-33.6°/-30.0°
	δ_{DiCE}	0.0°/-30.0°
SSTF n = 1.2 g's θ = 30.0°	% T	-80.0%
	δ_{LaCE}	-0.8°/-20.0°
	δ_{LoCE}	-934.0°/-30.0°
	δ_{DiCE}	2.3°/30.0°
SSRP p = 10.0°/s β = 0.0°	% T	65.5%
	δ_{LaCE}	1.2°/20.0°
	δ_{LoCE}	-31.6°/-30.0°
	δ_{DiCE}	368.4°/30.0°
SSRP δLaCE = 30.0° ϕ = 20.0°	Time to Bank	**1.26**
Stability	SM	**-0.86**

Figure 5.7 Control power assessment charts and trimmed aerodynamic data for the stall conditions IC and LM

The approach and landing flare segments are modeled at maximum cross wind conditions. Maximum cross wind conditions typically size the Lateral Control Effectors (LaCE) and Directional Control Effectors (DiCE). However, zero wind conditions are assumed in the equations of motion used in *AeroMech* (Chudoba 2001). In order to account for crosswind, the velocity vectors are used to compute equivalent sideslip angles and true airspeed. The results of the control power assessment for the maximum crosswind conditions are shown in Figure 5.8. The key observations are:

1. Only the LaCE saturate during the approach.
2. Both the LoCE and LaCE saturate during the landing flare. This is because the introduction of the ground effect reduced the thrust requirements; therefore, more directional control deflection is required.

Recommendations for future studies based on these results are:

1. The ailerons need to be resized for more lateral control power.
2. The engine location needs to be traded the find the position that provides the most directional control power.
3. Small aerodynamic rudders could be added to increase directional control power.

The section is the last of the steady state control power results. It explores the control power at engine thrust limits.

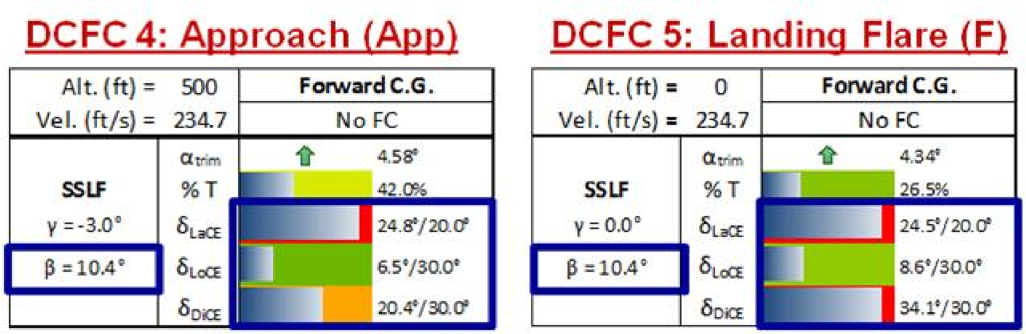


Figure 5.8 Control power assessment chart for segments with maximum crosswinds

The engine limit conditions include the ceiling altitude cruise and the one engine inoperable (OEI) condition (i.e. HS-ETOPS and LM-ETOPS). The results of the steady state control analysis on these conditions are shown in Figure 5.9. Key observations from these results are:

1. there is insufficient thrust for all the maneuvers with one engine inoperable pull-up being the most demanding;
2. the LM-ETOPS condition suffers from the same pitch break problems as the stall conditions.

Recommendations for further study include:

1. the thrust requirement for the one engine inoperable (OEI) pull up at HS-ETOPS should be used to size the engines in a successive sizing iteration loop;
2. a wing redesign or an angle attack limiter is required to curb the pitching moment problem near the stall condition.

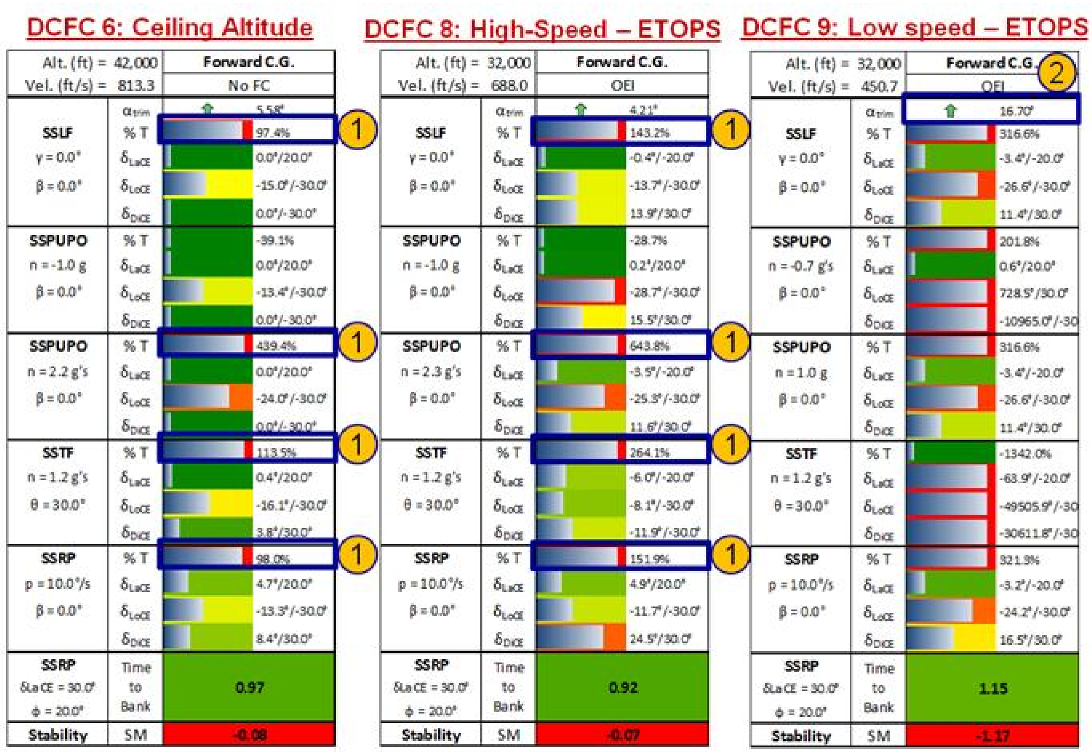


Figure 5.9 Control power assessment chart for engine limit conditions

5.3.5 Steady State Control Power Study Summary and Conclusions

A summary of all the observations and recommendations from the TVC commercial transport steady state control power analysis are shown in Table 5.2. These results show that the TVC commercial transport has many stability and deficiencies that need to be further evaluated. Further studies, based on the recommendations provided, are necessary in order to determine the true feasibility and certifiability of such a vehicle. However, these recommendations include a lot of system complexity penalties that will negate most of the benefits the vehicle has over a conventional TAC commercial transport.

Another conclusion that can be drawn from the results is on the physics of relaxed static stability. The Grumman X-29 possesses enhanced pitch maneuverability because of its -35% SM (Webster, Purifoy, and AIR FORCE FLIGHT TEST CENTER EDWARDS AFB CA. 1991). However, the TVC commercial transport demonstrates insufficient control power at -17%SM and has marginal characteristics at -5%SM. The reason for this difference is that the lever arm of the X-29 virtually increases the more negative the static margin becomes because of its Tail First Configuration (TFC). However, since the TVC transport resembles a Tail Aft Configuration (TAC), decreasing the static margin also decrease the lever arm of the thrust vector; clearly, there is an optimum point where the control power requirement reach a minimum. In this study, this point is at a static margin of -5%. The dynamic stability analysis of the TVC transport using the FCS module is discussed next.

Table 5.2 Summary of TVC transport steady state control power assessment results

Test Cases	Observations	Recommendations
Statically stable configuration (HS)	Large SM travel between forward and aft cg locations.	Use a fuel transfer system to keep the cg at the forward location
	Insufficient control power for almost all the maneuvers.	Increase LoCE control power by relocating the wings
Wing Location Trades (HS)	There are control power issues at extremely negative SM	Increase the thrust requirement for cruise in a later sizing study
	The most control power is available at -5% SM	Assess control power at off design conditions using the -5% SM wing location
	Insufficient thrust available at -5% SM	
Stall Performance (IC, LM)	LoCEs saturate during most of the maneuvers in both conditions	Redesign the wing to delay the pitch break
		Use an angle of attack limiter to constrain angle of attack to safe pitching moment regions
Crosswind Performance (A, F)	Only the LaCEs saturate during the approach	Resize the ailerons for more lateral control power
	Both the LoCE and LaCE saturate during the landing flare	Relocate engines for more directional control power
		Add undersized rudders
Engine Limit Conditions (C, HS-ETOPS, LM-ETOPS)	Insufficient thrust available for all the maneuvers with the HS-ETOPS pull-up being the most demanding	Use the thrust requirement for the HS-ETOPS pull-up to size the engines
	The LM-ETOPS condition suffers from the same pitch break problems as the stall conditions	Wing redesign or an angle attack limiter is required to curb the pitching moment problem near stall

5.4 Dynamic Stability Analysis Using the Integrated FCS Module

Although some modifications are required for the TVC transport to meet steady state control power requirements, due to time constrains these changes were not implemented before the dynamic analysis. Thus, this dynamic analysis only serves as a proof of the capabilities of the FCS module in *AeroMech*. The capabilities demonstrated are:

1. open loop dynamic stability analysis

2. design of 2 longitudinal flight control systems and the dynamic stability analysis of the resulting closed loop systems
3. design of 1 lateral flight control system and the dynamic stability analysis of the resulting closed loop system
4. comparison between a stable TVC transport configuration and an unstable TVC transport configuration to demonstrate the concept of flight vehicle hardware plus FCS shaping for good stability characteristics as a prelude for good handling qualities shaping.

The open loop dynamic stability analysis is described in the next section.

5.4.1 Open Loop Dynamic Stability Analysis

The open loop dynamic analysis involves comparing the Eigenvalues of the open loop system to the flying qualities requirements in (Anonymous1986). Various requirements are given depending on the vehicle size and mission phase. A commercial transport of this size during cruise is classified as a class III vehicle in phase B (Roskam 2001). The longitudinal and lateral-directional flying qualities requirements for this classification are shown in Table 5.3 and Table 5.4 respectively.

Table 5.3 Longitudinal flying qualities for a class III vehicle in phase B from (Anonymous1986)

Mode	**Level 1**	**Level 2**	**Level 3**
Phugoid	$\zeta \geq 0.04$	$\zeta \geq 0.0$	$T_{double} \geq 55.0$
Short Period	$0.3 \leq \zeta \leq 2.00$	$0.20 \leq \zeta \leq 2.00$	$\zeta \geq 0.15$
	$0.085 \leq \omega_n^2/n_\alpha \leq 3.60$	$0.038 \leq \omega_n^2/n_\alpha \leq 10.0$	$\omega_n^2/n_\alpha \geq 0.38$

Table 5.4 Lateral-directional flying qualities for a class III vehicle in phase B from (Anonymous1986)

Mode	Level 1	Level 2	Level 3
Spiral	$T_{double} \geq 20$ s	$T_{double} \geq 8$ s	$T_{double} \geq 4$ s
Dutch Roll	$\zeta \geq 0.08$	$\zeta \geq 0.02$	$\zeta \geq 0.02$
	$\zeta\omega_n \geq 0.15$	$\zeta\omega_n \geq 0.05$	no limit on $\zeta\omega_n$
	$\omega_n \geq 0.4$	$\omega_n \geq 0.4$	$\omega_n \geq 0.04$
Roll	$\tau \leq 1.4$ s	$\tau \leq 3.0$ s	$\tau \leq 10$ s

The longitudinal and lateral open loop Eigenvalues computed for the TVC commercial transport during high speed cruise are shown in Table 5.5 and Table 5.6 respectively. These results cannot be correlated with flying qualities requirements because the vehicle is so unstable that there is no definite phugoid or short period mode. The instability is seen in the poles plot shown with Figure 5.10 and the time response to doublet inputs shown in Figure 5.11 to Figure 5.13. There are two open loop poles on the Right Hand Side (RHS) of the imaginary axis in the poles plot which signifies that the system will diverge. This divergence is very severe as evidenced by the magnitudes of the time responses after 15 seconds. This vehicle has terrible flying qualities which need to be fixed by using a Flight Control System. Two longitudinal FCS are designed and the resulting closed loop systems are analyzed in the next section.

Table 5.5 Longitudinal open loop Eignenvalues of the TVC commercial transport

Real ($-\zeta\omega_n$)	**Imaginary (ω_d)**	**ω_n**	**ζ**	**T_{double}**	**ω_n^2/n_α**
0.2502	0	0.2502	-1	2.77	-
0.0307	0	0.0307	-1	22.58	-
-0.0280	0	0.0280	1	-	0.0001
-0.6641	0	0.6641	1	-	0.048

Table 5.6 Lateral-directional Eigenvalues of the TVC commercial transport

Real ($-\zeta\omega_n$)	Imaginary (ω_d)	ω_n	ζ	T_{double}	τ
0.9287	0	0.9287	-1	0.75	1.08
0.0115	0	0.0115	-1	60.27	86.96
-0.7983	0	0.7983	1	-	-1.25
-1.3319	0	1.3319	1	-	-0.75

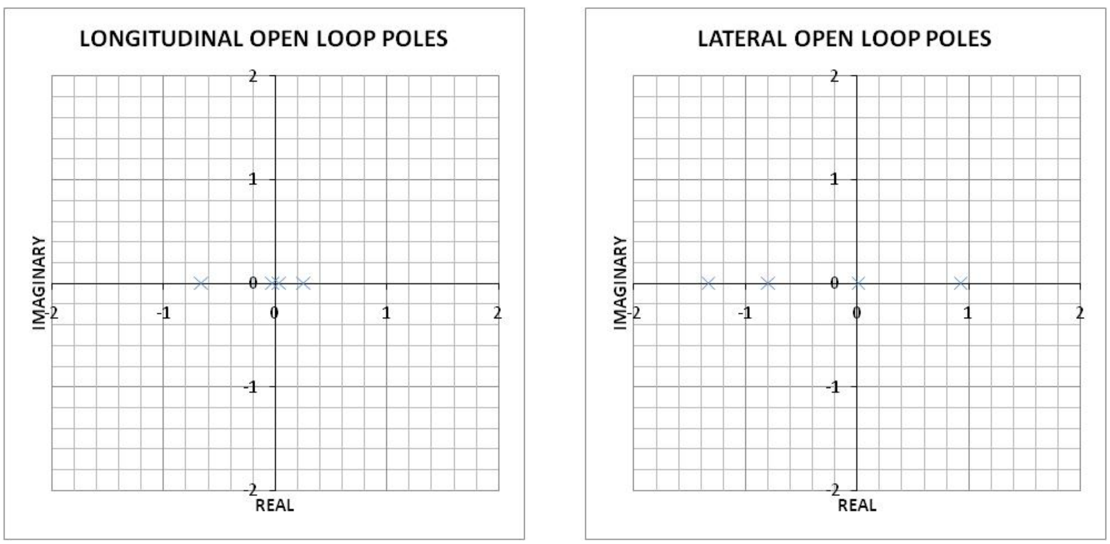


Figure 5.10 Longitudinal and lateral-directional open loop poles plots for the TVC transport

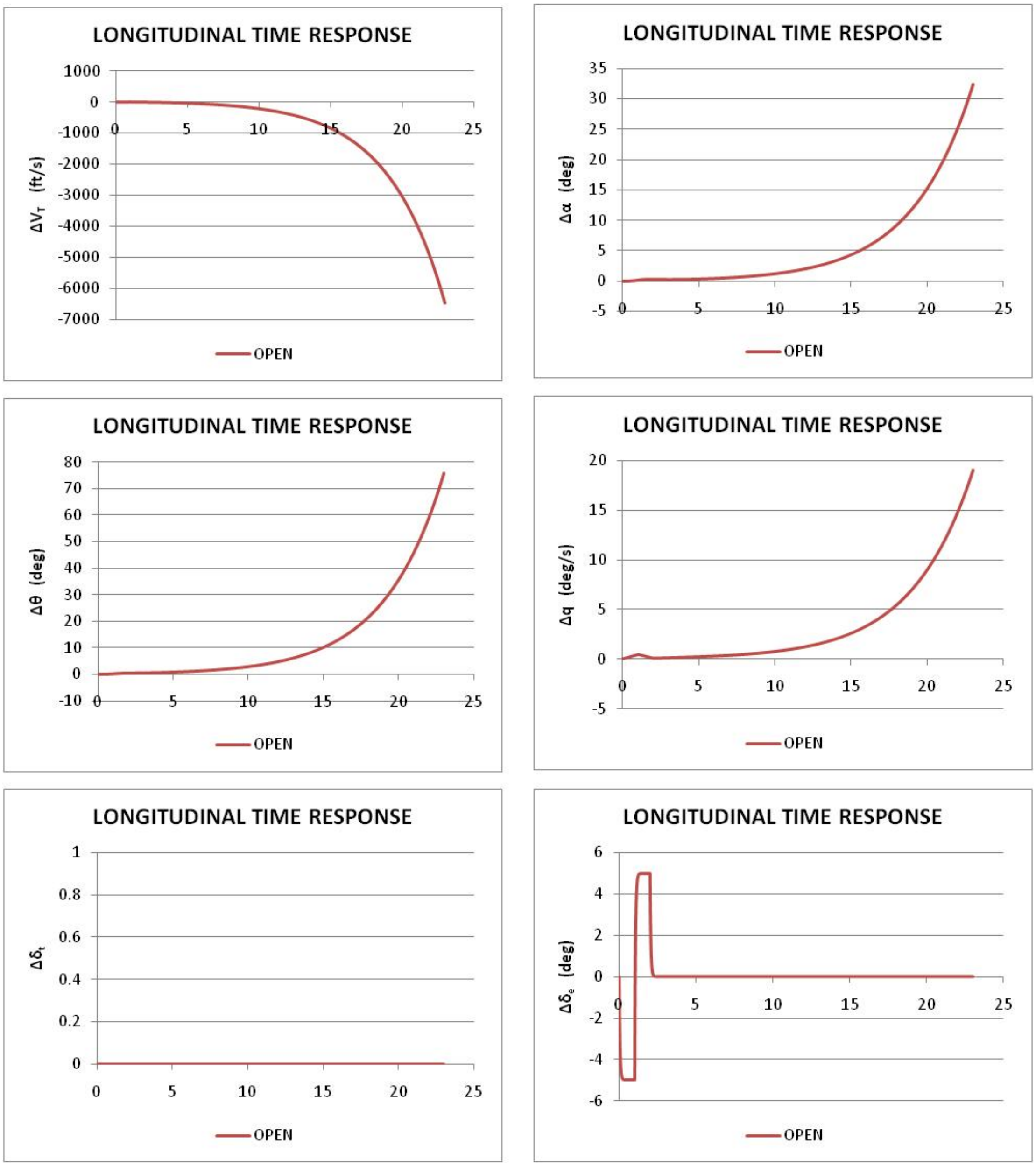


Figure 5.11 Open loop time response of the TVC transport to a 5 deg elevator doublet

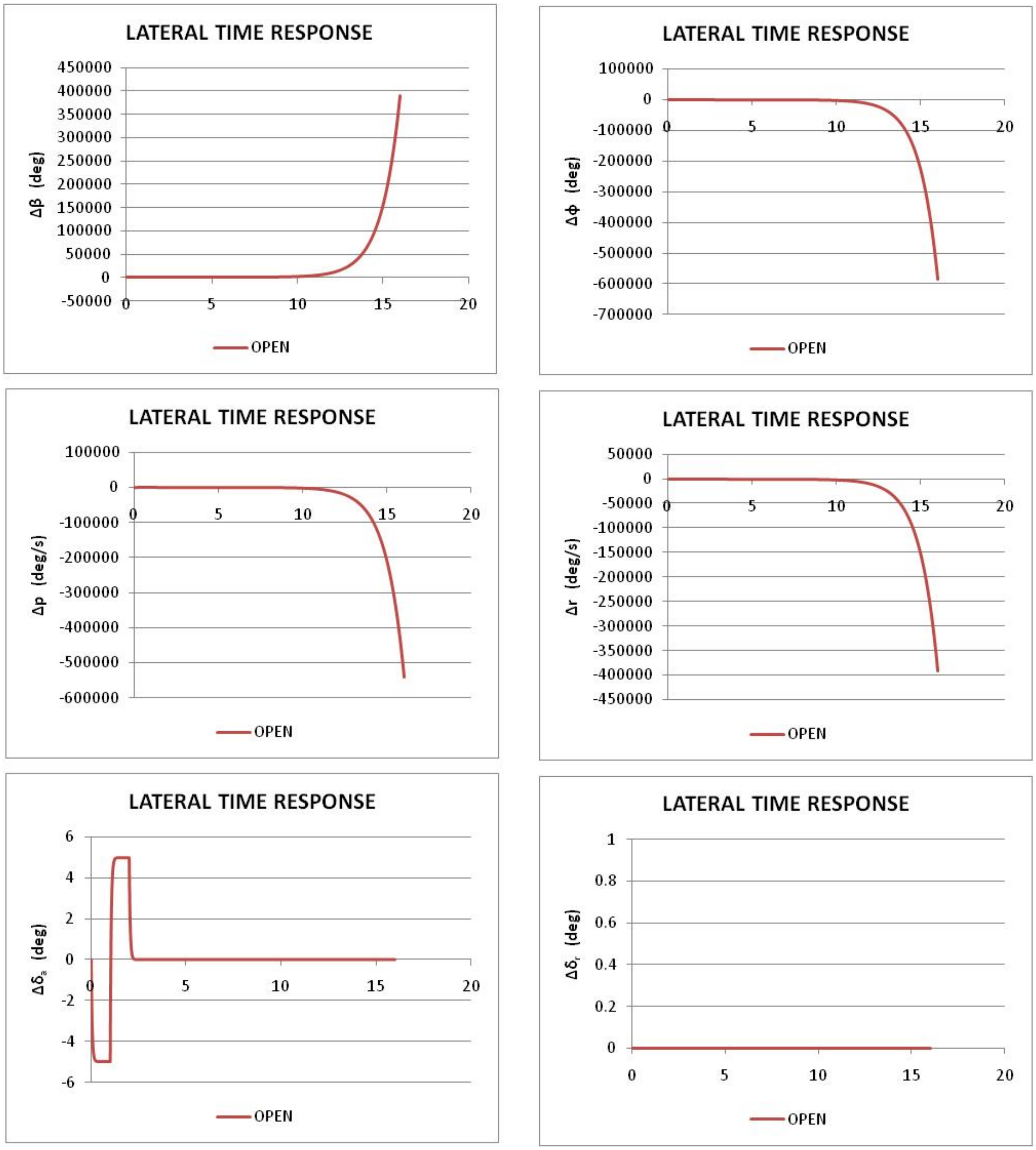


Figure 5.12 Open loop time response of the TVC transport to a 5 deg aileron doublet

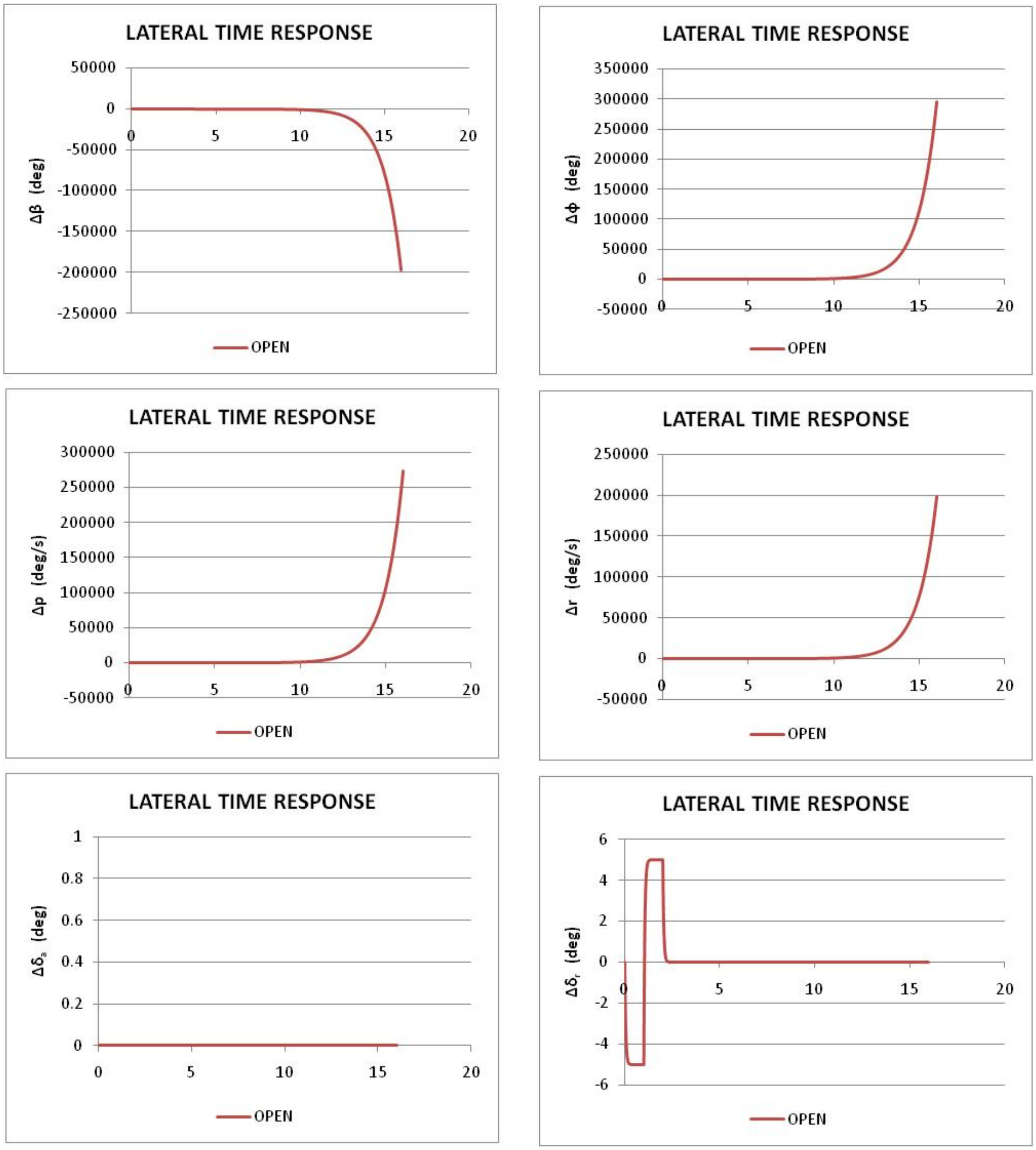


Figure 5.13 Open loop time response of the TVC transport to a 5 deg rudder doublet

5.4.2 Longitudinal FCS Design and Dynamic Analysis of the Resulting Closed Loop System

Before designing the FCS, it is necessary to define time constants for the angle of attack filter and the LoCE and throttle actuators. The selections for this case study are shown in Table 5.7. These time constants are needed because they help the model better represent the dynamics of a real aircraft. A filter is placed in the angle of attack feedback loop because angle of attack sensors are very susceptible to noise, thus requiring a filter for noise attenuation. The ability to model a filter in the angle of attack feedback channel is one of the reasons why the Linear Quadratic Regulator (LQR) with output feedback design is selected for this research (see Chapters 2 and 3). Another reason for choosing this design technique is because it allows the selection of any desired control structure. As discussed in Chapter 4, 15 different longitudinal feedback structures are programmed in the FCS module integrated in *AeroMech*. For the sake of brevity, only two longitudinal control structures are used in this case study. They are:

1. angle of attack feedback,
2. pitch rate plus angle of attack feedback.

The dynamic analysis of the closed loop systems resulting from these control structures are discussed next.

Table 5.7 Time constants used in the longitudinal model

	1/τ (1/s)
Angle of Attack Filter	10.0
Throttle Actuator	0.2
LoCE Actuator	20.2

As discussed in Chapter 3, the LQR output feedback problem has been formulated to have only two dependent variables, ρ and k. Selecting $k = 2$, is sufficient to overcome the observability problem, thus, only ρ needs to be adjusted to give the desired system response. Also, previously discussed is the fact that the introduction of controller dynamics invalidates the use of the regulations in (Anonymous1986) as a measure of handling qualities. However, since the specification of handling qualities criteria has been reserved for a later study, it is assumed that the seventh order closed loop dynamics is similar to a fourth order system in (Anonymous1986). Therefore, the dynamic analysis is done by comparing the Eigenvalues of the closed loop system to the regulations in (Anonymous1986).

After some adjustments of ρ, it is discovered that the system cannot be stabilized to level 1 flying qualities by using an angle of attack feedback gain below 10 deg/s. In order to confirm this, a root locus plot made using MATLAB is shown in Figure 5.14[2]. From this figure, the maximum possible phugoid mode damping ratio is 0.0151 which occurs at a gain of 10 deg/s. The level 1 requirement is for a ratio greater than 0.4. Table shows results for a gain of -4.85 deg/s. This gain is selected because it is below 5 deg/s minimum specified in Chapter 2.

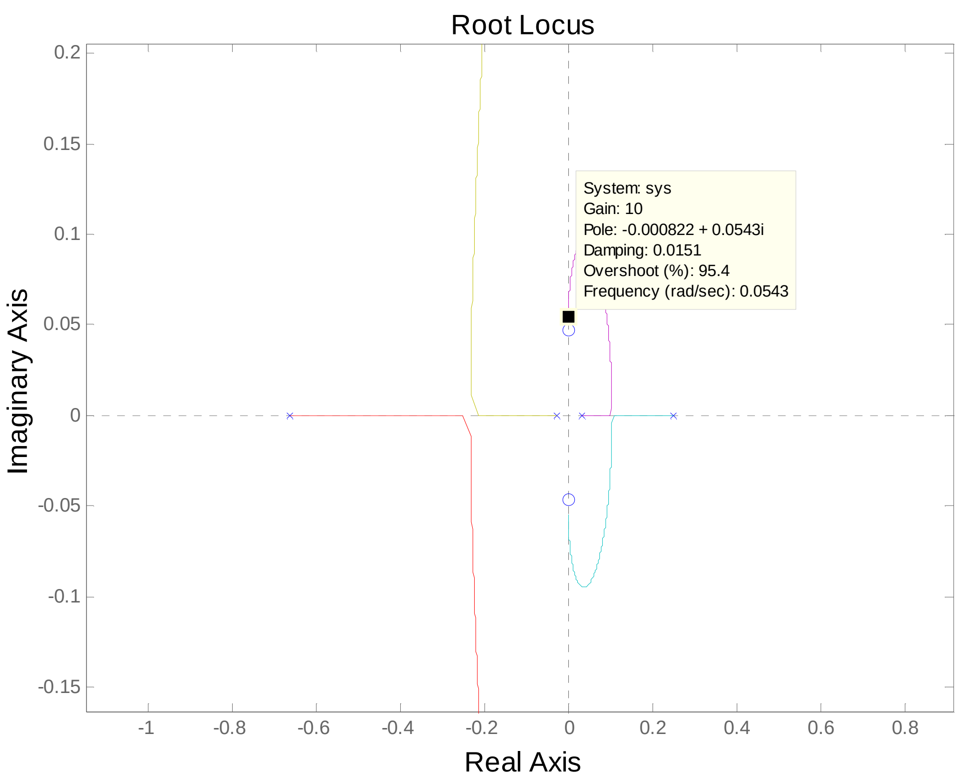


Figure 5.14 Root locus plot

[2] It is possible to use root locus because this is a Single Input Single Output System

Table 5.8 Longitudinal closed loop Eigenvalues of the TVC transport with α feedback

Real ($-\zeta\omega_n$)	**Imag. (ω_d)**	**ω_n**	**ζ**	**T_{double}**	**ω_n^2/n_α**
0.001	-0.0653	0.0653	-0.0151	693.15	
0.001	0.0653	0.0653	-0.0151	693.15	
-0.1787	-0.4675	0.5005	0.357		0.027
-0.1787	0.4675	0.5005	0.357		0.027
-0.2	0	0.2	1		0.004
-10.0672	0	10.0672	1		
-20.1886	0	20.1886	1		

From the results, the closed loop system has level 1 short period flying qualities and level 3 phugoid flying qualities. In addition, if a disturbance causes an angle of attack deviation of 2 deg, then the additional control power required is 10 degrees of LoCE deflection. Since, the vehicle trimmed at 18 deg (see Figure 5.6), the total control power requirement is 28 deg. This does not saturate the control effector. In the next section, an angle of attack plus pitch rate feedback controller is designed.

It is possible to obtain level 1 flying qualities using angle of attack plus pitch rate feedback. After some adjustments, it is discovered that a ρ of 15.0 gives the desire response. The corresponding gains and Eigenvalues are shown in Table 5.9 and Table 5.10.

Table 5.9 Longitudinal closed loop Eignenvalues of the TVC transport with α plus q feedback

Real ($-\zeta\omega_n$)	**Imag. (ω_d)**	**ω_n**	**ζ**	**T_{double}**	**ω_n^2/n_α**
-0.0008	-0.0412	0.0412	0.0198		0.000
-0.0008	0.0412	0.0412	0.0198		0.000
-0.2	0	0.2	1		0.004
-0.2585	0	0.2585	1		0.007
-0.9997	0	0.9997	1		0.108
-10.0241	0	10.0241	1		
-19.3274	0	19.3274	1		

Table 5.10 Angle of attack and pitch rate feedback gains

$K_{\delta_{LoCE}/\alpha_f}$	$K_{\delta_{LoCE}/q}$
-1.4424	-9.4367

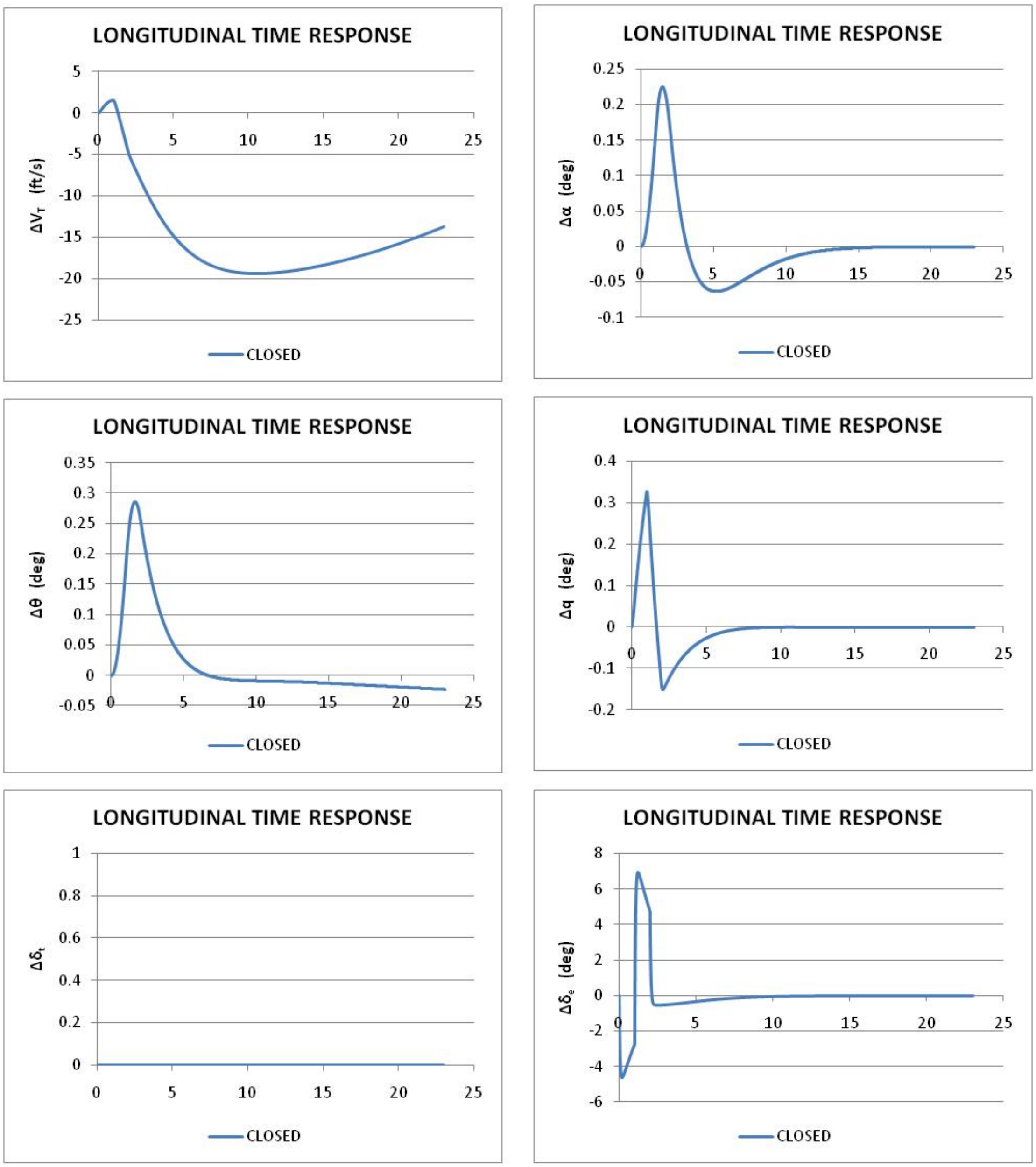


Figure 5.15 Closed loop time response of the TVC transport to a 5 deg elevator doublet

The closed loop time responses to a 5 deg elevator doublet at this gain setting are shown in Figure 5.15. This response converges quickly unlike the open loop system shown in Figure 5.11 which does diverge. The down side to this gain selection is that the pitch rate feedback gain exceeds the 2 deg limit specified in Chapter 2. If a disturbance induces both a 2 deg angle of attack and a 2 deg/s pitch rate, the resulting additional control power is 21.8 degrees of LoCE deflection. Since, the vehicle is trimming at 18 deg (see Figure 5.6), the total control power requirement is 39.8 deg. This deflection will saturate the control effector and is therefore unacceptable. This example demonstrates the interaction between flight vehicle choice and resulting FCS difficulties trying to stabilize a statically unstable vehicle. The next section discusses the design of a lateral FCS for this vehicle.

5.4.3 Lateral-Directional FCS Design and Dynamic Analysis of the Resulting Closed Loop System

In the same manner as for the longitudinal case, it is necessary to define time constants for the yaw washout filter and the aileron and DiCE actuators. The values selected for this study are shown in Table 5.11. The control structure selected after some iterations is a washed-out yaw rate plus sideslip angle plus roll rate plus bank angle feedback. $\rho = 98$ is selected. The resulting gains and Eigenvalues are shown in Table 5.12 and Table 5.13 respectively.

Table 5.11 Time constants used in the lateral model

	1/τ (1/s)
Yaw Washout Filter	0.25
LaCE Actuator	20.2
DiCE Actuator	20.2

Table 5.12 Lateral feedback gains

K_{δ_{DiCE}/r_w}	$K_{\delta_{LaE}/p}$	$K_{\delta_{DiCE}/\beta}$	$K_{\delta_{LaCE}/\phi}$
0	0.175	0	-0.625
-15.782	0	33.642	0

Table 5.13 Lateral-directional closed loop Eigenvalues of the TVC transport

Real ($-\zeta\omega_n$)	**Imaginary (ω_d)**	**ω_n**	**ζ**	**T_{double}**	**τ**
-0.2606	-1.2902	1.3162	0.198		
-0.2606	1.2902	1.3162	0.198		
-0.3129	0	0.3129	1		3.20
-0.5298	-0.8143	0.9715	0.5454		
-0.5298	0.8143	0.9715	0.5454		
-19.2247	0	19.2247	1		
-20.7215	0	20.7215	1		

The system is directionally highly unstable, thus it requires large sideslip and yaw rate gains. This instability exists because of the lack of side area aft of the center of gravity. A similar problem is identified in (Colgren and Loschke 2008, 1441-1449). One solution is to add light weight vertical fins aft of the cg to increase directional stability. This example also demonstrates another major challenge for the TVC transport aircraft. In the next section, a longitudinal FCS is designed for a stable TVC configuration to indentify a balance between FCS and hardware shaping.

5.4.4 Longitudinal FCS Design and for a Statically Stable TVC Configuration

In order to demonstrate the concept of hardware plus flight control system shaping, the following case is considered. An FCS is designed for the statically stable vehicle configuration show in Figure 5.5. The control gain and Eigenvalues are shown in Table 5.14 and Table 5.15 respectively.

Table 5.14 Angle of attack plus pitch rate feedback gains for the stable TVC configuration

$K_{\delta_{LoCE}/\alpha_f}$	$K_{\delta_{LoCE}/q}$
0	0
0.6877	-0.7948

Table 5.15 Longitudinal closed loop Eigenvalues of the stable TVC configuration

Real ($-\zeta\omega_n$)	**Imag. (ω_d)**	**ω_n**	**ζ**	**T_{double}**	**ω_n^2/n_α**
-0.0004	-0.034	0.034	0.0119		0.000
-0.0004	0.034	0.034	0.0119		0.000
-0.2	0	0.2	1		0.004
-0.2553	-0.5429	0.5999	0.4256		0.039
-0.2553	0.5429	0.5999	0.4256		0.039
-9.9907	0	9.9907	1		
-20.1328	0	20.1328	1		

These Eigenvalues correspond to a level 2 performance as with the unstable configuration. However, the required gains are reduced, thus the additional control power decrease and so does the total control power requirement for this case.

Having discussed flying quality and FCS trades, it can be concluded that this research envisions a situation where handling quality requirements are used to judge the effectiveness of the gains resultin in design trades to be performed where the flight vehicle hardware and the flight control system are traded to fulfill a global objecyive function.

5.5 Chapter Summary

In conclusion, this chapter applies the knowledge and resources gained throughout this research undertaking. The design application chosen has been the challenging stability and control analysis trades towards a Thrust Vector Control (TVC) commercial transport concept. The TVC transport study has been the primary vehicle chosen due to theactive research assignment by NASA Langley Research Center (LaRC). Although the research for a certifiable TVC transport has not yet been finished, the steady state control power assessment of the TVC transport using *AeroMech* is vividly demonstrating the power of the approach to generate physical design insights into flight vehicle hardware and FCS shaping. Clearly, the results of this assessment show the sensitivities to arrive at a performance-superior TVC vehicle consisting of a balanced flight vehicle hardware and FCS design. It is beyond the present MS research investigation to further probe the feasibility of a commercially viable TVC transport. However, the preliminary TVC transport results demonstrate that the open loop longitudinal and lateral system is highly unstable. It is also shown that angle of attack feedback and pitch rate plus angle of attack FCS structures are not sufficient to produce a system with flying quality characteristic complying with regulations. Additionally, the gains required to stabilize the system have been demonstrated to increase the control power requirement drastically. Finally, at the end of the chapter, a trade between static margin and FCS gains has highlighted the significance of the implemented controlled onto the design opportunities offered with Control-Configured Vehicles (CCVs).

CHAPTER 6

CONTRIBUTIONS, RECOMMENDATIONS AND REFLECTION

6.1 Contributions Summary

This current research undertaking is in essence an approach to Flight Control System (FCS) design from an aircraft conceptual designer's point of view. In approaching the problem from this perspective, a broader objective is realized in which the goals of flight control system design are married with key objectives of conceptual design. The purposes of flight control systems, specifically stability augmentation systems, directly translate into artificially making aircraft have desired handling qualities. On the other hand, the objective of conceptual design is to ascertain the best vehicle concepts for a specified mission. The merger of these two objectives births the idea of designing the total aircraft, including airframe and flight control system, to possess desirable handling qualities while meeting mission requirements. There are five major contributions of this current research undertaking which are:

1. the identification of a unique research problem;
2. the development of specifications for a flight control system design technique that is suitable for conceptual design to meet handling qualities requirements;
3. the systematic formulation, programming and validation of a standalone flight control system design technique for use in conceptual design by compiling ideas from various literary sources;
4. the identification of key interfaces required for integrating an FCS design technique into a conceptual design synthesis environment;
5. the stability and control analysis of a Thrust Vector Control (TVC) commercial transport aircraft.

These contributions as expounded in the following subsections.

6.1.1 The Identification of a Unique Research Problem

The concept of shaping the aircraft airframe and flight control system (FCS) concurrently to provide desired handling qualities is a unique idea not discusses in any conceptual design literature. A literature review of various conceptual design texts reveals that aircraft FCS is typically not considered in any aircraft sizing methodologies. In the cases where it is, the FCS is used to augment flying qualities without the use of specific handling qualities criteria for augmented aircraft. The closest attempt to simultaneously shape the airframe and FCS is described in (MORRIS 1992). In this methodology, the airframe and handling qualities are shaped simultaneously using an unconstrained penalty and optimization process. However, there is no direct use of commonly accepted handling qualities criteria in the formulation. Handling qualities are accounted for in the formulation by introducing a penalty function which weights the deviation of the designed aircraft from a prescribed model with good handling qualities. This formulation is a step towards handling qualities shaping. However, without the use of specific handling qualities criteria; it does not address the problem directly. The uniqueness of this topic is further confirmed by a personal communication between Chudoba and William Mason from Virginia Tech:

> "Bernd, thanks for email… I would agree that the configuration should work together with the FCS to produce a really good airplane. I don't have an example showing this though, but I bet it would be possible." (Mason 2010)

In an attempt to address this research problem systematic, it has quickly become obvious that the topic is beyond the scope of a single master's thesis. Therefore, this current research undertaking does focus on the FCS design technique which represents as the interface between the conceptual design objective and the handling qualities goal.

6.1.2 The Development of Specifications for a Flight Control System

The development of a dedicated set of research specifications requires a clear understanding of the underlying cause-effect parameters involved between airframe and FCS design. The background research performed resulted in a clear identification of key effects and implications of FCS during the conceptual design phase. In this context, a ‘best practice’ flight control system design guideline has been formulated. Although the Linear Quadratic Regulator (LQR) with output feedback design technique is not typically used for preliminary design, this survey shows that it is the correct one for the conceptual design phase.

6.1.3 Formulation, Programming and Validation of a Standalone FCS Design Technique

There are many different formulations of LQR with output feedback design available in literature. In order to arrive at a formulation that is most suitable for the specifics of the conceptual design phase, a derivation of desired elements of the FCS design methodology has been required. These derivations are individually, however, available in literature. The contribution of this research has been to correctly select the appropriate modules such as the weight penalty constraint method, the expansion of implied elements, and the control of a similar system into the dedicated conceptual design level methodology.

6.1.4 The Integration of an FCS Design Technique into a Conceptual Design Environment

The contribution related to the integration exercise is the identification of key interfaces required between the FCS module and a conceptual design stability and control tool. The candidate stability and control tool for conceptual design is *AeroMech*. *AeroMech* is a dedicated control effector sizing methodology and software, capable of assessing control power, trimmed aerodynamics and analyzing static and dynamic stability and control characteristics of any aircraft configuration concept (Chudoba 2001; Coleman 2007, 283; Chudoba and others 2008, 293). The interfaces required for the integration of a modern FCS system design methodology have been identified.

6.1.5 Stability and Control Analysis of a TVC Commercial Transport Aircraft

The contribution summary from the resulting total aircraft stability and control analysis include a steady state control power assessment and a dynamic stability assessment of a unique aircraft configuration. The steady state control power assessment of the TVC transport using *AeroMech* vividly demonstrates the power of the approach to generate physical design insights into flight vehicle hardware and FCS shaping. The results of this assessment show the sensitivities to arrive at a performance-superior TVC vehicle consisting of a balanced flight vehicle hardware and FCS design. Note that it is beyond the present MS research investigation to further probe the feasibility of a commercially viable TVC transport.

6.2 Recommendations for Future Work

Although the intermediate goal of selecting and integrating a handling quality-capable FCS module into the conceptual design phase is complete, there are still broader objectives which are still pending resolution. These recommendations for future studies include:

1. A detailed review of all available handling qualities texts. The review of a few texts for this current research undertaking alluded to the identification of the handling qualities problem at hand. A more in-depth understanding of the problem is required in order to arrive at a solution.
2. The integrated FCS module prototype needs to be used in designing and analyzing the Flight Control System of many vehicles of different configurations in order to fully understand its capabilities. The FCS module developed has been only validated for the test cases described in this text due to time constraints and the need to address the unique stability concerns of the TVC transport.
3. The FCS module needs an expansion in order to design the control laws for asymmetric configurations such as the oblique-wing aircraft and for asymmetric flight conditions such as turning flight. The presence of the numerical linearization subroutines and the ability of the implemented design technique to allow the design of any desired control structure will be valuable to such a research. However, a lot of research is required in order to define the control schemes for such novel vehicles.
4. Finally, a systematic study is required to arrive at a specification, selection, implementation and validation of a methodology and process for shaping aircraft

hardware and flight control systems for desired handling qualities during the conceptual design phase.

6.3 Reflection on the Research Experience

This research endeavor has been both challenging and rewarding to this author. One of the challenges was approaching a broad topic such as designing for handling qualities during the conceptual design phase. It is not clear whether there is enough resolution in conceptual design to evaluate handling qualities; however, this is a worthwhile topic to research. Another challenge was difficulties that exist in interfacing between the conceptual design and preliminary design engineers. The lesson learned is that both these groups are passionate about what they do. And it is important to stress that goal of using preliminary design techniques during conceptual design is not to replace the preliminary design engineers but to compliment them by factoring in their concerns into early decision making.

APPENDIX A

THE EFFECT OF TIME WEIGHTING PARAMETERS

Table A.1 Variation of gains with the weighting parameter ρ

ρ	K
0.01	$\begin{bmatrix} 0 & -2.00 & 0 & -10.5 \\ -3.90 & 0 & 16.8 & 0 \end{bmatrix}$
0.1	$\begin{bmatrix} 0 & -0.95 & 0 & -4.32 \\ -2.17 & 0 & 5.37 & 0 \end{bmatrix}$
1	$\begin{bmatrix} 0 & -0.61 & 0 & -2.00 \\ -1.16 & 0 & 1.38 & 0 \end{bmatrix}$
10	$\begin{bmatrix} 0 & -0.50 & 0 & -0.91 \\ -0.56 & 0 & 0.38 & 0 \end{bmatrix}$
100	$\begin{bmatrix} 0 & -0.27 & 0 & -0.29 \\ -0.26 & 0 & 0.12 & 0 \end{bmatrix}$

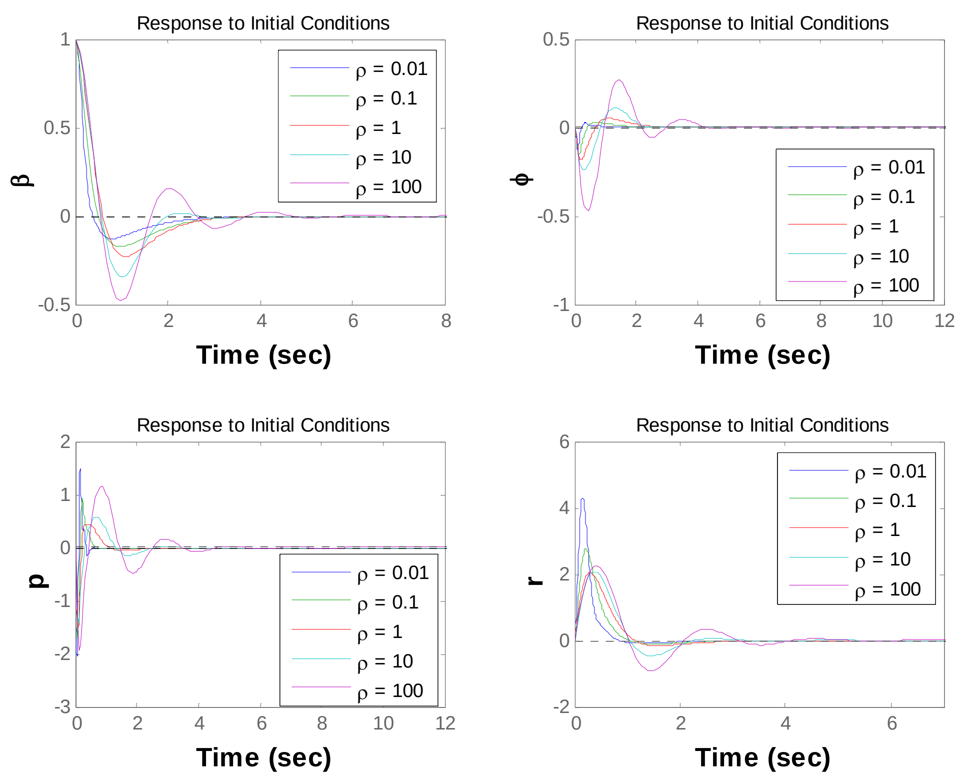


Figure A.1 Variation in time responses with the weighting parameter ρ

Table A.2 Variation of gains with the time power constant k

$\boldsymbol{k}$	$\boldsymbol{K}$
0	$\begin{bmatrix} 0 & -1.17 & 0 & -1.10 \\ -0.82 & 0 & -0.10 & 0 \end{bmatrix}$
1	$\begin{bmatrix} 0 & -0.81 & 0 & -1.61 \\ -1.09 & 0 & 0.50 & 0 \end{bmatrix}$
2	$\begin{bmatrix} 0 & -0.61 & 0 & -2.00 \\ -1.16 & 0 & 1.38 & 0 \end{bmatrix}$
3	$\begin{bmatrix} 0 & -0.60 & 0 & -2.10 \\ -1.18 & 0 & 2.36 & 0 \end{bmatrix}$
4	$\begin{bmatrix} 0 & -0.67 & 0 & -2.32 \\ -1.07 & 0 & 3.12 & 0 \end{bmatrix}$

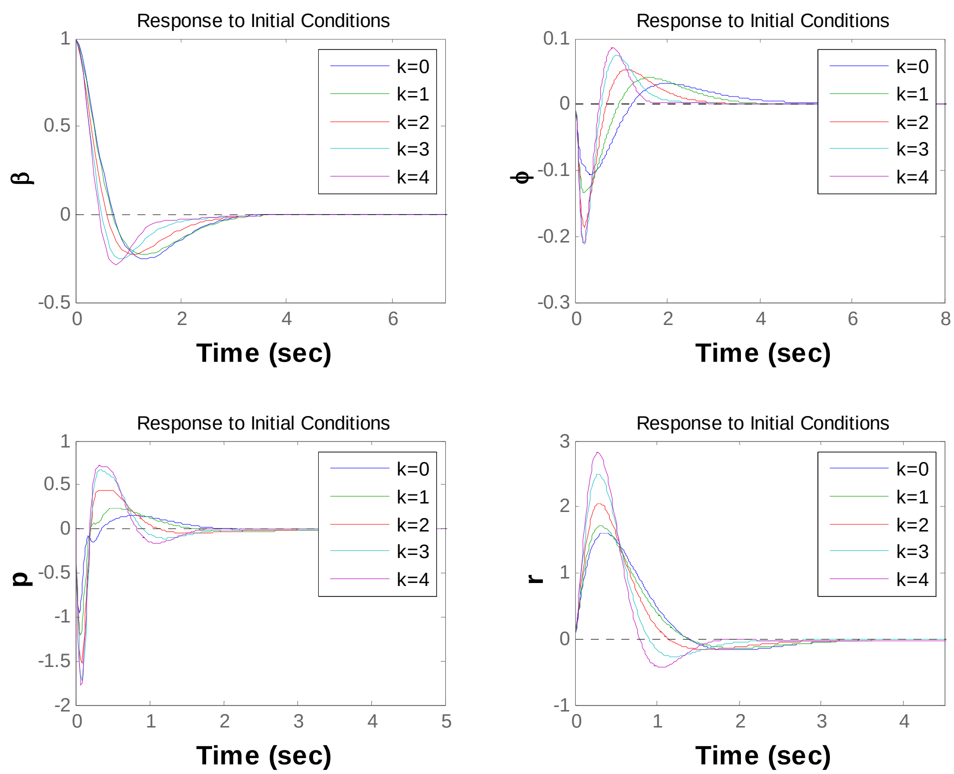


Figure A.2 Variation in time responses with the time power constnt k

APPENDIX B

LIST OF SPECIALISTS CONTACTED

Table B.1 Industry and Academia flight dynamicist and Designers contacted

Name	Company	Tool	Text
Abzug L. Malcom	Retired	ILOCS	Computational Flight Dynamics
Arthur Rizzi			
Bailey, Roger	Cranfield		
Bernard Etkin	University of Toronto		Dynamics of Atmospheric Flight
Bowcutt, Kevin G	Boeing		
Brian D. Anderson	Australian National Universtiy		Optimal Control Linear Quadratic Methods
Brian Stevens	Georgia Tech	JACOB, TRIMMER	Aircraft Control and Simulation
Buckley B. Stams	Lockheed Martin		
Carty, Atherton	Lockheed Martin		
Chaput, Armand J	University of Texas Austin		
Chris Coting	Virginia Polytechnic		
Clay M. Thompson			
Cook, Mike	Cranfield		
Dan DeLaurentis	Georgia Tech		
David Klyde	STI		
Dunbrack, Harold	Wyle		
Edmund Field	Boeing		
Engelbeck, Ranald M	Boeing		
Frank Lewis	University of Texas Arlington	JACOB, TRIMMER	
Gerald Blausey	Lockheed Martin		
Green, Lawrence L.	NASA		
Guynn, Mark D.	NASA		
Hahn, Andrew S.	NASA		
Irving Ashkenas			
Ivan Burdun			
Jacob Kay	Bihrle Applied Research		
James D. Blight	Northrop Grumman Corporation		Practical control law design for aircraft using multivariable techniques
John C. Doyle	Cal. Tech		Essentials of robust control
John Gibson			Development of a methodology for excellence in handling qualities design for fly by wire aircraft
John Hodgkinson	Boeing		Aircraft Handling Qualities
John Valasek	Texas A&M		
Kenneth T. Moore	NASA	MaSCoT	
Leavitt, Laurence D.	NASA		
Leland Nicholai	Lockheed		

Table B.1 – *Continued*

	Martin		
Liebeck, Robert H	NASA		
Lloyd Duff Reid	University of Toronto		Dynamics of Flight: Stability and Control
Mark B. Tischler	ATCOM, Ames research Cener	CONDUIT	CONDUIT—A NEW MULTIDISCIPLINARY INTEGRATION ENVIRONMENTFOR FLIGHT CONTROL DEVELOPMENT,
Mark Dreier	Bell		
Mark More	NASA LRC		
Mike Butler	Team-ADSI		
Neal Pfeiffer	Hawker Beech Craft		
Nickol, Craig L.	NASA		
Oliver Brieger	DLR		
Paul Czysz			
Peter H. Zipfel	University of Florida	CADAC	Modeling and Simulation of Aerospace Vehicle Dynamics
Prof. Voit-Nitschmann			
Rob Wolz	Gulfstream		
Robert F. Stengel	Princeton University	Flight and Survey	Flight Dynamics
Robert G. "Bob" Hoey	USAF, Testing Center		
Schieck, Florian			
Sid Banerjee	Bantec		
Svoboda, Charles	Boeing		
Sylvain POUILLARD			
Warren F Philips	Utah State University		Mechanics of flight
William Mason	Virginia Polytechnic		
Wolf Roeger			
Wolf, Gerhard	Airbus		

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BIOGRAPHICAL INFORMATION

Amen Omoragbon is a God fearing follower of Jesus Christ. He was born in 1988 in his native homeland, Nigeria, which is a country on the west of Africa. This is where he grew up and received his secondary education diploma from Corona Secondary School Agbara in 2003. It was during his secondary school days that he fell in love with airplanes. Enticed by their curves and the physics behind their gravity defying motion, he made a decision that he was going to eventually design these vehicles.

He later came to the United States of America for a college degree. He began his colligate career at the University of North Texas during the spring of 2004. Then transferred the following year to the University of Texas at Arlington (UTA), where he earned a Bachelor of Science degree in Aerospace Engineering in 2008. During his undergraduate studies, he joined various engineering honors societies, such as Tau Beta Pi and Pi Tau Sigma, and was elected as the chapter president of the Nation Society of Black Engineers. In addition to these activities, He was a member of the Autonomous Vehicle Laboratory at UTA which won multiple awards at the International AUVSI Unmanned Air Systems competitions.

He continued with his graduate studies at the University of Texas at Arlington in the fall of 2008 and joined the Aerospace Vehicle Design (AVD) laboratory the following summer. His decision to join the AVD laboratory was because the lab gives him an avenue to realize his childhood dream of designing airplanes. As a member of the lab, in addition to this current research undertaking, has supported in a NASA initiative to design the future long-haul commercial transport and has been the primary stability and control analyst for a thrust vector control commercial transport feasibility study performed for NASA.

Following this Master of Science degree, he plans to continue working at the AVD lab with a Doctor of Philosophy (Ph.D.) research on handling qualities in conceptual design.