ABSTRACT

VOSBURG, VICTOR JAY. The Stability and Control of an Aircraft with an Adaptive Wing. (Under the direction of Dr. Ashok Gopalarathnam.)

With increasing interest in the use of adaptive lifting surfaces for improved aircraft performance, it is necessary to study the stability and control characteristics of an aircraft with an adaptive wing. This research builds on recent development of an automated cruise flap for adapting a wing shape to achieve low drag over a large lift range. Such an automated cruise flap system was shown to have unusual lift and pitching moment curves, which prompted the need for studying the effect on aircraft stability and control. In this thesis, the static stability considerations are used to show that when an automated cruise flap is used on an airfoil to continuously adjust the flap to the optimum angle, there is a need for an accompanying controller that achieves the desired lift coefficient by adjusting the airfoil angle of attack. Likewise, when an automated flap is used on an aircraft wing, there is a need for an accompanying airspeed controller to adjust the elevator. The thesis presents Simulink models for analyzing the dynamic behavior of an aircraft with an automated flap. Two schemes were studied for the flap and elevator controllers. The most desirable results were obtained when the flap and elevator controllers were coupled so that trim changes due to the flap are immediately compensated by adjustment of the elevator angle. The results of the simulation show that the aircraft does not exhibit any undesirable behavior with the automated cruise flap. The study, therefore, provides the confidence needed to implement the system on an uninhabited aerial vehicle.

The Stability and Control of an Aircraft with an Adaptive wing

by

Victor J. Vosburg

A thesis submitted to the Graduate Faculty of
North Carolina State University
in partial fulfillment of the
requirements for the Degree of
Master of Science

Aerospace Engineering

Raleigh, NC
2004

APPROVED BY:

Dr. Ashok Gopalarathnam
Advisory Committee Chairman

Dr. Charles E. Hall, Jr.
Advisory Committee Member

Dr. Arkady Kheyfets
Advisory Committee Minor Rep.

To all my friends and family

whose continued support aid in the pursuit of all my goals.

BIOGRAPHY

Victor Jay Vosburg was born in Wilmington, NC on May 12, 1979, to Robert and Barbara Vosburg. He was raised in Ruffin, NC with his younger brother Ross. He graduated from Rockingham High School in 1997 and started his studies at North Carolina State University. He received a BS in Aerospace Engineering on May 15, 2002 with a Minor in Business Management.

His graduate career began in the spring of 2003 when pursuing his Masters in Aerospace Engineering with a Minor in Mathematics. Throughout his collegiate career he has been a volunteer member of Western Wake Fire & Rescue where he has achieved the position of Captain. After completing his first semester he was appointed as a teaching assistant where he graded assignments and assisted students. After having classes with Dr. Ashok Gopalarathnam, Victor decided to work under his direction for the remainder of his graduate studies. Victor's research was geared toward the stability and control of an aircraft that implements the use of an auto-adaptive wing, culminating in an AIAA conference presentation. After graduation, Victor plans on working within aerospace hopefully with the design, development, and testing of UAV's.

ACKNOWLEDGEMENTS

This thesis would not have been possible without the guidance and assistance of my advisor, Dr. Ashok Gopalarathnam. His dedication, support, high expectations of his students have provided me with the ability to successfully complete my research.

I would also like to thank Dr. Charles Hall and Dr. Arkady Kheyfets for agreeing to be on my advisory committee. I am privileged to have taken classes with both of them and appreciate the knowledge that they have provided.

I would like to thank The Clair Hunnicutt Scholarship for its support in providing my education.

I would like to acknowledge Unmanned Dynamics for the educational use of the AeroSim blockset. This software greatly helped me in the pursuit of my research goals.

Finally, I would like to give a special thanks to friends and family for their continued love, support, and patience as I pursue my many goals.

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Nomenclature

AR	aspect ratio
b	wing span
c	chord length
C_d	airfoil drag coefficient
C_{D0}	aircraft min drag coefficient
C_L	aircraft lift coefficient
C_l	airfoil lift coefficient
C_{L0}	aircraft lift coefficient for min drag
C_M	aircraft pitching moment coefficient
C_m	airfoil pitching moment coefficient
C_p	airfoil pressure coefficient
$\Delta C'_p$	pressure coefficient Eq. 1.1
i	incidence
K	3D correction factor
$\bar{l}_t$	tail moment arm measured to ac of the wing
M	Mach number
mac	mean chord length
p	pressure
Re	Reynolds number
S	area
t	time

$\overline{V}_h$	horizontal tail volume ratio referenced to ac
V_m	max velocity
x	x-coordinate
α	angle of attack
δ	control surface deflection
ϵ	downwash
η	efficiency factor for tail
ϕ	bank angle
θ_f	angular location of flap hinge in radians

Subscripts

ac	aerodynamic center
a	aileron
cg	center of gravity
e	elevator
f	flap
$flapped$	flapped
fus	fuselage
$ideal$	ideal condition from thin airfoil theory
l	location near mid chord on the lower surface
ll	location near leading edge on the lower surface
n	neutral point
p	propulsion system
ref	reference, wing
r	rudder
t	tail
u	location near mid chord on the upper surface
uu	location near leading edge on the upper surface
w	wing

Chapter 1

Introduction

A well-known means of adapting an airfoil to achieve low drag over a large lift range is by means of a trailing-edge cruise flap. Recent research has provided a means of automating the cruise flap to achieve the benefits of improved aircraft performance without an increase in pilot workload. In this research, the stability and control aspects of an aircraft with such an adaptive wing are studied.

1.1 Background

1.1.1 Airfoil Adaptation Using a Camber-Changing TE Flap

Adaptive wings are receiving increasing attention for their potential to reduce aircraft drag over a large speed range.^{1,2} One of the elementary forms of wing shape adaptation is the adaptation of airfoil camber by use of a small TE flap on the wing (see Fig. 1.1(a)). In its simplest implementation, this flap works at the level of two-dimensional flow and is deflected uniformly over the entire wing span. Such a flap, often referred to as a “cruise flap,” is widely used on high-performance sailplanes to adjust the airfoil drag polar so that the drag bucket or low-drag range (LDR) brackets the airfoil lift coefficient for a given flight condi-

Figure 1.1: (a) Geometry of airfoil with a TE flap, and typical locations of the four pressure sensors referred to in Section 1.1.2, and (b) drag polars showing effectiveness of TE flap in adapting the drag polar to suit a wide range of lift coefficients.

tion.³ Figure 1.1(b), from Refs. 4 and 5, shows how a cruise flap results in an effectively larger drag bucket when compared to the zero-degree flap setting. The results were obtained using the XFOIL code,⁶ and were validated using experimental data in Ref. 4. On sailplanes, the flap is set to the correct angle by the pilot for any given airspeed using some form of a lookup table and the reading of the airspeed indicator. While this increased pilot workload is tolerated in the world of competition sailplanes, this increased workload is believed to be an important reason why cruise flaps have not become popular on general aviation and other commercial aircraft.

1.1.2 Recent Auto-Adaptive Airfoil Research at NCSU

With an ultimate objective of bringing the performance benefits of cruise flaps to current-day piloted aircraft and enabling their use on uninhabited aerial vehicles (UAVs) and future autonomous aircraft, recent NCSU research by McAvoy and Gopalarathnam sought to automate the cruise flap based on the flow conditions sensed at the airfoil. The results of the concept development, estimated benefits,^{4,5} and wind-tunnel closed-loop demonstration⁴ are presented in this subsection in an abbreviated form as they provide important background material for the research

Figure 1.2: Variation of the stagnation-point location with airfoil lift coefficient for flap deflections of -10 , -5 , 0 , 5 , and 10 degrees.

Figure 1.3: Geometry of the airfoil with inset showing desired range for stagnation-point location.

described in this thesis.

It has been known for some time that a cruise flap works by bringing the stagnation point to a small desirable region near the leading edge, thus resulting in favorable velocity gradients on both the upper and lower surfaces of the airfoil.^{3,7} This stagnation-point behavior is illustrated in Figs. 1.2 and 1.3. It is clearly seen that as long as the stagnation-point location is maintained within a small region, the airfoil is operating within the low-drag region independent of the flap angle. The implications of this stagnation-point behavior are discussed in detail and are

Figure 1.4: $\Delta C'_p$ vs. C_l “calibration” curves for different flap angles showing desirable range of $\Delta C'_p$ that results in low drag.

quantified in Refs. 4 and 5. As a result of this behavior, the boundary layers on both the upper and lower surfaces experience favorable pressure gradients, thus promoting laminar flow (where transition is governed by the growth of T-S waves) over a wide range of lift coefficients.

For automating the cruise flap, a pressure-based sensing approach was developed by hypothesizing that if the stagnation point is indeed located within a small region when the flap is set to the optimum angle, then the pressure distribution in the vicinity of the leading edge should also be quite similar. A systematic computational study with XFOIL was used to develop a scheme where pressure measurement at just four locations on the airfoil surface could be used to sense the airfoil C_l and also set the flap to the correct angle. The four pressure measurements are used to define a non-dimensional $\Delta C'_p$ that is shown in Eq. 1.1. Figure 1.1(a) shows the typical locations of the four pressure ports on the airfoil, and Fig. 1.1(c) shows the variation of $\Delta C'_p$ with C_l for various flap settings. Two important observations can be made, both of which were verified^{4,5} using experimental data of pressure measurements on two flapped natural-laminar-flow (NLF) airfoils from NASA wind tunnel tests:^{3,8}

Figure 1.5: Comparison of the drag polars and lift and pitching-moment curves for the example NLF airfoil with automated cruise flaps and without flaps.

1. For any given C_l if the flap angle is set to maintain the $\Delta C'_p$ within a desirable range, then the airfoil operates close to the minimum drag for that condition.
2. The $\Delta C'_p$ curves in Fig. 1.1(c) can be used as “calibration curves” so that if the flap angle and the $\Delta C'_p$ are known, then the airfoil C_l can be determined.

$$\Delta C'_p = \frac{p_{lu} - p_{ll}}{|p_u - p_l|} = \frac{C_{p_{lu}} - C_{p_{ll}}}{|C_{p_u} - C_{p_l}|} \quad (1.1)$$

Closed-loop control of the TE flap on an airfoil model was demonstrated in a wind tunnel study. The goal was that the control system would determine and automatically set the airfoil angle of attack α and flap angle δ_f in order to achieve desired values of C_l and $\Delta C'_p$. Details of this study are presented in Ref. 4.

1.1.3 Issues for Aircraft Stability and Control

The resulting drag polar for an NLF airfoil with an automated cruise flap system is compared in Fig. 1.5 to the drag polar of the same airfoil without a cruise flap. The

polars for this airfoil have been plotted at a constant value of reduced Reynolds number $Re\sqrt{C_l}$ of 2 million. The use of a constant reduced Re ensures that the changes in Re with C_l due to changes in the flight velocity are automatically taken into consideration. From Fig. 1.5, while the large increase in the C_l range for low drag is clear, it is seen that for the automated-flap case, the change in α required for an increase in C_l is a small negative value. This differs from the usual lift-curve slope of approximately 2π per radian for the airfoil without the cruise flap. Additionally, the airfoil C_m about the quarter chord has a large variation when an automated cruise flap is used instead of a nearly constant C_m for an airfoil without a cruise flap. The nearly constant α for a wide C_l range for the automated-flap case may prove quite beneficial because the fuselage and nacelles can be optimized for a small variation in the aircraft α . The unusual variations in the C_l and C_m for the automated-flap case prompted the current study to examine of how they affect the aircraft stability and control.

1.2 Research Motivation & Objectives

An automated cruise flap has high potential for achieving improved aircraft performance over a large speed range. This is a result of the effectively having a larger airfoil drag bucket. The previous section demonstrated the unusual behavior of both the C_l and C_m verses α curves. This unusual behavior prompted the need to study and understand the effects of this automated flap on the aircraft stability and control.

In the pursuit of understanding the effects on an aircraft, the two primary objectives of the current research were to:

1. Provide an improved understanding of the effects of an automated flap on the aircraft static and dynamic stability and control characteristics.

2. Simulate an aircraft with an automated flap and demonstrate the implementation and benefits of such a system, which will help promote the applicability of the adaptive wing to commercial aviation, general aviation, and UAV's.

1.3 Brief Outline

This chapter has provided both the recent research in the area of auto-adaptive wings and the motivation for the current research. Having provided all the necessary background information, Chapter 2 will introduce the aircraft that was selected for this study and the changes that were necessary for an accurate aircraft model. Chapter 3 will progress to the approach for the various static stability analysis that were performed. The static results are used to show the need for further investigation into the dynamics of the aircraft. Dynamic simulations were used to determine the effects of the auto-adaptive wing on the aircraft and are presented in Chapter 4. The first step in the dynamic analysis was to determine that the aircraft was dynamically stable. Having established the aircraft's stability a series of open-loop simulations were used to determine both the linear and non-linear aircraft response to a step change in the adaptive flap. The linear models were then used in the development of the various-closed loop controllers. The complete closed-loop simulations for the aircraft were then performed. Lastly, Chapter 5 presents the conclusions of the current work as well as suggestions for future research in this area.

Chapter 2

Aircraft Characteristics

2.1 Aircraft Configuration

The Aerosonde UAV configuration was chosen as the base platform for this research. This aircraft was chosen due to the amount of aerodynamic and stability data that was available. In addition it fit one possible application for an auto-adaptive wing. To elaborate, it has been shown that an aircraft can have an increase in both range and endurance over a larger speed range.⁹ The Aerosonde's applications include high endurance and long range missions. This makes it an applicable platform since the auto-adaptive wing can increase aircraft performance during both of these missions.

The basic characteristics of the Aerosonde are listed in Table 2.1, and the UAV is assumed to have the wing-tail planform as shown in Fig. 2.1. The AeroSim¹⁰ blockset was chosen to assist in the dynamic modeling and will be discussed in more detail in Chapter 4. This blockset requires a full aerodynamic, stability, propulsion, and mass properties model. For this reason a more complete aircraft model was required. The majority of the data for the complete aircraft model was available as a demo that was included with the AeroSim blockset. The data is arranged in a format that is required to interface with the simulation blocks. It is important

Table 2.1: Basic characteristics for the Aerosonde UAV.

Parameter	Value
Gross Weight (W)	107.9 N (24.3 lbf)
Mean Aerodynamic Chord (mac)	0.19 m (0.62 ft)
Reference area (S_{ref})	0.55 m ² (5.92 sq.ft.)
Wing aspect ratio (AR)	15.24
Tail area (S_t)	0.11 m ² (1.17 sq.ft.)
Tail moment arm ($\bar{l}_t$)	0.91 m (2.99 ft)
Static margin (sm)	8 % mac

Figure 2.1: Planview showing the right-side geometry of the wing and tail for the Aerosonde UAV used in this paper.

to note that the coefficients are taken about a reference point. This allows for changes to the aircraft center of gravity (cg) throughout the flight. The reference point for this aircraft is the aerodynamic center of the wing. The format file has been changed slightly from the one that is provided with AeroSim, this was done to reflect the changes that were required for the modeling of the adaptive flap. These changes will be discussed in more detail in the subsequent sections.

2.2 Calculating the Adaptive Surface

Characteristics

The AeroSim blockset is a general-purpose blockset for dynamic simulations of an aircraft. The aircraft model in this blockset was modified for the current research to handle an adaptive wing. Relations from thin airfoil theory (TAT) were used to assist with modeling of the adaptive wing. The background for TAT is presented in Appendix A and relations used will be discussed in more detail in both Chapters 3 & 4.

One of the required parameters for modeling a trailing-edge flap in TAT is the hinge-line location. The adaptive surface, for the Aerosonde aircraft, was chosen to be a full-span flap at 80 % of the chord (x_f/c). This is supplied in the configuration file and then converted to an angular (θ_f) position using Eq. 2.1. This angular coordinates are used in TAT modeling.

$$\theta_f = \cos^{-1} \left(1 - 2 \frac{x_f}{c} \right) \quad (2.1)$$

Aside from the angular position of the hinge line it is also necessary to determine the new change in lift ($C_{L_{\delta_f}}$) and the change in pitching moment coefficients ($C_{M_{\delta_f}}$) due to adaptive surface deflections. Using the following equations the coefficients for lift and pitching moment for the adaptive flap can be determined: The following two equations represent the effects of the flap on the airfoil.

$$C_{l_{\delta_f}} = [2(\pi - \theta_f) + 2 \sin \theta_f] \quad (2.2)$$

$$C_{m_{\delta_f}} = [\frac{1}{4} \sin 2\theta_f - \frac{1}{2} \sin \theta_f] \quad (2.3)$$

The airfoil effects can then be modified for 3D effects using the following equations.

For this aircraft the assumption was made that the adaptive surface is a full span flap.

$$C_{L_{\delta_f}} = 0.9(K_f)(C_{l_{\delta_f}})\frac{S_{flapped}}{S_{ref}} \quad (2.4)$$

$$C_{M_{\delta_f}} = C_{m_{\delta_f}} - \eta(V_H)(C_{L_{\alpha_t}})(-\frac{\partial \epsilon}{\partial \delta_f}) \quad (2.5)$$

$$\frac{\partial \epsilon}{\partial \delta_f} \approx \frac{2C_{L_{\delta_f}}}{\pi AR} \quad (2.6)$$

Where $K_f = 0.9$ and $S_{flapped}/S_{ref} = 1$ for the full span flap.

Using these equations, the flap terms in the configuration script were modified to account for the adaptive wing.

2.3 Determining The Aircraft Drag Buildup

For this research it was also of interest to examine the drag benefits due to the adaptive wing. This is an area of interest, so that the benefits of the adaptive wing could be observed. The method used to calculate drag by the AeroSim blockset is shown by Eq. 2.7. Figure 2.2 is a plot of the drag polar for Eq. 2.7. This figure shows the shift of the drag polar with three flap deflections. Here we see that the drag polar shifts horizontally to the right for both the negative and positive flap deflection and they fall on top of each other. There is no vertical movement, which is expected from Section 1.1, of the drag polar with this drag model. This occurs because the drag model that is supplied with AeroSim is not created to capture the benefits of an adaptive wing. This does not accurately model how the aircraft drag would respond with an adaptive wing and therefore needed to be modified.

Figure 2.2: Demonstration of the drag polar shift due to a change in the flap angle for the AeroSim method (Eq. 2.7).

$$\begin{aligned}
 C_D = C_{D0} &+ \frac{2(C_L - C_{L0})}{\pi A Re} + C_{D_{\delta_f}} |\delta_f| \\
 &+ C_{D_{\delta_e}} |\delta_e| + C_{D_{\delta_a}} |\delta_a| \\
 &+ C_{D_{\delta_r}} |\delta_r| + C_{DM} M
 \end{aligned} \tag{2.7}$$

A new drag buildup was implemented that more accurately represents the shift of the drag polar that is expected for an adaptive wing. The new drag method was formulated by examining the changes to the drag polar of typical low Reynolds number airfoils and adding the contributions due to the induced and parasite drag of the aircraft. The drag buildup is expressed in Eq. 2.8 and the drag polars are seen in Fig. 2.3. As we see in this figure the drag polar does follow the trend

presented for the adaptive airfoil in Fig. 1.1(b). The drag polar now shifts both vertically and horizontally for a flap deflection. This demonstrates an increase in the LDR region for the adaptive wing. However, for this model the LDR is not as pronounced as the one shown for the airfoil in Chapter 1.1, which occurs for two reasons. First the Aerosonde aircraft is not specifically designed for an adaptive wing and secondly the drag coefficient of the fuselage and K_{2D} term were directly derived from the AeroSim model. Therefore this new method more accurately models the expected drag polar shifts for the adaptive aircraft but does not show substantial benefits of the adaptive flap.

$$\begin{aligned}
C_D = & C_{D0} + K_{2D}(C_L - C_{L_{mind}} - 2 \sin(\theta_f)\delta_f)^2 \\
& + \frac{(C_L)^2}{\pi A R e} + C_{D, fus} + C_{D_{\delta_e}}|\delta_e| \\
& + C_{D_{\delta_a}}|\delta_a| + C_{D_{\delta_r}}|\delta_r| + C_{DM}M
\end{aligned} \tag{2.8}$$

Figure 2.3: Demonstration of the drag drag polar shift due to a change in the flap angle for the new method (Eq. 2.8) .

Chapter 3

Static Longitudinal Stability Analysis

The first logical step in investigating the effects of an auto-adaptive wing on the stability of an aircraft is a static stability analysis. This chapter will present the methodology and the results of this analysis. The methods described in this chapter are a combination of traditional stability equations and applications of thin airfoil theory.

The static analysis and results are presented in the following three sections:

1. Section 3.1 presents the static stability characteristics of the modified Aerosonde aircraft. There will also be a brief discussion of the effects of the flap on the $C_{M_{\delta_f}}$ of the aircraft.
2. Section 3.2 presents the effect of an automated cruise flap on an airfoil. This is used to help explain the expected behavior the adaptive wing will have on the aircraft.
3. Section 3.3 presents the finding of the static stability analysis and what the implications for the current research.

3.1 Aircraft Analysis

The first step in the aircraft analysis was to determine if the aircraft is statically stable. This typically is done by looking at the C_{M_α} curve for the aircraft. The following method describes the process of evaluating the static stability. The aircraft $C_{M,cg}$ vs. α variation for this case was constructed by first calculating the C_L and the downwash angle (ϵ) at the tail for a given α . It is assumed that the primary contribution to the aircraft C_L comes from the wing C_L . Using the notation of Etkin and Reid,¹¹ the following equations are obtained.

$$C_L = C_{L,0} + \frac{\partial C_L}{\partial \alpha} \alpha \quad (3.1)$$

$$\epsilon \approx \frac{2C_L}{\pi AR} \quad (3.2)$$

$$C_{L,t} = \frac{\partial C_{L,t}}{\partial \alpha_t} (\alpha_w - i_w - \epsilon + i_t) + \frac{\partial C_{L,t}}{\partial \delta_e} \delta_e \quad (3.3)$$

$$C_{M,cg} = C_{M,ac,wb} + C_L(h - h_{n,w}) - \eta V_H C_{L,t} + C_{M,p} \quad (3.4)$$

The resulting aircraft $C_{M,cg}$ vs. α variation for two flap settings and for a fixed elevator angle is shown in Fig. 3.1. The negative slope for the C_M - α curves show that the aircraft is statically stable and very traditional results are achieved. These two values were chosen to compare with the results from the dynamic stability studies and will be discussed in Chapter 4.

Having established the static stability of the aircraft alone, it is of interest to examine the pitching moment effects the flap will have on the aircraft. Equation 2.5 which was discussed earlier shows the method that was used to determine the $C_{M_{\delta_f}}$ for the flap. Also for the following discussion it is also important to

Figure 3.1: $C_{M,cg}$ vs. α comparison for fixed elevator for two flap settings.

remember that this term is calculated about the reference point which for this aircraft is the aerodynamic center of the wing. The effects the two terms of the $C_{M_{\delta f}}$ equation have on the stability can give some insight to the possible response of an aircraft to the automated adaptive wing. To briefly investigate these two components the wing-only and wing-tail combined effect will be examined.

At first only the wing component of Eq. 2.5, which is the first term $C_{m_{\delta f}}$, is considered. It is well known that a positive flap deflection will increase the nose down pitching moment for an airfoil. That would lead to reason that the wing-only contribution would tend to pitch nose down after a positive flap deflection and vice versa for the negative deflection. For any flap deflection the wing-only contribution has a stabilizing effect on the aircraft. This does not mean however pitching moment of the wing-only contribution would be the required amount to re-trim the aircraft for steady flight with the new flap setting.

When both the wing and tail contributions are examined an interesting phe-

nomenon occurs with the second term of Eq 2.5. When a flap is positively deflected there is an increase in the C_L of the aircraft, which corresponds to a increase in the downwash at the tail of the aircraft. This effectively decreases the α the tail experiences and thus decreases the nose down pitching moment the tail creates. This counteracts the wing-only contribution and thus makes the stabilizing effects of the wing-tail configuration less effective. This is only a brief description of the effects the adaptive wing would have on the stability of an aircraft.

3.2 Airfoil Analysis

To examine the static stability of an aircraft with an automated cruise flap, it is useful to study the stability of the flap controller on an airfoil. The function of the flap controller is to set the flap angle so that the airfoil is operating within the low-drag range. When the automation is done using sensed pressures, as described in Sec. 1.1.2, the flap controller adjusts the flap angle with the aim of getting the $\Delta C'_p$ to the target value. At this condition, the airfoil is said to be operating at its ideal condition for that flap angle and the stagnation point is at the optimum location at the leading edge.

For the purpose of stability analysis and simulations, it is necessary to model the sensor inputs to the flap controller. A convenient approach to developing such a model is to use relations from thin airfoil theory for the ideal C_l and ideal α for an airfoil with a trailing-edge flap. These relations are shown in Eq. 3.5 and 3.6 for an airfoil with a flap angle of δ_f , with all angles in radians.

$$C_{l_{ideal}} = C_{l_{ideal}, \delta_f=0} + 2\delta_f \sin \theta_f \quad (3.5)$$

$$\alpha_{ideal} = -\frac{\pi - \theta_f}{\pi} \delta_f \quad (3.6)$$

Figure 3.2: Airfoil C_l vs. α curve for two flap angles to illustrate flap-controller behavior.

Thus, an appropriate model for the fast-reacting flap controller could be one that uses the measured C_l of the airfoil and determines the desired flap angle for that C_l :

$$\delta_f = \frac{C_l - C_{l_{ideal, \delta_f=0}}}{2 \sin \theta_f} \quad (3.7)$$

Figure 3.2 shows the C_l - α curves for a typical airfoil for two flap angles along with the points on these curves that correspond to various conditions of interest for this discussion. For the purpose of examining the static stability of the flap controller on this airfoil, it is assumed that the system is operating in a trimmed state at point A, which corresponds to zero flap and the ideal condition for this flap, i.e. $\alpha = 0$ deg. In order to examine the static stability of the system, a perturbation to the α to point B is considered and the response of the flap is

examined. When the system is perturbed to point B, the C_l increases. As a result the flap controller reacts by increasing the flap angle, as per Eq. 3.7. The result of the increase in flap angle is that the operating point now shifts to C, which is at a higher C_l . By carrying out this argument further, it is clear that the flap angle will quickly diverge until it reaches the maximum deflection. Thus, it is seen that the flap controller is unstable.

In order to make this system stable, a controller is needed that will adjust the α in order to maintain a target C_l . It is for this reason, that the wind-tunnel experiments of an airfoil with an automated cruise flap system, described in Ref. 4, used a controller in which both the flap angle and α were simultaneously adjusted at every time step to maintain desired values of $\Delta C'_p$ and C_l . In a similar manner, an automated cruise flap system implemented on an aircraft wing will need an accompanying system for control of α to maintain the desired C_L . This control of α is best achieved by adjusting the elevator to maintain the desired C_L . Thus an automated cruise flap system on an aircraft needs to be integrated with some form of trim controller in order for the entire system to be stable and achieve the desired response.

3.3 Summary of Static Stability Results

First the static stability and control characteristics were studied for an aircraft with a trailing-edge flap on the wing. When the flap is fixed at a specified angle, the stability characteristics are similar to those of a conventional aircraft. Also observed were the characteristics of both components of the $C_{M_{\delta_f}}$ equation. It was shown that the wing-only contribution would have the tendency to stabilize the effects of the aircraft. With the addition of the horizontal tail the wing contribution is lessened due to the increased downwash the tail experiences. Having this

understanding it was of interest to further look at the effects of the automated system on an airfoil. When the automated cruise flap is used on an airfoil, it is shown that the automated flap, by itself, exhibits a unstable, divergent behavior when there is a perturbation to the angle of attack. This divergent behavior can be avoided if an additional controller is incorporated to achieve a desired lift coefficient by adjusting the angle of attack. This result pointed out that when an automated flap is incorporated on the wing of an aircraft, there is a need for an accompanying elevator controller to maintain the desired airspeed. This led to the need for further investigation into the dynamics and controls of the conventional tail Aerosonde aircraft.

Chapter 4

Dynamic Longitudinal Stability Analysis

The dynamic stability and control analysis was performed using conventional control methods. The models and block diagrams used for this analysis were built using the AeroSim blockset¹⁰ in the Simulink environment. This blockset contains the complete nonlinear six degree-of-freedom (6-DOF) aircraft models and other supporting blocks that allow for accurate modeling of the aircraft dynamics. The focus of the effort was on the longitudinal dynamics, and only symmetric flight was considered. The UAV, shown earlier in Fig. 2.1, was used for this analysis and different approaches were studied for automating the adaptive wing.

The dynamic analysis and results are presented in the following five sections:

1. Section 4.1 presents the dynamic stability characteristics of the modified Aerosonde aircraft. There will also be a brief background into the linearization method used for the AeroSim non-linear model and how it compared to conventional linear control theory.
2. Section 4.2 presents the block diagrams for studying the open-loop response of the aircraft to a step input to the flap angle. This study was carried out using both the complete 6-DOF nonlinear model as well as a linear model. The results of the step response were also used to validate the models by

comparing the steady-state response to the predictions of the static-stability analysis, thus proving the validity of using the open loops for controller design.

3. Section 4.3 presents the various controllers that were developed for this investigation. It will also provide the two controller schemes for controlling the flap and elevator angles.
4. Section 4.4 presents the closed-loop simulations that were performed to investigate the effects of the automated cruise flap. This will provide the insight to the implementation of the automated cruise flap on an aircraft.
5. Section 4.5 presents the results and finding of the dynamic simulations. It will reinforce the benefits of the various implementation schemes and their effect on the stability of the aircraft.

4.1 Dynamic Stability and Linearization

To understand the effects the automated flap will have on the dynamics of the aircraft it is first important to check the dynamic stability of the aircraft. This is performed so the effects of the flap can be isolated from any undesirable characteristics of the baseline aircraft. A traditional approach was used in which the aircraft was trimmed for various steady flight conditions and then analyzed for the appropriate stability characteristics and handling qualities.

A simple Matlab script was written using standard approximations to determine the aircraft state-space model. This script reads the aircraft configuration file then computes the state-space model about a given trim condition. Appendix B presents a copy of this script. This system was then evaluated for dynamic stability, controllability, and observability. This script was helpful in determining that

the aircraft itself was dynamically stable and had level-one handling qualities for the flight conditions where the adaptive wing is of interest. It was also determined that the aircraft was controllable and observable.

The script that was developed from conventional methods was compared to the linearization script that came packaged with the AeroSim blockset. The script provided with AeroSim linearized the non-linear model about a specific set of flight conditions. This script also developed a state-space model that could be compared to the previous method. Although both models compared favorably the linearization script that was packaged with AeroSim matched closer to the response of the nonlinear models, which will be discussed in the following section.

4.2 Open Loop Simulations

Having determined that the aircraft is dynamically stable, controllable, and observable it was of interest to see how the aircraft would respond to an input to the adaptive flap. For this several open-loop models were developed to compare and validate the longitudinal model for the UAV. The results from the open-loop simulations were compared with those from the static stability analysis. Fig. 4.1 shows, as an example, the nonlinear model used for determining the response to a step input to the flap angle. Linear models for the longitudinal motion were derived from the 6-DOF model for designing the various controllers that will be described in the following section. The block diagram for studying the open-loop response to the flap using the linear model is shown in Fig. 4.2.

The results of the open-loop response of the aircraft forward velocity and the α for a step input to the flap angle are presented in Fig. 4.3. Both the nonlinear and the linear models are presented against the trim values of velocity and α from the static stability analysis, shown earlier in Fig. 3.1. It is seen that the results

Figure 4.1: Nonlinear open-loop model for studying the step response to the flap angle.

Figure 4.2: Linear open-loop model for studying the step response to the flap angle.

Figure 4.3: Comparison of Linear, Nonlinear, and Static results.(a) Flap Angle,(b) Velocity response, and (c) Alpha response.

of the two open-loop simulations agree well with each other and with the results of the trimmed static stability calculations. This good agreement validates the use of the longitudinal linear dynamics model for use in controller design for the UAV.

4.3 Controller Design

Figure 4.4 shows the block diagram for the simulation of the aircraft motion using the complete nonlinear 6-DOF equations of motion. The main aim of this

model was to evaluate different options for incorporating the automated cruise flap. Because only the longitudinal motion is of interest, a bank-angle controller was incorporated to maintain zero bank angle. An altitude-hold and airspeed controller were also developed and incorporated to maintain the desired flight path. These controllers are shown in Fig. 4.4. The controllers were designed using conventional methods and the linear aircraft models were used for controller validation and robustness verification.

Figure 4.4: Model for closed loop simulation.

4.3.1 Bank, Airspeed, and Altitude Controllers

Since the primary interest was to study the longitudinal effects of the adaptive wing it was necessary to isolate the longitudinal dynamics. This was achieved with a bank angle (ϕ) controller set to achieve and maintain zero bank angle.

A proportional and integral (PI) controller was designed and implemented for maintaining zero bank angle.

With a controller to maintain zero bank angle, the next step was to maintain a desired forward airspeed. A proportional, integral, and derivative (PID) controller was used to close the airspeed to elevator loop. For the various flight conditions the appropriate gains were determined and implemented to maintain the desired response.

The last control loop required to maintain the desired flight conditions was an altitude hold. This loop was closed using a closed-loop throttle controller. A proportional controller was used to maintain the desired altitude-hold response. These three feedback controllers are used to maintain constant flight speed, altitude, and symmetric flight when the flap is dynamically adjusted for minimizing drag.

4.3.2 Flap and Elevator Controllers

As shown earlier, when an automated cruise flap is incorporated on an aircraft, there is a need for an accompanying controller that will maintain the desired aircraft C_L . This C_L control is best achieved by adjusting the elevator angle. When changes are made to the flap angle, the elevator controller will then adjust the elevator to re-trim the aircraft at the desired C_L , thus maintaining the desired airspeed. The controllers for the flap and elevator are shown as a single block in Fig. 4.4. The elevator is controlled using the airspeed controller that was described in Section 4.3.1. In the current effort, two options were evaluated for the flap and elevator control: decoupled and coupled. These are described in the following subsections.

Figure 4.5: Flap and elevator controllers with no coupling.

Decoupled flap and elevator controllers

In this option, the controllers for the flap and elevator are independent, as shown in Fig. 4.5. The elevator controller is the PID controller described earlier that maintains the desired airspeed. The flap controller adjusts the flap angle in an effort to maintain the wing at its ideal condition, by shifting the drag polar of the wing. For the purpose of the simulation, the aircraft C_L was used as input to the flap controller to determine the ideal flap angle (δ_f) using Eq. 4.1. The relation in Eq. 4.1 for the aircraft flap controller is similar to that shown earlier in Eq. 3.7 for the flap controller on an airfoil.

$$\delta_f = \frac{C_L - C_{L_{ideal, \delta_f=0}}}{2 \sin \theta_f} \quad (4.1)$$

On an aircraft, the C_L can be deduced from the measured airspeed. In the current work, the C_L used by flap controller at any instant was determined from a windowed average of the measured C_L . The windowed average was taken over the past 30 seconds of the simulation. This was done in order to prevent the flap from reacting to high-frequency C_L oscillations. In addition, the model was designed so that flap-actuation frequency was a user-specified variable.

The exploded view of the flap controller is depicted in Fig. 4.6. In this block diagram the aircraft C_L is buffered and averaged over the thirty second buffer. It

Figure 4.6: Exploded block diagram of the flap controller.

is then fed into a subsystem that can be enabled at sometime after the simulation is started to simulate the system being turned on during flight. After the system is switched on, it can be actuated at various frequencies using the trigger. When triggered, Eq. 4.1 calculates the optimum flap angle which is then rate limited and bounded by the limits of the maximum flap deflections. The final commanded flap deflection is then sent to the input of the aircraft model.

Coupled flap and elevator controllers

In the second option, the flap and elevator controllers were coupled. In order to automatically compensate for the trim changes due to flap. The flap-to-elevator coupling is illustrated in Fig. 4.7. For every change made to the flap angle, a change was automatically made to the elevator angle using a predetermined gain. This gain is aircraft specific. The sensitivity of this gain to changes in various parameters such as airspeed and cg shifts is unknown. However, for the simulations that were performed during this study it remained constant. This coupling was included in order to automatically compensate for the trim changes due to a dynamically-changing flap.

Figure 4.7: Illustration of the flap-to-elevator coupling.

4.4 Closed Loop Simulations

The results of the dynamic simulations are presented for two cases. In the first case, the effect of the flap controller on the response of the aircraft is presented. The results are shown for both with and without the flap-elevator coupling. In the second case, the results are presented for the response of the aircraft to a step change in the desired airspeed. These results are presented for four different flap-actuation frequencies. The benefits of the auto-adaptive wing were also investigated.

4.4.1 Flap-Elevator Coupling Comparison Study

Figure 4.8 shows the response of the aircraft from a simulation in which the flap controller, that is initially off, is turned on at $t = 120$ s. The responses are compared for both with and without the flap-to-elevator coupling. In these simulations, the desired airspeed is set at 30 m/s for the entire duration. The flap angle is initially set to 0 deg and the flap controller is off. At $t = 30$ s, the aircraft reaches steady-state conditions, as shown in the plots in Fig. 4.8 (a).

At $t = 120$ s, the flap controller is switched on. Because the optimum flap angle for this airspeed is approximately 4 deg, it is seen that the flap controller immediately sets the flap to that angle. When there is no flap-elevator coupling,

Figure 4.8: Comparison of aircraft response with and without flap-elevator coupling: (a) velocity, (b) flap angle, and (c) elevator angle

the elevator responds only after the airspeed decreases because of the change in flap angle. With the coupling, on the other hand, the change to the flap also results in an immediate change to the elevator angle. The responses of the elevator angle for the decoupled and coupled scenarios are compared in Fig. 4.8(c). The associated airspeed response is shown in Fig. 4.8(a). It is clearly seen that both the decoupled and coupled control schemes are successful in achieving the steady-state conditions for the airspeed soon after the flap controller is switched on. However, the coupled control scheme avoids the spike in the velocity that is seen for the decoupled case.

Figure 4.9: Response of aircraft to step change in the desired velocity for several flap-actuation frequencies: (a) velocity (b) flap angle (c) altitude (d) throttle. Both (e) and (f) are the zoomed in views of the velocity and throttle after the change in desired velocity.

4.4.2 Change in Desired Airspeed

In the second case study, the effect of a step input to the desired airspeed is studied for different flap-actuation frequencies. In this simulation, the airspeed is initially set to 30 m/s with the flap set to zero deg and the flap controller turned off. At $t = 120$ s, the flap controller is switched on and the aircraft is allowed to reach steady-state conditions. Subsequently, at $t = 200$ s, a step input is given to the desired airspeed to change it to 33 m/s. The response plots for the velocity and flap angle are presented in Figs. 4.9(a),(b) for four different flap-actuation frequencies.

It is seen that for all the four actuation rates, the aircraft responds to the change in desired velocity by adjusting the flap angle appropriately and settles quickly to the new steady-state condition. When the flap is actuated at high frequency (every 1 s), the response is smoother than when the flap is actuated at low frequency (every 120 s). In both cases, however, there is no undesirable behavior. These response plots confirm that as long as there is a controller that adjusts the elevator angle to maintain the desired airspeed, the automated cruise flap system is successful in continuously setting the optimum flap angle on the wing without any undesirable flight characteristics.

4.4.3 Benefits of Adaptive Wing

In the current research, it was also of interest to determine the benefits of the automated flap during the simulation. With the use of the new drag model described in Section 2.3, the benefits of the adaptive wing were examined. Figure 4.10 is a demonstration of the benefits for one of the simulations that was performed. The automated flap system is turned on at $t = 120$ s and the flight velocity and altitude were maintained with their appropriate closed-loop controllers. The C_D ,

Figure 4.10: Demonstration of drag benefits of the adaptive aircraft during simulation. The figures are: (a) flap deflection, (b) Coefficient of Drag (C_D), (c) throttle setting, (d) fuel consumption, (e) velocity, and (f) altitude

in Fig. 4.10 (b), for the aircraft was plotted versus time to demonstrate that the drag is reduced after the automated flap optimizes the wing. Figure 4.10 (c) shows the throttle reduction after the flap system is switched on. Much like the throttle, there is also a reduction in the fuel consumption indicating that the aircraft with the adaptive wing is indeed operating more efficiently at the same desired flight condition. From these plots it is clear that the auto-adaptive aircraft has drag benefits. The drag savings can be increased if the aircraft and airfoil are tailored for wing adaptation. Such a decrease in drag translates to a large benefit on the range, endurance, and V_{max} of an aircraft. Future work in this area could provide a better understanding of the aircraft drag with a auto-adaptive aircraft.

4.5 Summary of Dynamic Stability and Control Results

In the interest of better understanding the effects of an adaptive wing on an aircraft, a dynamic stability and control analysis was performed. As described in this chapter, a number of non-linear closed loop models were studied. Two controller schemes were developed for comparing the effects of re-trimming the aircraft after a change in the adaptive flap. It was observed that if the elevator was automatically deflected for every flap defection by some predetermined amount a more desirable response was achieved. When the flap was not coupled to the elevator, the aircraft took longer to re-trim after changes in the flap, although the response was still quite acceptable.

To determine if the automatic flap controller responded appropriately to a change in the desired flight conditions a step change to velocity was simulated. For a step change in the desired velocity the automatic flap readjusted the flap angle to the optimal setting and the aircraft was re-trimmed for the desired flight

condition. The system therefore responded appropriately to changes to the desired flight conditions.

Lastly the benefits of the adaptive surface were shown. For this the modified drag model was investigated. Here the benefits of the adaptive wing were observed by looking at the drag, throttle, and fuel consumption responses after the auto-adaptive wing was turned on. There was a drop in all three of these values after the automated flap is turned on, thus showing that the auto-adaptive wing does optimize the wing for the desired flight condition.

Chapter 5

Concluding Remarks

5.1 Summary of Results

With increasing interest in the use of adaptive lifting surfaces for improved aircraft performance, it is of interest to study the aircraft stability and control characteristics with adaptive wings. This work builds on a recent development of an automated cruise flap for adapting an airfoil to achieve low drag over a large lift range. Such an automated cruise flap system was shown to have unusual lift and pitching moment curves, which prompted the need for studying the effect on aircraft stability and control. Therefore an aircraft was selected and analyzed to determine the effects the adaptive wing on the aircraft stability and control.

In the first part of this work, the static stability and control characteristics were studied for an aircraft with a trailing-edge flap on the wing. When the flap is fixed at a specified angle, the stability characteristics are similar to those of a conventional aircraft. When the automated cruise flap is used on an airfoil, it is shown that the automated flap, by itself, exhibits a unstable, divergent behavior when there is a perturbation to the angle of attack. This divergent behavior can be avoided if an additional controller is incorporated to achieve a desired lift coefficient by adjusting the angle of attack. This result pointed out that when an automated flap is incorporated on the wing of an aircraft, there is a need for an

accompanying elevator controller to maintain the desired airspeed.

In the second part of the research effort, dynamic analysis and simulations were conducted using Simulink and the AeroSim blockset. The dynamic models were validated by verifying that the steady-state results from open-loop simulations agreed with the results of the trim conditions from the static-stability analysis. Two schemes were developed for the flap and elevator controllers. In the first scheme, there is no coupling between the flap controller and that for the elevator. In the second scheme, flap-elevator coupling was achieved by making a change to the elevator angle using a predetermined ratio whenever the flap controller makes a change to the flap angle.

The results of the simulations show that the automated flap is successful in adjusting the flap angle when accompanied by an elevator controller that maintains the desired airspeed. However, a small velocity spike was observed when the flap controller was switched on. When flap-elevator coupling was used, this velocity spike was eliminated. The flap and elevator controllers were also successful in tracking changes to the desired airspeed and no undesirable behavior was noticed for a wide range of flap actuation frequencies. The benefits of this type of system were also captured while performing the dynamic analysis. These results provide confidence in the implementation of the automated-flap concept on an aircraft wing.

This study, therefore, serves as a good stepping stone for taking the technology to the next level by flight testing the concept on an uninhabited aerial vehicle. It also opens the door for other possible applications of the automated cruise flap technology by providing the benefits without the increased pilot workload. This study provides only a step in the right direction to the successful implementation of a system that can increase aircraft performance over a wider range of operating conditions.

5.2 Recommendations for Future Work

While studying the effects of the adaptive aircraft a few areas of interest for future investigation have surfaced. The following three topics would be areas to investigate in more depth:

1. First an investigation into the drag buildup and modeling, which is one area where improvements can be made to more accurately model an aircraft with an adaptive wing. As shown in Section 2.3, the drag polar did not shift as expected for an aircraft with an adaptive wing and required a new model to be developed for this investigation. There are also questions about the differences in fuselage profile drag for an aircraft without and with an adaptive wing.
2. Second it would be useful to study an aircraft that is specifically designed for this technology. The Aerosonde was not specifically designed for this type of technology and there were some assumptions made to determine aircraft performance. An aircraft that is designed for this type of technology would be more successful in showing the benefits of the adaptive wing. To get a better idea of the benefits of the adaptive wing, a complete aircraft model would have to be created for an aircraft designed for the adaptive wing.
3. The third topic would be a comparison of the conventional aircraft with other configurations. It would be of interest to study the effects of an adaptive wing on unconventional configurations such as a flying wing.

Chapter 6

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Appendix A

Thin Airfoil Theory Analysis

This appendix briefly presents the thin airfoil theory calculations used in the modeling of the auto-adaptive wing.

It is well known¹² that the total circulation, Γ , of an airfoil can be calculated from thin airfoil theory using Eq. A.1.

$$\Gamma = \int_0^c \gamma(x) dx \quad (\text{A.1})$$

Using the coordinate transformation shown in Eq. A.2 the total circulation becomes Eq. A.3.

$$x = \frac{c}{2}(1 - \cos \theta) \quad (\text{A.2})$$

$$\Gamma = \frac{c}{2} \int_0^\pi \gamma(\theta) \sin \theta d\theta \quad (\text{A.3})$$

Where $\gamma(\theta)$ can be presented as the Fourier series shown in Eq. A.4.

$$\gamma(\theta) = 2V_\infty [A_0 \frac{1 + \cos \theta}{\sin \theta} + \sum_{n=1}^{\infty} A_n \sin n\theta] \quad (\text{A.4})$$

As it turns out using thin airfoil theory for an airfoil with small trailing-edge flap

deflections, the Fourier coefficients can be calculated from Eqs. A.5 and A.6.

$$A_0 = \alpha + \frac{1}{\pi} \int_{\theta_f}^{\pi} \delta_f d\theta = \alpha + \frac{\delta_f}{\pi} (\pi - \theta_f) \quad (\text{A.5})$$

$$A_n = -\frac{2}{\pi} \int_{\theta_f}^{\pi} \delta_f \cos n\theta d\theta = \frac{2\delta_f}{\pi} \frac{\sin n\theta_f}{n} \quad (\text{A.6})$$

Where δ_f and θ_f are the flap angle and flap hinge location, respectively, in radians. It also turns out from thin airfoil theory that the approximate C_l of the airfoil can be calculated from only the first two coefficients using Eq. A.7.

$$C_l = 2\pi A_0 + \pi A_1 \quad (\text{A.7})$$

Finally, to compute the $C_{l_{ideal}}$ corresponding to the angle-of-attack at which the stagnation point is located exactly at the leading edge of the thin airfoil, one must find the C_l where there are no singularities (or infinite suction peaks) on either the upper or lower surface. This $C_{l_{ideal}}$ corresponds to the C_l where the Fourier coefficient, A_0 , goes to zero. Thus the $C_{l_{ideal}}$ can be calculated using Eq. A.8.

$$C_{l_{ideal}} = \pi A_1 = 2\delta_f \sin \theta_f \quad (\text{A.8})$$

Appendix B

Linear Longitudinal Dynamic Stability Analysis Script

This appendix briefly presents the construction of the states based model and the stability analysis that was created from traditional stability calculations.

```
% Longitudinal Dynamic Analysis
% 2-10-04
% Victor Vosburg

% Inputs

load aerosondecfg.mat

U0= 30; % Flight Velocity (consistent units with config file m/s)
m=11;
gamma=1.4;
R=287; %J/(kg K)
T=286.85; %Standard Day
asound=339.53;
M=U0/asound;
rho=1.2017; %density sea level
Q= 1/2*rho*U0^2;
alpha=0;
AR=b^2/S;
g=9.81; %gravity m/s^2
IC=[0 0 0 0]

% Calculate the REFERENCE CL and CD
```

```

alpha=1.31*pi/180;
deltae=0.0404;
deltaf=0;

CLREF=CL0+CLa*alpha+CLdf*deltaf+CLde*deltae+CLM*M;
CDREF=CD0+(CLREF-CL0)^2/(pi*osw*AR)+CDdf*deltaf ...
      +CDde*deltae+CLM*M;

% Calculate A MATRIX

% X Forces
Xu=-(M*CDM+2*CDREF)*Q*S/(m*U0);
Xw=-((2*CLREF*CLa/(pi*osw*AR))-CLREF)*Q*S/(m*U0);

% Z Forces
Zu=-(M^2/(1-M^2)*CLREF+2*CLREF)*Q*S/(m*U0);
Zw=-(CLa+CDREF)*Q*S/(m*U0);
Zwdot=-CLalphadot*MAC*Q*S/(2*U0*U0*m);
Za=U0*Zw;
Zadot=U0*Zwdot;
Zq=-CLq*MAC*Q*S/(2*U0*m);

% Moments
Mu=CmM*M*Q*S*MAC/(U0*Iy);
Mw=Cma*Q*S*MAC/(U0*Iy);
Mwdot=Cmalphadot*MAC*Q*S*MAC/(2*U0*U0*Iy);
Ma=U0*Mw;
Madot=U0*Mwdot;
Mq=Cmq*MAC*Q*S*MAC/(2*U0*Iy);

A=[Xu Xw 0 -g; ...
   Zu Zw U0 0; ...
   Mu+Mwdot*Zu Mw+Mwdot*Zw Mq+Mwdot*U0 0; ...
   0 0 1 0];

% Calculated B matrix

% X forces
Xde=-CDde*Q*S/m;
Xdf=-CDdf*Q*S/m;
Xdt=0;

% Z forces
Zde=(-CLde)*Q*S/m;

```

```

Zdf=(-CLdf)*Q*S/m;
Zdt=0;

% Moments
Mde=Cmde*Q*S*MAC/Iy;
Mdf=Cmdf*Q*S*MAC/Iy;
Mdt=0;

B=[Xde Xdf Xdt; ...
   Zde Zdf Zdt; ...
   Mde Mdf Mdt; ...
   0 0 0];

%C and D matrices

C=[1 0 0 0;0 1 0 0;0 0 1 0;0 0 0 1];
D=[0 0 0 0;0 0 0 0;0 0 0 0];

long=ss(A,B,C,D)
damp(long)

co=ctrb(long);
ob=obsv(long);

Controllability= rank(co)
Observability= rank(ob)

```