

CRANFIELD UNIVERSITY

DA QING YANG

Stability and Control Effectiveness of a Seamless Aeroelastic Wing Aircraft

SCHOOL OF ENGINEERING
Department of Aerospace Engineering

PhD THESIS
ACADEMIC YEAR 2010-11

Supervisor: Dr Shijun Guo
Dr Craig Lawson

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This thesis is submitted in partial fulfilment of the requirements for
the degree of Doctor of Philosophy

ABSTRACT

Today's aerospace engineers can design a much lighter aircraft than before thanks to the improved accuracy in the estimation of aerodynamic load and structural strength. But reduction of airframe material has also made the aircraft more flexible, giving rise to aeroelastic phenomena which would have profound effect on flight stability and control.

From stability point of view, redistributed aerodynamic force due to wing deformation would influence the static stability of the aircraft. The structural flexibility and dynamic behaviour reduces the flight dynamic damping and influences the flight dynamic modes, which in turn could cause dynamic stability problems.

From a control effectiveness perspective, control surface deflection exerts local aerodynamic force and deformation, which generally reduces intended control authority. The magnitude of control effectiveness reduction mainly depends on the structure and control surfaces design. An aircraft of special planform such as swept back wing makes the problem more complicated.

This research project was motivated by the Active Aeroelastic Wing Technology (AAWT) demonstrated by the F/A-18AAW (later known as NASA X-53). The proposed research aircraft was a small flying wing type aircraft with 40 degree swept back angle and centrally mounted high swept fin. Powered by a single micro jet engine, this technology demonstrator features a Seamless Aeroelastic Wing (SAW). This study follows the AAWT approach, but extends the AAWT discrete rigid body control surfaces to spanwise continuous and chordwise hingeless control surfaces, integrated with internal actuation system.

This research is aimed to achieve the stability and controllability of an aircraft with a specifically designed flexible seamless aeroelastic wing, which had control effectiveness problem. Firstly an investigation was made to mitigate the stability problems of the SAW aircraft. Then secondly, based on the SAW feature, the research proposes a systematic solution to the control effectiveness problem of this SAW aircraft. The solution would combine multidisciplinary efforts including aeroelastic tailoring, novel control surface actuation system design, distributed flight control management and active flight control algorithm.

In the study, the effect of structural flexibility and dynamic behaviour of the wing on the flight dynamics has been considered. This thesis covers design, modelling, simulation and control of the SAW aircraft. The design, modelling, simulation, prototype construction and testing of the SAW were undertaken with collaboration of another PhD student and part of the work is also included in this thesis. The study has shown that the stability of the SAW aircraft can satisfy flying quality requirements with the compensation of stability augmentation system. The control effectiveness can be increased by SAW technology comparing with a conventional hinged control surface design. Flight envelope can be extended beyond control reversal speed with the aid of successfully designed gain scheduling control algorithm.

The SAW aircraft shares many common aeroelastic problems with other flexible aircraft, but also has aeroelastic problems specific to its own design. If the design is changed, detailed solution would be altered accordingly, but the problem solving approach would be applicable. This research views these problems from a system level and endeavour to lead the way forward in achieving and demonstrating a systematic approach to a series of aeroelastic problems.

Keywords:

Flight dynamics, structural dynamics, morphing wing, control surface actuation mechanism, aeroelastic control reversal, flying wing flight control, flying quality and stability augmentation.

ACKNOWLEDGEMENTS

I would like to express my sincerest gratitude to my supervisor, Dr Shijun Guo for his continual advice, guidance and support throughout the past four years. My thanks also go to Dr Xiang Zhang for her meticulous support, academically and spiritually in my life. Thanks to Dr Mike Cook, Dr Alastair Cook, Dr James Whidbourne and Dr Craig Lawson for their support on specific academic areas and guide me through all the bewilderment and doubts.

I would like to offer also my deepest gratitude for my family, without their love and support, it would be impossible for me to finish my study away abroad for all these years. To my mother Yan Sun, my father Jin Yang and my girlfriend Kathy Tian and all family members, I thank you all.

I also wish to express my thanks to:

My colleagues Daochun Li, Andrew Stuart, Naveed Rahman, David Richardson, Analisa, Guido, Melanie, Ryoko, Natalia, Debra, Pierre, Mudassir, Guido Montazino and Shiraz for their help and advice.

My friends DaoHua Hu, Huange Wang, Zhen jie Wang, Xi xia Liu, Shi Tu Li, Ze Xu Sun, Tian Xiang Xiong, Ting Ding, Wei Guo, Hong Mei Gao, Xiao Wei Sun and Sun Xin Shi for their assistance and encouragement.

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NOMENCLATURE

English Alphabet

A	States matrix
AR	Aspect Ratio
B	Control matrix
CD_0	Zero-lift drag coefficient
$C_{L\alpha}$	Lift curve slope
C_D	Drag coefficient
C_L	Body axes rolling moment coefficient
C_M	Body axes pitching moment coefficient
C_N	Body axes yawing moment coefficient
C_X	Body axes X force coefficient
C_Y	Body axes Y force coefficient
C_Z	Body axes Z force coefficient
$C(k)$	Theodorsen function
K_n	Static margin
R_{air}	Air specific gas constant
T_0	Thrust at sea level
T_{ISA}	Temperature corresponding to ISA model
V_{EAS}	Equivalent Airspeed
V_{TAS}	True Airspeed

X_{AC}	X-axis position of the aerodynamic centre
X_{CG}	X-axis position of the centre of gravity
Y_{CG}	Y-axis position of the centre of gravity
Z_{CG}	Z-axis position of the centre of gravity
h	Altitude
I_{xx}	Moment of inertia about x-axis
I_{xy}	Product of inertia about x-axis and y-axis
I_{xz}	Product of inertia about x-axis and z-axis
I_{yy}	Moment of inertia about y-axis
I_{yz}	Product of inertia about y-axis and z-axis
I_{zz}	Moment of inertia about z-axis
J_v	Bessel function of first kind
$[K]$	Stiffness matrix
K	Gain matrix
K_{lat}	Lateral SAS gain
K_{long}	Longitudinal SAS gain
L	Moment about x-axis or rolling moment
$[M]$	Mass matrix
M	Moment about y-axis or pitching moment
Ma	Mach number
N	Moment about z-axis or yawing moment

P	Rotation about body x-axis (roll rate)
Q	Rotation about body y-axis (pitch rate)
R	Rotation about body z-axis (yaw rate)
Re	Reynolds number
Sref	Reference wing area
T	Temperature
U	Body x-axis velocity
V	Body y-axis velocity
W	Body z-axis velocity
X	Force along body x-axis
Y	Force along body y-axis
Yv	Bessel function of second kind
Z	Force along body z-axis
a	Pitch axis location relative to the mid-chord
a_{air}	Speed of sound
b	Semi-chord
bref	Reference wingspan
cref	Reference chord
g	Acceleration due to gravity
h	Vertical displacement
qbar	Dynamic pressure

Creek Alphabet

Λ	Leading edge sweep angle
Γ	Dihedral
Δ	Increment
α	Angle of attack or incidence angle
β	Sideslip angle
γ	Flight path angle
δ	Control angle
ζ	Rudder angle
η	Elevators angle
θ	Attitude angle
λ	Taper ratio
μ	Dynamic viscosity
ξ	Elevon angle
ρ	Air density
τ	Throttle level
ψ	Yaw or heading angle
ϕ	Bank angle

Abbreviations

AC	Aerodynamic Centre
AOA	Angle Of Attack or incidence

BAC	Body Axes Centre
BWB	Blended Wing Body
CFD	Computational Fluid Dynamics
CG	Centre of gravity
CTTAM	Curved Torque Tube Actuation Mechanism
DARPA	Defence Advanced Research Projects Agency
DOF	Degrees of Freedom
EAP	Electro-active Polymers
EHA	Electro-Hydrostatic Actuators
EMA	Electro-Mechanical Actuators
EOM	Equations of Motion
FCS	Flight Control System
ISA	International Standard Atmosphere
LE	Leading Edge
MAC	Mean Aerodynamic Chord
MTOW	Maximum Take-Off Weight
PZT	Piezoelectric translator
SAF	Seamless Aeroelastic Fin
SAS	Stability Augmentation System
SAW	Seamless Aeroelastic Wing
TE	Trailing Edge

Chapter 1 Introduction

Modern airplane structures are flexible, and aeroelastic phenomena arise when structural deformations induce changes in aerodynamic forces. The additional aerodynamic forces cause yet more structural deformation, which can lead to greater aerodynamic forces. Such closed-loop aeroelastic interaction may cause a detrimental effect on flying qualities and control effectiveness even though the flight speed is well below any critical limit. Furthermore it may result in catastrophic consequences such as divergence or flutter when the state of equilibrium is breached.

In order to improve the aeroelastic stability of a wing, the conventional approach is simply to add more structural material to increase its stiffness. Instead of paying the weight penalty to make the structure stiffer, one of the innovative design concepts is to allow the wing to be flexible hence light such that it can be made to deform in a desirable shape in either passive or active manner.

The passive methods exploit non-homogeneous properties of the composite material, this allows its fibre orientation to be tailored to induce coupled bending–twist deformation, which reduce the negative aeroelastic effect and maintain the desired wing shape.

Active methods control the morphing of the airfoil shape to:

- a) Redistribute pressure on the wing to mitigate the negative aeroelastic effect, hence maintain flight,
- b) Encourage desired wing twist to promote specific lift distribution actively, hence achieve flight control.

The active method of turning aeroelastic effects into a beneficial flight control effort is known as active aeroelastic wing (AAW) technology.

The study of AAW is a multidisciplinary subject in that it integrates flight dynamics, unsteady aerodynamics, structural dynamics and control theory. In principle, the AAW employs the energy of the air stream for a desirable wing twist to enhance the flight performance. Multiple control surfaces of smaller size and less actuation motion can be achieved compared to a conventional design. Consequently the wing produces the

required control forces more effectively. The feasibility of this technology has been demonstrated through the use of multiple leading and trailing edge control surfaces activated by a digital flight control system in the NASA F/A-18AAW fighter experiment [1][2][3]. At higher dynamic pressures, the control surfaces were used as aerodynamic tabs that promoted wing twist for added control force and enhanced manoeuvres. Large amounts of control power can be generated at these pressures. Overall the AAW wing is expected to experience less twist than a conventional wing, which twists in opposition to the control force generation. Potential benefits of AAW technology to future systems include substantially increased control power, reduced aerodynamic drag, reduced manoeuvre loads and reduced structural and takeoff gross weight, and an overall increased design latitude in terms of wing span, sweep, and thickness due to relaxed stiffness requirement. AAW design techniques have been shown to reduce aircraft takeoff gross weight by 5-20% [1]. Specifically from a control system perspective, AAW technology also has these four interlinked major benefits:

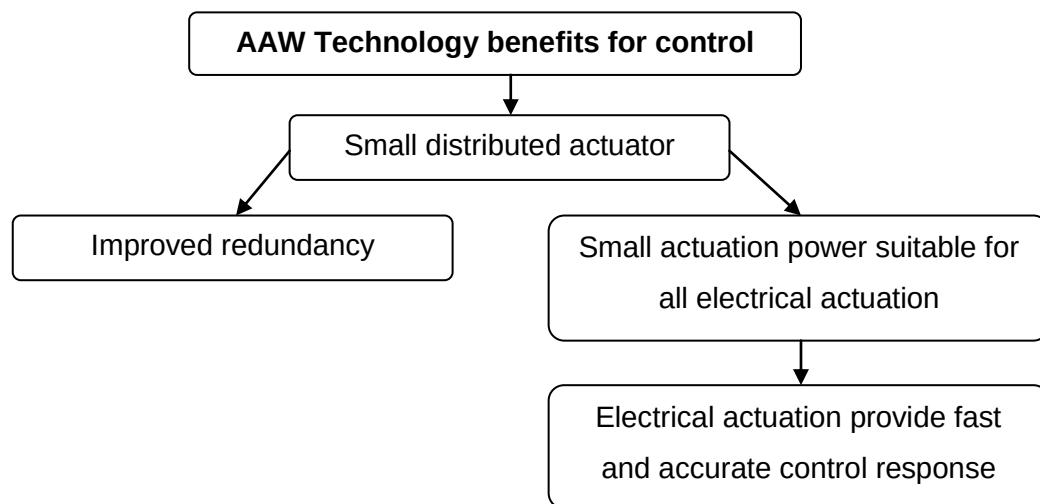


Figure 1-1 AAW technology benefits for control

This technology is especially synergistic with thin, high aspect ratio wings and gives designers more freedom to exploit higher aerodynamic efficiency especially when operating in a broad range of subsonic, transonic and supersonic conditions. It can be used, not only to improve the capabilities of existing wing forms, but also to reduce the conflict in requirements for a new wing design [4].

Following the development of AAW, some research efforts were made to extend the AAW concept to the design of a flexible and variable camber wing. For small scale wing and fin structures, smart materials were considered in developing an efficient actuating control system. Recently a couple of research projects have been undertaken at Cranfield University to develop a seamless aeroelastic wing (SAW) and seamless aeroelastic fin (SAF) structures for achieving flapless flight control [5]. A SAW or SAF is a one piece, continuous lifting or control surface, which subjected to an initial, controlled chord-wise geometry changes and subsequently provokes full span wing warping for flying control. The primitive form of this particular concept can be traced back to the beginning of aviation. Wing warping was used by the Wright Brothers in their historic first manned powered flight in 1903 as roll control method [6]. The aviation industry has come a long way since then. We have flown faster, higher and further. With the development of new materials, technology and computation schemes, today we are at a stage to revisit the wing warping concept and upgrade the old technique to a new technology. It is a new challenge to develop a suitable active control system to achieve this goal.

1.1 Aim and objective

The aim of this research is to investigate the stability and controllability of a flying wing aircraft with flexible seamless aeroelastic wing. Flight control is achieved by actively controlled wing deformation that exploits possible aeroelastic beneficial effect.

The key objectives for this research are:

- Design and numerical modelling of a rigid flying wing aircraft in the MATLAB/Simulink environment.
- Airfoil morphing shape and mechanism design.
- Numerical model development for seamless aeroelastic wing (SAW).
- SAW aircraft model construction and simulation in the MATLAB/Simulink environment.
- Performance, stability and control evaluation of aeroelastic wing aircraft in comparison with rigid aircraft.

1.2 Contribution to knowledge

This thesis has the following contributions to knowledge:

- Construction of a generic numerical model of a flexible flying wing aircraft, including both unsteady aerodynamics and structural dynamics, embedded with flight control system and stability augmentation system. The model is capable of pilot-in-the-loop simulation, and suitable for modern control law development.
- A study into the impact of aeroelastic effects on swept back wing aircraft performance, stability and control characteristics.
- An attempt to use flexible wing aeroelastic deformation to actively produce flight control.

1.3 Structure of thesis

In summary, this thesis describes the development of a seamless aeroelastic wing aircraft model and its application in the MATLAB/Simulink environment. The impact of aeroelastic effects on the flight dynamic stability and control characteristics of the aircraft are assessed and reviewed.

Chapter 2 presents a review of the historic notes of warping wing, AAWT development history, leading edge and trailing edge combination for flight control, exploration of aeroelasticity in aircraft design, seamless control surface design, morphing concept and development, available and potential actuators, and Cranfield university Demon project.

Chapter 3 presents analytical methods used in this study. This includes definition of the equations of motion for a flexible airplane, mathematical foundation of structural dynamics and unsteady aerodynamics.

Chapter 4 derives the mathematical model of the seamless aeroelastic wing (SAW). The lift and pitching moment characteristics of the wing are analysed in both quasi-static and dynamic terms in the MATLAB/Simulink environment.

Chapter 5 presents design and mathematical modelling of the aerodynamic, environmental and propulsive forces acting on the baseline rigid airplane D-47. This includes the definition of a state-space unsteady aerodynamic model suitable for real-time simulation.

Chapter 6 combines the SAW model with the D-47 rigid aircraft's fuselage and creates the flexible airplane D-47SAW, the impacts of flexibility on the aircraft performance, stability and control was investigated.

Chapter 7 consists of concluding remarks and recommendations for further work.

Appendices A to D provide the structural, geometric and aerodynamic data, which describe the implementation of D-47SAW in MATLAB Simulink as well as flying quality requirements.

Chapter 2 Literature Review

2.1 Historic notes of wing warping and twisting

The idea of achieving a useful amount of control power by deforming the lifting surfaces is far from new. [6]

Before the dawn of human beings, birds have been controlling their flight gracefully by warping their feather-covered wings for centuries. Human dreamed about being able to fly like birds ever since they first saw one. By observing and imitating bird, human began to design and construct machines that one day may take them into the sky.

For his historic first powered flight on December 17, 1903, Orville Wright became the first human who took off into the air. He used the movements of his hips in the airplane's "saddle", in which he laid prone, to twist or warp either the left or right wingtip. This provided roll flight control without the use of ailerons or flaps. (Figure 2-1) [6]

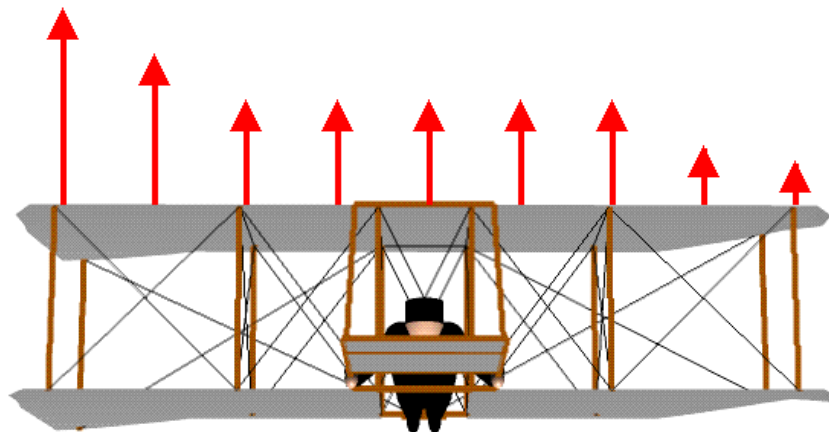


Figure 2-1 Wright flyer one uses wing warping to generate roll moment

Thus, from both a conceptual and an applied point of view, wing warping was developed earlier than the "conventional" control methodology based on hinged trailing edge moving surfaces.

Both hinged trailing edge control and wing warping control change effective airfoil camber to develop stronger local air circulation to gain additional lift. A key difference between them is wing warping control needs to overcome wing structural strength. As aircraft's weight and flight speed increases, soft airframe materials such as wood and fabrics were replaced by stronger metals like aluminium and steel. Wing warping control becomes impossible for human power. In fact even hinged control surface needs power boost for human to manage when aerodynamic load becomes really high.

To make wing warping control possible (with the aid of actuator), wing structure has to have relatively low chord-wise stiffness while maintain its sufficient strength. Not to mention the difficulty of designing wing of this kind itself, the bigger problem is such wing structural design would also have low torsional stiffness, leading to significant aeroelastic twisting of the wing at high speed and loss of control authority which is unacceptable. Early design philosophy has used wing twist to distribute load and improve aircraft stall character, but treated aeroelastic twist (additional wing twist after loading) as unwanted structural uncertainties. Engineers have done everything possible to counteract it, from physically stiffening the wings to incorporating other control surfaces that produce less twist. In the past, the response to aeroelastic twisting has been to reduce it.

2.2 F/A-18 AAW technology demonstrator (X-53)

The F/A-18 active aeroelastic wing (AAW) aircraft program was a joint research effort involving NASA, the Air Force Research Lab (AFRL), the United States Navy, and Boeing Phantom Works. The program looked to develop the multidisciplinary technologies needed to integrate aircraft aerodynamics, active controls, and structural aeroelastic behaviour to maximize vehicle performance. Control laws were designed for a light, flexible-winged aircraft that would improve roll performance by exploiting the elastic characteristics of the wing. [1] [3] [4]

An F/A-18A, operated by NASA Dryden tail number 853, shown in Figure 2-2, was selected as the test aircraft for several reasons.



Figure 2-2 Active Aeroelastic Wing research aircraft [4]

First, NASA already had a full F/A-18 simulation and the team already had a wealth of knowledge about the aircraft and its systems. Secondly, the leading-edge flap (LEF) of the F/A-18 is split into an inboard and an outboard. A standard F/A-18 LEF is operated as one unit driven by a single actuator. Additional hardware could be added to allow

the inboard section to operate independently of the outboard section. This could provide an additional set of surfaces with which to control the twist, aerodynamics, and load distribution of the wing.

Thirdly, the history of F/A-18 was an interesting one. During preproduction performance testing it was discovered that the wings, which had aluminium aft box wing panels and composite materials with honeycomb cores were too flexible and did not meet the specified roll rates performance criteria at high speeds, because of a loss of control effectiveness and subsequent aileron reversal. For this reason, production version F/A-18 was redesigned with a stronger, heavier wing. This helped to improve the F/A-18 roll performance, but it resulted in a heavier aircraft, reducing range and payload capability.

For the AAW technology demonstration purpose, preproduction version F/A-18 provided a test bed for modification, while production version F/A-18 became the performance target for the AAW technology to compete and compare.

Two modifications were made to the aircraft for the AAW program.

Firstly, the wings were returned to the preproduction torsional stiffness levels to provide greater flexibility for flight testing. Secondly, modifications were made to separate the leading-edge flap (LEF) drive system into independently controlled inboard and outboard flaps as well as increase their upward deflection limits. A schematic illustrating the control surfaces can be seen in Figure 2-3. Figure 2-4 shows photos of modified F/A-18A undergoing AAW wing assembly and torsion testing in the Flight Dynamics Lab at NASA-Dryden.[1]

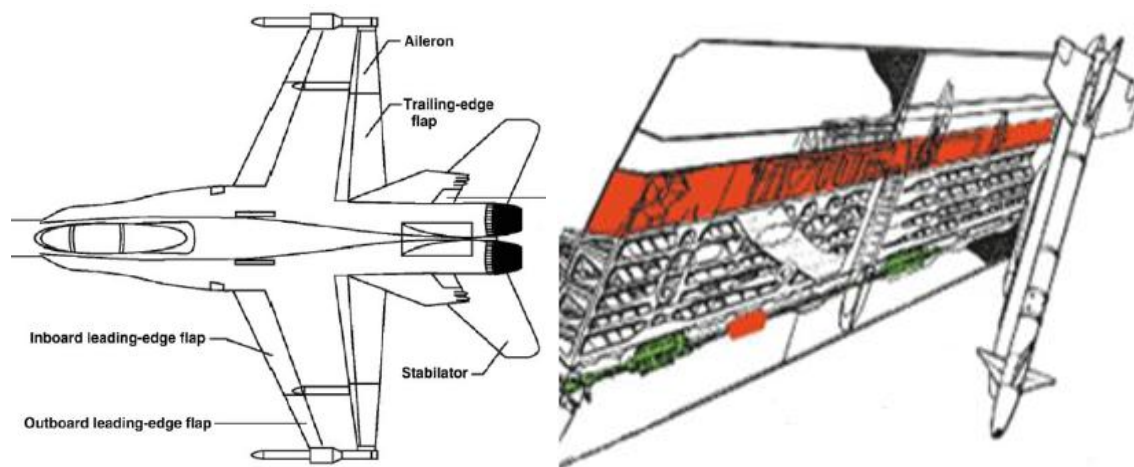


Figure 2-3 F/A-18 AAW Wing modifications [1]




 NASA Dryden Flight Research Center Photo Collection
<http://www.dfrc.nasa.gov/gallery/photo/index.html>
 NASA Photo: EC01-0076-1 Date: March 15, 2001 Photo by: Tom Tschida
 Structural loads testing on the Active Aeroelastic Wing F-18 in the Flight Loads Laboratory at NASA's Dryden
 flight Research Center, Edwards, California




 NASA Dryden Flight Research Center Photo Collection
<http://www.dfrc.nasa.gov/gallery/photo/index.html>
 NASA Photo: EC01-0112-2 Date: April 10, 2001 Photo by: Tom Tschida
 A modified F/A-18A undergoes wing torsion testing in the Flight Dynamics Laboratory at NASA's Dryden Flight
 Research Center, Edwards, California

Figure 2-4 F/A-18 AAW wing assembly and torsion test

The LEF drive system range of motion as can be seen in Table 2-1.

Table 2-1 Control surface rates and limits

Control surface	Rate, deg/sec	Upward travel, deg	Downward travel, deg
Aileron	100	25	45
Inboard leading-edge flap	15	3	34
Trailing-edge flaps	18	18	45
Stabilator	40	10.5	24
Rudder	56	30 (left)	30 (right)
Baseline outboard leading-edge flap	15	3	34
Research flight control system outboard leading-edge flap	45	10	34

The AAW test matrix was made up of 18 separate test points: nine subsonic (Mach 0.85, 0.9, and 0.95) and nine supersonic (Mach 1.1, 1.2, and 1.3). Dynamic pressures ranged from 600 to 1500 psf. Figure 2-5 displays the target AAW test matrix and flight envelope. [3]

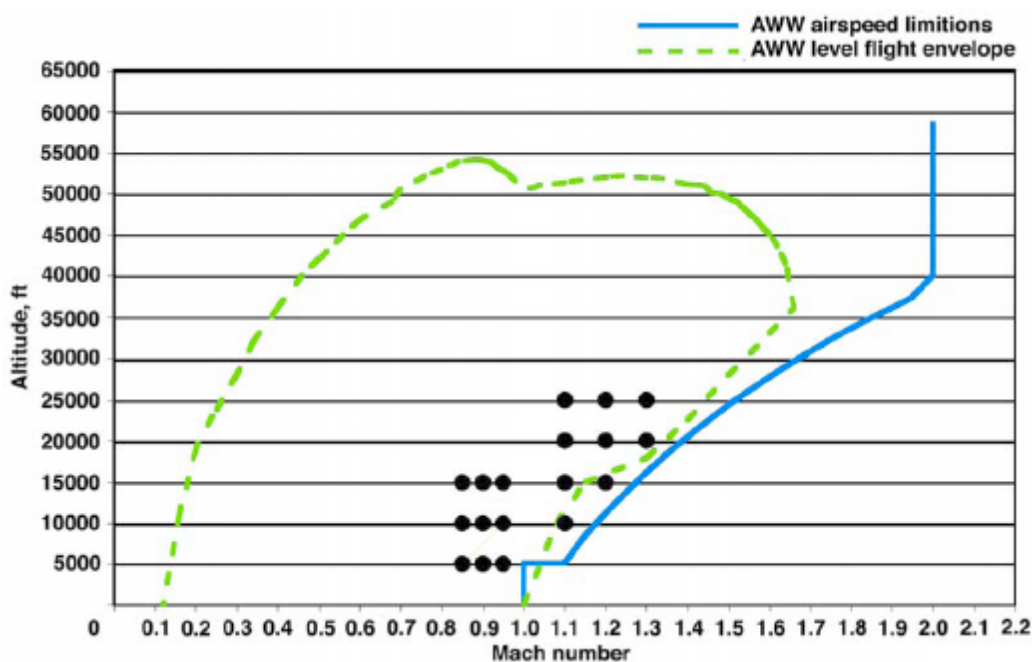


Figure 2-5 Active Aeroelastic Wing test matrix. [4]

However, detailed test results are still confidential, but as Denis Bessette, NASA's AAW Flight Research Program manager, mentioned, the F/A-18 AAW achieved roll rates up to about 200°/sec, which is somewhat less than what a conventional F-18 can

achieve. But studies showed that by applying an AAW solution to a transonic airplane of that type, a 7-10% savings in gross weight is achievable, maintaining a fixed performance.[1]

2.3 LE, TE deflection with aerodynamic consideration

When maximum C_L is desired from a defined 2D airfoil, slat and flap combination is designed for different flight cases. A combination of slat and flap interaction on F-16 fighter plane during different flight phases was presented in Figure 2-6. As one can see that flap was deployed to increase C_L no matter what flight phase the aircraft was in. Slat was deployed to help increase C_L only when high angle of attack (AOA) was encountered. Or we could consider slat mainly helps prevent flow separation above the wing at high α .

If add on the aeroelastic deformation of the wing in 3 dimensional respects. Wing bending won't affect the span wise lift distribution very much, but wing twist has a major impact. If we look at Figure 2-6 again, when maximum C_L and adequate C_M are desired from a defined 2D airfoil, which slat and flap combination is preferred at different flight cases?

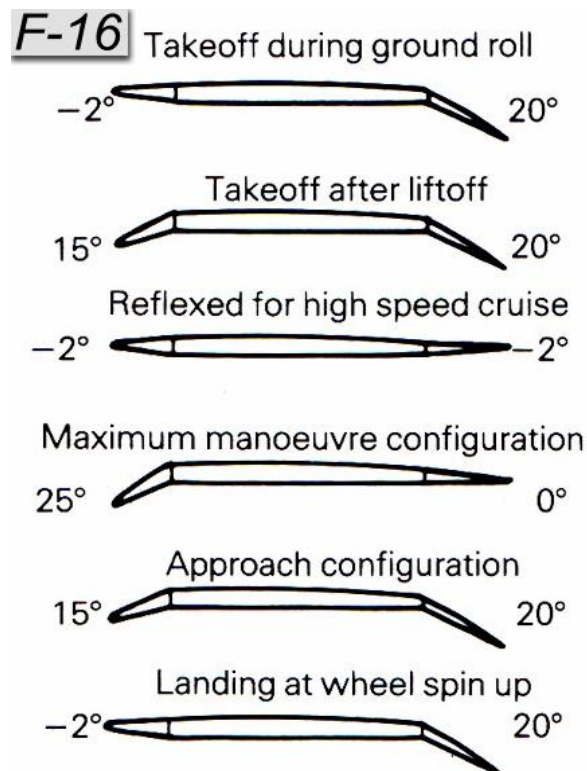
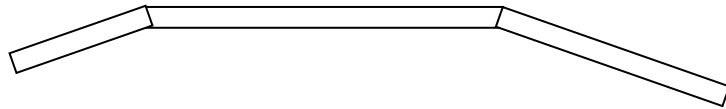


Figure 2-6 Slat and flap combination on F-16 fighter plane [7]

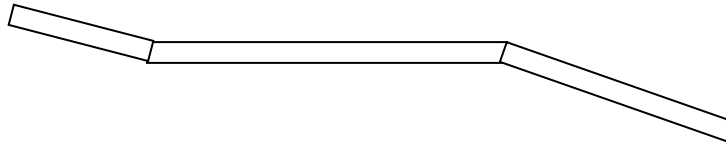
If leading edge slat was considered, there are two control surfaces, each has two deflection modes: up and down. Therefore, there are four combinations to get C_L increase with different C_M variations.

They are: (L for leading edge, T for trailing edge)

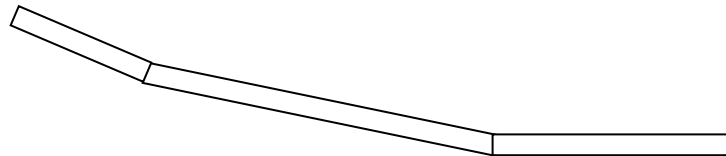
Option 1: L ↓, T ↓



Option 2: L ↑, T ↓



Option 3: L ↑, T ↑



Option 4: L ↓, T ↑

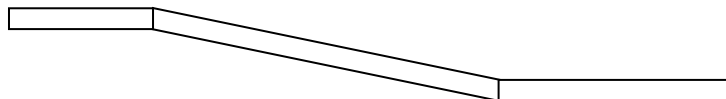


Figure 2-7 Four combinations of leading and trailing edge device deployment

Option 1 and 2 wing centre sections were level, option 3 and 4 add twist to the wing.

It is extremely important to understand the aerodynamic characteristics of the above four wing options, especially the C_M variation, because it would greatly influence the active aeroelastic control strategies in future system design. A detailed 2D

aerodynamic study on this subject is carried out with the aid of Java-foil, results were presented in Chapter 4.



Figure 2-8 Test video captured F/A-18 AAW aircraft control surface motion while rolling

From the test video of F/A-18 AAW aircraft in roll motion (Figure 2-8), we can see the port side wing creates down force with leading and trailing edge combination. This is similar to an inverted option 2 deflection.

2.4 Aeroelasticity

Aeroelasticity is perhaps best described as an interaction between solid and fluid mechanics, with dynamics serving as the adhesive [8][9]. In aerospace applications, divergence, control reversal (in static term), and flutter, dynamic response, buffeting (in dynamic terms) are the main aeroelastic problems that engineers have to overcome before a new aircraft can take its first flight [10][11][12][13]. These aeroelastic phenomena are conveniently summarised in Collar's Triangle of Forces. [14]

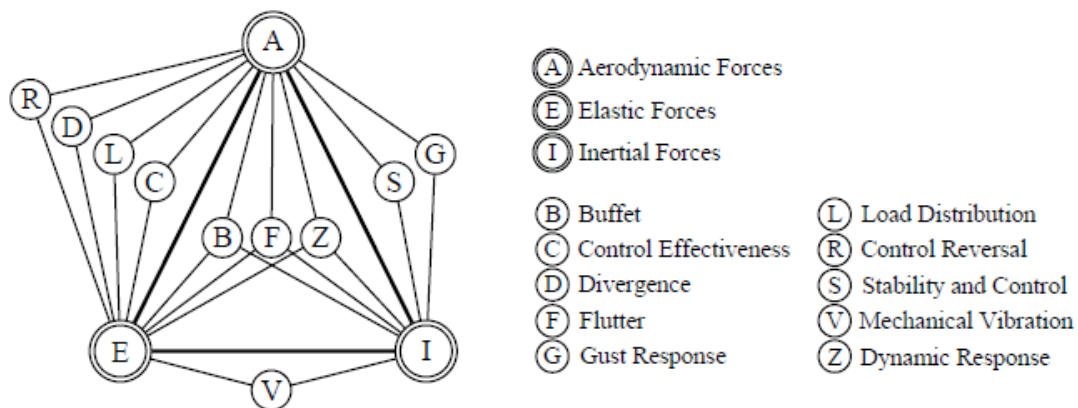


Figure 2-9 Triangle of Forces in Aeroelasticity [8]

From a traditional engineering point of view, aeroelastic phenomena have been treated as a negative factor to be avoided in aircraft structural design. Aeroelastic beneficial effect is rarely properly defined in the academic context but it is the interest of this project. Therefore the review was conducted to cover previous design efforts which treat aeroelasticity as both negative and positive factor.

2.4.1 Design for clearance of aeroelastic negative effect

Divergence and flutter can happen on any part of the aircraft, from main lifting surface to control surfaces and even the fuselage. Control reversal is mainly caused by the control surfaces that are attached to a structure with insufficient stiffness. A good example is the wing, where all three problems (divergence, flutter and control reversal) persist.

In the past, these problems were solved by increasing of wing box stiffness, adding mass or shifting the wing box elastic centre and natural frequency. But these physical changes normally come with weight penalty. More recently, composite tailoring opened a new approach towards an optimal structural solution [15][16]. As this project is focused on aeroelastic aircraft modelling and control system design, composite tailoring is not included.

The recent technological advances made in the field of control systems and the increased reliability of control system components offer a new way of treating aeroelastic problems by utilising control surface aerodynamic forces to combat divergence or flutter instability. The deflection of the control surfaces is constantly adjusted by the control system to provide an artificial stability.

However, the genesis or basement of these approaches differs. One approach is based on the structural instability theory; the control is applied with the feedback of stress and displacement from various sensors in time domain or feedback of oscillation modes and frequencies in frequency domain. The other approach is based on aerodynamic energy considerations, which is mainly used for solving dynamic problems.

When based on the structural instability theory, in recent years, active controls mainly in conjunction with problems associated with gust alleviation (the objective being to

reduce rigid-body response to turbulence) and with mode stabilization (the objective being to reduce structural dynamic response to gusts). [17]

The study, design and test of gust-alleviation systems have been explored over a number of decades and although they have been shown to be effective and technically feasible [18][19][20], they have not been employed very widely.

Mode stabilization, to reduce structural response, is similar to the problem of flutter suppression, which was studied widely during the 60s and 70s.

No attempt will be made to review the extensive literature on gust alleviation, since it's the dynamic response of the wing unrelated to the objective of generating control force.

During the literature review, the study on flutter was most encountered; numerous ideas and attempts were raised and made for decades to suppress flutter.

Theodorsen founded theories for unsteady aerodynamics, which introduced vibration frequency and mode terms to help explain the interaction between aerodynamic force and vibrating structural system [21]. Based on his method, many researches were conducted on flutter suppression; flutter margin augmentation and flapping wings [15][22][23][24][25][26]. Though it has the limitation of small oscillation and flat surface assumptions, in recent years, new methods like 'Deformable Wavy Wake' and 'Unsteady Panel Method' were developed [27]. Solving flutter problem itself is not the focus of this study.

Based on aerodynamic energy considerations. A better understanding on flutter suppression emerged.

Because flutter is a self-excited oscillation in which energy is absorbed by the lifting surface from the air stream. The state of stability of the system is defined by the sign of the work per cycle when the lift in surface undergoes an arbitrary oscillatory motion.

The use of energy techniques to investigate the stability of an aeroelastic system is not new; however, a contribution to the area of flutter suppression is the development of an aerodynamic energy criterion by Nissim in 1971[28]. This criterion states that a necessary and sufficient condition for the prevention of flutter is that for all allowable oscillatory motions of and elastic system in an air stream, positive work must be done by the system on the surrounding air stream.

Maynard and Sandford used this method on a delta-wing to study the transonic flutter suppression in Langley Research Centre [29], they applied three different control laws to the leading and trailing edges, increase of aerodynamic pressure around 22% to 30% is achieved.

2.4.2 Design for Aeroelastic beneficial effect

In terms of utilizing aeroelastic beneficial effect, exploitation on flight surface twist to add aerodynamic force and aid aircraft lateral performance was most commonly studied, typically as part of the wing, which was made more flexible. F/A-18 AAW is the latest project to apply this concept to a real flight.

Stephen and Corey developed and validated the AAW model and controlled it with a PID controller. [4]

Ryan, Michael and Robert developed and tested the full control law for F/A-18 AAW which was used in the test flight. [1]

More academically, Ying Chen and Hsin-Pian Chen used μ -Synthesis to develop the control law which copes better with uncertainties from unsteady aerodynamic loads, nonlinear structural stiffness and variable structural damping. [27]

Andersen and Forster did a work on the multiple control surface utilization; they showed a combination of leading and trailing edge effort to achieve specified roll motion. But the results were totally from simulation. [17]

Martin and Carin tested an elastic wing with 8 control surfaces in the wind tunnel, they used a PD controller to minimize the control effort, with a step input, and they tried several deflection combinations and acquired simulation result similar to wind tunnel test. One discovery from their work is that although with more control surfaces discrete along the span the better aerodynamic efficiency can be achieved. It was those surfaces close to the wing tip contribute the most. Therefore wing shape configuration and control surface layout becomes a serious consideration when trade study on weight saving and power consumption was necessary.[30][31]

Up to this point, we can see that using multiple distributed small control surfaces to deform the wing along the span is going to be the ideal way of achieving active aeroelastic flight control.

The difference of this project compared to the previous studies is to explore the impact to flight control when seamless control surface replaces the conventional hinged surface.

As mentioned earlier, the study on aeroelastic phenomenon helps better understanding of the boundary condition of this project; the active control with multiple control surface optimizations will be carried out below these boundary conditions.

2.4.3 Active Aeroelastic control

The idea of controlling aerodynamic performance by active, mechanically controlled distortion of wings is not new. Elber filed a patent entitled “Means for Controlling Aerodynamically Induced Twist” [32]. His controller was an active torque tube located inside the wing such that it could twist the wing tip relative to the root to control the tendency of swept wings to lose lift effectiveness. He did not claim to directly control drag. At about the same time, programs like the Transonic Aircraft Technology Program (TACT) had performance optimisation objectives involving limit load manoeuvring: this required actively reshaping the spanwise lift distribution with an internal mechanism for chordwise wing bending to force variable camber shapes.

Through camber control, drag polars were reshaped to reduce drag at operational airspeeds and to reduce critical wing bending moments. The constant span drag reduction objective translates into making the spanwise lift distribution as close to elliptical as possible, as long as strength requirements are not violated. The internal mechanisms required to do this reshaping were heavy and literally outweighed their benefits. [33]

Two other innovative approaches to active/passive control of aerodynamic performance are worth mentioning. The first is the use of active materials and adaptive actuators for pitching and rolling performance as well as drag reduction.

Secondly, there has been interest in improving lateral roll authority by intentionally reducing wing structural stiffness (the Active Flexible Wing/Active Aeroelastic Wing) and then controlling the aeroelastic response.

Aircraft control approach allows a wing control system to be so flexible that it will reverse and then be actively controlled. A recent study found that, over a broad range of airspeeds, wing induced drag can be reduced with either passive wing aeroelastic tailoring or active control actuators. When used in combination, induced drag can be minimised with minimal actuator input if the structure is tailored properly. Thus, a combination of structural tailoring and control surface design can actively reduce drag if aeroelastic features are exploited. [34]

2.5 Seamless control surfaces

One step beyond the F/A-18 AAW technology is to replace the current slat and flap arrangement with seamless control surfaces. When deflected, the control surfaces optimise the pressure distribution around the wing to give the desired C_L & C_M relationship which reducing the drag compared to hinged control surface. Many design attempts were made to achieve a seamless morphing mechanism.

One design is so called “finger” concept. The flexible trailing edge is achieved by combining separate plate like elements with joints. The kinematics is shown in Figure 2-10.[35]

Another design called “belt-rib” concept comes from German Aerospace Centre (DLR). The belt-rib structure which replaces the conventional rib was connected to skin by hinges; actuator can push the belt-rib to vary the aerofoil camber (Figure 2-11). [36]

Musgrove suggested a curved tube eccentric actuator to transmit rotation to bending motion. It is a mechanical design that has the potential for aerospace application (Figure 2-12). [37]

A more successful design comes from Flexsys Inc. They demonstrated smart material actuated adaptive morphing wing in 2006. The company claimed that their adaptive flaps were aeroelastically equivalent to conventional control surfaces, strong enough to

withstand repeated lifecycle loading, and competitive in terms of weight, force, and power consumption (See Figure 2-13). [38]

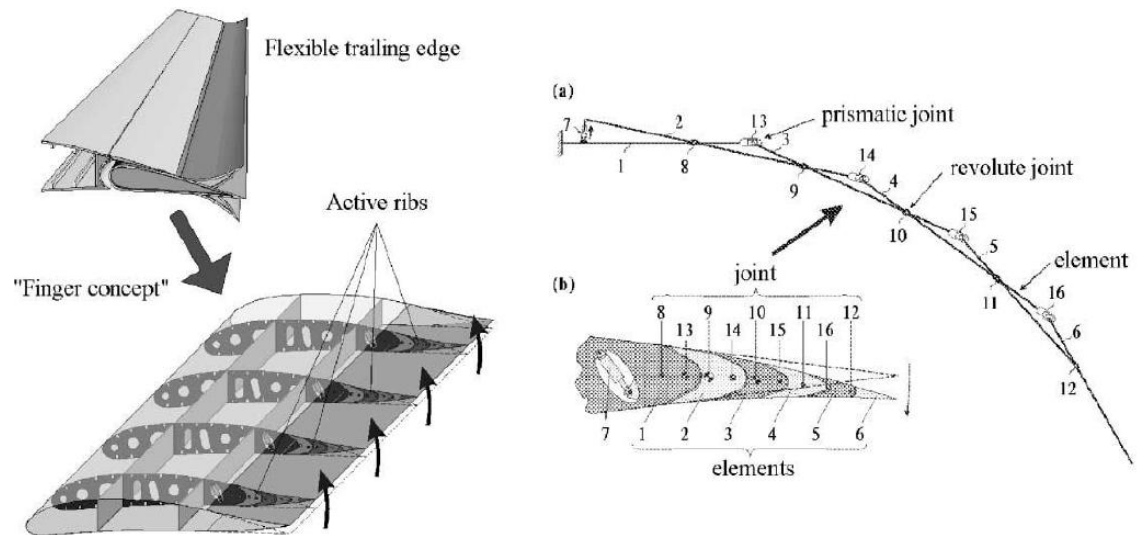


Figure 2-10 Finger concept seamless trailing edge



Figure 2-11 Belt-rib morphing concept

The seamless actuation mechanism designed in this research is inspired by all the above concepts but also has its own unique feature. Detailed design information is presented in chapter 4.

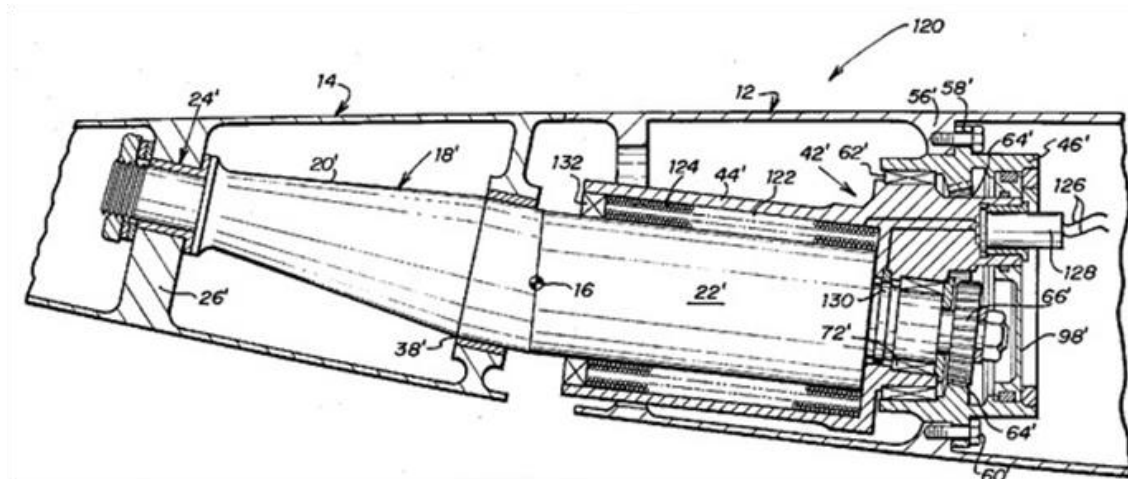


Figure 2-12 Eccentric actuator schematics

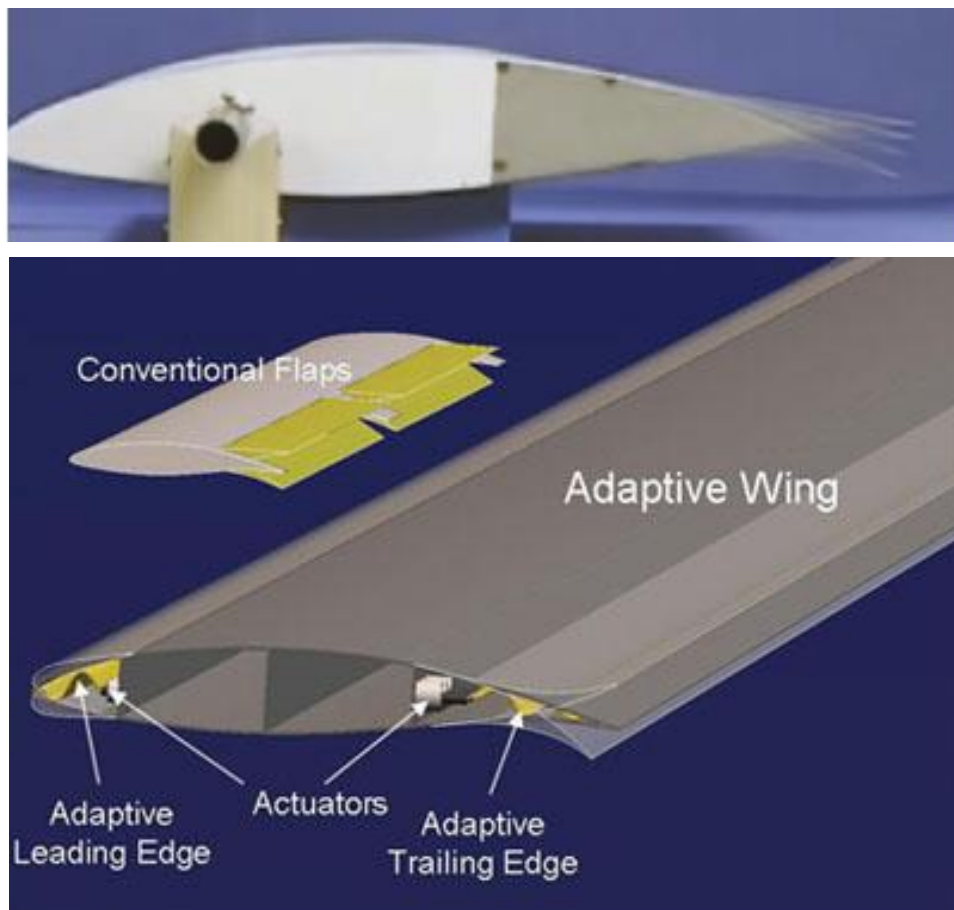


Figure 2-13 Adaptive wing seamless control surface

2.6 Unmanned Dilemma on Flying and Handling Qualities Assessment

The flying and handling qualities of an aeroplane are those properties which describe the ease and effectiveness with which it responds to pilot commands in the execution of some flight task. The flying quality is regarded as being task related, whereas the handling quality being response related. [39][40][41]

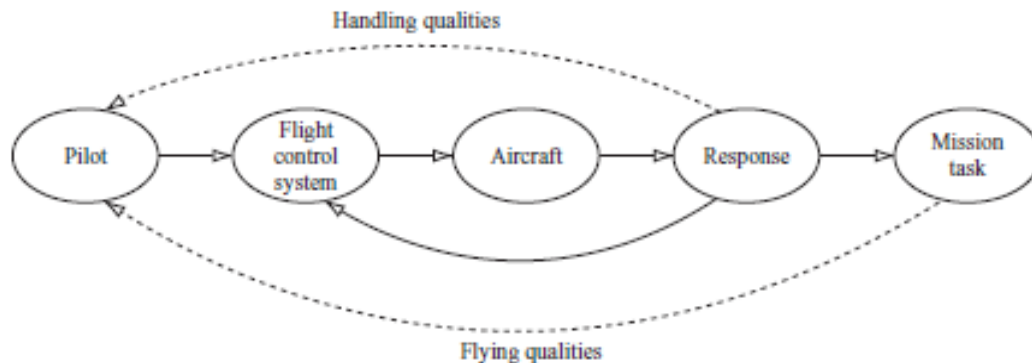


Figure 2-14 Flying and handling qualities of fly-by-wire aircraft

In the unmanned air vehicle, the non-existence of human pilot would render all handling quality criteria irrelevant. Mission tasks can still be accomplished by auto-pilot with insufficient damping in much wider bandwidth. But as flight controller design is traditionally based on the well established regulations such as DEF-STAN-00-970 or MIL-F-8785C. It is still worthwhile to use these standards to check flying quality of unmanned aircraft.

2.7 Available & Potential Actuators

A collection of actuators were reviewed, some of them may be used in AAW technology.

2.7.1 Hydraulic Actuators

Hydraulic actuation in aircraft refers to actuation achieved using actuators powered by the aircraft's hydraulic main circuit. This actuation may be controlled mechanically or electrically, depending upon particular system architecture.

Hydraulic actuator with mechanical control is relatively simple, in that control surface movement demands are met by hydraulically power assisted actuation. In these types of systems, actuators are often able to accept control surface movement demands from

two separate channels of the hydraulic systems. Thus providing control system redundancy.

Actuators have a failure reversion mode that allows aerodynamic centring to move the control surfaces into neutral positions under the aerodynamic loads experienced during flight. However, most modern aircraft use hydraulically powered actuation controlled by electrical signalling. Such systems work in a similar manner to the mechanically controlled systems. But in this case, control surface movement demands are passed to the actuator electrically. There are generally two types of electrical control signals in the type of system architecture. The first is a straight forward demand signal initiated from the pilot's controls in the cockpit. The second type is an auto-stabilisation signal initiated from a control computer.

2.7.2 Pneumatic actuators

In the past, pneumatic power has been used for the actuation of control surfaces. However, today this is rare because of the high mass and volume of ducting that would be required to provide sufficient actuation power. Also the relatively long response time of the pneumatic actuator meant it was more suitable for non-primary control surfaces, for example the slats on the Boeing 747, which can be actuated pneumatically.

But as technology advances, this is about to change. A new pneumatic actuation device was invented recently, it is called Air Muscle. Air Muscle is a simple yet powerful device for providing a pulling force. It behaves in a very similar way to a biological muscle. When actuated with a supply of compressed air, they contract by up to 40% of its original length. The force it provides decreases as it contracts, and the first few percent of the contraction is very powerful indeed. See Figure 2-15. [42]



Figure 2-15 Air muscle extension and extraction

2.7.3 Electro-Hydrostatic Actuators (EHA)

EHA is an actuation implementation under active development. In this method, power is provided to an actuator electrically. Each actuator has an electric motor which drives a localised hydraulic system, including a hydraulic pump and reservoir. Such technology includes the use of microprocessors for digital control, which improves system performance.

2.7.4 Electro-Mechanical Actuators (EMA)

EMA is an emerging technology for control surface actuation applications. An EMA dispenses with hydraulics altogether and uses an electric motor and a gearbox to implement the actuation. While EMA has been used for many years on lower powered actuation tasks such as trim actuation and powered doors, recent advances in technology are making high powered actuation implementations feasible[43].

2.7.5 Piezoelectric translator (PZT)

A Piezoelectric translator (linear actuator) is a solid-state ceramic actuator which converts electrical energy directly into linear motion (mechanical energy) with virtually unlimited resolution.

Types of piezoelectric motor include the well-known travelling-wave motor used for auto-focus in reflex cameras, inchworm motors for linear motion, and rectangular four-quadrant motors with high power density (2.5 watt/cm^3) and speed ranging from 10 nm/s to 800 mm/s. All these motors work on the same principle. Driven by dual orthogonal vibration modes with a phase shift of 90° , the contact point between two surfaces vibrates in an elliptical path, producing a frictional force between the surfaces. Usually, one surface is fixed causing the other to move. In most piezoelectric motors the piezoelectric crystal is excited by a sine wave signal at the resonant frequency of the motor. Using the resonance effect, a much lower voltage can be used to produce a high vibration amplitude

Advantages of PI Piezoelectric Actuators are :

- Sub-Nanometre resolution
- Large force generation (up to 50,000 N and more)
- Microsecond-range response

- No backlash, stiction or friction
- Immune to magnetic fields
- Extremely low steady-state power consumption
- No wear and tear
- Vacuum and clean-room compatibility
- Operation at cryogenic temperatures possible

2.7.6 Electro-active Polymers (EAP)

Electro-active Polymers or EAPs are polymers whose shape is modified when a voltage is applied to them. They can be used as actuators or sensors. As actuators, they are characterized by the fact that they can undergo a large amount of deformation while sustaining large forces. Due to the similarities with biological tissues in terms of achievable stress and force, they are often called artificial muscles, and have the potential for application in the field of robotics, where large linear movement is often needed.

EAP can have several configurations, but are generally divided in two principal classes:

Dielectric EAPs, in which actuation is caused by electrostatic forces between two electrodes which squeeze the polymer. This kind of EAP is characterized by a large actuation voltage (several thousand volts), but very low electrical power consumption. Dielectric EAPs require no power to keep the actuator at a given position. Examples are Electrostrictive Polymers and Dielectric elastomers.

Ionic EAPs, in which actuation is caused by the displacement of ions inside the polymer. Only a few volts are needed for actuation, but the ionic flow implies a higher electrical power needed for actuation, and energy are needed to keep the actuator at a given position. Examples of ionic EAPS are Conducting Polymers, Ionic Polymer Metal Composites (IPMC), and Responsive Gels.

2.7.7 Shape Memory Alloys

A shape memory alloy (SMA), also known as a smart alloy or memory metal) is an alloy that "remembers" its geometry. After a sample of SMA has been deformed from its original crystallographic configuration, it regains its original geometry by itself during heating (one-way effect) or, at higher ambient temperatures, simply during unloading (superelasticity). These extraordinary properties are due to a temperature-dependent martensitic phase transformation from a low-symmetry to a highly symmetric crystallographic structure. Those crystal structures are known as martensite (at lower temperatures, highly twinned crystalline structure) and austenite (at higher temperatures, body-centred cubic structure).

The three main types of SMA are the copper-zinc-aluminium, copper-aluminium-nickel, and nickel-titanium (NiTi) alloys. NiTi alloys are generally more expensive and possess superior mechanical properties when compared to copper-based SMAs. The temperatures at which the SMA changes its crystallographic structure are characteristic of the alloy, and can be tuned by varying the elemental ratios.

2.7.8 Summary

Electro-mechanical actuator seems to be the most reliable, low-risk choice. But PZT actuators, with their unique advantages, are becoming more and more popular in industry in recent years. Although lacking of large travel actuation, a row of properly designed small distributed control surfaces along the wing span may just need that much actuation displacement. Especially for those seamless control surfaces, the actuator maybe embedded just in the wing skin. In such case, the size and force that PZT actuator could offer makes it an appealing candidate.

Chapter 3 Theory and Analysis Methods

The theory, equations and analytical methods used in this thesis are presented in this chapter. Necessary assumptions made to validate their applicability are also presented.

3.1 Axes system

3.1.1 Axes definition (Mean or Local)

Axes system is a appropriate coordinate system in which the equations of motion can be developed. It is therefore fundamental and crucial for modelling and analysis of flight dynamics. In the context of flexible aircraft modelling and analysis, motions of both aircraft (as single mass point) and its wing structure (as beam element) are all considered, but the axes system is not shared.

The body axes system for aircraft modelling regards CG as its origin. Beam axes system for the wing structure regards each beam mass nodal position as its origin. The definition of positive direction is also different (See Table 3-1 and Figure 3-1).

Table 3-1 Axes difference aircraft and structure model

<i>Axes System</i>	<i>Aircraft</i>	<i>Wing Structure</i>
Datum Point	Centre of Gravity	Mass Element Node
X Positive	Forward	Backward
Y Positive	Right	Outboard
Z Positive	Down	Up
Rotation	Right Handed	Left Handed
Variables	Acceleration, Velocity	Acceleration, Velocity, Displacement

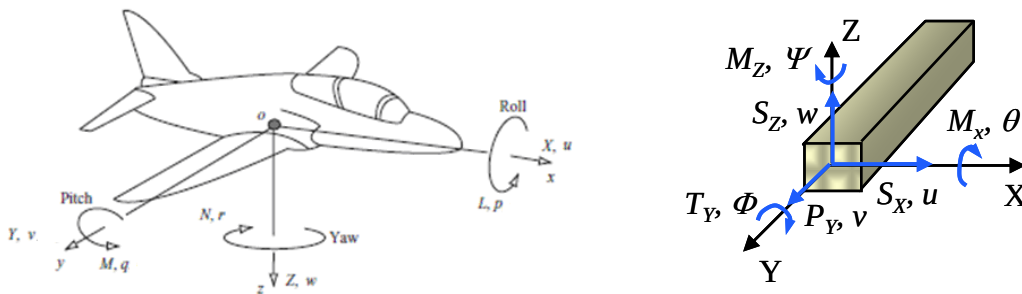


Figure 3-1 Axes system and variable notation for rigid aircraft and structures

For flexible aircraft, it is common to use mean-axes system to develop equation of motion. The mean-axes system is defined such that it does not remain fixed at some material point within the body, but is free to float relative to the body. Relative angular and linear momentum with respect to the body axes is zero at every instant in time in mean-axes system.

However, in the real aircraft, flight control relies on the flight parameter feedback from the sensor package (Inertia Measurement Unit IMU), which is physically fixed on a mass point on the aircraft.

In order to reflect the control challenge caused by inertial coupling between the overall body reference frame and the elastic deformation of the aircraft body, local axes system was adopted. The origin of the local axes system is on the mass node corresponds to IMU location in the aircraft fuselage. Force and moment inputs are hence not only include aerodynamic and gravitational terms but also the inertial term of fuselage.

As one may notice the displacement term in structure axes system does not exist in aircraft axes system, it is because all nodes in the structure have same axes orientation including the fixed node (boundary condition). In aircraft axes system, the fixed reference point is earth which has different axes orientation. Axes transformation is needed for further flight dynamic calculation.

3.1.2 Axes transformation

It is not necessary to transform local axes to mean axes since all force and moment calculated on the fuselage node can be directly solved by equation of motion for flight response with a negative sign. But for gravity calculation and navigation purposes, it is important to know the aircraft's attitude and position in space relative to the earth. Direction Cosine Matrix (DCM body to earth) is hence introduced.

$$DCM_{be} = \begin{bmatrix} \cos \theta \cos \psi & \cos \theta \sin \psi & -\sin \theta \\ \sin \phi \sin \theta \cos \psi - \cos \phi \sin \psi & \sin \phi \sin \theta \sin \psi + \cos \phi \cos \psi & \sin \phi \cos \theta \\ \cos \phi \sin \theta \cos \psi + \sin \phi \sin \psi & \cos \phi \sin \theta \sin \psi - \sin \phi \cos \psi & \cos \phi \cos \theta \end{bmatrix}$$

Equation 3-1

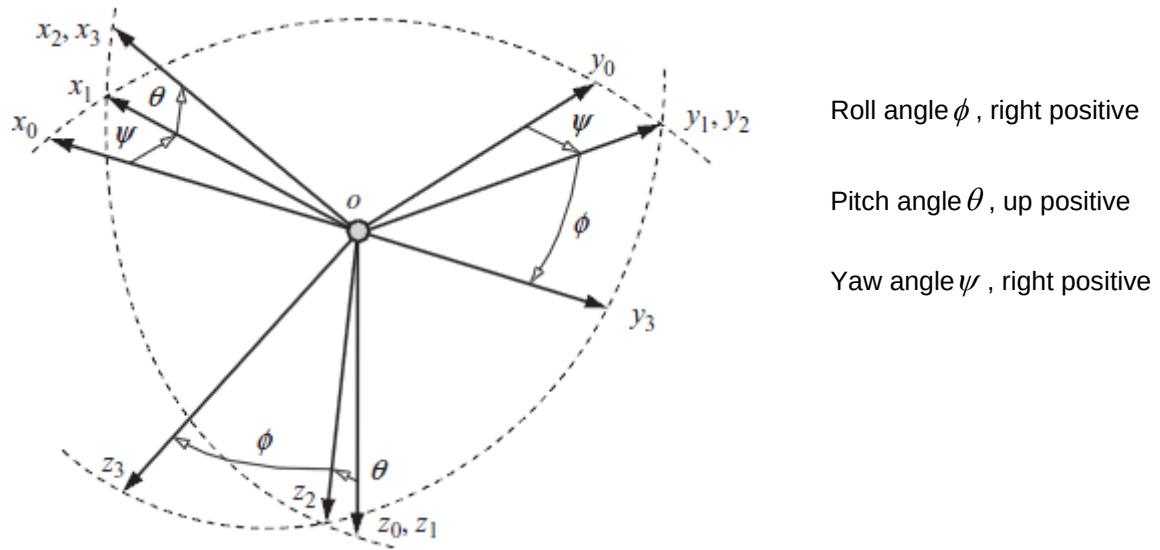


Figure 3-2 The Euler angles [44]

Aerodynamic forces and moments are calculated relative to the flight velocity vector i.e. wind axes, which has differences in angle of attack α and side slip angle β relative to body axes (see Figure 3-3). Transformation from wind axes to body axes is:

$$DCM_{wb} = \begin{bmatrix} \cos \alpha \cos \beta & \sin \beta & \sin \alpha \cos \beta \\ -\cos \alpha \sin \beta & \cos \beta & -\sin \alpha \sin \beta \\ \sin \alpha & 0 & \cos \alpha \end{bmatrix}$$

Equation 3-2

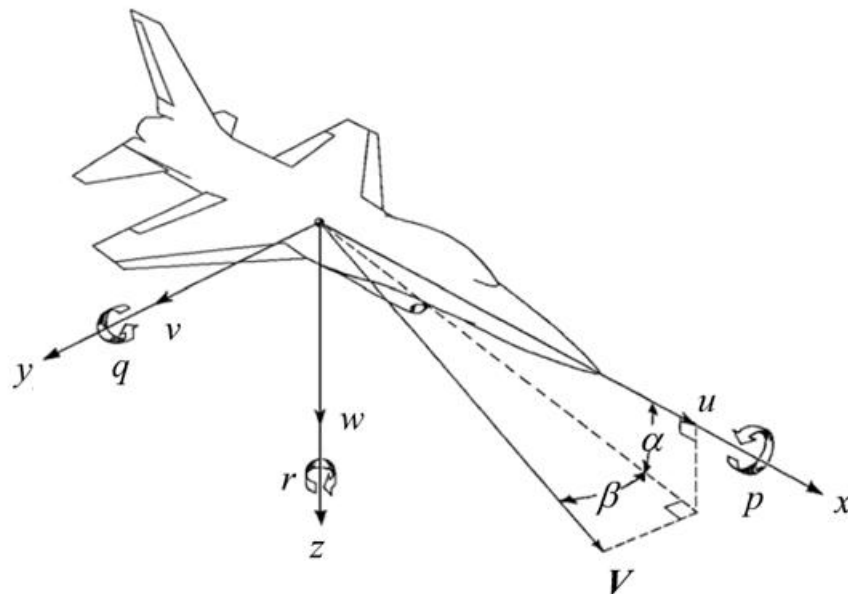


Figure 3-3 Body fixed axis system (rigid body model)

3.2 Equations of motion (EOM)

3.2.1 Rigid body aircraft EOM

Newton's second law of motion for each of the six degrees of freedom forms the foundation of rigid aircraft equation of motion.

$$\text{Mass} \times \text{Acceleration} = \text{Force}$$

And for rotary motion:

$$\text{Moment of Inertia} \times \text{Angular acceleration} = \text{Moment}$$

Definition of motion variables in the derivation is shown in Figure 3-4, Table 3-2 and Table 3-3

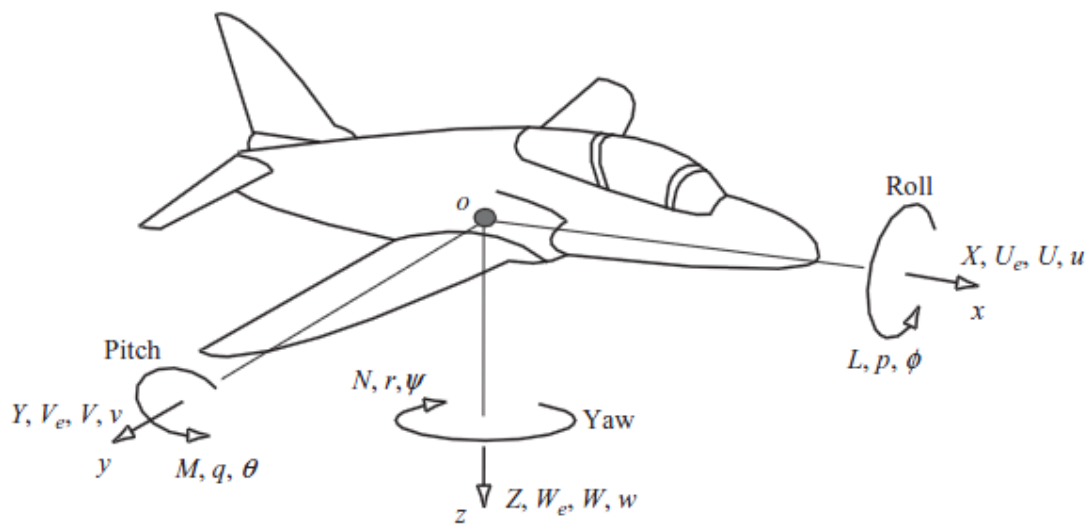


Figure 3-4 Body axis motion variables notation [44]

Table 3-2 Motion variables

	<i>Trimmed equilibrium</i>			<i>Perturbed</i>		
Aircraft axis	ox	oy	oz	ox	oy	oz
Force	0	0	0	X	Y	Z
Moment	0	0	0	L	M	N
Linear velocity	U_e	V_e	W_e	U	V	W
Angular velocity	0	0	0	p	q	r
Attitude	0	θ_e	0	ϕ	θ	ψ

Table 3-3 Perturbation variables

X	Axial “drag” force	Sum of the components of aerodynamic, thrust and weight forces
Y	Side force	
Z	Normal “lift” force	
L	Rolling moment	Sum of the components of aerodynamic, thrust and weight moments
M	Pitching moment	
N	Yawing moment	
p	Roll rate	Components of angular velocity
q	Pitch rate	
r	Yaw rate	
U	Axial velocity	Total linear velocity components of the cg
V	Lateral velocity	
W	Normal velocity	

Mass, second moment of inertia and product moment of inertia terms are:

$$m, I_x, I_y, I_z, I_{xy}, I_{xz}, I_{yz}$$

In the force equation, acceleration terms are express as

$$\begin{aligned} a_x &= \dot{U} - rV + qW - x(q^2 + r^2) + y(pq - \dot{r}) + z(pr + \dot{q}) \\ a_y &= \dot{V} - pW + rU + x(pq + \dot{r}) - y(p^2 + r^2) + z(qr - \dot{p}) \\ a_z &= \dot{W} - qU + pV + x(pr - \dot{q}) + y(qr + \dot{p}) - z(p^2 + q^2) \end{aligned}$$

Equation 3-3

If axis datum at CG, then $x = y = z = 0$, total force on the rigid body are given by

$$\begin{aligned} m(\dot{U} - rV + qW) &= X \\ m(\dot{V} - pW + rU) &= Y \\ m(\dot{W} - qU + pV) &= Z \end{aligned}$$

Equation 3-4

Angular accelerations are derived from linear counterparts [39].

The total moments acting on the rigid body are:

$$\begin{aligned}
I_x \dot{p} - (I_y - I_z)qr + I_{xy}(pr - \dot{q}) - I_{xz}(pq + \dot{r}) + I_{yz}(r^2 - q^2) &= L \\
I_y \dot{q} + (I_x - I_z)pr - I_{xy}(qr + \dot{p}) + I_{xz}(p^2 - r^2) + I_{yz}(pq - \dot{r}) &= M \\
I_z \dot{r} - (I_x - I_y)pq + I_{xy}(q^2 - p^2) + I_{xz}(qr - \dot{p}) - I_{yz}(pr + \dot{q}) &= N
\end{aligned}$$

Equation 3-5

If the aeroplane is symmetric about o - x - z plane and mass is uniformly distributed, $I_{xy} = I_{yz} = 0$, moment equation simplify to be:

$$\begin{aligned}
I_x \dot{p} - (I_y - I_z)qr - I_{xz}(pq + \dot{r}) &= L \\
I_y \dot{q} + (I_x - I_z)pr + I_{xz}(p^2 - r^2) &= M \\
I_z \dot{r} - (I_x - I_y)pq + I_{xz}(qr - \dot{p}) &= N
\end{aligned}$$

Equation 3-6

Aircraft control forces were imposed by deflection of aileron ξ , elevator η and rudder ζ . The positive sign of deflections were defined as right aileron down, elevator down and rudder left (see Figure 3-5). Engine thrust variable is τ .

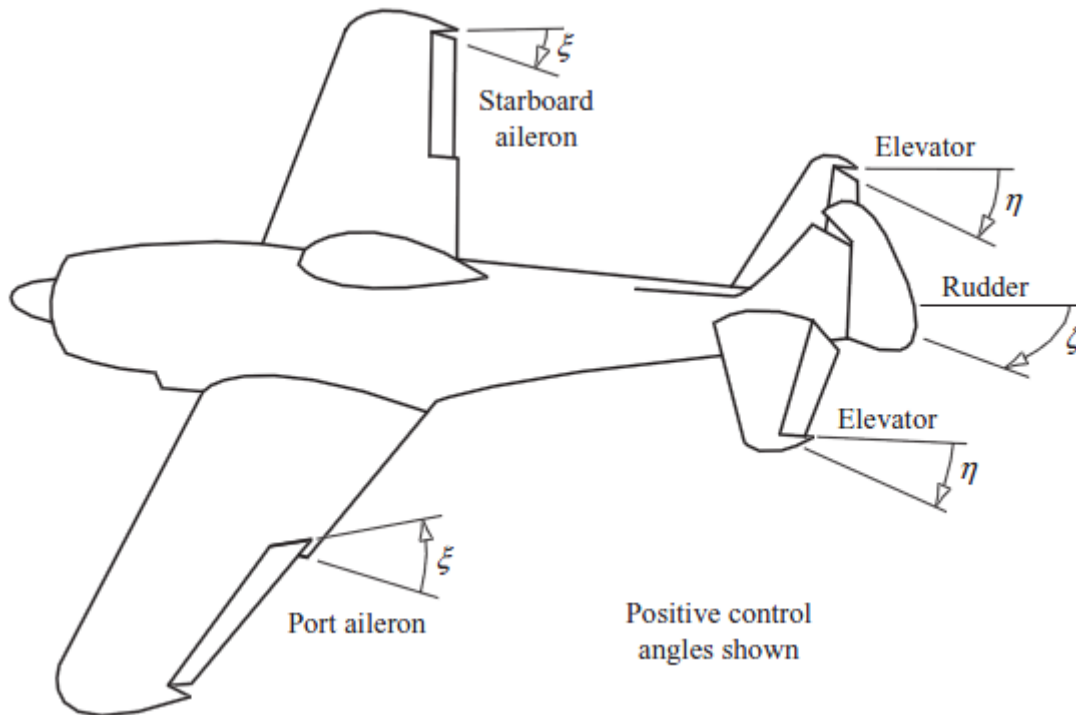


Figure 3-5 Aerodynamic control sign convention

The force and moment terms on the right hand side (RHS) of the equation consists: Aerodynamic effects, Gravitational effects, movement of aerodynamic Controls, Propulsion effect and atmospheric Disturbances. Modelling of these forces and moments will be discussed in chapter 4 modelling of rigid aircraft.

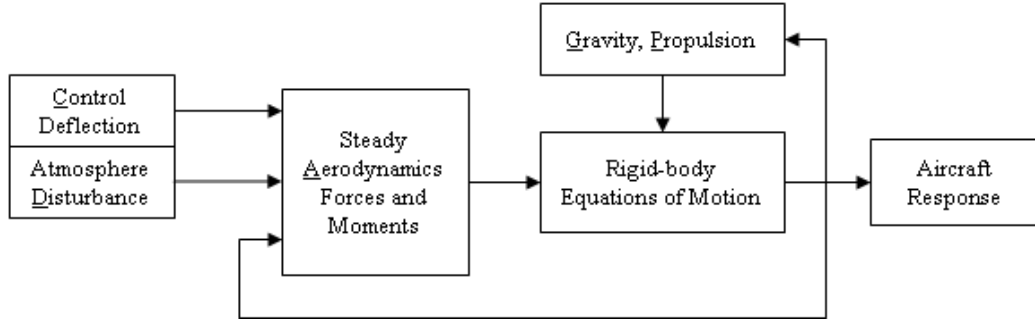


Figure 3-6 Rigid aircraft equation of motion

3.2.2 Decoupled EOM (state-space expression)

3.2.2.1 Assumptions

Dynamic modes of an aeroplane can be obtained by assuming that the influence of longitudinal motion on the lateral and directional forces and moment is negligible and vice-versa. (This assumption is reasonable for rigid aircraft but not so good for flexible aircraft of very large size.)

Therefore, forces and moments due to translational and rotational velocity coupling effect, such as $X_v, X_p, X_r, Z_v, Z_p, Z_r, M_v, M_p, M_r, Y_u, Y_w, Y_q, L_u, L_w, L_q, N_u, N_w, N_q$ are assumed to be zero. Equally it is necessary to assume that the CG offset is zero and that the aircraft is symmetric about the xz-plane. That is CG offset distance $d_x, d_y, d_z = 0$ and $I_{xy}, I_{yz} = 0$.

In addition, down wash effect is not present for tailless aircraft, hence $X_{\dot{w}}, Z_{\dot{w}}, M_{\dot{w}} = 0$

3.2.2.2 Equation of longitudinal motion

Applying the assumptions indicated above, EOM was decoupled into longitudinal and lateral-directional expressions. They can be written into state space format. The longitudinal EOM becomes:

$$M_{mlong} \dot{x} = M_{along} x + M_{clong} u$$

Equation 3-7

Where: M_{mlong} is mass matrix, M_{along} is aerodynamic matrix and M_{clong} is control matrix, $\dot{x}$ contains state derivatives, x contains state variable and ui contains control inputs.

$$\dot{x} = \begin{bmatrix} \dot{u} \\ \dot{w} \\ \dot{q} \\ \dot{\theta} \end{bmatrix} \quad x = \begin{bmatrix} u \\ w \\ q \\ \theta \end{bmatrix} \quad ui = \begin{bmatrix} \eta \\ \tau \end{bmatrix}$$

$$M_{mlong} = \begin{bmatrix} m & 0 & 0 & 0 \\ 0 & m & 0 & 0 \\ 0 & 0 & I_{yy} & 0 \\ 0 & 0 & 0 & 1 \end{bmatrix} \quad M_{clong} = \begin{bmatrix} X_{\eta} & X_{\tau} \\ Z_{\eta} & Z_{\tau} \\ M_{\eta} & M_{\tau} \\ 0 & 0 \end{bmatrix}$$

$$M_{along} = \begin{bmatrix} X_u & X_w & (X_q - mW_e) & -mg \cos \theta_e \\ Z_u & Z_w & (Z_q + mU_e) & -mg \sin \theta_e \\ M_u & M_w & M_q & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix}$$

3.2.2.3 Equation of lateral and directional motion

Similarly, applying the assumptions indicated in paragraph one:

$$M_{mlat-dir} \dot{x} = M_{alat-dir} x + M_{clat-dir} ui$$

Equation 3-8

Where:

$$\dot{x} = \begin{bmatrix} \dot{v} \\ \dot{p} \\ \dot{r} \\ \dot{\phi} \\ \dot{\psi} \end{bmatrix} \quad x = \begin{bmatrix} v \\ p \\ r \\ \phi \\ \psi \end{bmatrix} \quad ui = \begin{bmatrix} \xi \\ \zeta \end{bmatrix}$$

$$M_{mlat-dir} = \begin{bmatrix} m & 0 & 0 & 0 & 0 \\ 0 & I_{xx} & I_{xz} & 0 & 0 \\ 0 & I_{xz} & I_{zz} & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 0 & 1 \end{bmatrix} M_{clat-dir} = \begin{bmatrix} Y_\xi & Y_\zeta \\ L_\xi & L_\zeta \\ N_\xi & N_\zeta \\ 0 & 0 \\ 0 & 0 \end{bmatrix}$$

$$M_{alat-dir} = \begin{bmatrix} Y_v & (Y_p + mW_e) & (Y_r - mU_e) & mg \cos \theta_e & mg \sin \theta_e \\ L_v & L_p & L_r & 0 & 0 \\ N_v & N_p & N_r & 0 & 0 \\ 0 & 1 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 \end{bmatrix}$$

3.2.3 Flexible body aircraft EOM

Flexible body aircraft EOM retains most components of rigid body aircraft EOM but introduces an aero-structural interaction inner loop. The aeroelastic inner loop consists of structural dynamic model and unsteady aerodynamic model, calculation of force and deformation iterates on until equilibrium between aero force and structural elastic force is reached. Theoretical foundation of the aeroelastic inner loop will be detailed in the later paragraph; the author would like to emphasis the impact of the inner loop here.

The input of control force and disturbance will excite the dynamic response of the structure as well as response of aircraft in flight; this coupled response is of great interest from flight and handling quality point of view. Many aeroelastic problems will arise regarding certain aircraft design layout, each problem will have substantial influence on flight control law design.

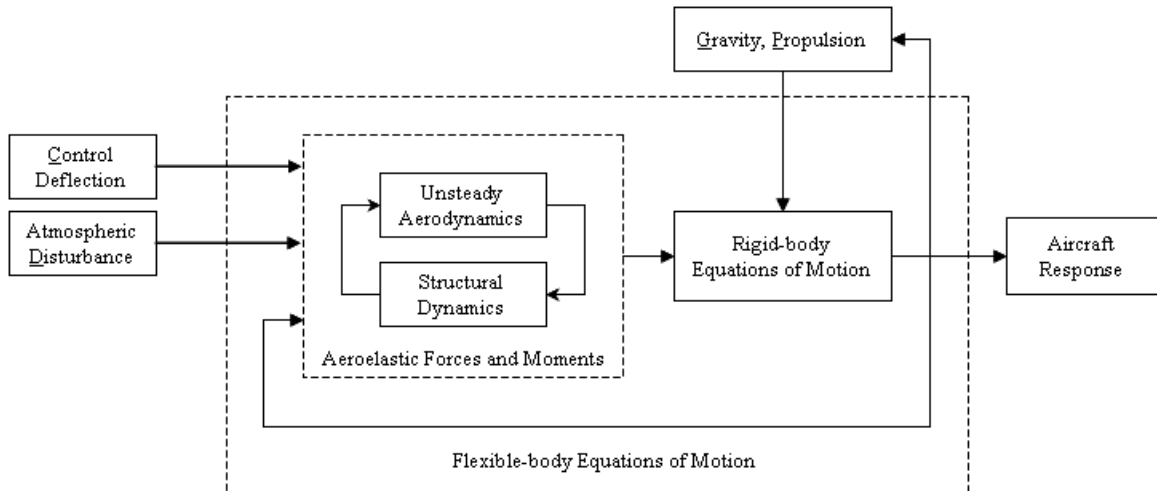


Figure 3-7 Flexible aircraft equation of motion

3.3 Elastic Beam Theory

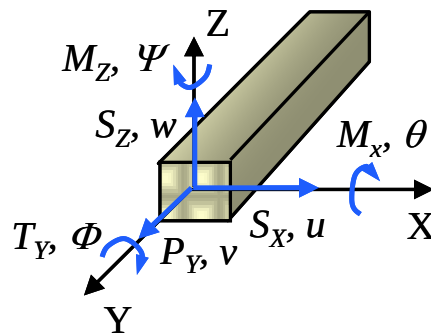


Figure 3-8 Beam axes variables

For a single beam element of length l , cross-sectional area A , effective shear area A_x and A_z , second moment of area I_x and I_z , the beam itself is weightless, its mass is concentrated on two mass nodes on each end of the beam, the node has a mass of M , mass inertia I_{xx} , I_{yy} , I_{zz} , I_{xy} , I_{yz} , I_{xz} and torsional inertia J . Material properties are Young's modulus E , shear modulus G .

Single beam element dynamic equation is

$$[M]\ddot{x} + [C]\dot{x} + [K]x = [F_A]$$

Equation 3-9

Where $[M]$ is the mass matrix, $[C]$ is the damping matrix, $[K]$ is the stiffness matrix and $[F_A]$ is applied load.

Mass matrix $[M]$ is:

$$[M] = \begin{bmatrix} M & 0 & MZ & 0 & 0 & -MY \\ 0 & M & -MX & 0 & MY & 0 \\ MZ & -MX & I_{yy} & 0 & -I_{xy} & -I_{yz} \\ 0 & 0 & 0 & M & MZ & -MX \\ 0 & MY & -I_{xy} & MZ & I_{xx} & -I_{xz} \\ -MY & 0 & -I_{yz} & -MX & -I_{xz} & I_{zz} \end{bmatrix}$$

Equation 3-10

Damping matrix $[C] = \zeta \times [K] / \omega$, ζ is 2% for most structures internal damping, ω is the 1st order natural frequency of the beam.

Definition of the stiffness matrix $[K]$ is much more complicated, it is derived from flexibility matrix $[F]$.

Considering the beam is subject to principle load P_Y , shear load S_X and S_Z , torque T_Y , bending moment M_X and M_Z . The deformations with respect to the loadings are linear displacements v , u , w and rotational angle ϕ , θ , ψ . Deformation and loading were related by flexibility matrix $[F]$.

$$\begin{bmatrix} u \\ w \\ \phi \\ v \\ \theta \\ \psi \end{bmatrix} = [F] \begin{bmatrix} S_X \\ S_Z \\ T_Y \\ P_Y \\ M_X \\ M_Z \end{bmatrix}$$

Equation 3-11

The flexibility matrix $[F]$ (6×6) for the single beam element is:

$$[F] = \begin{bmatrix} F_{11} & 0 & 0 & 0 & 0 & F_{61} \\ 0 & F_{22} & 0 & 0 & F_{52} & 0 \\ 0 & 0 & F_{33} & 0 & 0 & 0 \\ 0 & 0 & 0 & F_{44} & 0 & 0 \\ 0 & F_{52} & 0 & 0 & F_{55} & 0 \\ F_{61} & 0 & 0 & 0 & 0 & F_{66} \end{bmatrix}$$

Equation 3-12

Where

$$F_{11} = \frac{l^3}{3EI_z} + \frac{l}{GA_x}$$

$$F_{22} = \frac{l^3}{3EI_x} + \frac{l}{GA_z}$$

$$F_{33} = \frac{l}{GJ}$$

$$F_{44} = \frac{l}{EA}$$

$$F_{55} = \frac{l}{EI_x}$$

$$F_{66} = \frac{l}{EI_z}$$

$$F_{52} = \frac{l^2}{2EI_x}$$

$$F_{61} = -\frac{l^2}{2EI_z}$$

For a single beam element the local stiffness matrix $[K_i]$ (12×12) is

$$[K_i] = [C_E]^T [F]^{-1} [C_E]$$

Equation 3-13

$[C_E]$ is expanding matrix to account for degree of freedom of both nodes

$$[C_E] = \begin{bmatrix} 1 & 0 & 0 & 0 & 0 & -l & -1 & 0 & 0 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 & l & 0 & 0 & -1 & 0 & 0 & 0 & 0 \\ 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & -1 & 0 & 0 & 0 \\ 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & -1 & 0 & 0 \\ 0 & 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & -1 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 & 0 & 0 & 0 & 0 & 0 & -1 \end{bmatrix}$$

Equation 3-14

Since the beam is orientated arbitrarily with respect to the global axis, a transformation matrix $[T]$ (6×6) is multiplied to the leading diagonal of $[K_i]$ to get global stiffness matrix $[K]$. Definition of $[T]$ refers to Appendix.

$$[K] = \begin{bmatrix} [T]^T & 0 \\ 0 & [T]^T \end{bmatrix} [K_i] \begin{bmatrix} [T] & 0 \\ 0 & [T] \end{bmatrix}$$

Equation 3-15

The derivation from structural geometry and material property to mass and stiffness matrix is accomplished by an in-house FORTRAN code called 'Beam-3D'. Beam-3D is developed and improved by Dr Guo et al in Cranfield University. [45]

3.4 Unsteady Aerodynamics

3.4.1 Unsteady Aerodynamics (Theodorsen's Theory 2D)

The problem of finding the aerodynamic loads on an oscillating airfoil was first tackled by Glauert, but was properly solved by Theodorsen[46].

Assume disturbance is small, for a thin 2D harmonically oscillating airfoil in inviscid, incompressible flow, the unsteady lift L and pitching moment M contains both non-circulatory and circulatory terms which are given by,

$$L = \pi \rho b^2 (\ddot{h} + V \dot{\alpha} - ba \ddot{\alpha}) + 2\pi \rho V b C(k) [\dot{h} + V \alpha + b(\frac{1}{2} - a) \dot{\alpha}]$$

$$M_{1/2} = -\pi \rho b^2 [(\frac{1}{2} - a) V b \dot{\alpha} + b^2 (\frac{1}{8} + a^2) \ddot{\alpha} - ab \ddot{h}] + 2\pi \rho V b^2 (a + \frac{1}{2}) C(k) [\dot{h} + V \alpha + b \dot{\alpha} (\frac{1}{2} - a)]$$

Equation 3-16

Where a is the pitch axis location relative to the mid-chord of the airfoil and is measured in terms of the semi-chord b , α is angle of attack, h is vertical displacement relative to neutral position.

The 1st term of L and M is non-circulatory, depending upon the accelerations of the airfoil. The 2nd term of L and M is the circulatory part with the effect of shed vortices multiplied by $C(k)$.

Theodorsen's function $C(k)$ is a complex valued function of reduced frequency k , $k = (\omega \times cref)/(2 \times TAS)$, where ω is the cyclic frequency, $cref$ is the reference chord length and TAS is the dimensional air velocity. $C(k)$ is expressed in terms of Hankel functions H , where

$$C(k) = F(k) + iG(k) = \frac{H_1^{(2)}(k)}{H_1^{(2)}(k) + iH_0^{(2)}(k)}$$

Equation 3-17

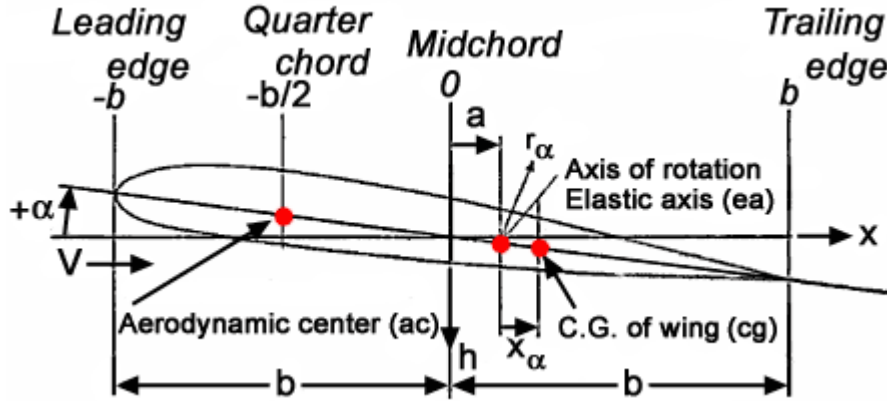


Figure 3-9 Airfoil definition for unsteady aerodynamic analysis

The Hankel function is defined as $H_v = J_v - iY_v$, with J_v and Y_v being Bessel functions of the first and second kind, respectively. Each Bessel function has an argument k , and then the real part and imaginary part of $C(k)$ can be written as:

$$F(k) = \frac{J_1(J_1 + Y_0) + Y_1(Y_1 - J_0)}{(J_1 + Y_0)^2 + (J_0 - Y_1)^2}$$

$$G(k) = \frac{Y_1Y_0 + J_1J_0}{(J_1 + Y_0)^2 + (J_0 - Y_1)^2}$$

Equation 3-18

The amplitude and phase of Theodorsen' function are given by:

$$|C(k)| = \sqrt{F^2 + G^2} \quad \phi = \tan^{-1}\left(\frac{G}{F}\right)$$

Equation 3-19

The complex number in the $C(k)$ term physically represent a delay of lift due to circulation building up process. In Theodorsen's theory, the lift-curve slope = $2\pi = 6.2831$ as it assumes flat plate, but to represent NACA 0006 and NACA 0015 airfoil used in this study, the lift-curve slope was 4.3285 from panel method results. A

correction factor of 68.89% was added to the Theodorsen's theory to modify the lift-curve slope so it represents the airfoil instead of a flat plate.

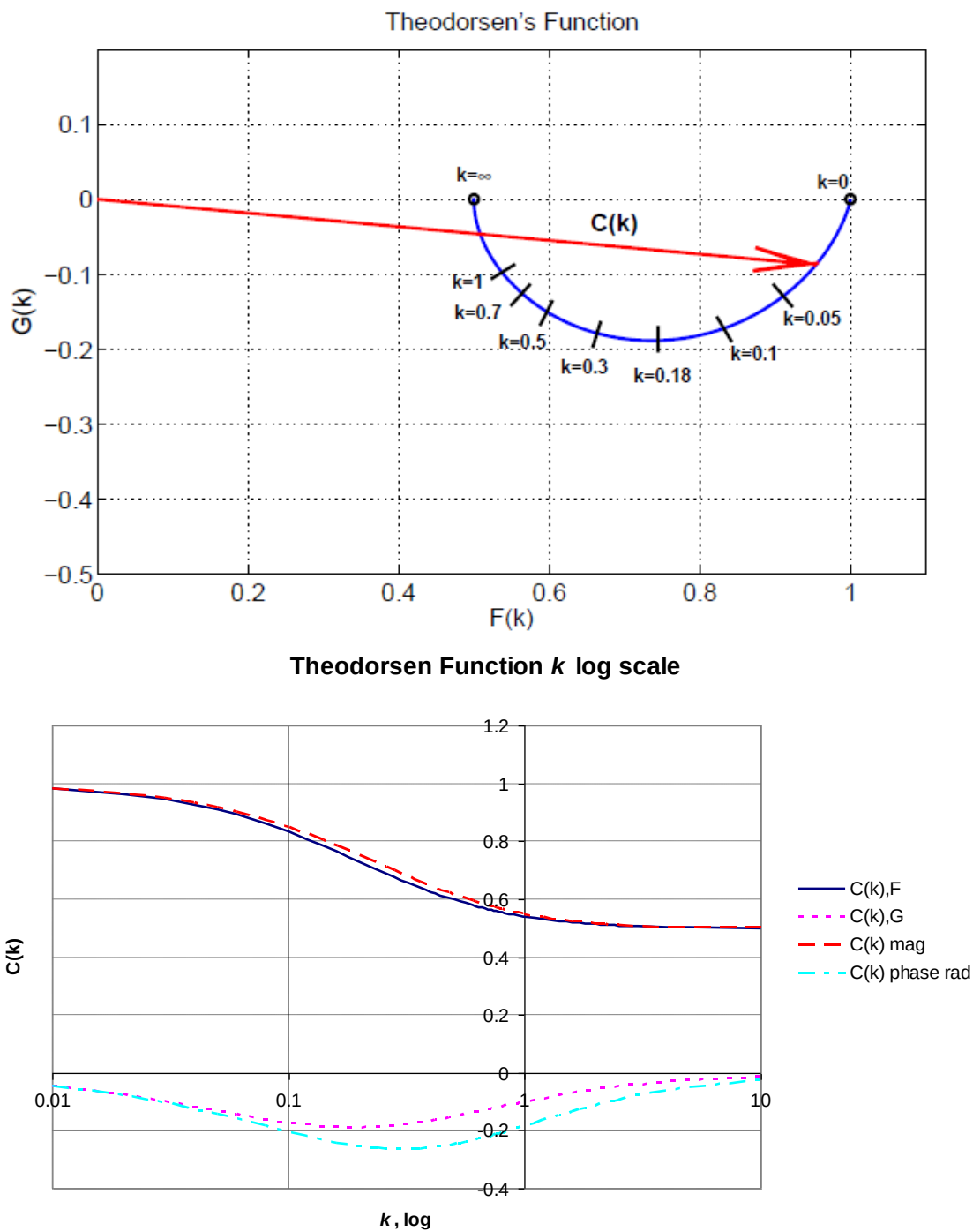


Figure 3-10 Theodorsen function graphic representation

3.4.2 Strip theory aerodynamics for 3D wing

Strip theory, also known as blade element theory, involves dividing an aircraft's wing geometry into discrete stream-wise segments and computing aerodynamic forces and moments on those segments based on their local velocities.

Forces and moments are then summed across all segments to arrive at the total force on the aircraft's centre of gravity (CG) and the total moment about it.

This method of modelling an aircraft is general enough that users can specify main wings, stabilizer surfaces, and even rotary wings in a similar manner. Thus, aircraft designers can employ the strip theory flight model as a rapid development and behavioural estimation tool for a variety of aircraft geometries.

3.4.3 Vortex Lattice Method for 3D wing

Tornado VLM is a 3D-vortex lattice method computer program written in Matlab. It was jointly developed by Dr. Tomas Melin and Adrien Berard, University of Bristol. [47][48]

Theory

Tornado is based on standard vortex lattice theory, stemming from potential flow theory. The wake coming off the trailing edge of every lifting surface is flexible and changes shape according to the flight condition. The classical "horse-shoe" arrangement of other vortex-lattice programs has been replaced with a "vortex-sling" arrangement. It basically works in the same way as the "horseshoe" procedure with the exception that the legs of the shoe are flexible and consist of seven (instead of three) vortices of equal strength.

Application

Tornado is meant to be used primarily in the conceptual design stage of an aircraft project or in training and education. Tornado supports multi-wing designs with swept, tapered, cambered, twisted and cranked wings with or without dihedral.

Any number of wings may be utilised as well as any number of control surfaces. Canards, flaps, ailerons, elevators and rudders may be modelled. Winglets, fences and engine mounts may also be incorporated in the design.

Limitations

One of the primary assumptions in the vortex lattice theory is the small angle of attack. Therefore, caution is advised when examining large angles as well as large rotational speeds.

Compressibility effects are neglected, as are thickness effects of the lifting surfaces.

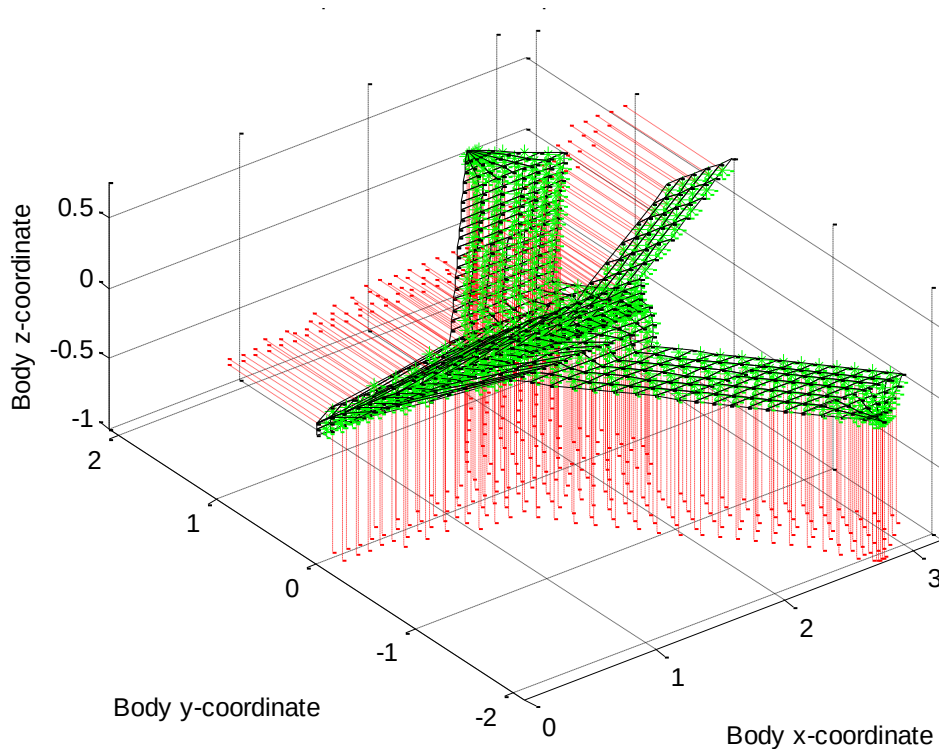


Figure 3-11 Tornado 3D vortex lattice panels representing D-47 aircraft

Other aerodynamic property estimation software were considered, examples are AVL and Digital DATCOM. However, Tornado is Matlab based tool which is easier for data transfer and manipulation within simulation iterations. Therefore it is more suitable for flight control design and flight simulation purpose.

3.5 Aeroelastic control reversal

When an aileron is rotated through ξ , it produces an additional δL_A and δM_A directly on a wing section and a subsequent wing twist $\delta\theta$, which results in a δL_W (could be opposite to δL_A) and δM_W (Figure 3-12).

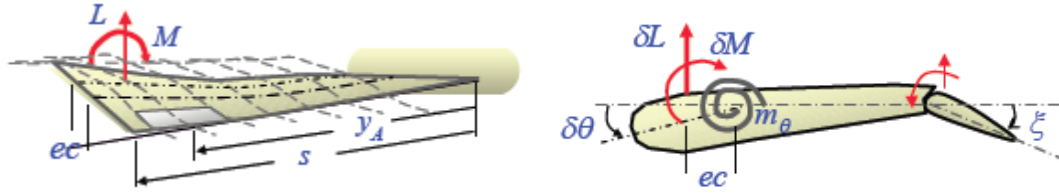


Figure 3-12 Illustration of aileron control reversal

Total lift and moment increments are

$$\begin{aligned}\delta L &= \delta L_W + \delta L_A = \frac{1}{2} \rho V^2 c (C_{L\alpha} \cdot \delta\theta + C_{L\xi} \cdot \xi) \\ \delta M &= \delta M_W + \delta M_A = \delta L \cdot ec + \frac{1}{2} \rho V^2 c^2 \cdot C_{m\xi} \cdot \xi\end{aligned}$$

Equation 3-20

Where, ρ is air density, V is velocity, c is chord length, e is distance between aerodynamic centre and structural elastic axis in ratio to chord, CL_α is lift gradient with respect to AOA, CL_ξ is lift gradient due to aileron and Cm_ξ is pitching moment gradient due to aileron deflection.

Since $\delta M = \delta\theta \cdot \frac{GJ}{0.7s} = \delta\theta \cdot m_\theta$ (s is semi-span, linear mode assumes pressure centre of aileron is at 70% of semi-span, m_θ is torsional rigidity from wing root to 70% of semi-span), then $\delta\theta = \frac{1}{m_\theta} (\delta L \cdot ec + \frac{1}{2} \rho V^2 c^2 \cdot C_{m\xi} \cdot \xi)$, substitute into above force equation,

$$\frac{\delta L}{\xi} = \frac{1}{2} \rho V^2 c \cdot C_{L\xi} \left(\frac{m_\theta + \frac{1}{2} \rho V^2 c^2 \frac{C_{L\alpha} \cdot C_{m\xi}}{C_{L\xi}}}{m_\theta - \frac{1}{2} \rho V^2 c^2 e \cdot C_{L\alpha}} \right).$$

Because wing divergence velocity $V_{DIV} = \sqrt{\frac{2m_\theta}{\rho c^2 e \cdot CL_\alpha}}$ and control reversal velocity

$$V_R = \sqrt{\frac{2m_\theta \cdot CL_\xi}{\rho c^2 \cdot CL_\alpha \cdot Cm_\xi}}$$

Lift change due to aileron deflection can be simplified as:

$$\frac{\delta L}{\xi} = \frac{1}{2} \rho V^2 c \cdot CL_\xi \left[\frac{1 - (V/V_R)^2}{1 - (V/V_{DIV})^2} \right]$$

Equation 3-21

Lift coefficient due to aileron deflection $CL_\xi \left[\frac{1 - (V/V_R)^2}{1 - (V/V_{DIV})^2} \right]$ decreases to zero exponentially as flexible aircraft flies faster. However, control effectiveness refers to lift increment $\frac{\delta L}{\xi}$ and it exhibits an interesting feature. Figure 3-13 shows aileron control force increases from zero up to a peak value at about 80% of V_R , then quickly decreases until aileron effectiveness reaches zero at V_R .

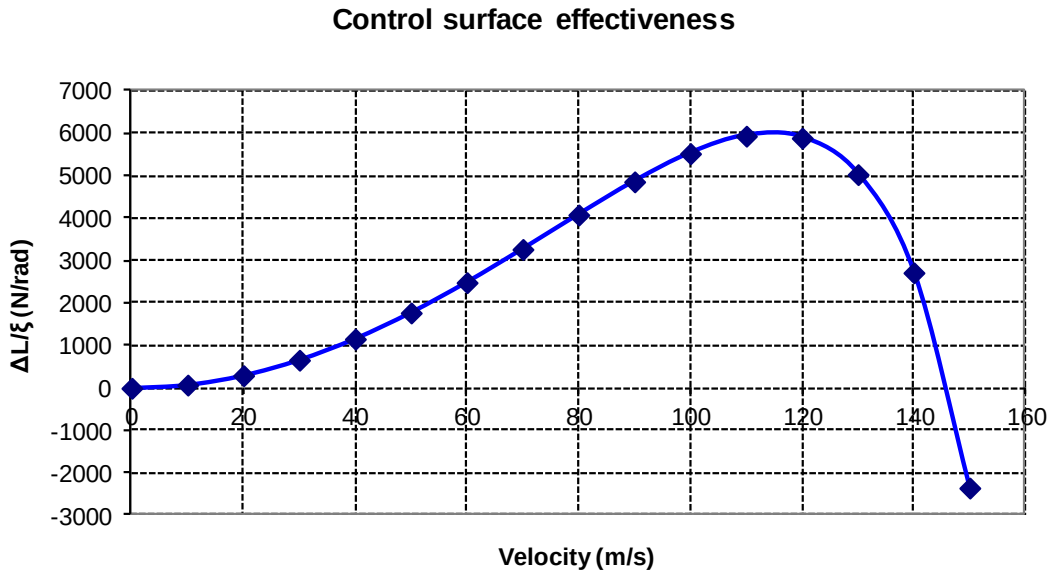


Figure 3-13 Control effectiveness vs. flight velocity

3.6 Summary

Up to now, sufficient theories and analysis methods were prepared for modelling of a flexible aircraft. Following chapters will be focused on designing an innovative seamless morphing flight control surface. Then apply it to a modified unmanned aircraft, which is purposely designed with a pair of very flexible, high swept-back-angle wings. These wings will be easily deformed under aerodynamic loading and demonstrate unacceptable roll control reversal problem at relatively low speed. Simulation of this flexible wing aircraft in flight will quantify the loss of roll control power and reduction of longitudinal stability due to aeroelastic wing twist. This research will attempt to use seamless morphing control surface together with an active control system to provide a practical solution to this engineering problem.

Since the seamless morphing control surface design and active control system will be used collusively to solve this aeroelastic problem, it is named as Seamless Active Aeroelastic Wing Technology (SAAWT), and the wing designed with this technology is known as Seamless Aeroelastic Wing (SAW).

Chapter 4 Seamless Aeroelastic Wing (SAW) Design, Modelling and Analysis

The concept of seamless aeroelastic wing is the first step towards the bird-like morphing wing design. It features variable camber airfoil constructed by seamless elastic skin and internal actuation mechanism. The increased wing flexibility would enable actively controlled wing twist to manipulate lift distribution for aircraft flight control.

This chapter will review the SAW design concept first, then apply the design to the example aircraft (D-47) described in the previous chapter. Numerical modelling and analysis of the SAW aeroelastic behaviour is presented in the end.

4.1 SAW Design Concept

4.1.1 SAW airfoil morphing shape

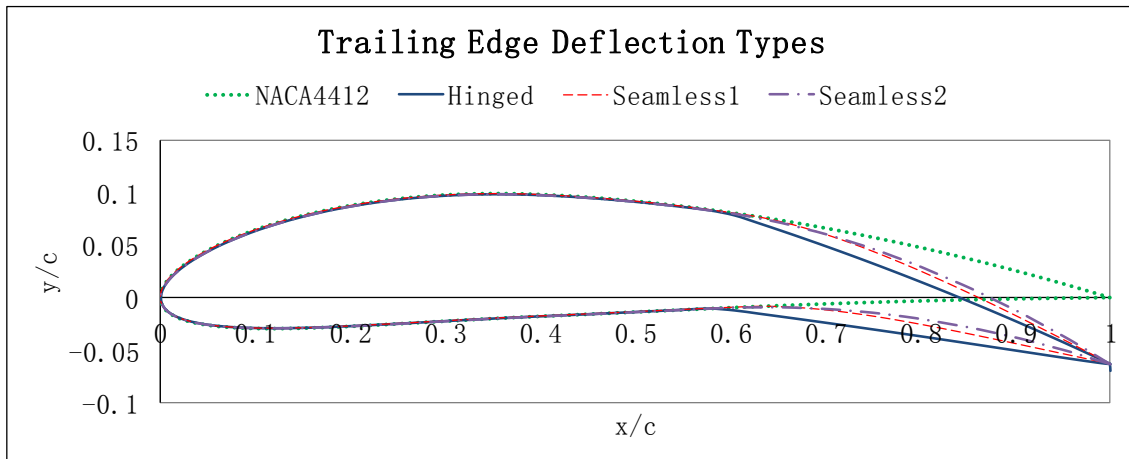


Figure 4-1 NACA 4412 morphing from 60%c and TE drop down 6.3%c

The SAW morphing is defined by the change of airfoil camber curvature for a certain proportion of the chord. For example NACA 4412 airfoil with the morphing shape starting from 60% of chord. Figure 4-1 shows the trailing edge (TE) angle of 9.1 degree.

Since there is no distinctive hinge line on the airfoil to measure the bending angle, it is more convenient to use TE vertical displacement in the percentage of chord to measure the morphing. In this example, the TE deflection is 6.3% of the chord.

The SAW design was initially focused on the morphed curvature of camber line. There are different ways to define the curvature, e.g. parabola, semi-arc, various order of polynomial, segmented beam and triangular load beam curve. The curvature design is aimed for aerodynamic benefits such as higher lift drag ratio and larger pitching moment for control, etc. Three deflection types were considered in this study. They are conventional hinged TE deflection, seamless deflection curvature option 1 and seamless deflection curvature option 2. Option 1 is triangular load beam curve. Option 2 is segmented beam curve. Their design principles were shown in Figure 4-2. Detailed TE curvature is show in Figure 4-3.

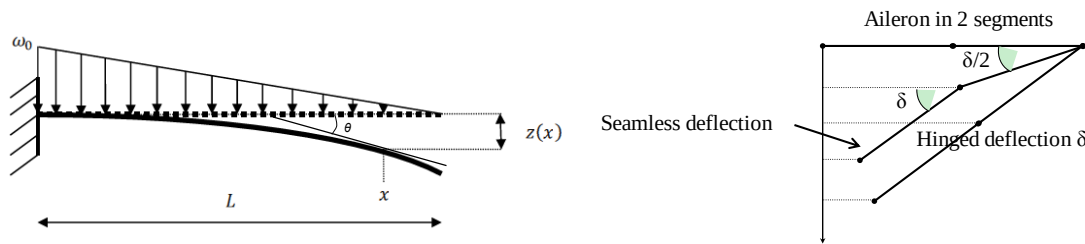


Figure 4-2 TE Morphing curvature design option 1 (left) and option 2 (right)

Visual difference between curvature option 1 and 2 maybe small but its impact on aerodynamic property is significant. Figure 4-4 shows pressure distribution of NACA4412 airfoil with hinged and two kinds of seamless control surfaces. Table 4-1 presented aerodynamic coefficients of plain NACA4412 airfoil and all three TE deflection types.

These results, calculated by 2D aerodynamic software Javafoil, indicated that seamless control surface can generate more lift compared to conventional design. However it also produces more drag and larger pitch down moment. Seamless curvature option 2 is more effective on both positive and negative accounts.

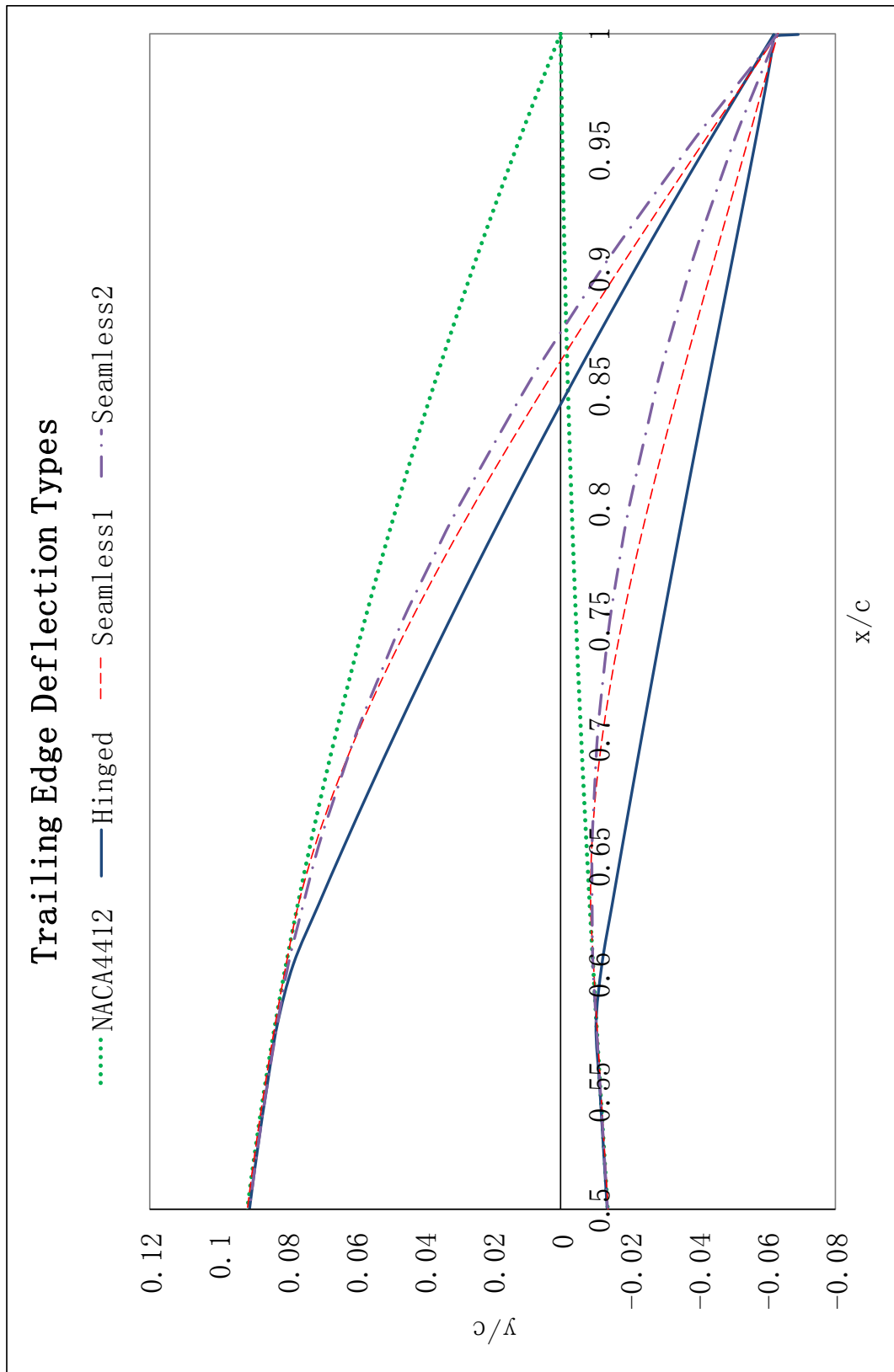


Figure 4-3 TE deflection curvature comparison

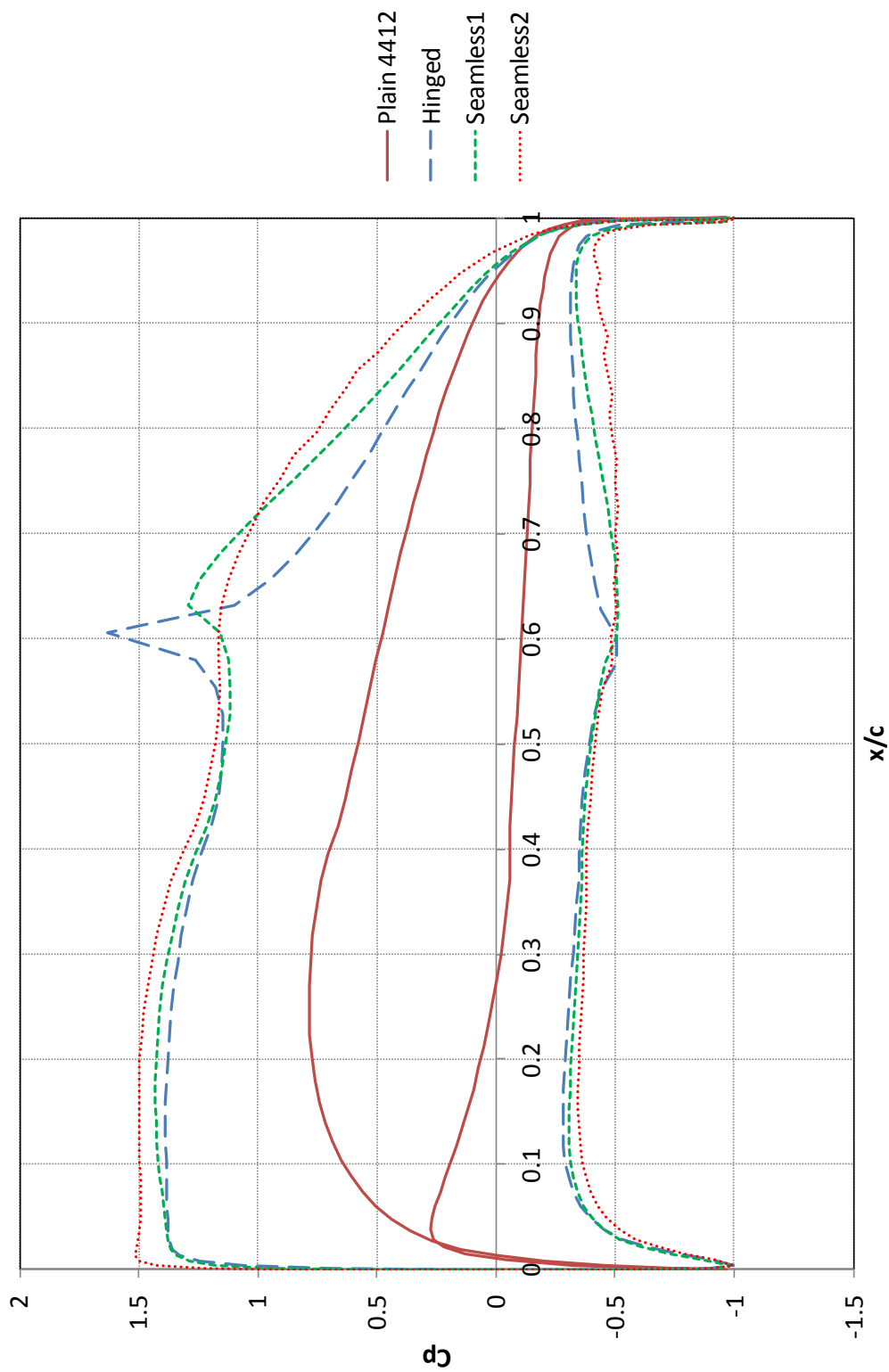


Figure 4-4 Pressure coefficients of a wing with hinged and seamless control surface deflect by 6.3% of chord

Table 4-1 Aerodynamic contribution of TE deflection of different types

<i>Aerodynamic Effect of TE Deflection 6.3%c</i>				
Coeff	NACA4412	Hinged	Seamless1	Seamless2
C_L	0.5185	1.3450	1.4253	1.5414
C_D	0.0089	0.0093	0.0135	0.0142
$C_{M0.25}$	-0.1107	-0.2136	-0.2475	-0.2814

4.1.2 SAW airfoil morphing mechanism

Based on the concept initiated in the DARPA smart wing project as well as US patent 4286761 [37], the design uses an eccentric curved beam to convert a rotational input motion into a vertical displacement along the beam. The required trailing edge (TE) camber shape is determined by the beam curvature at all rotating angles. The output end of the curved beam is in contact with the upper and lower skins of the TE which can be deflected up or down depending on the direction of the input rotation. (Figure 4-5)

A key design feature is that disks of different sizes mounted along the beam where their edges are in contact with the upper and lower skin stiffeners. The disks together with the beam work as an active rib to maintain the TE chordwise shape and force it to deform as the beam is rotated. The actuator torque is distributed to the disks and converted into compressive forces acting on the stiffeners to deflect the TE section.

One interesting design problem is that disk edges and skin stiffeners are moving in opposite direction during TE deflection (Figure 4-6). The disks have to be mounted at different angles to the beam axis in order to allow the disk edges move in a straight track spanwise and minimise the chordwise sliding motion and friction between the disk edge and the stiffener.

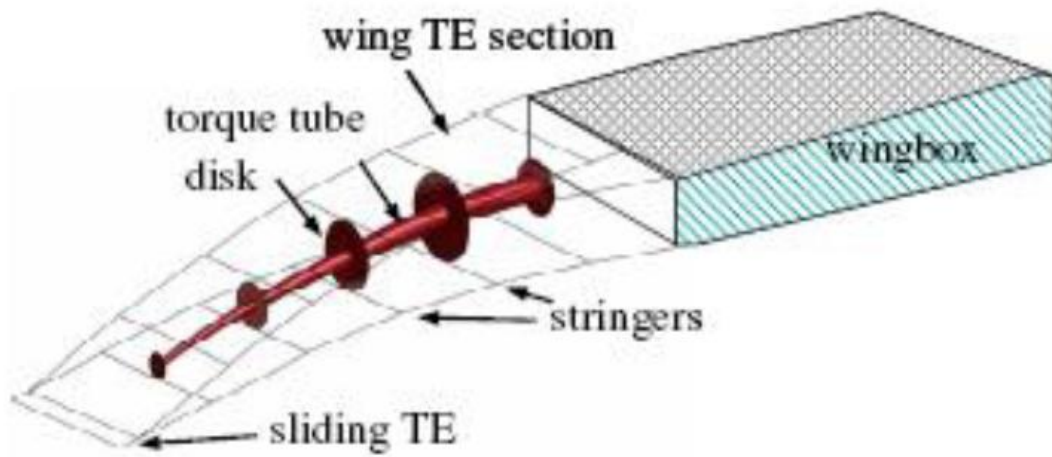
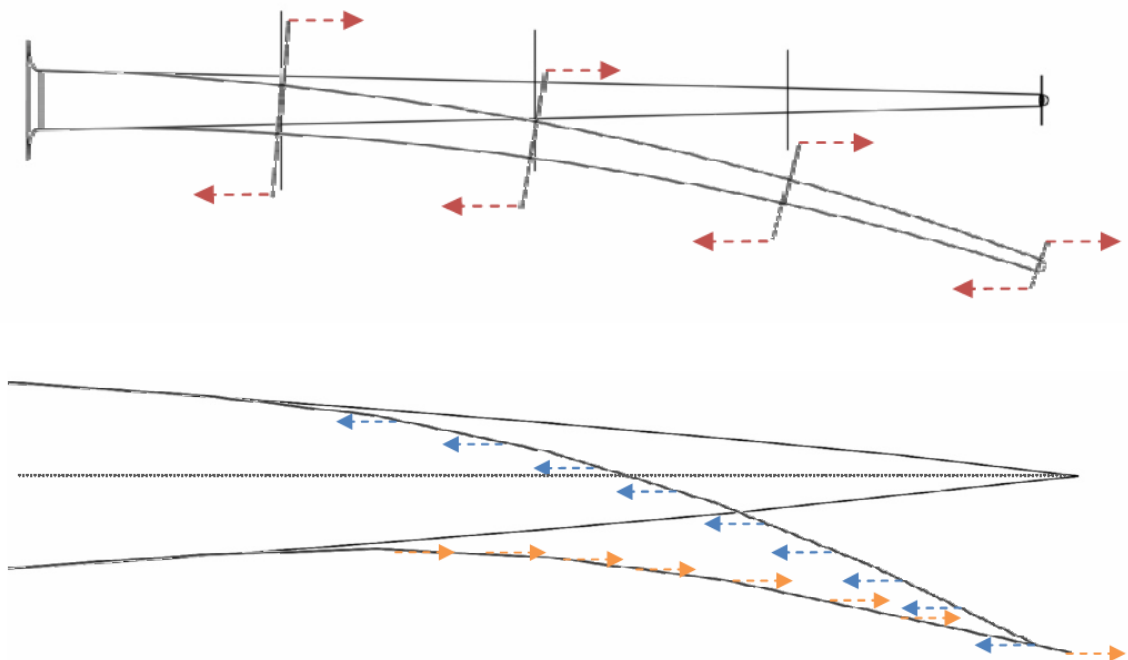


Figure 4-5 A SAW model with seamless TE actuation mechanism



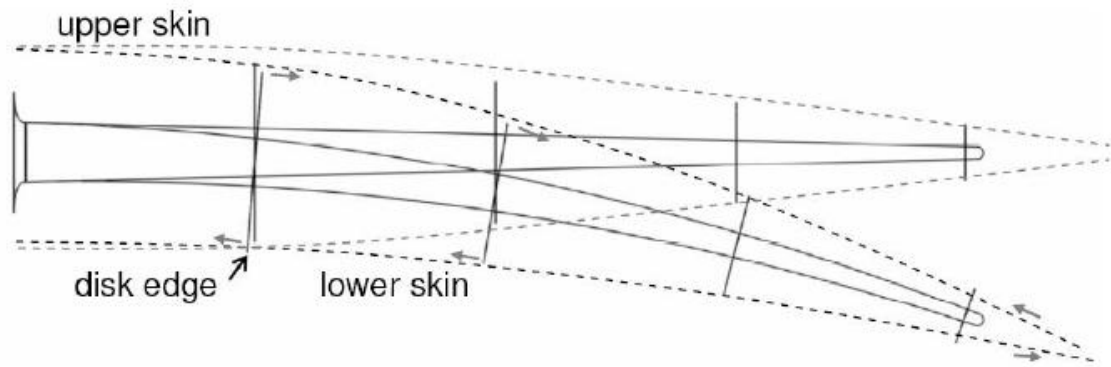
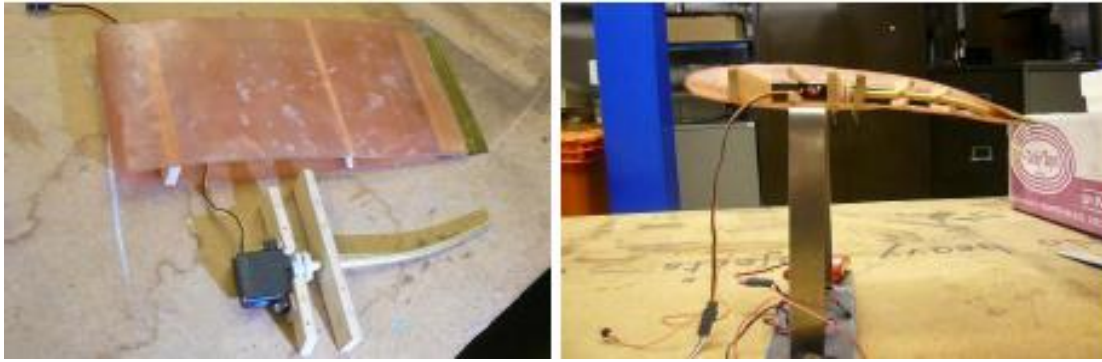


Figure 4-6 Disk edges and skin stiffeners motion during TE deflection

4.1.3 SAW concept demonstration model

To demonstrate the SAW design concept, a scale model was built together with another Cranfield PhD student. The shared results were presented below.

NACA0015 airfoil cross-section is 100 mm in span and 200 mm in chord. Skin is composed of four layers of polyester glass fibre pre-preg in 0/90 layup. Spar and the ribs are made of light wood. The positioning of the front spar is at 20% chord and the rear spar at 50% with a mid spar at 40% of chord. A servo motor installed in the mid-spar produce 0.37 Nm of torque at 4.8V. Stiffeners were attached above and below the motor to reinforce the mounting against the skin reaction torque. Two pairs of stiffeners were placed at 60% and 90% chord after the rear spar to strengthen the skin and support the disc-skin contact. A curved torque tube went through a bearing, connected to the motor at one end and supported by the rear spar at the other end. The tube without the disks contacts the stiffeners directly in the flexible TE section. These stiffeners attach to the rotating tube to transfer the actuating load to the skin. The required TE camber shape will follow the tube curvature at all rotating angles. Figure 4-7 (a) and (b) shows the model before assembly and when deformed by the internal actuation mechanism in place respectively. [49]



(a)

(b)

Figure 4-7 SAW model skin, spars motor and curved tube

A second model was constructed to carry out further structural and aerodynamic testing of the concept. The new model integrates the curved beam actuation mechanism, sliding TE and various sensors. The objectives of this test were:

- Verify the aerodynamic characteristics of the SAW design.
- Estimate stiffness contribution from actuator and actuation mechanism on the wing structure.
- Estimate power consumption of the actuation mechanism.

The skin made of glass fibre composite is shown in Figure 4-8(a). The wing model section is 600mm long and 300mm in chord. The skin was made of three layers of glass fibre polyester fabric in 0/45/90 degree layup. The front spar was made of wood positioned at 18% chord and the rear spar at 55% of chord from the leading edge. Two sets of Curved Torque Tube Actuation Mechanism (CTTAM) were made and mounted to the spars through ball bearings as shown in Figure 4-8(b). The two Hi-tech RS4549 digital servo motors were mounted to drive each of the curved torque tubes through 1 to 1 ratio gears independently. The tube's diameter was 5.56mm with two discs mounted along it. The material properties of the tube and discs were shown in Table 4-2. Each motor can produce a maximum torque of 0.85Nm operated at 6V. The actuation forces were delivered from the CTTAM to the stringers and deflect the skin up or down to achieve the designed TE shape. [49]

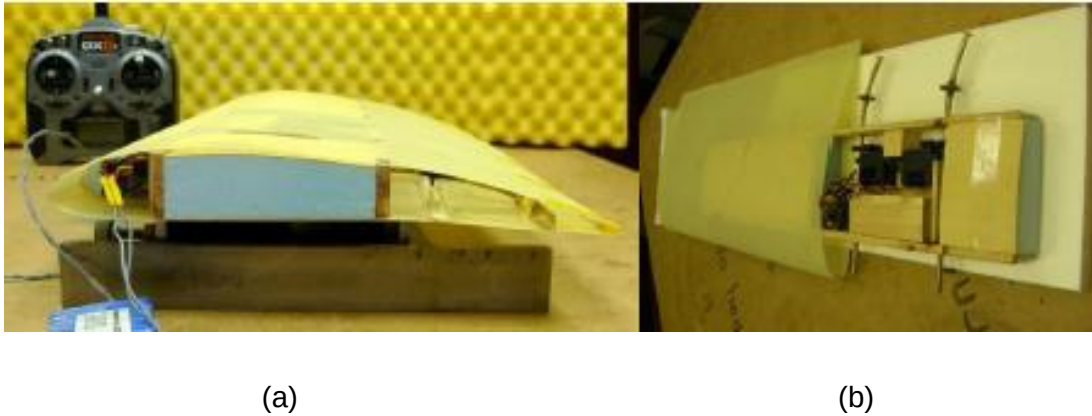


Figure 4-8 Assembled SAW model and CTTAM mounted inside SAW

Table 4-2 Material properties estimation of the torque tube and discs

	Elastic Modulus (GPa)	Poisson Ratio	Density (kg/m ³)
Beam	85	0.33	7250
Disc1	110	0.33	8600
Disc2	0.75	0.33	1100

CTTAM was integrated into the SAW as shown in Figure 4-8(a). The servomotor has very high stiffness (electromagnetic stiffness is close to infinity), the gears and the curved torque tubes connected to the motor maintains and modifies the airfoil shape, their function is analogous to active rib. The stiffness of CTTAM system is an important parameter in the SAW structure model for static and dynamic analysis. Aeroservoelastic analysis also needs actuation dynamics input. However this key parameter is not available due to the insufficient data from the servomotor manufacture. Experiment was therefore carried out to identify the stiffness of the CTTAM. [49]

Wind tunnel test was not carried out due to budget problem; no aerodynamic data were available to verify the SAW advantages except Java foil calculation results.

For stiffness estimation, a test rig was made to clamp the servomotor; a short aluminium alloy beam was connected to its output shaft as shown in Figure 4-9(b). The measured frequency of the model by performing a vibration test was $f=305\text{Hz}$. Since

the 1st bending mode frequency of the short beam alone clamped at one end was measured as 770Hz. The 305Hz frequency is identified as the torsional mode of the servomotor. The torsional stiffness of the motor is estimated to be 30.3Nm/rad. When the electric power was supplied to the servomotor, the resonant frequency increased to $f=373$ Hz as a result of an added larger effective stiffness from the motor.[49]

For power consumption estimation, no test was carried out but an analytical assessment was made, results were presented in the next section.

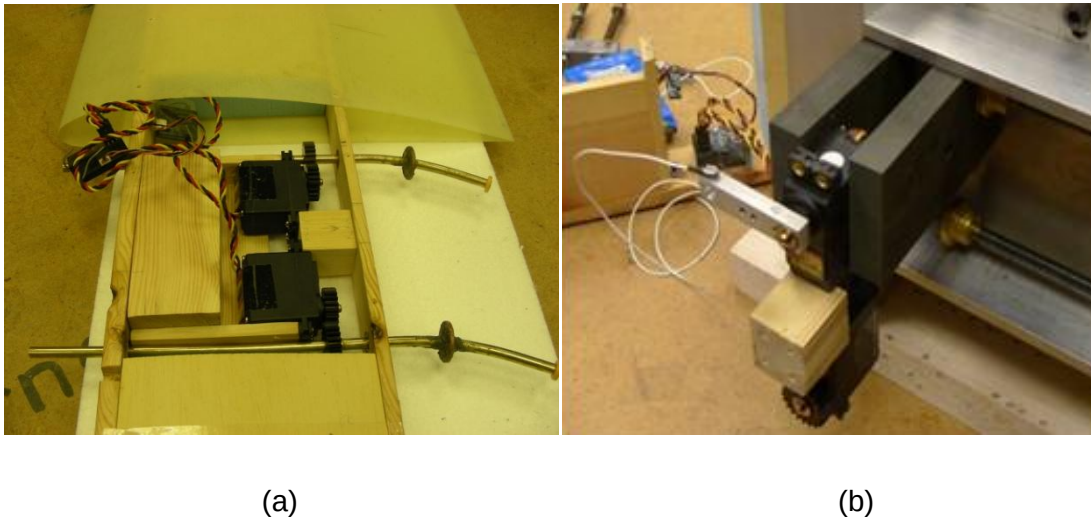


Figure 4-9 Servomotor test setup

4.1.4 SAW morphing actuation analysis

The actuator would act against the loadings from both aerodynamic and structural elastic forces through the actuation mechanism installed in the SAW control section.

Aerodynamic load was calculated by Java foil, lift force and bending moment of the TE from 60% of chord is shown in Figure 4-10.

Wing skin bending load was calculated by a method derived from beam bending theory. Since the experimental model has two disks on CTTAM, skin deformation was assumed to be a flat beam deflected by two different loads (F_1 and F_2) applied at different locations (L_1 and L_2) on the beam, the deflection contribution of each load (D_1 and D_2) were expressed as:

$$\begin{cases} \frac{F_1 L_1^3}{3EI} + \frac{F_2 L_1^2}{6EI} (3L_2 - L_1) = D_1 \\ (L_2 - L_1) \times \tan\left(\frac{F_1 L_1^2}{2EI}\right) + \frac{F_2 L_2^3}{3EI} = D_2 \end{cases} \begin{cases} y_{11} + y_{21} = D_1 \\ y_{12} + y_{22} = D_2 \end{cases}$$

Equation 4-1

Where EI is the bending stiffness, all other variables' definitions were shown in Figure 4-11.

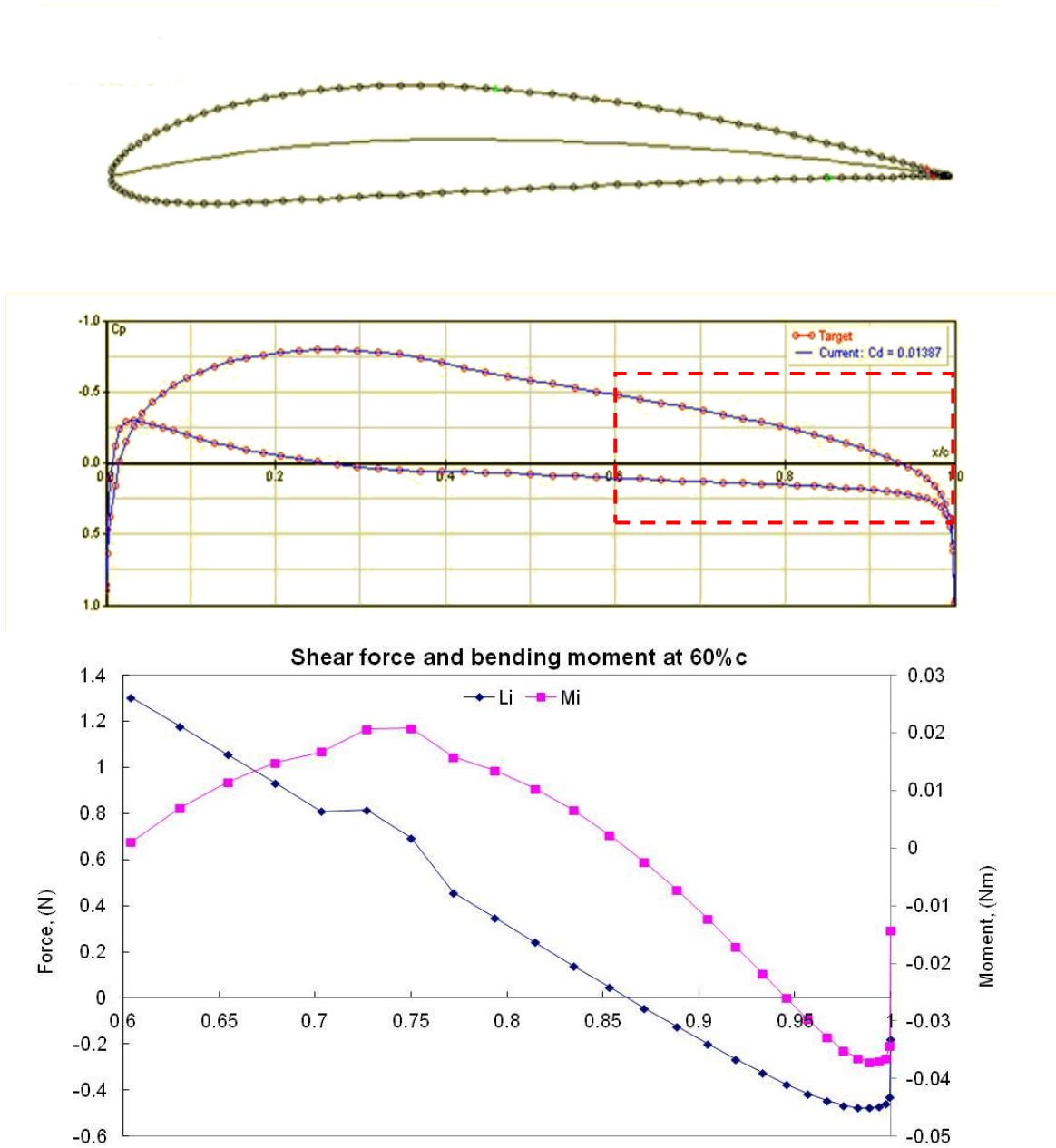


Figure 4-10 Javafoil analysis of seamless airfoil TE aerodynamic loading

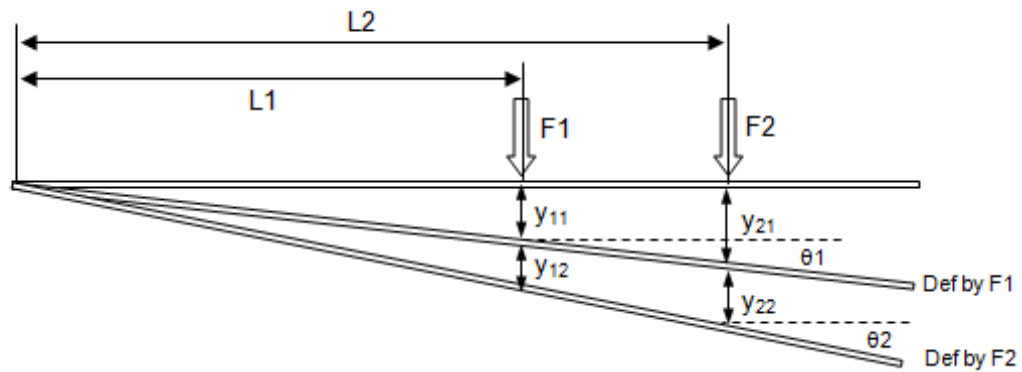


Figure 4-11 Skin bending load estimation variables

Due to the rotation actuation of the beam, chord wise loading was only taken by the actuator when the moment arm is not zero. To calculate the moment applied on the actuator, the curved tube's position has to be determined in order to calculate the moment arm that transfer the aero-structural loading. At 0 degree rotation, the tube has to follow the camber line hence it is not horizontal (see Figure 4-12). When actuator rotated 90 degree, the tube tip has passed the lowest position and the moment arm becomes negative value, which means required actuation torque is reversed (see Figure 4-13). Actuator torque requirement is calculated and presented in Table 4-3 and Table 4-4.

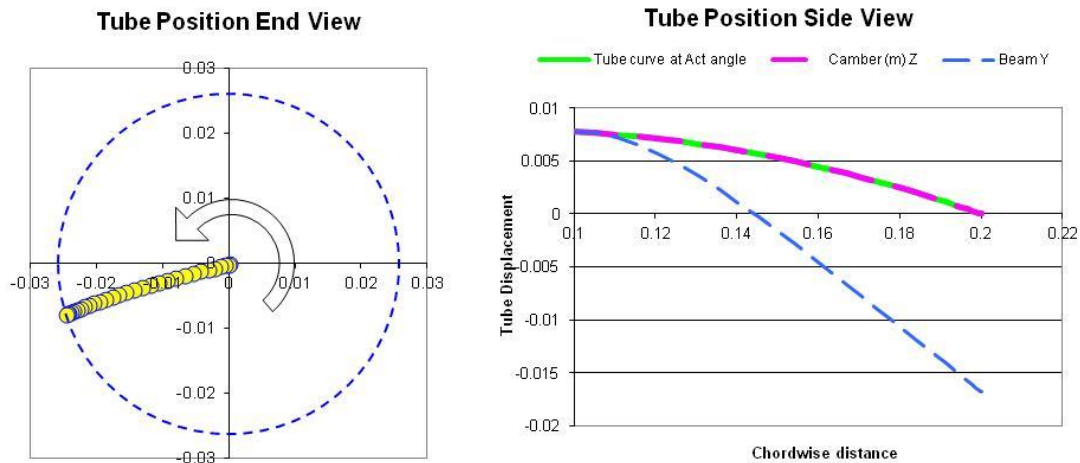


Figure 4-12 Curved tube position when actuator is neutral (Rotation angle = 0 degree).

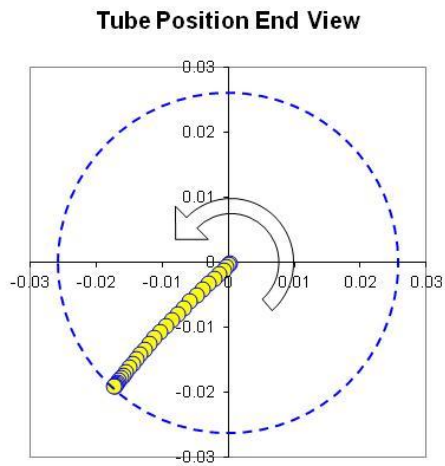
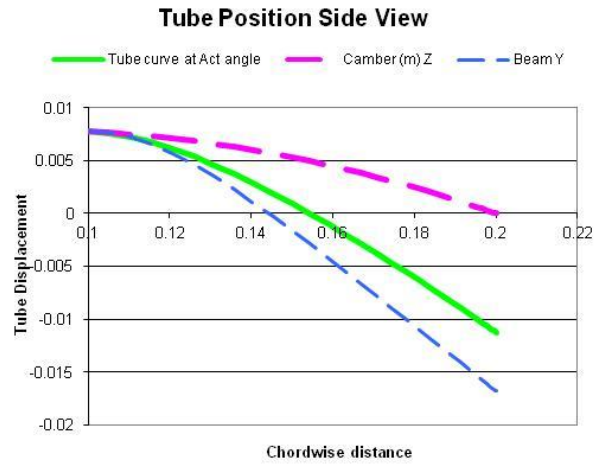
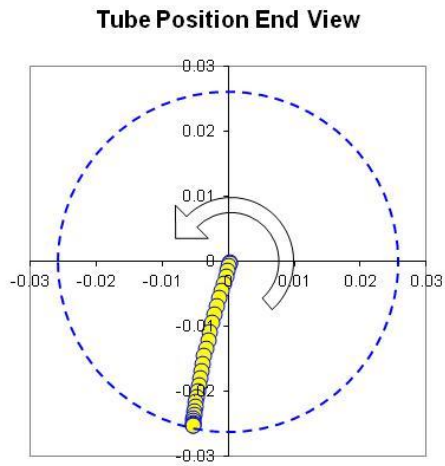
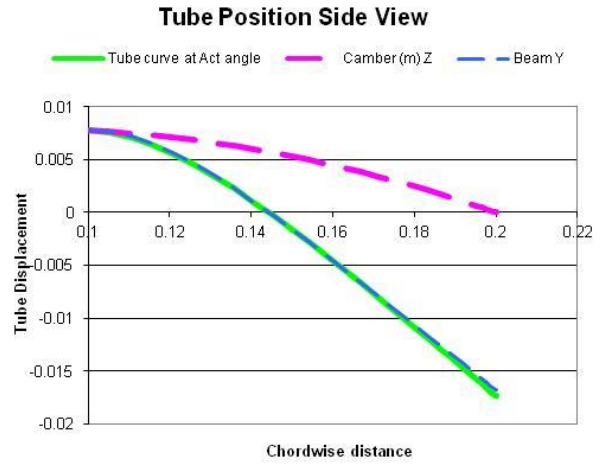
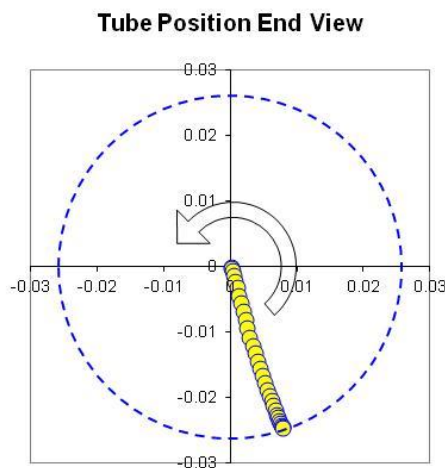
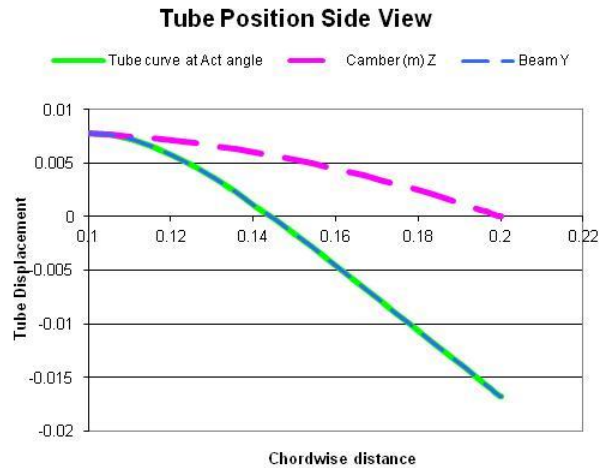


Figure 4-13 Curved tube position at 30, 60 and 90 degree actuator rotation

Table 4-3 Skin bending load on SAW actuator

Angle (deg)	Skin Bending Load					Two skins (Nm)
	Load 1 (N)	Arm 1 (m)	Load 2 (N)	Arm 2 (m)	Torque (Nm)	
0	0.0124	0.0068	0.0000	0.0187	0.0001	0.0002
10	5.2599	0.0064	0.0700	0.0175	0.0348	0.0696
20	10.0970	0.0058	0.1341	0.0157	0.0603	0.1207
30	14.3782	0.0050	0.1900	0.0135	0.0741	0.1483
40	17.9719	0.0040	0.2362	0.0108	0.0751	0.1502
50	20.7695	0.0030	0.2711	0.0079	0.0639	0.1278
60	22.6859	0.0018	0.2938	0.0047	0.0427	0.0854
70	23.6630	0.0006	0.3036	0.0013	0.0149	0.0298
80	23.6710	-0.0006	0.3001	-0.0021	-0.0152	-0.0303
90	22.7096	-0.0018	0.2835	-0.0054	-0.0429	-0.0859

Table 4-4 Aerodynamic load on SAW actuator

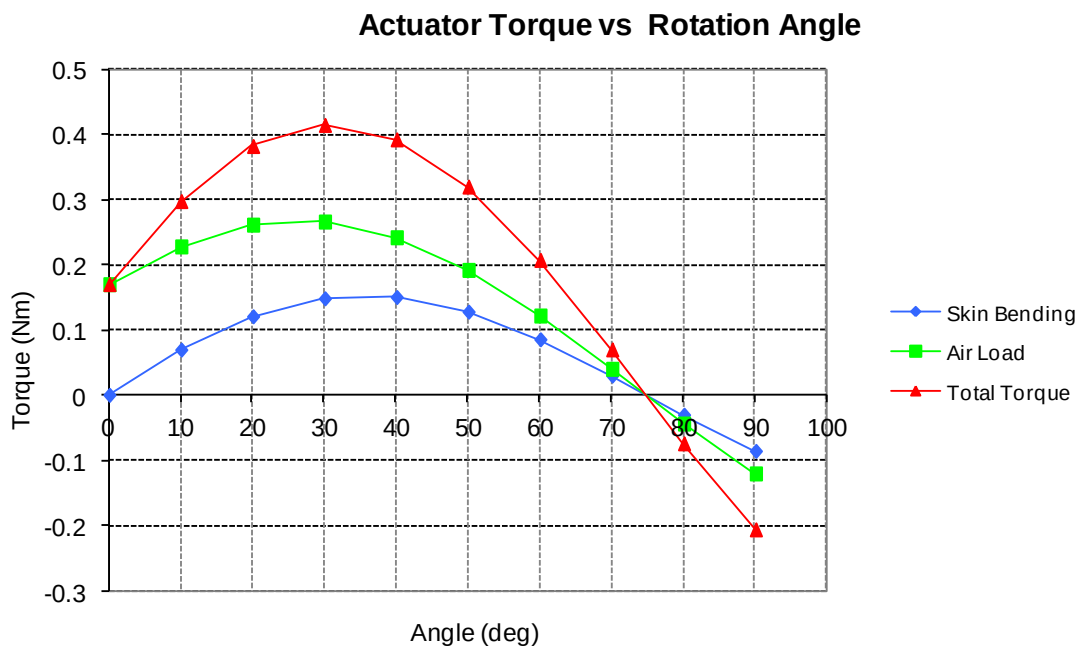
Angle (deg)	Aerodynamic Load		
	Load (N)	Arm (m)	Torque (Nm)
0	30.7298	0.0055	0.1700
10	43.9640	0.0052	0.2280
20	55.8238	0.0047	0.2613
30	65.9402	0.0040	0.2661
40	74.0646	0.0033	0.2420
50	79.9891	0.0024	0.1919
60	83.5762	0.0015	0.1219
70	84.7454	0.0005	0.0401
80	83.4674	-0.0005	-0.0439
90	79.7858	-0.0015	-0.1204

For the first SAW concept demonstration model shown in Figure 4-7, the actuation torque required at cruise speed 60m/s is shown in Figure 4-14 (a). As actuator has a maximum torque of 0.5 Nm should be adequate for the task.

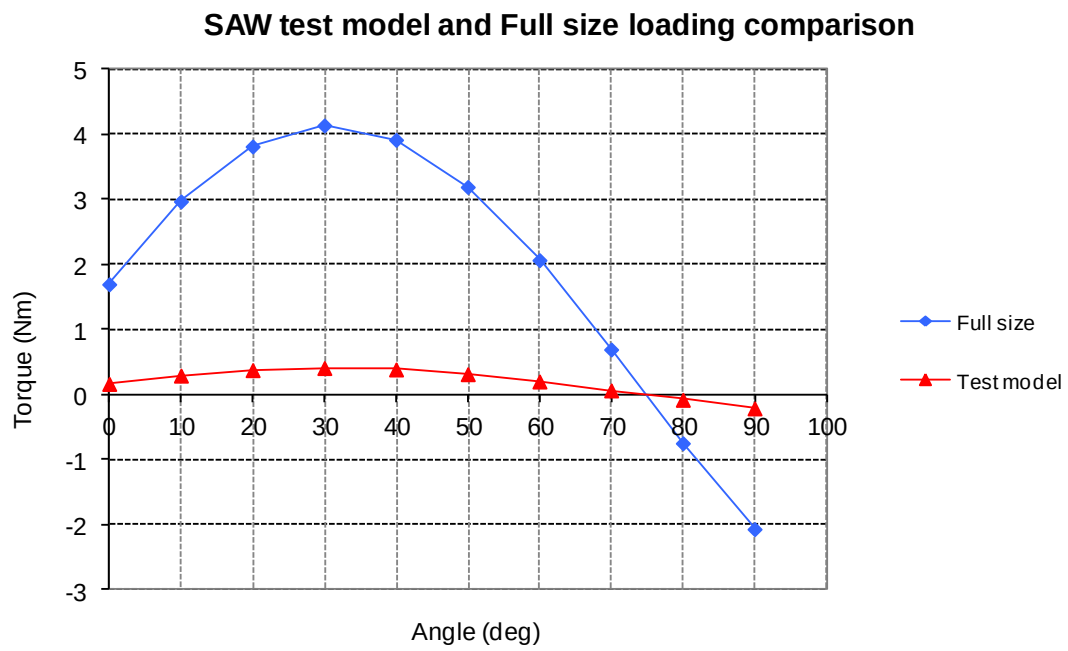
For the second test model shown in Figure 4-8, the SAW of 2m span and 0.6m chord has half of the span being seamless control surface. At dive speed of 90m/s, the loading is about 10 times of the above test model and the required actuation torque for the TE rotary position is shown in Figure 4-14 (b). The required actuation torque would be greater when the LE control is included.

Notice in Figure 4-14, x-axes were actuator rotation angles. In order to compare SAW with a conventional hinged control surface, it is more appropriate to use TE deflection angles at each actuator rotation. Figure 4-15 shows how the same torque result may appear differently with respect to TE deflection angle.

For a conventional hinged control surface, the required actuation torque would increase linearly with the TE deflection since the aerodynamic force generated torque acts at the hinge. Comparing with the hinged control design, the SAW actuation mechanism share the aerodynamic load between curved tube and actuator, it has the advantage of smaller overall torque demand especially at higher TE deflection positions as shown in Figure 4-16.



(a)



(b)

Figure 4-14 Required actuator torque vs. TE actuator rotation angle

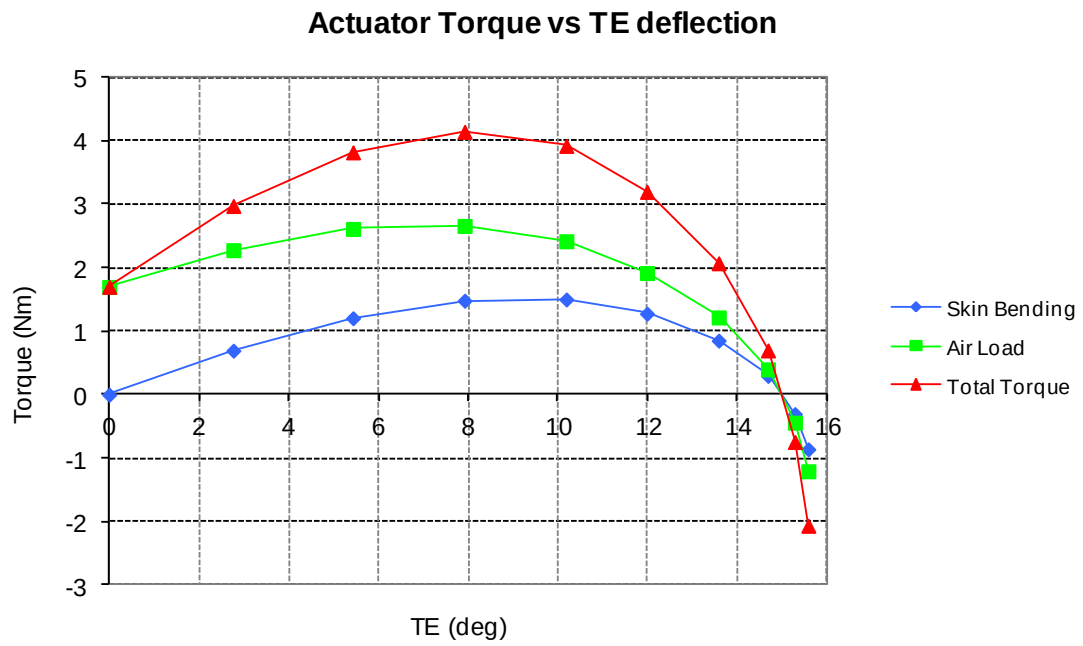


Figure 4-15 Required actuator torque vs. TE deflection angle

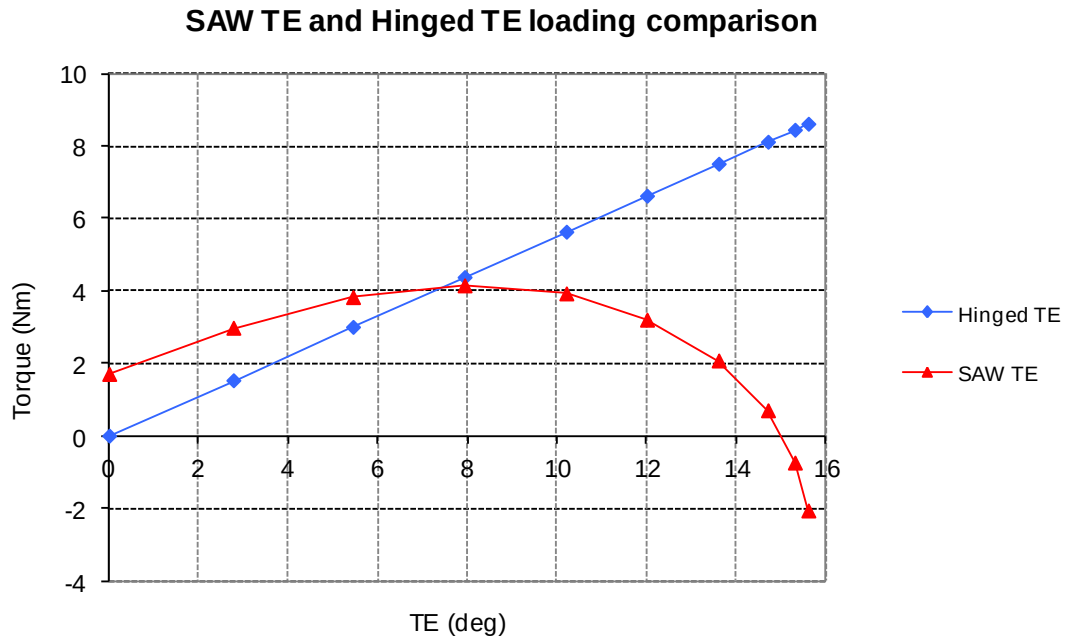


Figure 4-16 Torque comparison between the SAW and hinged control

4.2 Application of SAW design on D-47 aircraft

When the SAW concept is applied to the example aircraft (D-47), the wing structure details were designed according to wing spanwise loading. Numerical model was also constructed in Matlab Simulink environment to simulate SAW dynamic behaviour.

4.2.1 Morphing airfoil for D-47 Example

The symmetrical airfoil for the D-47 example has both leading edge (LE) and trailing edge (TE) control surfaces. This would make the aerodynamic features more complicated, morphing shape example NACA0009 is shown in Figure 4-17. Control derivatives such as C_L , C_D and C_M with respect to LE and TE deflection were calculated in Java foil, since Java foil cannot change LE deflection angle, each deflection case was given as individual airfoil input for the calculation. The aerodynamic results of hinged and seamless LE control surfaces were presented in Figure 4-18. TE results were presented in Figure 4-19. The deflection limits were -15 (down) and +15 (up) degree, 5 degree interval was chosen due to the concern of LE and TE combination

would increase the aerodynamic data dramatically (See Appendix C). Better result resolution was left to future CFD analysis.

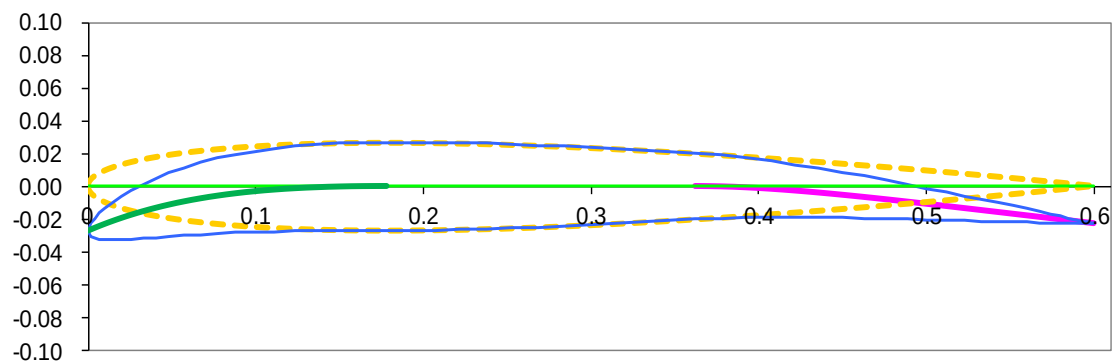
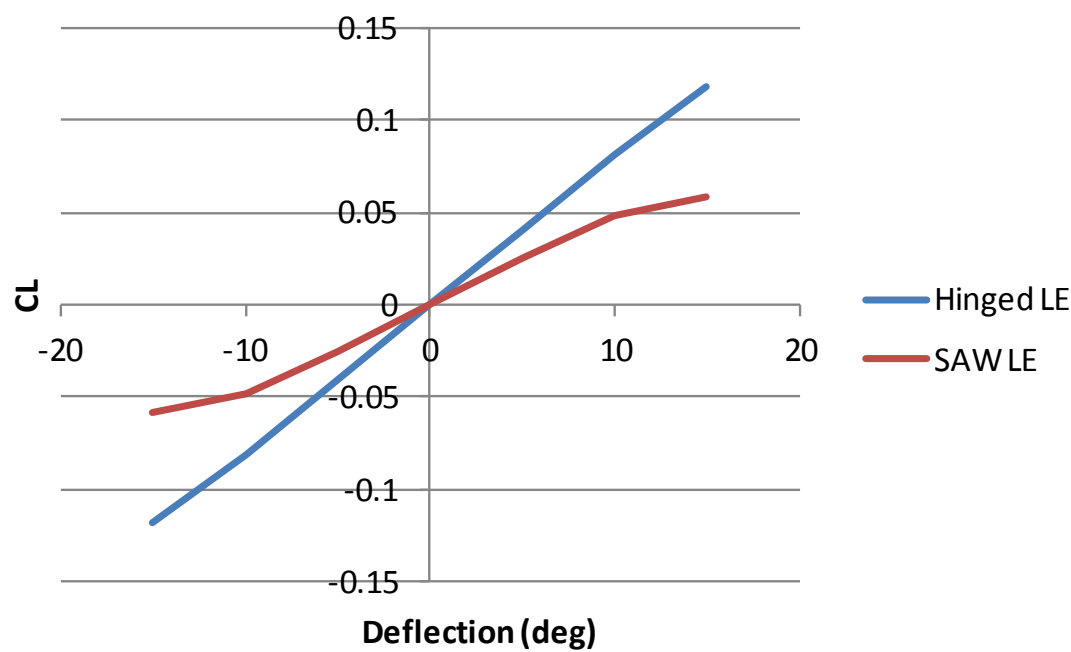


Figure 4-17 NACA 0009 morphing LE and TE.



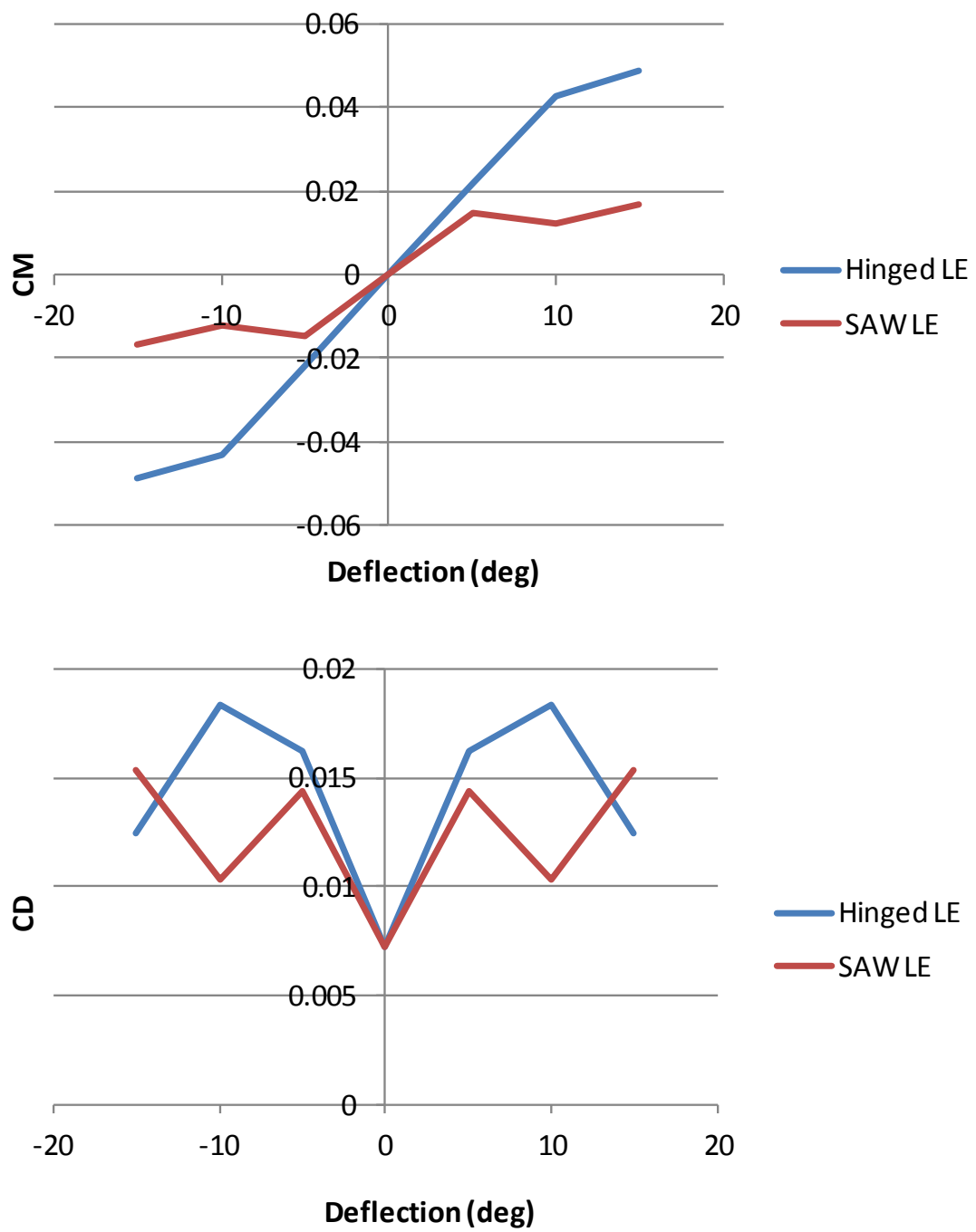
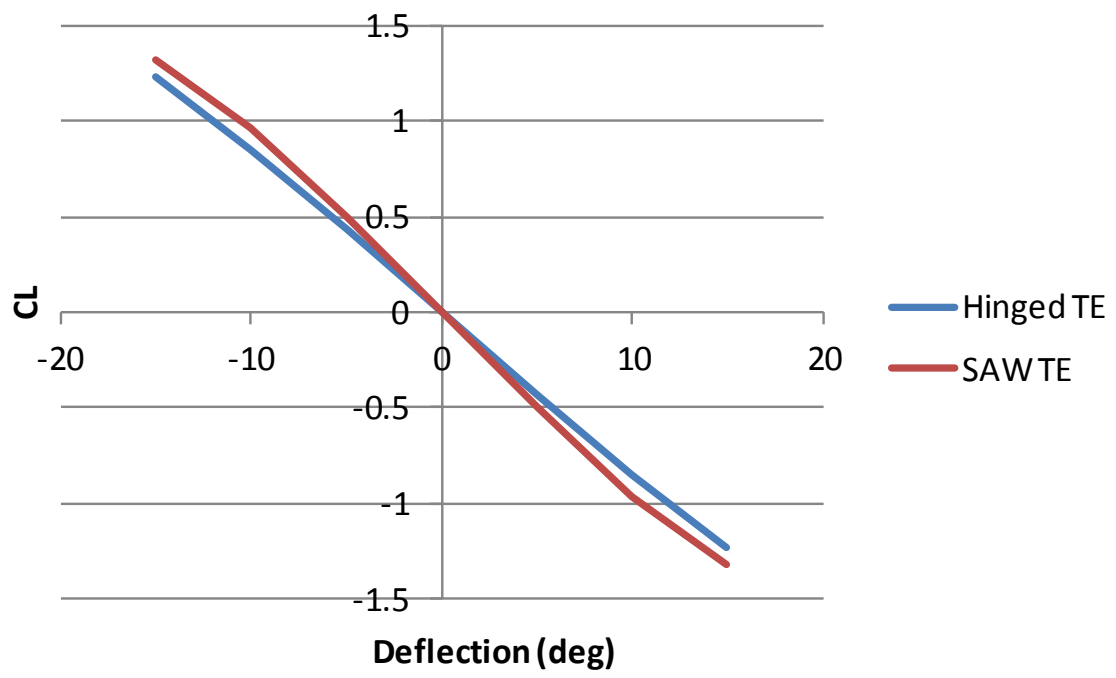
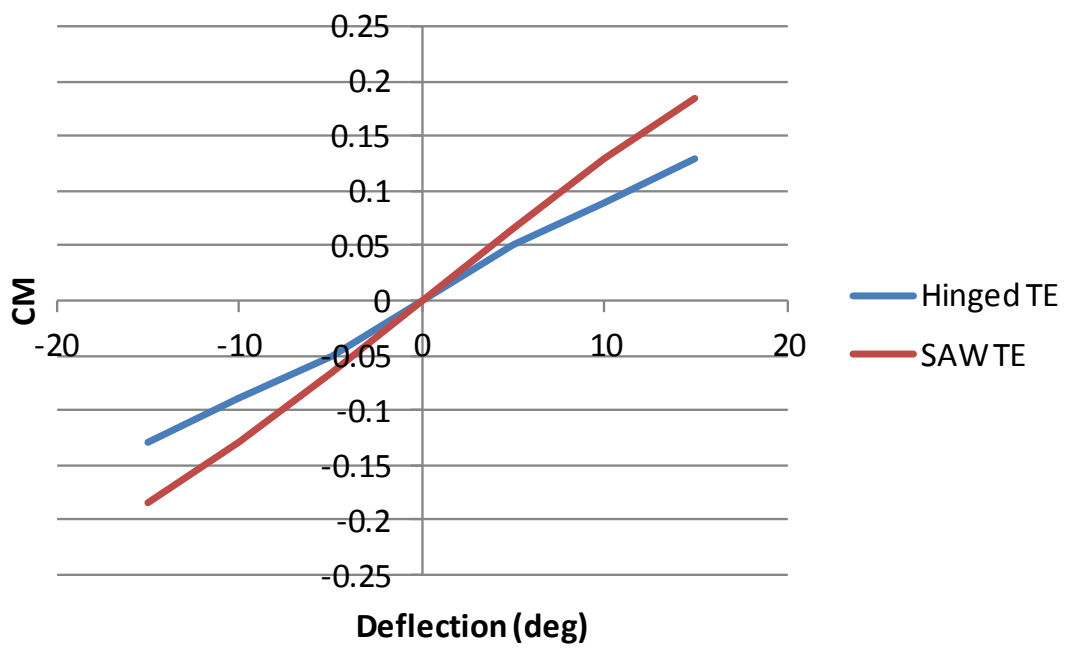


Figure 4-18 SAW airfoil hinged and seamless LE aerodynamic contribution



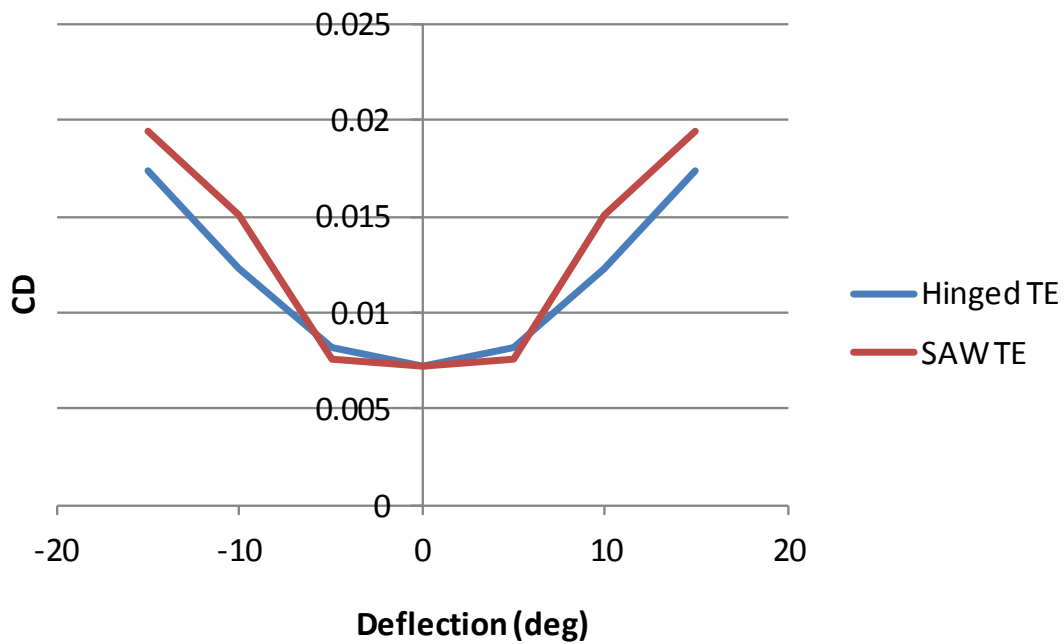


Figure 4-19 SAW airfoil hinged and seamless TE aerodynamic contribution

4.2.2 Static load from D-47 example rigid wing

Static loading on the wing comprises lift produced from the wing at level flight (1G) and the weight of the wing itself. Flight condition is 60m/s at AOA=1.8° and the total loading on the wing is 95.33N.

Spanwise lift distribution was calculated by Tornado VLM.

Lift on each node was calculated based on the lift distribution and spanwise strip width. Subtracted by the weight of each node, the loading on the semi-span wing was obtained as shown in Figure 4-20 and Table 4-5

Table 4-5 Semi-span loading derivation

<i>Nodes</i>	<i>Lift/m</i>	<i>Strip Width</i>	<i>Span Position</i>	<i>Lift (N)</i>	<i>Weight (N)</i>	<i>Load (N)</i>
1	232.236	0.2	0.5	46.447	-30.326	16.122
2	225.158	0.2	0.7	45.032	-32.646	12.386
3	222.339	0.2	0.9	44.468	-32.063	12.405
4	218.313	0.2	1.1	43.663	-30.871	12.792
5	212.132	0.2	1.3	42.426	-28.992	13.434
6	201.554	0.2	1.5	40.311	-26.303	14.008
7	173.243	0.2	1.7	34.649	-22.551	12.097
8	121.240	0.2	1.9	24.248	-19.071	5.177
9	57.988	0.1	2.1	5.799	-8.888	-3.089

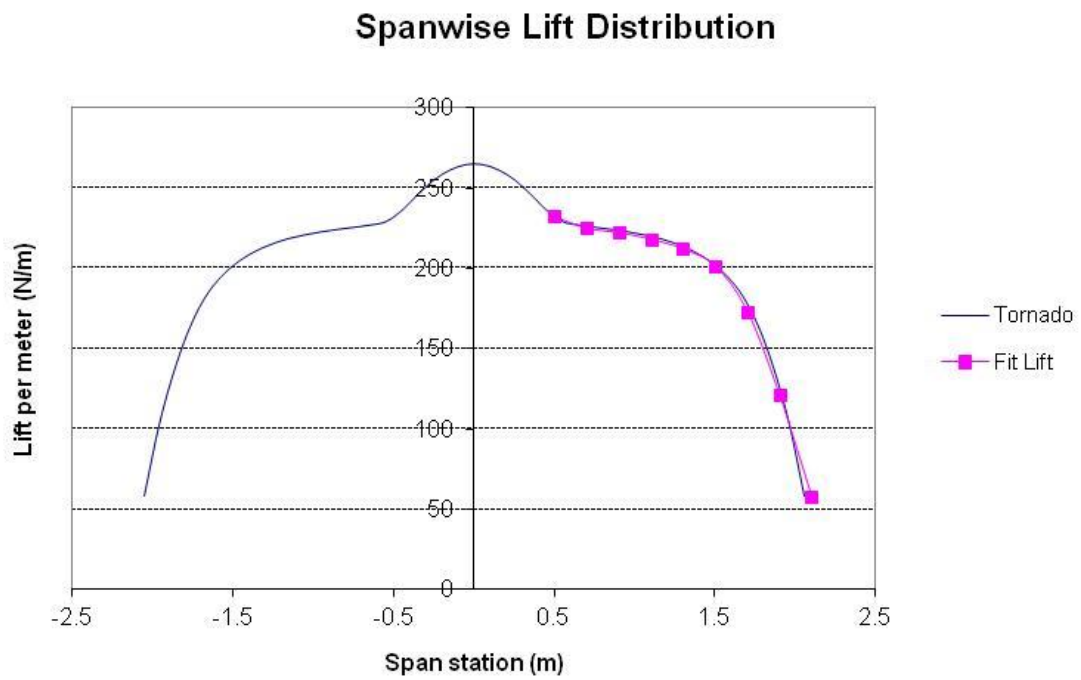
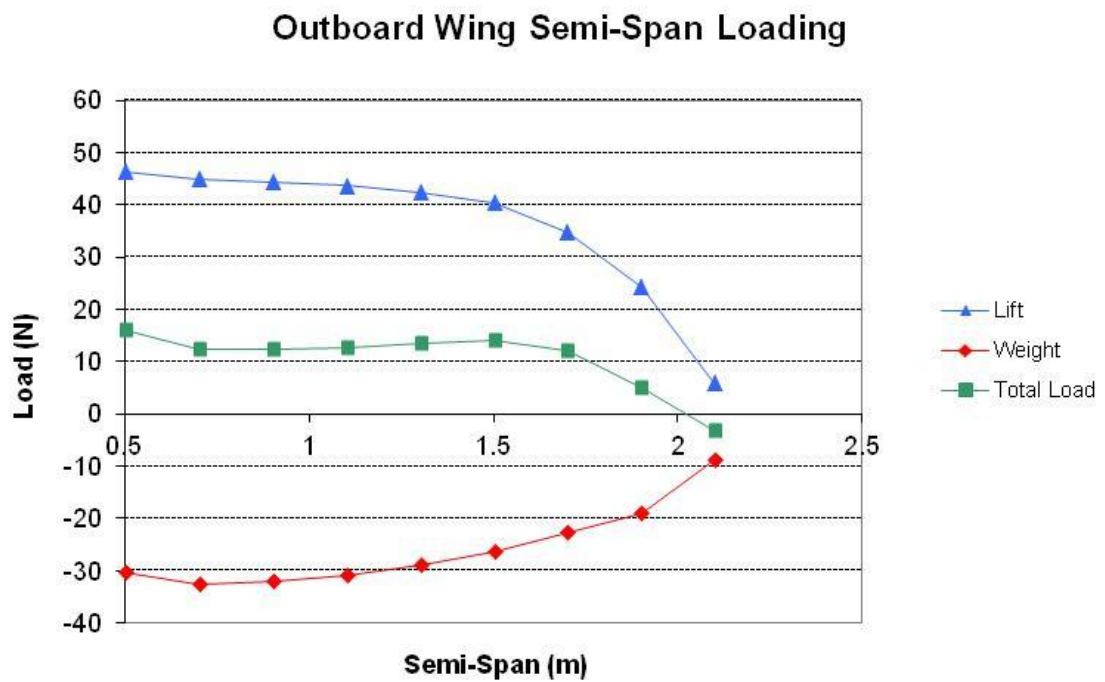


Figure 4-20 Lift distribution and outboard wing semi-span loading

4.2.3 SAW structural design

The SAW structure comprises of a centre wing box made of composite front and rear spars and skins reinforced by six T-shape stringers, three on the upper and three on the lower skin. The detailed geometry and dimensions of the SAW root and tip sections are shown in Figure 4-21.

The wing is 2.08m long by 0.6 m chord non-tapered with a 40 degree sweep. The thickness is tapered, lofting from NACA0015 to NACA0006 at the tip. It is made of glass fibre composite materials and modelled by using 8 span-wise single-cell thin-walled beams along the elastic axis. The elastic axis (EA) is located at 35% of the chord along the wing. The trailing edge control surface is positioned at the 7~8th wing section of span-wise length 0.45m. Unlike traditional aileron or elevon, the seamless control surface can be deflected from 60% of chord by internal actuators, which morph the wing camber smoothly.

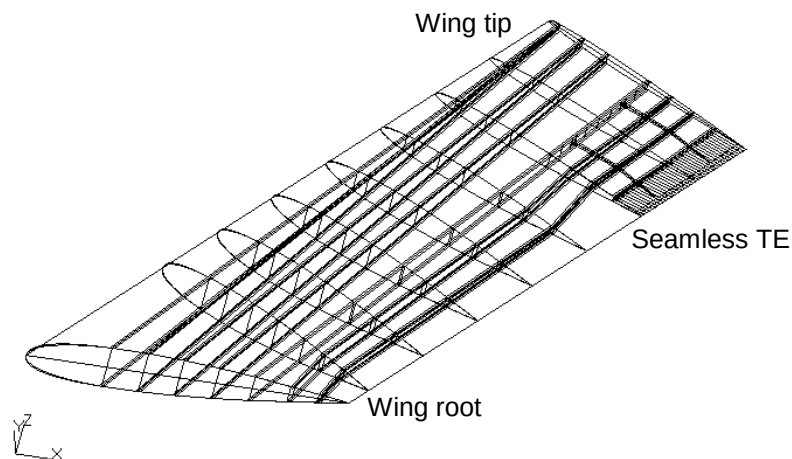


Figure 4-21 Cross-sectional details of the wing box

The front and rear spars of the SAW are made of 8-ply carbon/epoxy laminate of 1mm thickness with aeroelastically tailored layup for maximum flutter speed. For weight saving, the skin is optimised and reduced to only two layers of E-glass/epoxy laminate [+20/-20] of 0.5mm thickness due to manufacture constraint. This design was selected

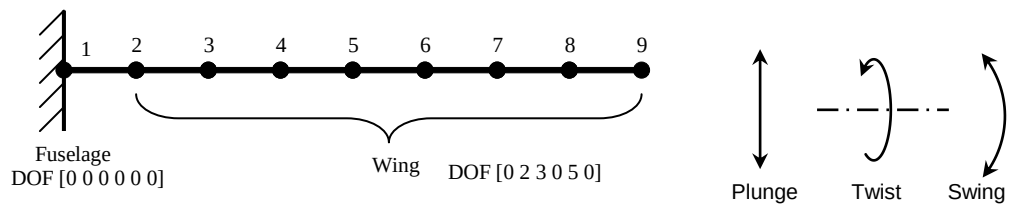
for its high control reversal speed of 85 m/s compared with other layup options [49]. The material properties of the E-glass/epoxy used in the analysis are supplied by the manufacturer and listed in Table 4-6.

Table 4-6 Mechanical properties of E-Glass/Epoxy

E_1 (GPa)	E_2 (GPa)	G_{12} (GPa)	ν_{12}	X_t (MPa)	X_c (MPa)	Y_t (MPa)	Y_c (MPa)	S (MPa)	ρ (Kg/m ³)
40	8	4	0.25	1000	600	30	110	40	1900

4.2.4 SAW structure simplification

Complete finite element analysis (FEA) of the SAW structure has been done by co-working PhD student [49]. FEA is ideal for deformation mode and frequency prediction. But in this study, structure equations have to be solved with unsteady aerodynamic equations and flight dynamic equations in time domain. A simplified structure model with mass and stiffness property from FEA was produced for this purpose. The simplified SAW structure was a multi-beam-element model which was defined by 9 concentrated mass nodes connected by 8 beams. Node 1 represents the half fuselage mass, nodes 2 to 9 represent distributed mass points of the semi-span wing with 3 degrees of freedom (DoF) for each node (plunge, twist and swing); node 1 is a fixed point (0 DoF) which provides the boundary condition of SAW structural dynamic systems.



The wing structural mass and stiffness properties are shown in the tables below. Structure damping was assumed to be $\zeta = 2\%$.

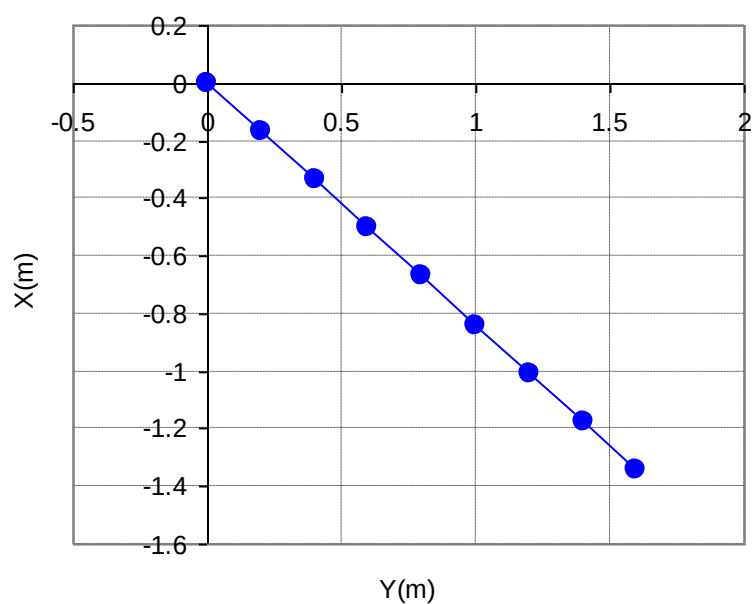


Figure 4-22 Beam model illustration

Nodes' mass and positions were shown in Table 4-7:

Table 4-7 Half wing beam model mass properties

Node No.	1	2	3	4	5	6	7	8	9
Mass (kg)	23.0913	3.3278	3.2684	3.1469	2.9554	2.6812	2.2988	1.944	0.906
X (m)	0	-0.16782	-0.33564	-0.50346	-0.67128	-0.8391	-1.00692	-1.17474	-1.34256
Y (m)	0	0.2	0.4	0.6	0.8	1	1.2	1.4	1.6

Connections amongst node 1 to 9 are elastic beam elements with stiffness properties presented in Table 4-8.

Table 4-8 Half wing beam model stiffness properties [+20/-20]

Section No.	1~2	2~3	3~4	4~5	5~6	6~7	7~8	8~9
$EI \times 10^3$	44.50	29.20	21.50	15.00	9.80	5.98	3.47	1.92
$GJ \times 10^3$	130.00	88.90	71.80	55.90	41.80	29.70	20.10	13.10
$CK \times 10^3$	-1.44	-1.00	-0.78	-0.58	-0.41	-0.26	-0.16	-0.10

4.2.5 Simulink implementation of beam model

The equation of motion for the SAW model is taken from section 3.3:

$$[M]\ddot{x} + [C]\dot{x} + [K]x = [F_A]$$

Where $[M]$ is the mass matrix, $[K]$ is the stiffness matrix, $[C]$ is the damping matrix and $[F_A]$ is applied load, x is displacement of all 9 nodes except node 1 which is fixed.

M and K were calculated by Beam 3D (see appendix A3), damping matrix $C = \zeta \times K / \omega$, where ζ is 2%, ω is the 1st order natural frequency of the beam.

Static deformation can be found by simply solving $[K]x = [F_A]$. The dynamic response requires the solution of the complete equation. The most common solution is in frequency domain in which x becomes mode shape η and frequency ω ; one can obtain the eigenvalues (vibration displacement) and eigenvectors (vibration mode shape), and then use the modal superposition method to acquire the final beam motion.

However, for the flexible aircraft, flight dynamic behaviour was concerned in this study, time history of the SAW beam motion and its influence on the aircraft is more interested, plus flight controller will be implemented in the later stage. Therefore, the SAW beam model was implemented in Matlab Simulink environment and numerically solved in time domain (Figure 4-23).

In the Simulink model, FM input is a matrix of force, twisting moment and bending moment for nodes 2-9 (fixed). Outputs are displacement, velocity and acceleration for plunge, twist and swing on all 8 nodes.

The input load F is a set of constant values of force and moment for static deformation calculation. But in dynamic response calculation, F can be a function of time (step, pulse, sine wave, etc). In the aeroelastic analysis F is a function of wing deformation. Thus the load input is affected by wing shape output forming an iteration loop that takes many cycles to converge to the solution.

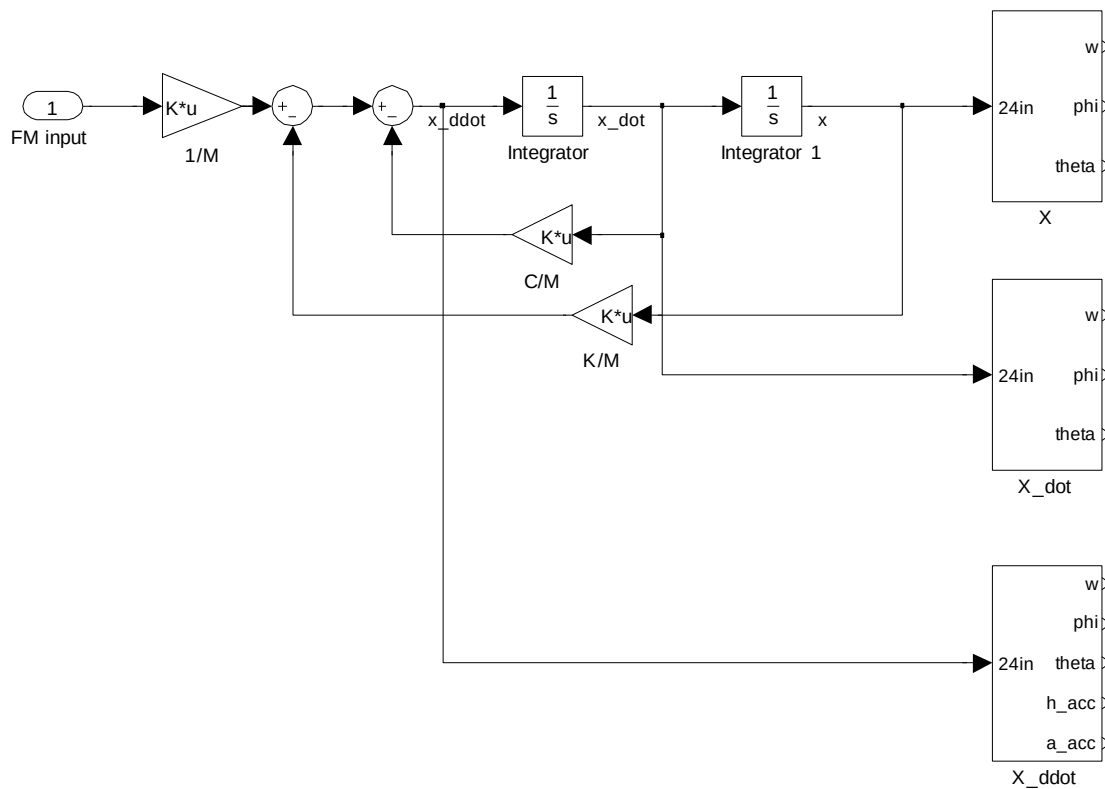


Figure 4-23 Beam model Simulink block diagram

4.2.6 Unsteady aerodynamic model in Simulink

The SAW unsteady aerodynamic model was built up from 2D Theodorsen's unsteady aerodynamic theory (section 3.4.1); where flat plate lift curve slope 2π is now scaled down to match the lift gradient of airfoil NACA0006, NACA0009 and NACA0015 as shown in Figure 4-24. Matlab code for this block is presented in appendix E. Note the outputs of pitching moment are relative to quarter chord line and elastic axis location respectively.

The strip theory was used to integrate all 2D sectional lifts along the span for 3D wing lift and superimpose the elliptical lift distribution to match total lift pattern and magnitude from the rigid wing. Although the beam consist 9 nodes, only 8 nodes were involved in the unsteady aerodynamic calculation since the root node was fixed, which is involved in steady aero-force only.

TE and LE control inputs on node 6 to 9 come from SAW airfoil design (section 4.1.1). The local incidence change was calculated by aircraft AOA and aeroelastic twist, which

exclude variable camber effect. Camber effects on aerodynamic coefficients from either hinged or seamless control surface were added to the local aerodynamic calculation. The combined AOA and camber effect were added together as control input.

By subtracting nodal weight (Table 4-7) from each lift station, the actual loading on each node can be obtained. Note that the nodal weight direction varies with the aircraft attitude. Hence a direction cosine matrix was added to calculate nodal gravity component as shown in Figure 4-25. Strictly speaking, local nodal weight direction should consider wing section attitude, but simplified beam element assumes CG was coincide with elastic axis, i.e. no moment due to gravity on wing section. Therefore twist angle was not added to weight calculation.

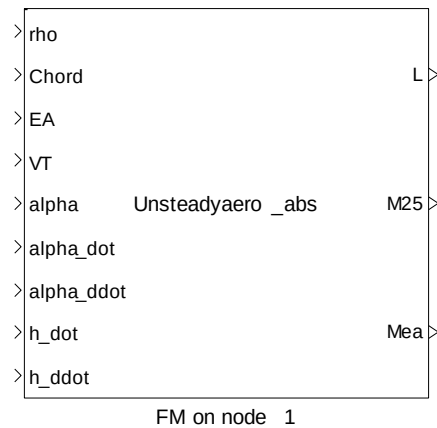


Figure 4-24 Theodorsen function and unsteady aerodynamic block

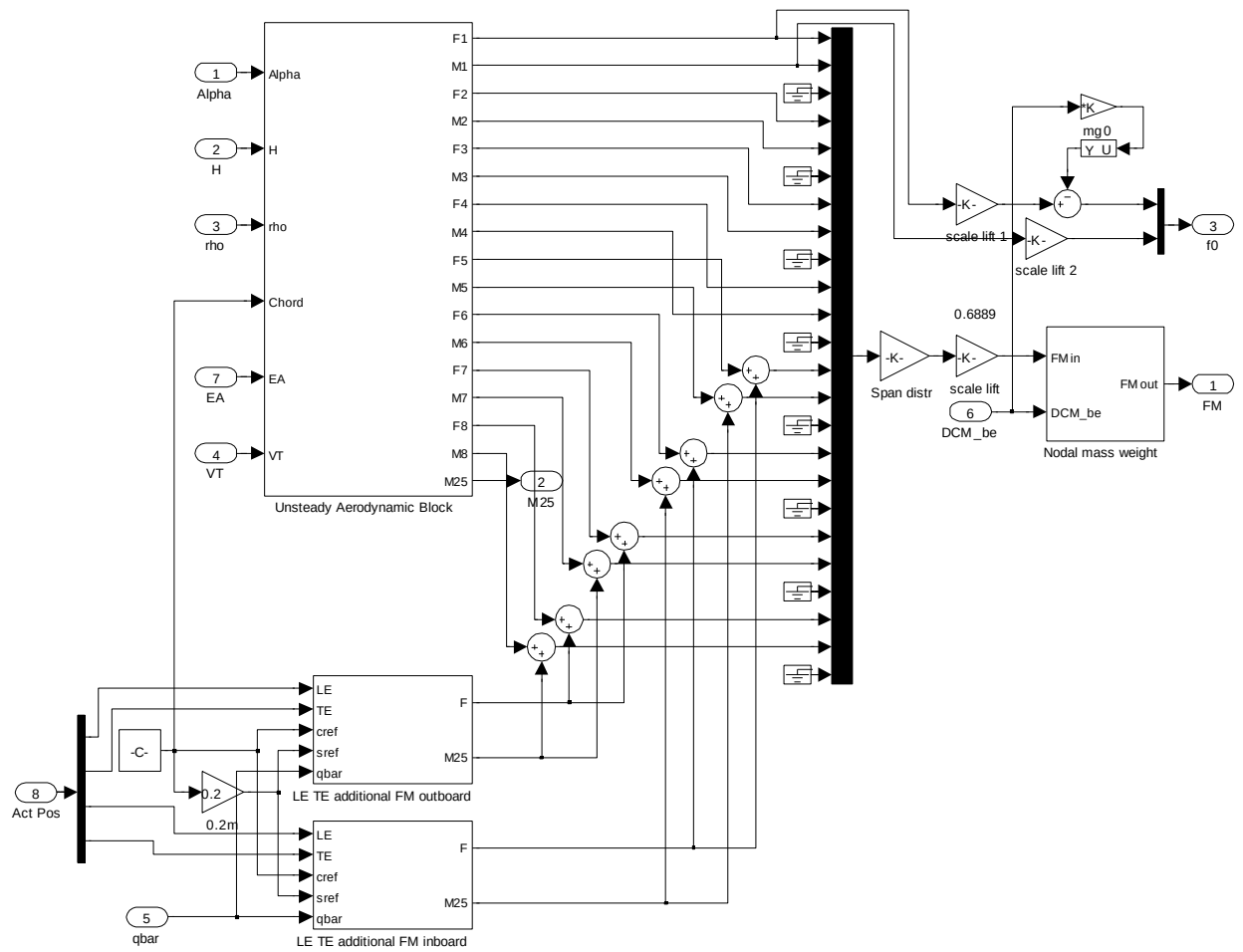


Figure 4-25 Unsteady aerodynamic model, LE TE control load and nodal gravity block

4.2.7 Combining two models

The unsteady aerodynamic model and structural dynamic model were combined to form SAW aero-structural interaction model.

The unsteady aerodynamic model took input from control surfaces; its loading output was the input to wing structure beam model, whose output of displacement, velocity and acceleration were sent back to unsteady aerodynamic model, thus completes the iteration loop. Meanwhile, unsteady aerodynamic output loading was also sent to 6DoF equation of motion (EOM) block to calculate flight motion.

Furthermore, fuselage node acceleration is sent out to calculate inertial load on the fuselage, which also add into 6DoF EOM to represent SAW elastic behaviour effect on IMU sensor pack (Figure 4-26)



4.2.8 Solving the SAW model

Matlab Simulink model of the SAW was solved numerically by Simulink ordinary differential equation solvers, interactions between aerodynamic and structure model iterated in variable calculation steps, steady state result maybe achieved, if solution converges.

The SAW model consists of a mixture of fast and slow changing variables, which is numerically stiff. Simulink default solver ode45 is not very suitable in this application. Deterministic solvers are more appropriate candidates, which include ode15s, ode23s, ode23t and ode23tb.

The ode15s is a multistep variable order solver based on the numerical differentiation formulas (NDFs). The ode23s solver is based on a modified Rosenbrock formula of order 2. Because it is a one-step solver, it may be more efficient than ode15s at crude tolerances. The ode23t solver is an implementation of the trapezoidal rule using a "free" interpolant. The ode23tb is an implementation of TR-BDF2, an implicit Runge-Kutta formula with a first stage that is a trapezoidal rule step and a second stage that is a backward differentiation formula of order 2. [50]

After several trial and error studies, ode15s was selected thanks to its sufficient accuracy and minimum calculation time. (Test runs ode45, 0.635 seconds. ode23s, 0.388 seconds. ode23t, 0.322 seconds. ode15s, 0.276 seconds)

It is important to record the solution steps as well as the time steps since there was more calculation steps at sharp gradient, to avoid data distortion; results need to be adjusted to reflect its time history. Figure 4-27 shows the difference between iteration step and time step results.

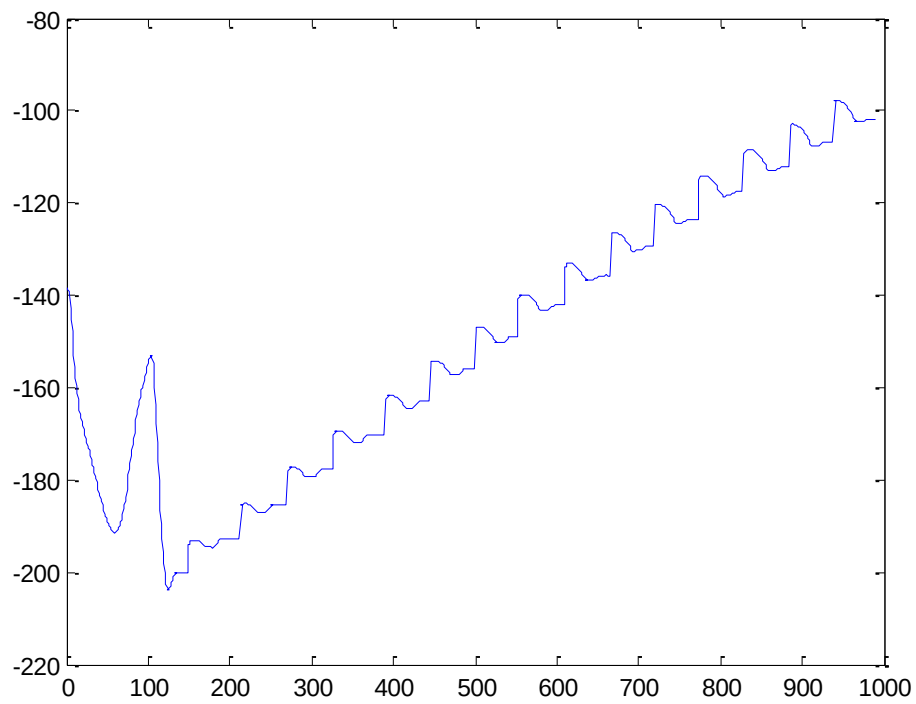
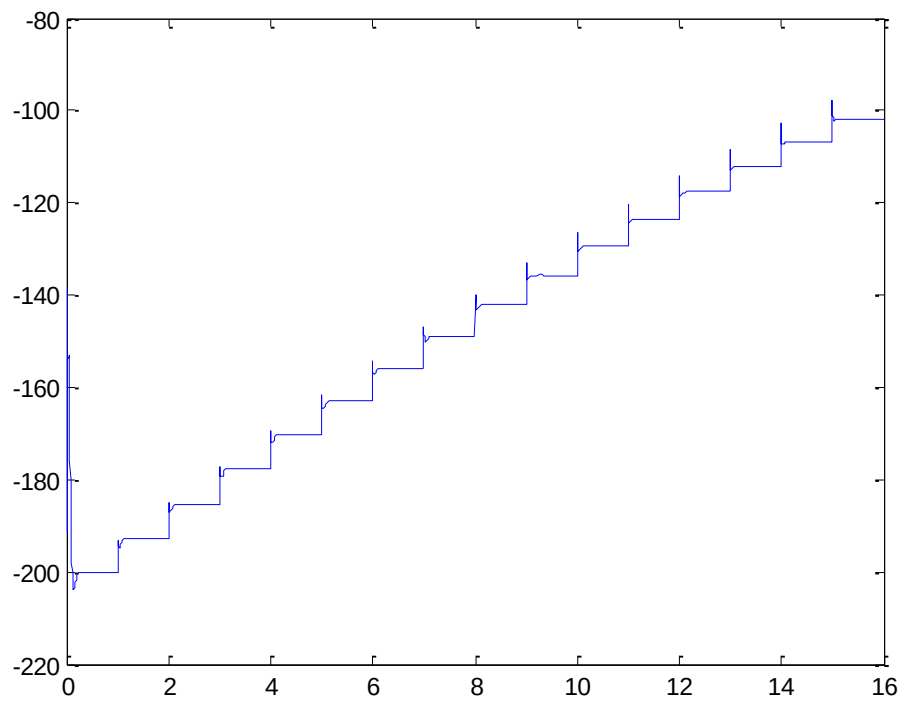


Figure 4-27 Simulation results in iteration steps (up) and time steps (down)

4.3 SAW Deformation and Loading Analysis

SAW under a given load performs a complex movement as the result of aero-structural interaction. This analysis focuses on steady state deformation and loading of the SAW.

4.3.1 Deformation and load at level flight

Static loading from the rigid D-47 example wing (section 4.2.2) was applied to the SAW. Deformation in plunge and twist were shown in Figure 4-28 and Table 4-9. The actual time spent on reaching this steady state deformation is not concerned in quasi-static analysis.

As a result of the wing deformation, the total lift will be less than what was needed to maintain level flight. This requires an increase in AOA. A comparative case study was made to investigate this in detail.

Table 4-9 Wing beam deformation in plunge and twist rigid case

Tornado 1.8deg 60m/s		Load does not change with deformation							
Static load and deformation		Total Load: 95.33 N							
Node No.	1	2	3	4	5	6	7	8	9
Load (N)	16.1216	12.3860	12.4048	12.7916	13.4338	14.0083	12.0975	5.1774	-3.0890
Plunge (m)	0.0000	0.0010	0.0041	0.0092	0.0160	0.0242	0.0331	0.0421	0.0507
Twist (rad)	0.0000	-0.0015	-0.0031	-0.0045	-0.0057	-0.0065	-0.0068	-0.0066	-0.0064

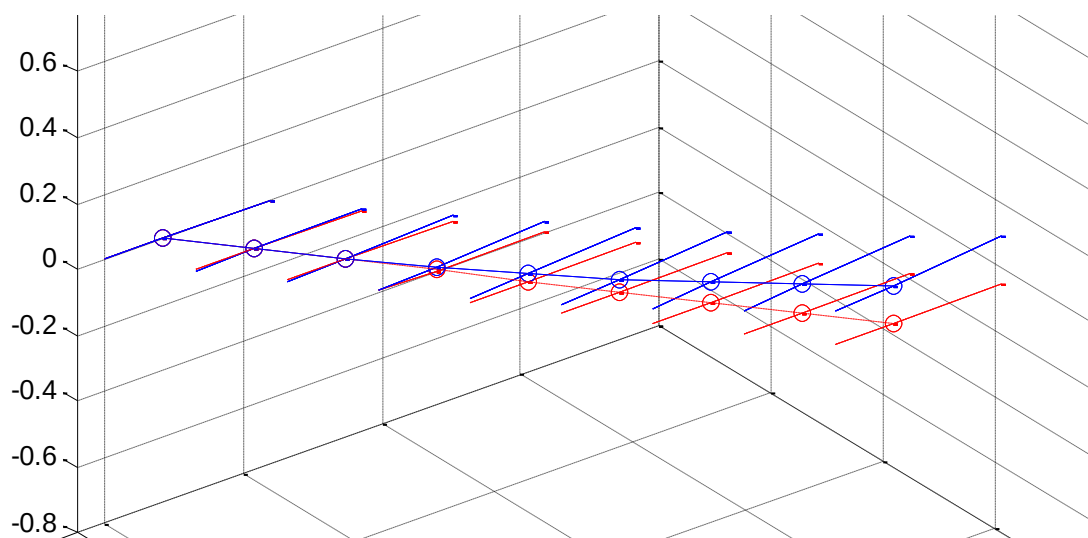


Figure 4-28 Beam nodes displacement

Case 1:

If initial flight condition was unchanged (60m/s, AOA=1.8°), the total loading on the deformed flexible wing will decrease to 71.15N compare to the rigid wing loading of 95.33N. (Figure 4-29) This is because of the outboard wing AOA reduction as a result of bending-twist coupling of the 40° swept back angle design. Wing deformation is also reduced (Table 4-10).

Since total lift is reduced, an increase of AOA is required if the flexible aircraft is to maintain the flight condition.

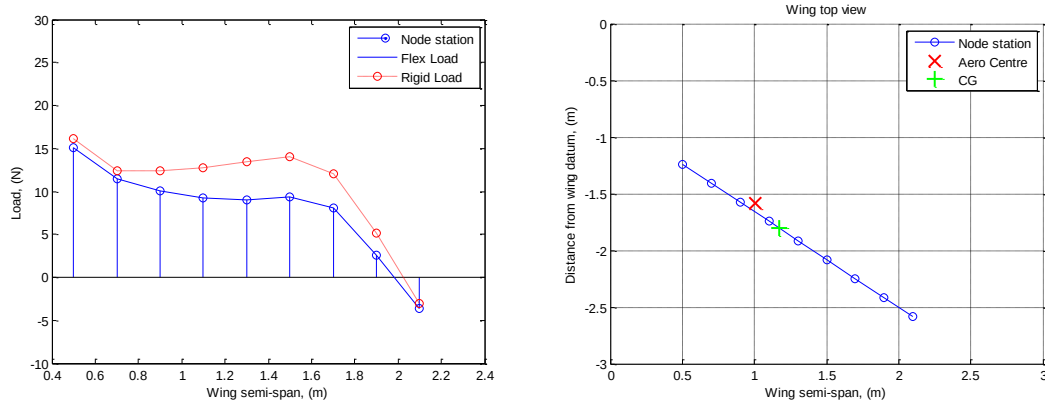


**Figure 4-29 Spanwise load distribution of rigid and flexible wing at AOA=1.8 deg
Vt=60m/s**

Table 4-10 Wing beam deformation in plunge and twist flex case 1

SAW 1.8deg 60m/s		Load changes with deformation							
Static load and deformation		Total Load: 71.15 N							
Node No.	1	2	3	4	5	6	7	8	9
Load (N)	15.1305	11.4251	10.0277	9.2394	9.0491	9.3181	8.0440	2.5830	-3.6699
Plunge (m)	0.0000	0.0007	0.0026	0.0059	0.0101	0.0150	0.0202	0.0252	0.0297
Twist (rad)	0.0000	-0.00101	-0.00201	-0.00287	-0.00356	-0.00396	-0.00398	-0.00367	-0.00345

Case 2:

Now AOA is increased to 2° at 60m/s, as shown in Figure 4-30. Total loading on the flexible wing becomes 95.36N which can maintain level flight again but lift distribution is very different from that of the rigid wing. More lift is produced inboard and less outboard, hence wing deformation shape and loading profile is changed, but the total load magnitude is the same (Table 4-11).

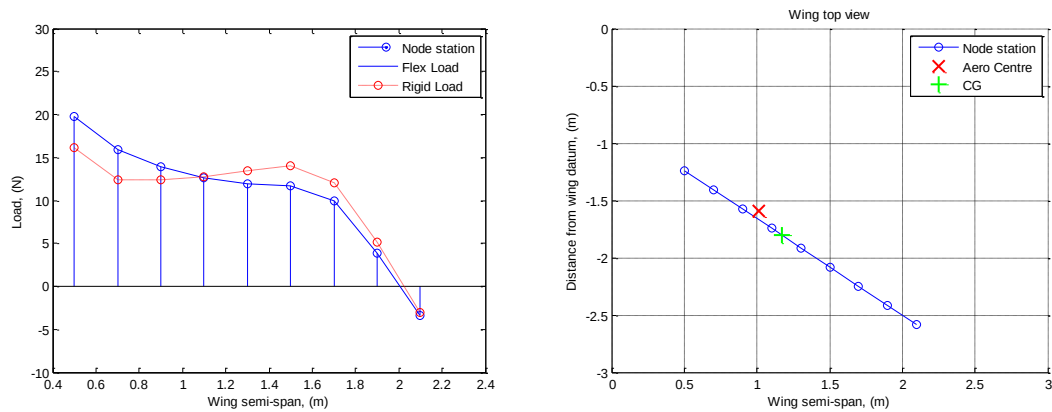


Figure 4-30 Spanwise load distribution of rigid and flexible wing at same total load

Table 4-11 Wing beam deformation in plunge and twist flex case 2

SAW 2.0deg 60m/s		Load changes with deformation							
Static load and deformation		Total Load: 95.36 N							
Node No.	1	2	3	4	5	6	7	8	9
Load (N)	19.7963	15.9487	13.9643	12.6312	11.9350	11.7336	9.9144	3.8156	-3.3792
Plunge (m)	0.0000	0.0009	0.0036	0.0080	0.0139	0.0208	0.0283	0.0358	0.0429
Twist (rad)	0.0000	-0.00138	-0.00275	-0.00395	-0.00494	-0.00559	-0.0058	-0.00558	-0.00538

Due to large swept back angle, inboard growth of lift shifts the wing aerodynamic centre both inward and forward. The longitudinal static stability of the aircraft will be greatly affected by this as aerodynamic centre (measured from nose datum) moved from -1.78m to -1.59m. Roll moment arm length on a single wing is also reduced from 1.09m to 1.01m.

Lift and pitching moment calculated by SAW model were converted to dimensionless terms to compare with that of the rigid wing. Results showed that SAW flexibility caused lift coefficient and pitching moment coefficient reduction, however they still maintain approximately a linear relationship with AOA (Figure 4-31) and the lift curve gradient CL_α decreased from 3.314 to 1.813, pitching moment curve gradient Cm_α changed from -0.633 to -0.418 (relative to wing root).

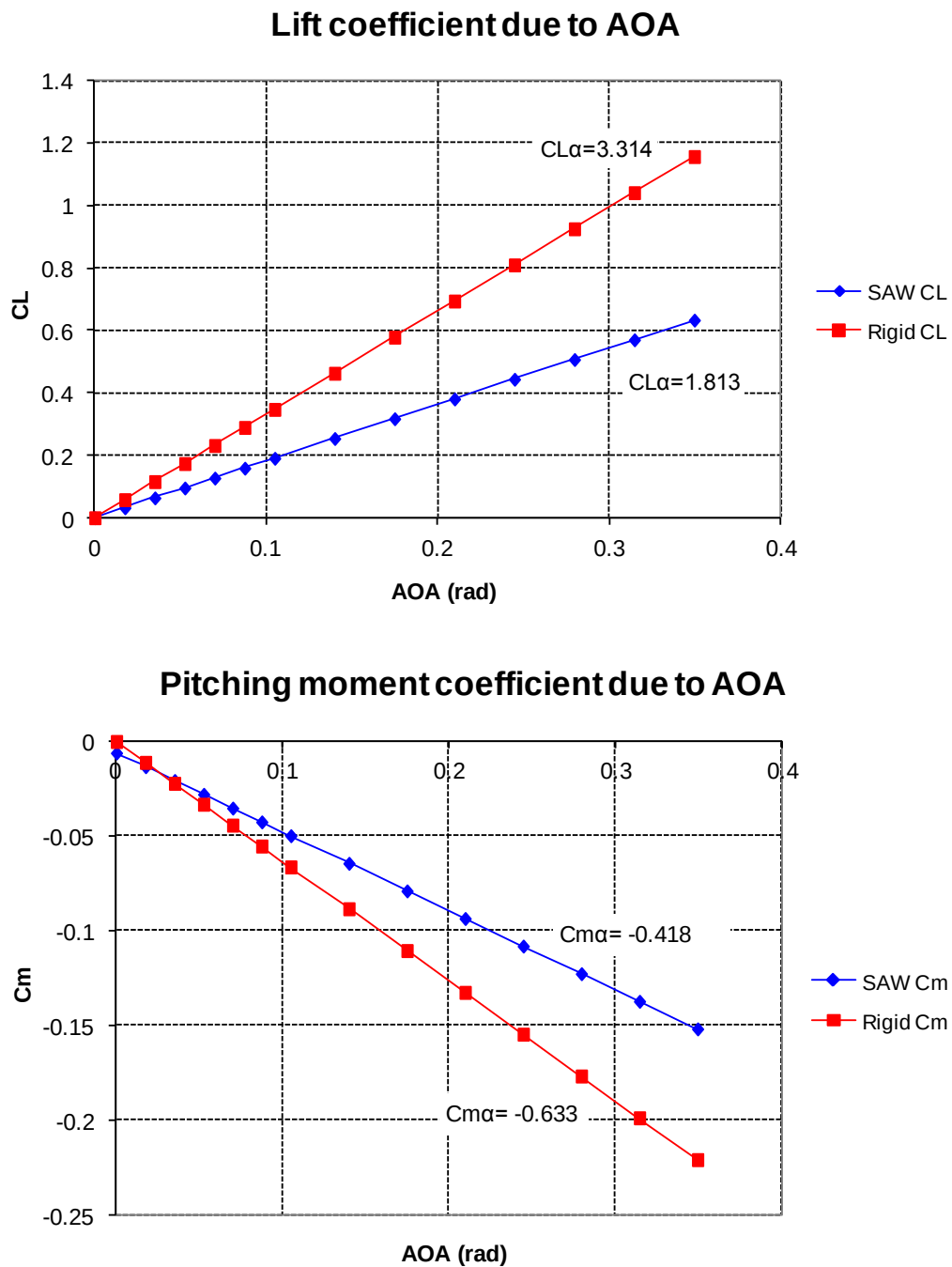


Figure 4-31 CL and Cm of SAW compared to that of rigid wing at 60m/s

4.3.2 Aeroelastic control reversal

Control reversal on a three dimensional wing is much more complicated than that on a two dimensional airfoil. Especially on the swept back wing, any increase of lift will

increase bending-twist coupling effect. Situation exacerbates when a trailing edge control surface is deflected, lift increase near the wing tip reduces effective incidence angle dramatically, control reversal speed V_R will be much less than structure design estimation.

When control input comes from both LE and TE control surfaces, lift and pitching moment characteristics are even more complicated since they all produce not only lift but also pitching moment. LE deflection contributes to both lift and pitch up moment but its effect is relatively small, TE deflection increase lift but pitch down moment, higher flight velocity rendered TE inoperable unless inverse TE deflection is programmed in the flight control law.

An example of SAW lift generation with ± 10 degree hinged control surface deflection (60m/s, 2 degree of AOA) was shown in Figure 4-32. $V_R = 76.5\text{m/s}$

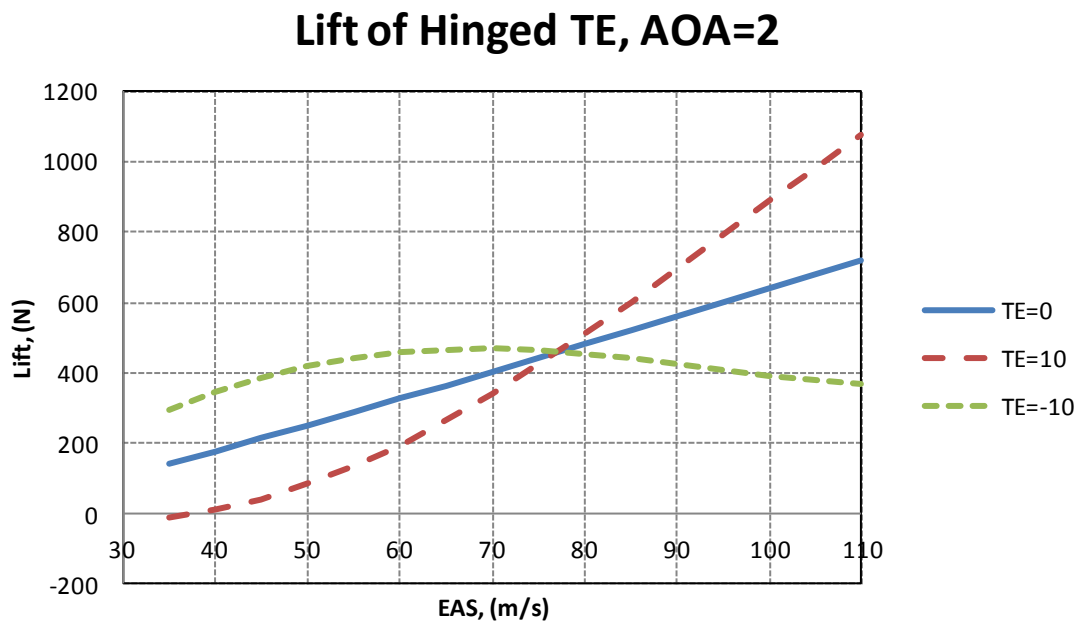


Figure 4-32 Aeroelastic control reversal example $V_R = 76.5\text{m/s}$

Deformation and loading at control reversal velocity is shown in Figure 4-33. Aerodynamic centre moved from $x=-1.78\text{m}$ $y=1.09\text{m}$ to $x=-1.95\text{m}$, $y=1.44\text{m}$.

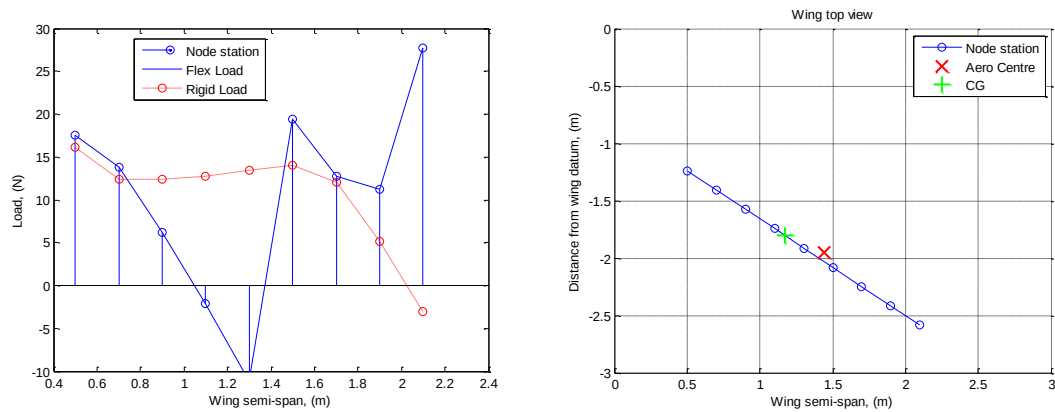


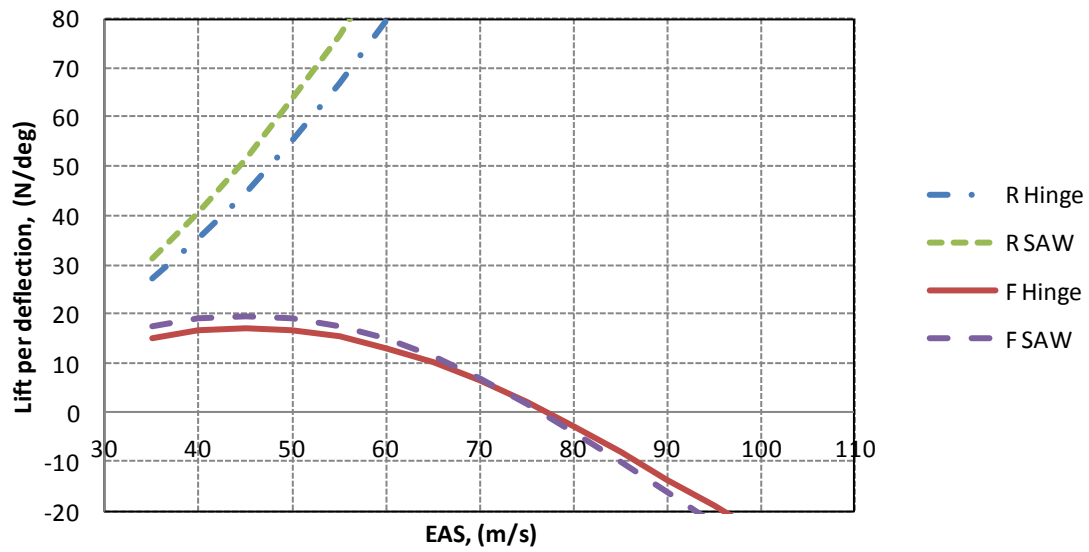
Figure 4-33 Deformation and loading at control reversal speed

D-47 design features flexible wing structure and seamless control surfaces. Unfortunately, these two design features influenced aeroelastic control reversal problem collusively. In order to obtain a comprehensive understanding of the impact from each design variable, 4 design cases were studied:

- Rigid wing with hinged control surface (R hinged)
- Rigid wing with seamless control surface (R SAW)
- Flexible wing with hinged control surface (F hinged)
- Flexible wing with seamless control surface (F SAW)

Control effectiveness of LE and TE for each case were analyzed and presented in Figure 4-34 and Figure 4-35. It can be observed that seamless control surface is more effective than hinged counterpart in both rigid and flexible cases. While flexibility of the wing completely changed the lift generation but pitch moment was only marginally affected. Nevertheless, the complex nature of SAW lift and moment generation and difficulties of using SAW for flight control poses a major challenge.

TE Lift control effectiveness



LE Lift control effectiveness

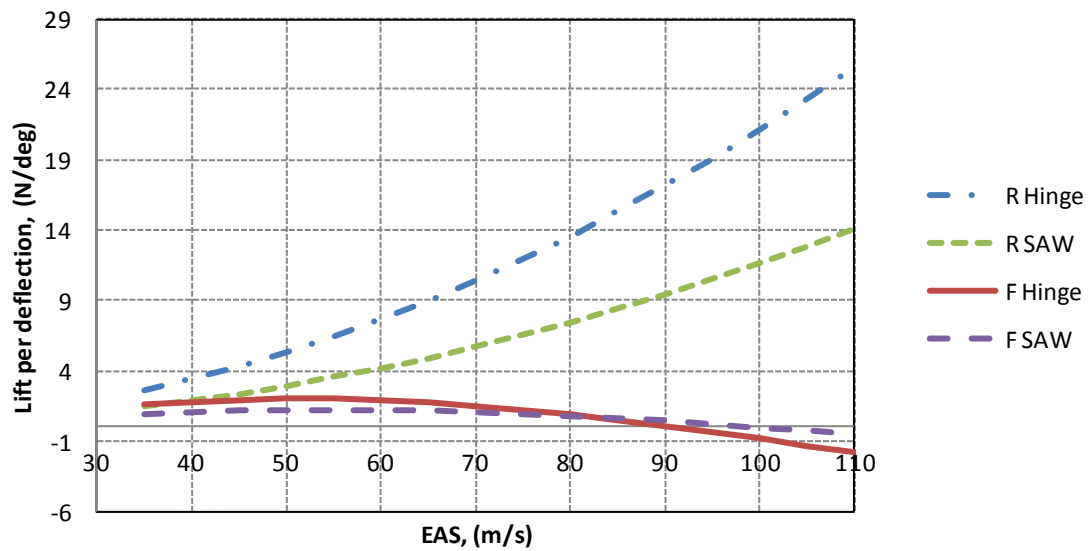
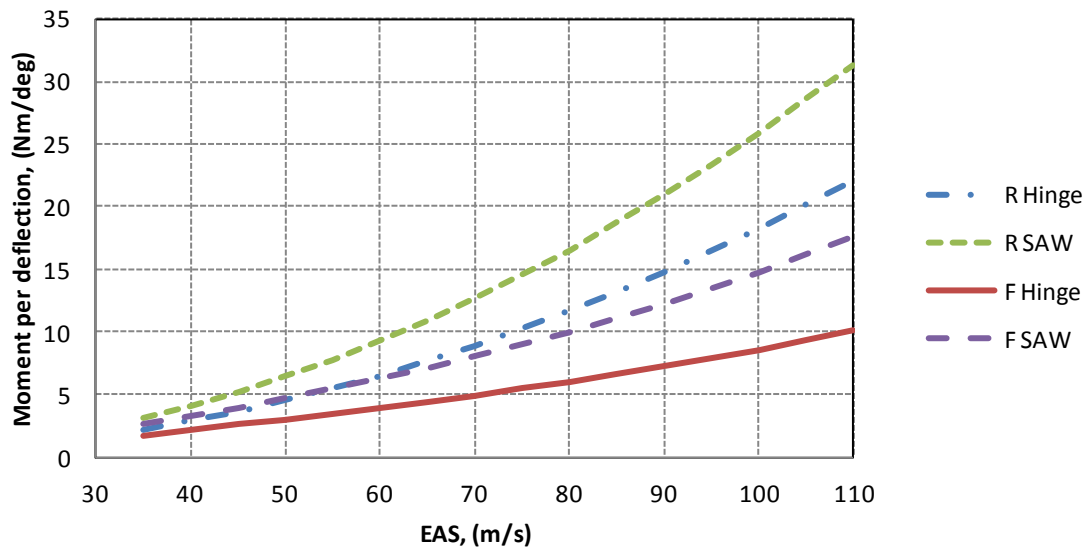


Figure 4-34 TE and LE lift control effectiveness.

TE moment control effectiveness



LE moment control effectiveness

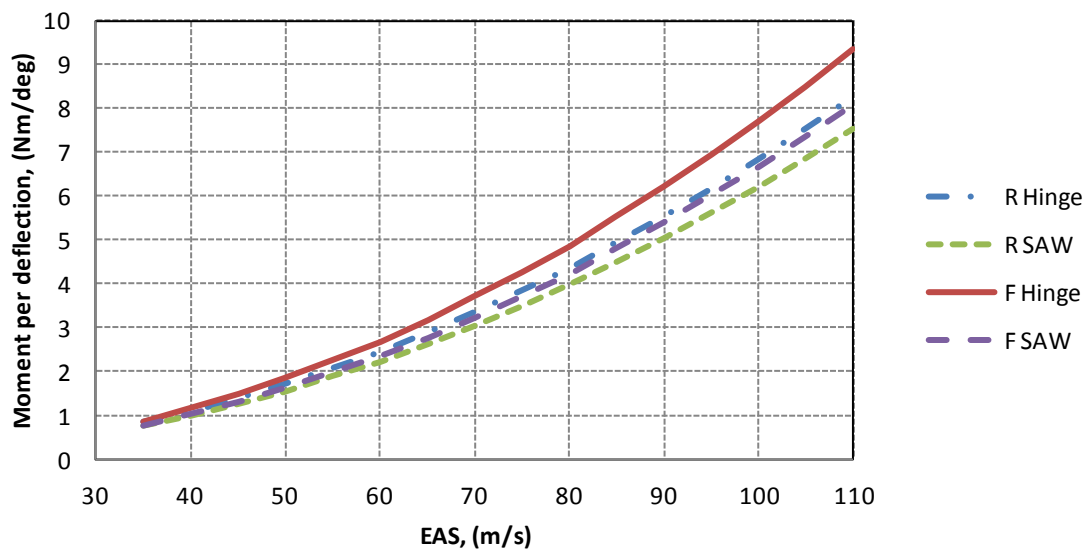


Figure 4-35. TE and LE pitching moment control effectiveness.

4.3.3 Outboard TE, inboard TE and LE deflection analysis

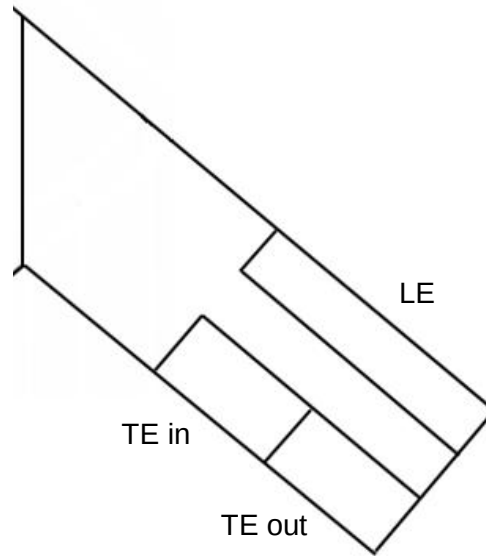


Figure 4-36 SAW control surfaces allocation

Another way of reducing aeroelastic control reversal effect is to allocate multiple control surfaces and deploy one or more of them at different flight condition. For D-47, one LE and two TE control surfaces were designed. They will be actuated independently and used for mixed pitch and roll control (Figure 4-36).

In level flight, semi-span of SAW was loaded 96.36N. TE control surface input 14.12N and -1.645Nm (pitch down) can be applied on either outboard (node 8 and 9) or inboard (node 6 and 7) portion of the wing. LE control surface inputs are 7.944N and 2.435Nm (pitch up) on node 6 to 9.

Outboard TE control surface causes local lift increase which bends up the wing further and twist wing down even more. The loading on the wing is however, reduced to 94.86N irrespective to local lift increase. This was because of excessive twist down which decreased overall lift. Pitch down moment change from -64Nm to -55Nm.

Spanwise loading, aerodynamic centre and loading time history are presented in Figure 4-37. Aero centre is located at 1.71m from aircraft nose datum.

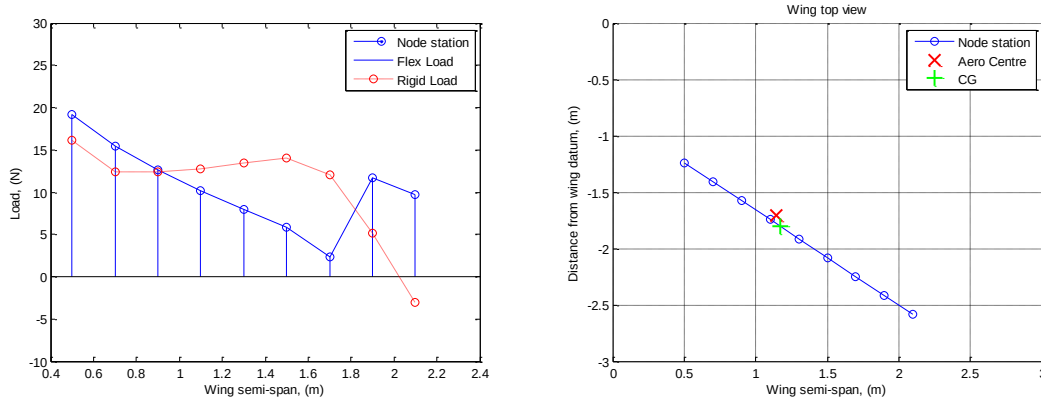


Figure 4-37 Lift results outboard TE deflection

Inboard TE control surface can provide same local lift increase with less bending and twist deformation, as the result, total lift of the wing is increased from 96.36N to 107.5N. Pitch down moment change from -64Nm to -59Nm. The spanwise loading, aerodynamic centre and loading time history are presented in Figure 4-38. Aero centre is located at 1.68m from aircraft nose datum

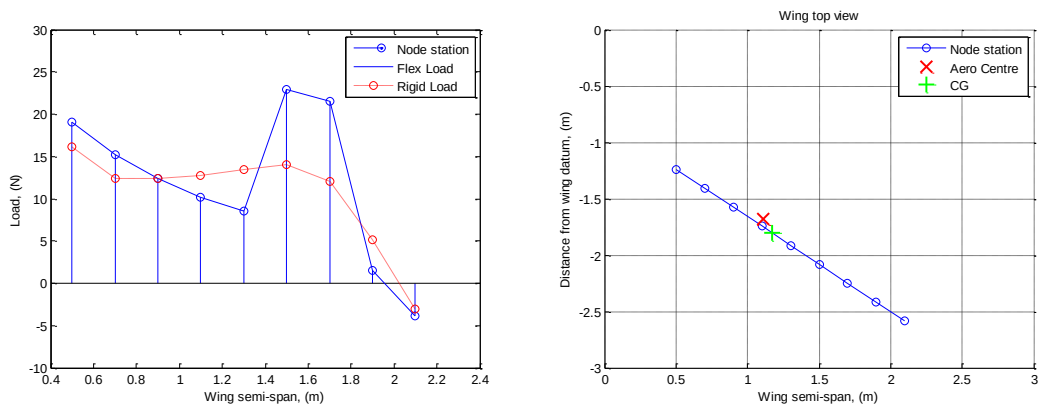


Figure 4-38 Lift results inboard TE deflection

LE control surface increases local lift as well as pitch up moment, as the result, total lift of the wing is increased from 96.36N to 101.7N. Pitch down moment change from -64Nm to -59.79Nm. The spanwise loading, aerodynamic centre and loading time history are presented in Figure 4-39. Aero centre is located at 1.6740m from aircraft nose datum

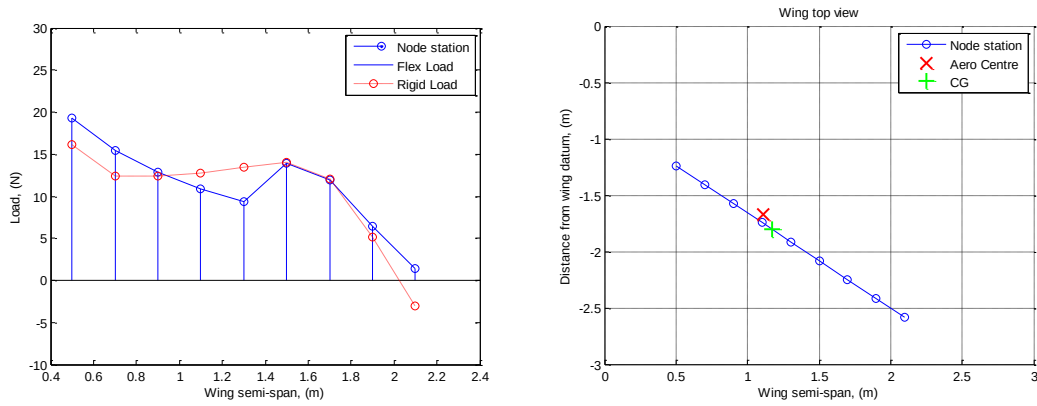


Figure 4-39 Lift results due to LE deflection

These results provide a guideline for flight control actuation management. Deterioration of control effectiveness with respect to flight velocity can be partially mitigated by using inboard TE control surfaces. Outboard control surface will be useful if flight control law can manage flight load actively.

4.4 SAW Dynamic Response Analysis

4.4.1 SAW dynamic response time history

The dynamic response of the flexible wing was initiated from the steady state deformation, where a step input of LE or TE control surface is introduced at 1 second, additional lift will act on the corresponding nodal station and excites wing motion.

Nodal displacement of the SAW was shown in Figure 4-40, aerodynamic lift and pitch moment produced by SAW is shown in Figure 4-41. Force and moment caused by inertia acceleration between wing and fuselage (or structural vibration internal force) is shown in Figure 4-42.

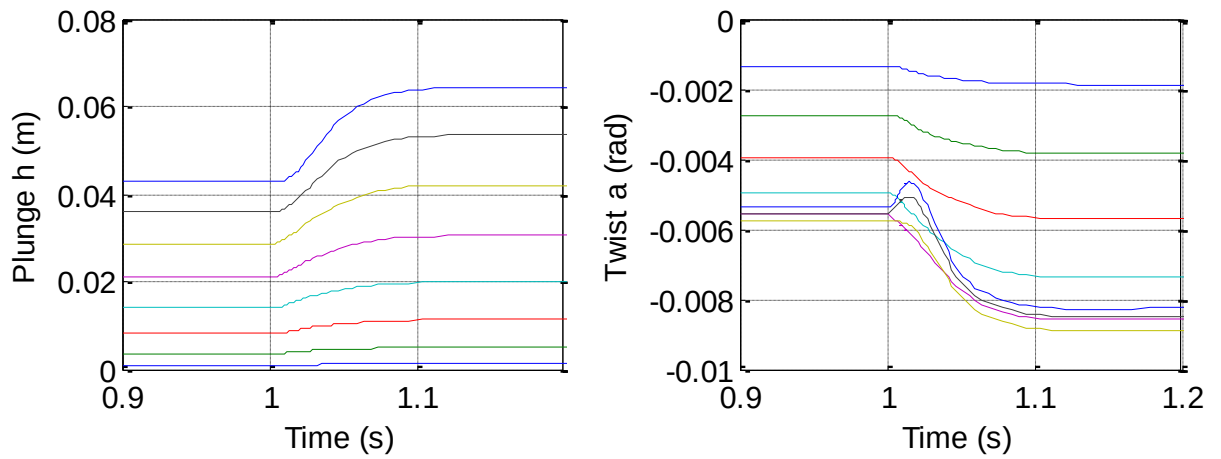


Figure 4-40 Plunge and twist of 8 nodes on SAW beam model

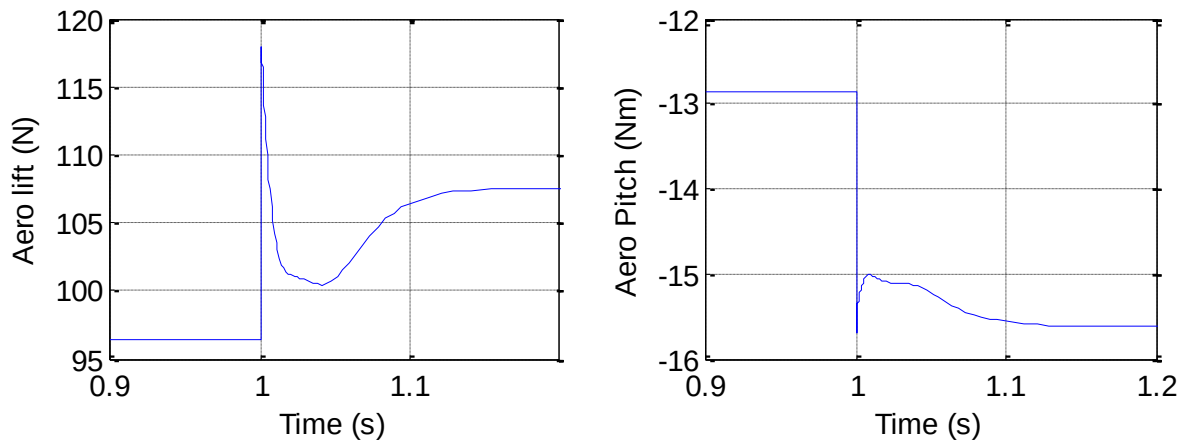


Figure 4-41 Aerodynamic lift and pitch moment on SAW

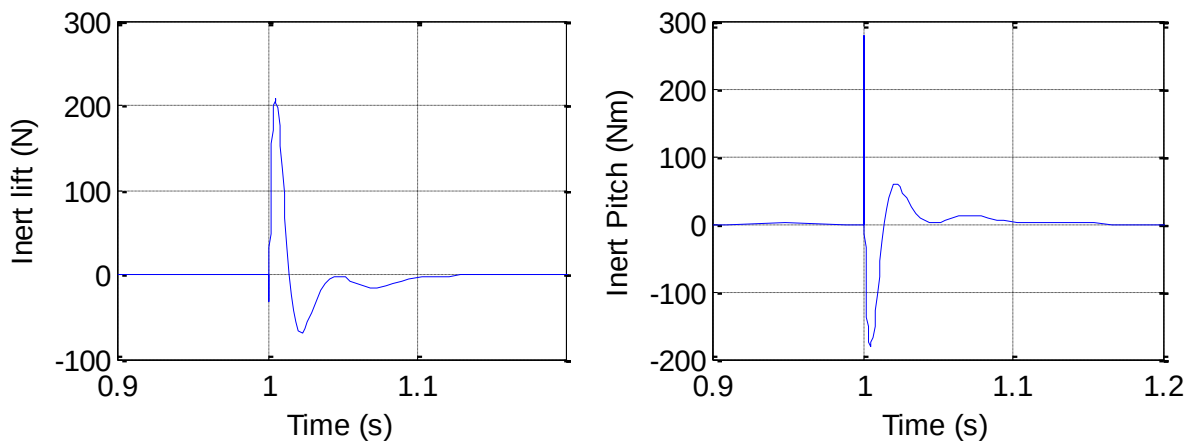


Figure 4-42 Inertial lift and pitch moment on SAW

4.4.2 Details of the unsteady aerodynamic load

At V=60m/s AOA=2 degree, The TE is deflected instantaneously 1 degree down at 1s. This step input could trigger response on all frequency range so that each component of the unsteady aerodynamic force can be revealed. The maximum deflection is the 9th node. Detailed examination shows proportion of different components in the unsteady aerodynamic load vary with respect to time (Table 4-12).

$$L = \underbrace{\pi \rho b^2 (\ddot{h} + V\dot{\alpha} - ba\ddot{\alpha})}_{\text{Mass effect}} + \underbrace{2\pi \rho V b C(k) [\dot{h} + V\alpha + b(\frac{1}{2} - a)\dot{\alpha}]}_{\text{Circulation effect}}$$

Table 4-12 Theodorsen lift components

<i>Time</i>	<i>Mass effect lift</i>			<i>Circulation effect lift</i>		
	<i>h ddot</i>	<i>alpha dot</i>	<i>alpha ddot</i>	<i>h dot</i>	<i>alpha</i>	<i>alpha dot</i>
0.0000	-30.34	0.00	7.80	0.00	141.06	0.00
6.2E-08	-30.30	0.00	7.78	0.00	141.06	0.00
1.24E-07	-30.26	0.00	7.76	0.00	141.06	0.00
.	.	.	.	.	.	.
.	.	.	.	.	.	.
.	.	.	.	.	.	.
0.016	31.78	1.96	1.70	84.09	160.71	3.13
0.017	30.99	0.63	1.78	98.45	161.31	1.00
0.019	29.28	-0.74	1.78	112.23	161.29	-1.18
.	.	.	.	.	.	.
.	.	.	.	.	.	.
.	.	.	.	.	.	.
0.51	0.00	0.00	0.00	0.00	82.67	0.00
.	.	.	.	.	.	.
.	.	.	.	.	.	.
.	.	.	.	.	.	.
1.00	0.00	0.00	0.00	0.00	82.67	0.00

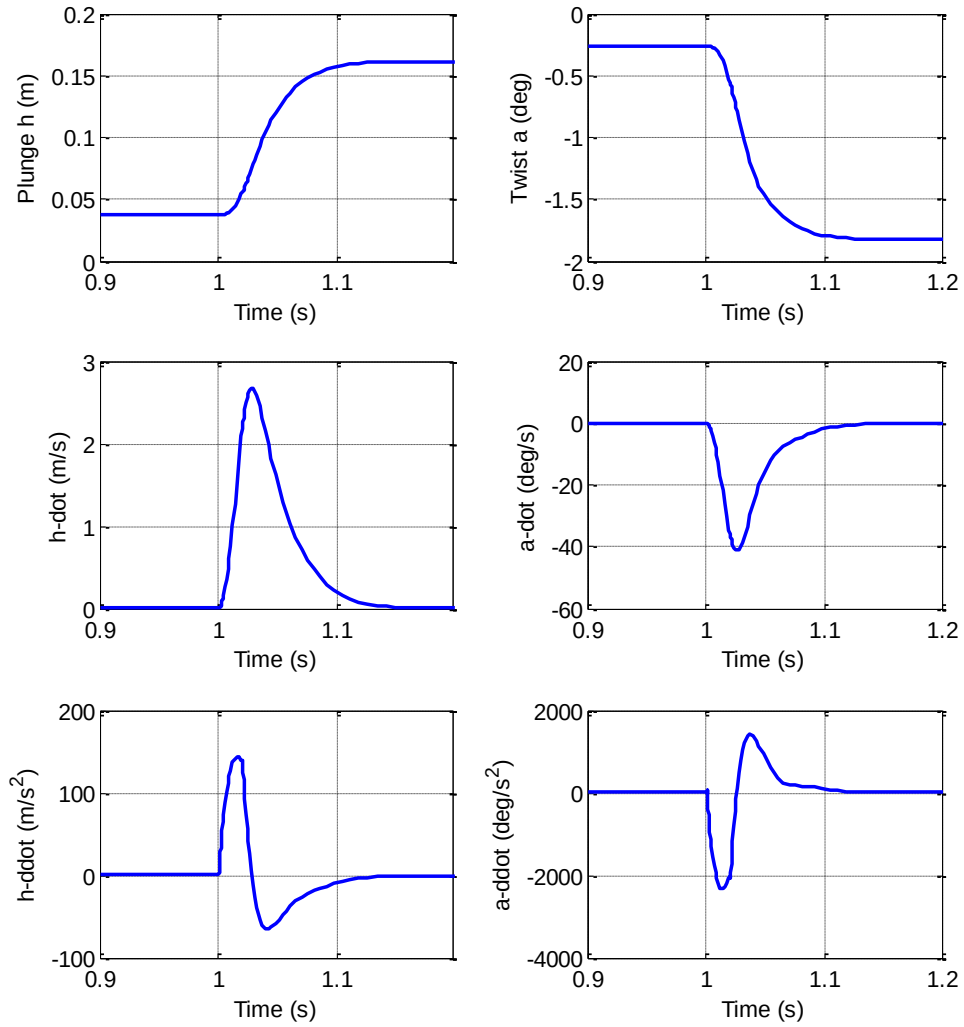


Figure 4-43 Displacement, velocity and acceleration on the 9th node

Full time history plot of Theodorsen unsteady aerodynamic load can reveal the effect of each contributor. When input of displacement, velocity and acceleration terms were given to the 9th node (Figure 4-43), unsteady air load was calculated and presented in Figure 4-44, it is apparent that α and $\dot{h}$ terms are most dominant, $\ddot{h}$ and $\dot{\alpha}$ terms are less influential, $\ddot{\alpha}$ makes virtually no effect on the magnitude of lift. If the input α is decreasing, $\ddot{h}$ and $\dot{h}$ terms are all acting as damping forces which prevent lift drop and vice versa. However, since $C(k)$ is complex number, a phase shift would be introduced

to the circulation terms which moves the peak of lift from $\dot{h}$ term. Consequently, this might change the nature of $\dot{h}$ damping effect to become excitation force.

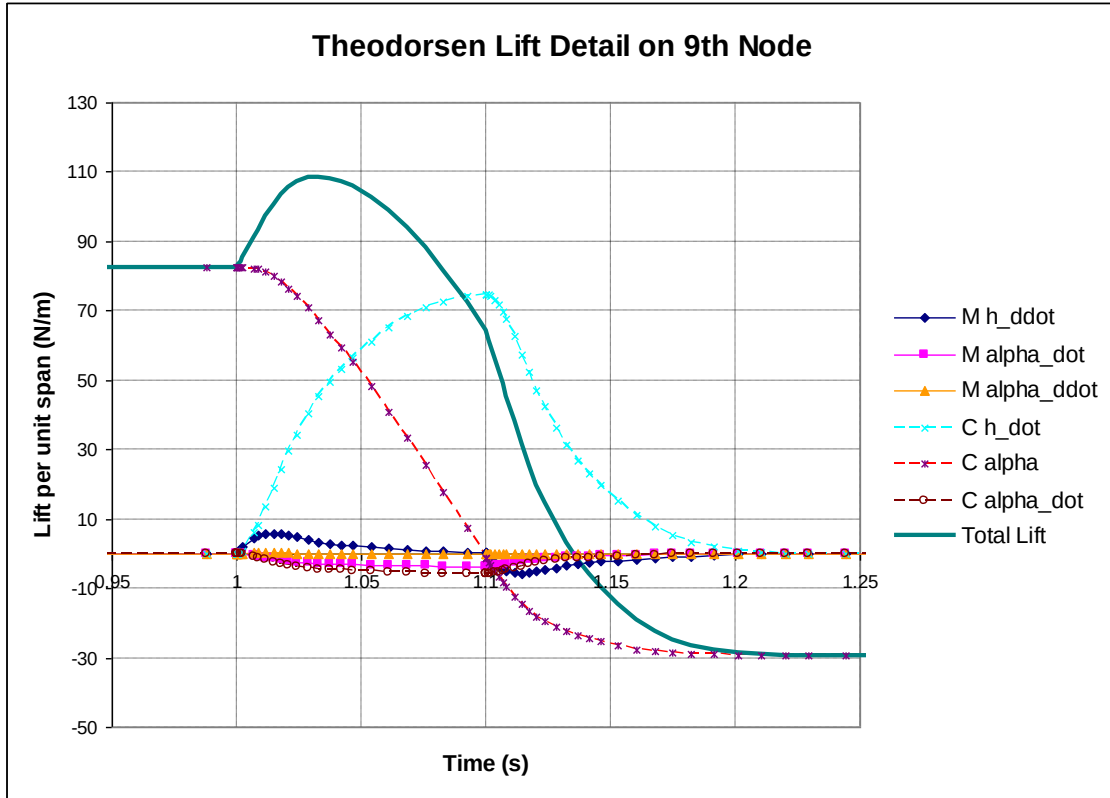


Figure 4-44 Theodorsen lift components

4.4.3 Damping effects of unsteady aerodynamic load

Damping effect of the unsteady aerodynamic load was due to the inclusion of velocity and acceleration terms. This was investigated by applying TE control surface input to generate a force of 14.12N and torque of -1.645Nm at 10th second. (It takes 10 second for vibration to settle without aero damping) The SAW was under 1G level flight load 96.36N.

Time histories of the SAW's response without and with aerodynamic damping are shown in Figure 4-45 and Figure 4-46. Different colours represent trace of 8 wing nodes.

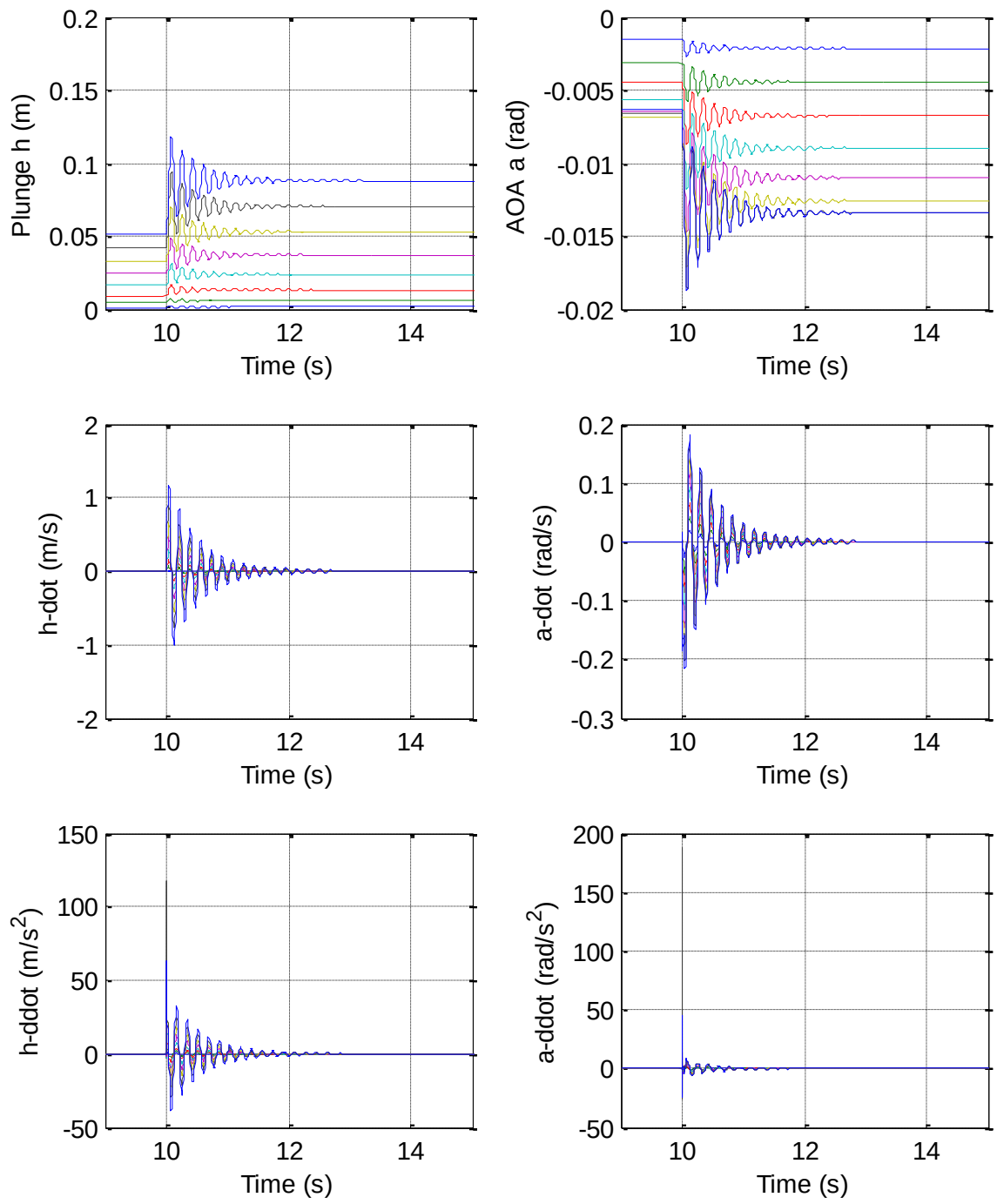


Figure 4-45 SAW dynamic responses without aero damping

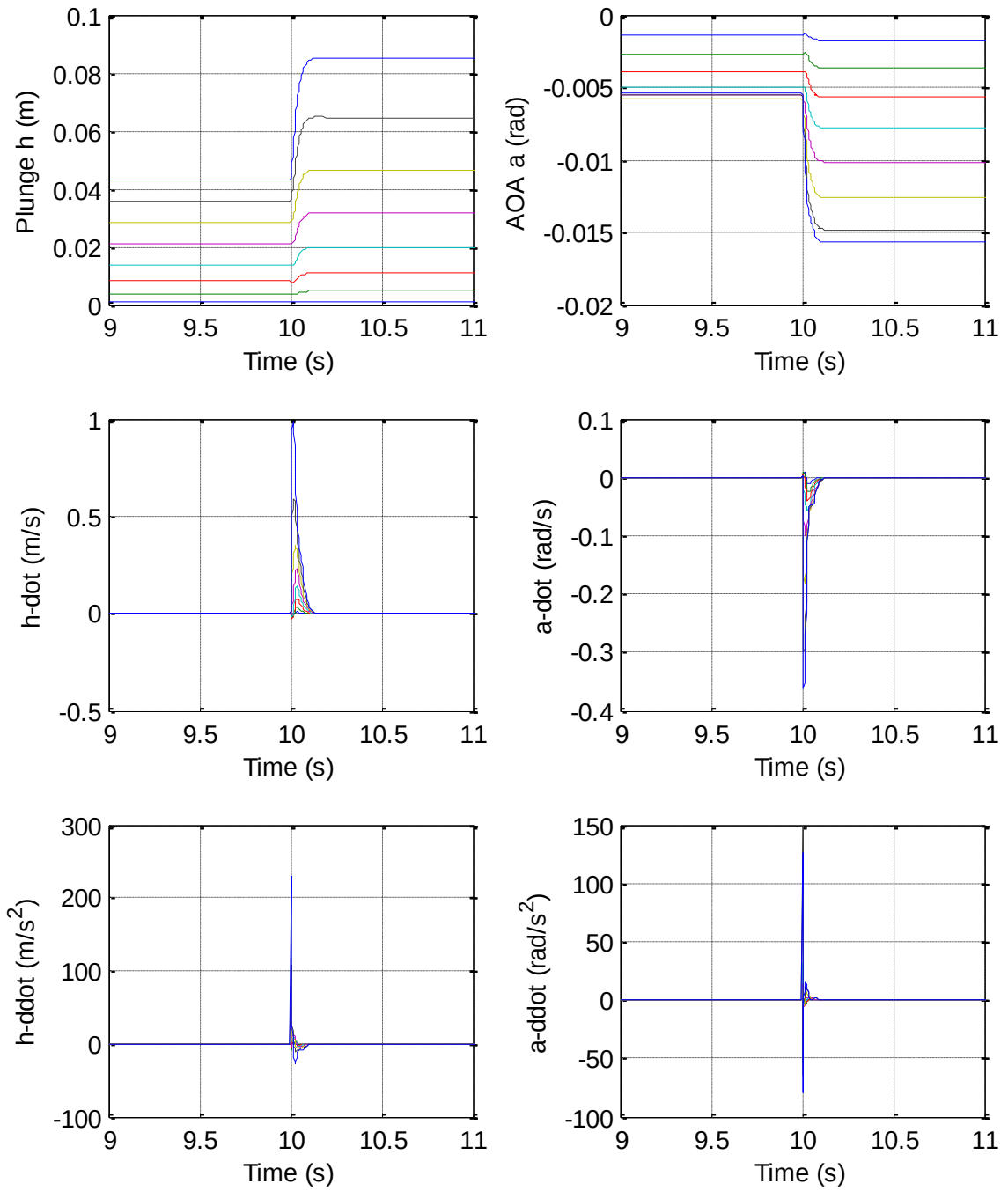


Figure 4-46 Dynamic responses with aero damping TE outboard

Unsteady aerodynamic force on the wing is subsequently changed which act as a damping force to the wing vibration. Aero-damping effect can be observed from the plunge and twist motion, damping ratio is estimated to be above 0.7.

4.4.4 Deflection rate analysis

Theoretically, this part of the study is in the scope of aero-servo-elasticity because of the inclusion of control actuator dynamics which defines the deflection rate of TE control surfaces. Typical actuator has a bandwidth up to 60 Hz, which is sufficient for flight control. But the low frequency mode of the flexible SAW may well be triggered by TE actuation, problem emerged when aircraft pitch or roll rate was sent back to flight control input. Sensors may not detect the pitch or roll rate built up since TE lift input bent up the wing at the execution time, controller will compensate this by over actuating the TE, by the time desired rate was achieved, control force was over-compensated and should the controller correct again, limit cycle oscillation is a possible outcome.

It is hence worthwhile to investigate the lift time history with TE input at various rates. TE input of 1 degree in 0.02s, 0.1s and 0.5s were tested. Lift results were shown in Figure 4-47, Figure 4-48 and Figure 4-49. Delay of total lift force reaching steady states changed from 0.15s to 0.6s.

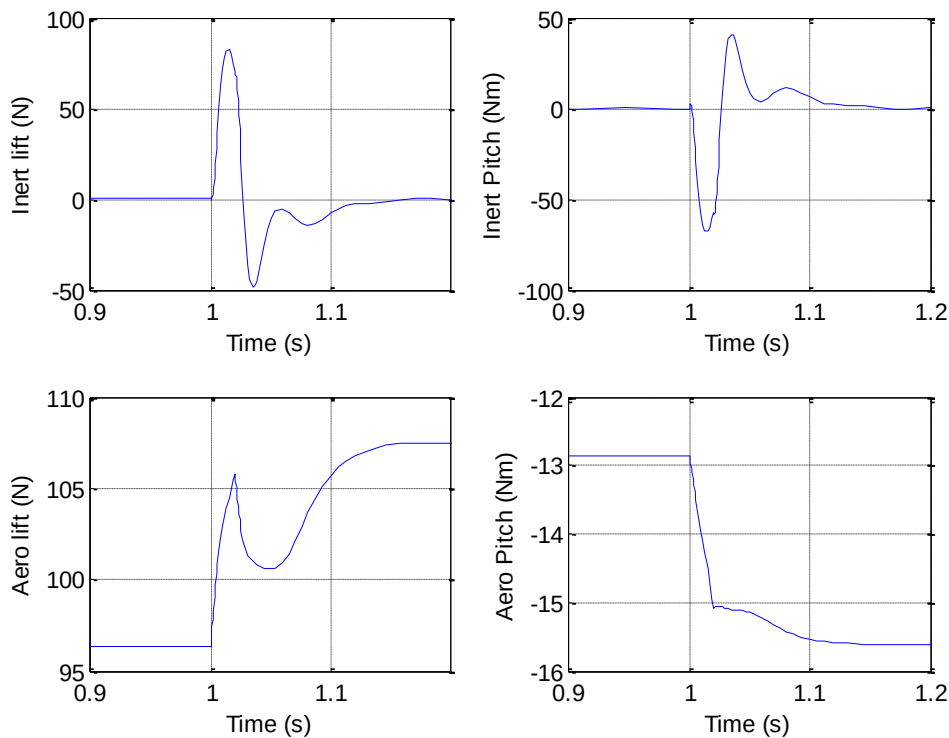


Figure 4-47 TE 1degree down ramp input in 0.02 second

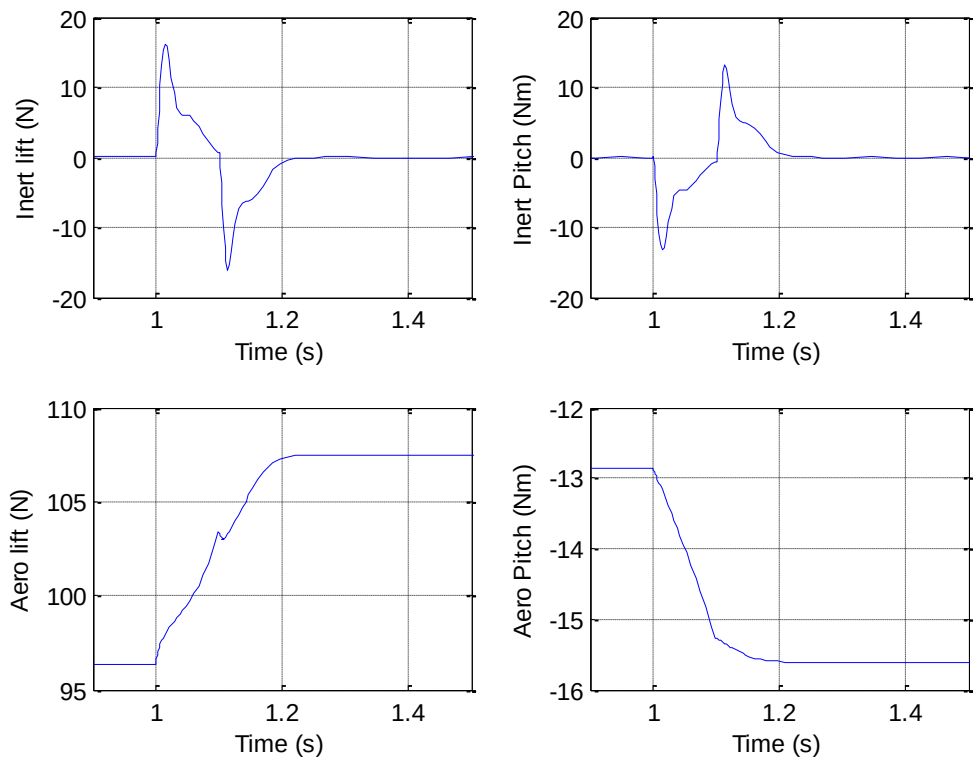


Figure 4-48 TE 1degree down ramp input in 0.1 second

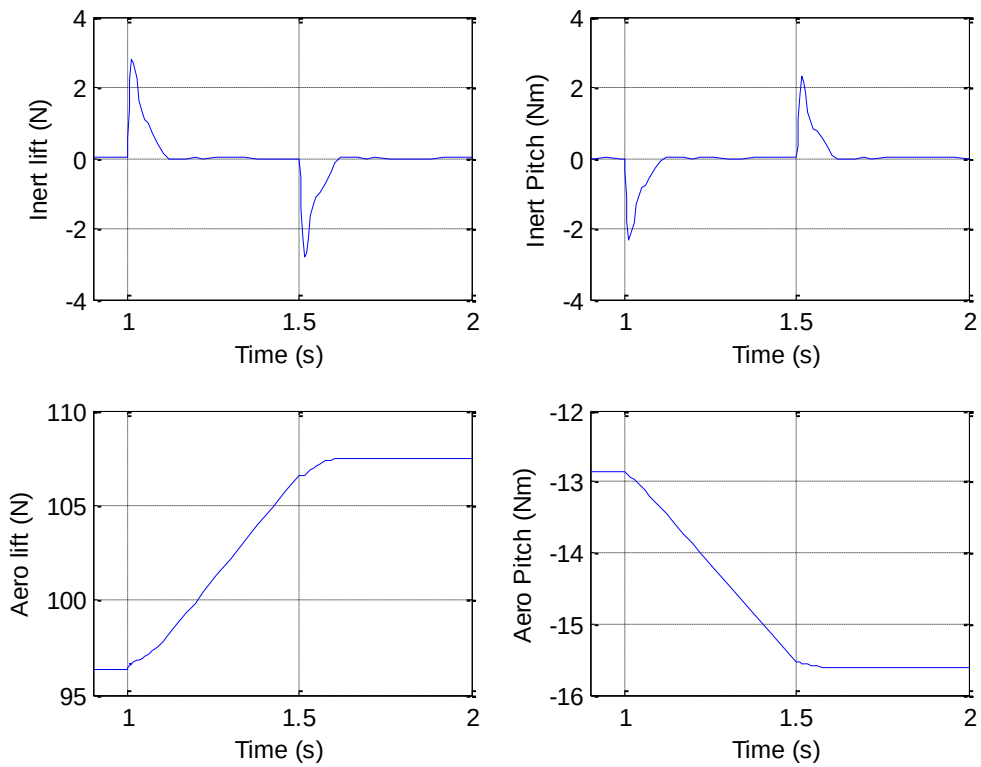


Figure 4-49 TE 1degree down ramp input in 0.5 second

Actuator dynamics are normally modeled by a second order system rather than simple ramp input. Bandwidth of actuator should be at least higher than the structure's 1st order natural frequency to ensure major structure vibration modes can be bypassed. In this way, aero-servo-elastic problem can be partially avoided. An eigen analysis of the SAW model can obtain its vibration mode and frequency (Table 4-13).

Table 4-13 1st order vibration mode frequency and time constant for SAW

1st Order	<i>Natural Freq</i> Hz	<i>Time Constant</i> s
<i>Plunge</i>	5.727	0.175
<i>Twist</i>	21.589	0.046
<i>Heave</i>	50.177	0.020

Further aero-servo-elastic analysis requires a closed loop model with aircraft attitude and rate feedbacks taken to calculate the flight control input. When this is considered, airframe flexibility induced response delay starts to destabilize flight control. Feedback signal filtering can be applied to prevent structural dynamic and actuator dynamic coupling. However, aero-servo-elasticity is not the focus of current study.

Closed loop flight dynamic analysis of the flexible aircraft will be investigated in this study, but a rigid body aircraft flight dynamic model has to be constructed beforehand to provide a analysis foundation and performance benchmark.

Chapter 5 Modelling and Analysis of a Rigid Body Aircraft

5.1 SAW aircraft initial design

The application of Seamless Aeroelastic Wing (SAW) technology to a small aircraft design is for low cost, light weight, high performance, etc. It uses a variable camber aerofoil to produce desirable twist on a lifting surface, consequently the required aerodynamic force and moment for aircraft manoeuvre. A SAW aircraft example is designated “D-47” which implies Cranfield University Demon aircraft modification in an “X-47” development manner. In order to investigate how performance, stability and control were affected by seamless control surfaces and structural flexibility of the SAW, both rigid and flexible cases models were developed with two kinds of control surfaces for comparison.

D-47A is the rigid flying wing aircraft with conventional hinged control surfaces. D-47B is the rigid flying wing aircraft with seamless control surfaces.

D-47HAW is the flexible flying wing aircraft with conventional hinged control surfaces. D-47SAW is the flexible flying wing aircraft with seamless control surfaces.

5.1.1 The example aircraft

The Demon aircraft was developed for demonstrating flapless flight control technology. Circulation control (CC) and fluidic thrust vectoring (FTV) were proposed to achieve this objective.

SAW technology would achieve flapless flight without heavy air compressor and pneumatic pipelines. In order to gain the maximum effect of SAW performance, a different aircraft planform has been selected. The un-tapered wing of adequate bending stiffness and desirable torsional stiffness has to be designed.

The new wing features:

1. Glass fibre skin with optimised layup for specific bending and torsional stiffness requirement
2. Seamless control surface mechanism design
3. Distributed control surface actuation with associated control law development.

The aircraft geometry breakdown and control surfaces' dimensions are shown in Table 5-1 and Table 5-2.

Table 5-1 SAW aircraft geometry breakdown

<i>D - 47 SAW Aircraft</i>			
Wing Data			
Span	4200	mm	
MAC	1175.6	mm	
Wing area	3.76	m ²	
Aspect ratio	4.7		
Taper ratio	1		
Quarter chord sweep	40	deg	
Root aerofoil section	NACA 0015		
Tip aerofoil section	NACA 0006		
Fueslage Data			
Max width	330	mm	
Max height	235.73	mm	
Length for	920	mm	
Length mid	1043	mm	
Length aft	540	mm	
Fueslage length	2503	mm	
Total length (inc fin)	3043	mm	
Total length (inc wing)	3044.2	mm	
Fin Data			
Height	600	mm	
MAC	503	mm	
Fin area	0.30	m ²	
Aspect ratio	1.19		
Taper ratio	1		
Quarter chord sweep	45	deg	
Root aerofoil section	NACA 0009		
Tip aerofoil section	NACA 0006		
Weight and Thrust			
Empty weight	66	kg	
Max takeoff weight	88	kg	
Max landing weight	80	kg	
Engine	Jet Cat P200SX		
	230	N	

Table 5-2 Control surface dimension

Control surfaces		
Elevon outboard		
Span (from wing tip)	453	mm
Chord (from trailing edge)	240	mm
Position of aero centre to CG	1618	mm
Deflection limits	±20	deg
Elevon inboard		
Span (from out elevon)	453	mm
Chord (from trailing edge)	240	mm
Position of aero centre to CG	1271	mm
Deflection limit	±20	deg
Leading edge		
Span (from wing tip)	906	mm
Chord (from trailing edge)	180	mm
Position of aero centre to CG	1695.2	mm
Deflection limits	±20	deg
Rudder		
Span (from fin tip)	580	mm
Chord (from trailing edge)	201.2	mm
Position of aero centre to CG	985.72	mm
Deflection limits	±20	deg

5.1.2 D-47 Mass and Inertia properties

The aircraft mass break down is listed in Table 5-3

Table 5-3 Mass break down for D-47 aircraft

Mass Breakdown	Original (kg)	Double (kg)	Modified (kg)
Structural items	11.94	23.88	36.38
Power supply, Sensors & Actuators	3.7	7.4	10
Main and nose landing gear	1.7	3.4	3.6
Avionics	3.6	7.2	7.2
Pneumatics system	4.3	8.6	0
Fuel system	1.2	2.4	2.4
Engine	2.5	5	5
Engine air intake	0.46	0.92	0.92
FTV system	2	4	0
Circulation Control system	1.3	2.6	0
Ballast (dead weight)	0.2	0.4	0.5
Total Empty Mass	32.9	65.8	66
Fuel	6.8	13.6	22
Maximum Take-Off Mass	39.7	79.4	88
Maximum Landing Mass	37.9	75.8	80

Since the freshly designed wing drastically changed aerodynamic centre (AC) and centre of gravity (CG) of the whole aircraft. It is crucially important to adjust the wing

5.1.2.2 Wing

The CG position and inertia of the wing is acquired as shown in Figure 5-1 by modelling a solid wing in Catia V5.

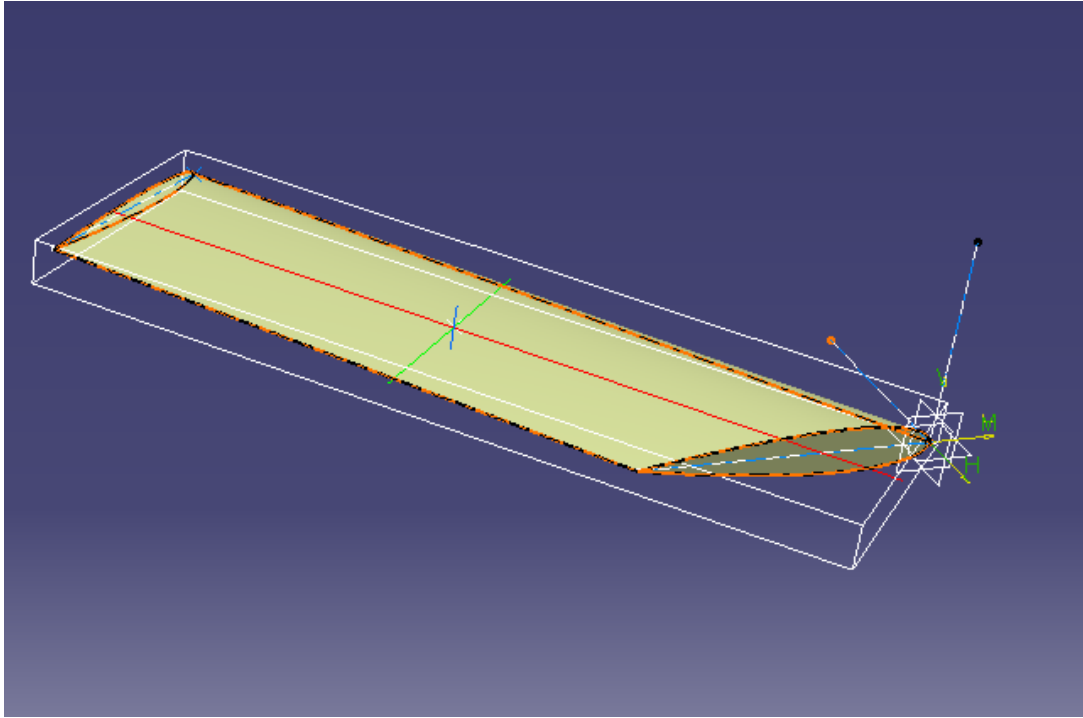


Figure 5-1 Catia inertia measurement output

Table 5-5 Wing mass and inertia properties

Component	Outer wing close	Unit
Area	2.24	m ²
Volume	2.19	m ³
Density	11	kg/m ³
Mass	24.06	kg/m ³
Gx	961.08	mm
Gy	1205.82	mm
Gz	0	mm
Ixx	3.86	kgm ²
Iyy	3.44	kgm ²
Izz	7.28	kgm ²
Ixy	-2.96	kgm ²
Ixz	0	kgm ²
Iyz	0	kgm ²

Because CG x position on the wing is relative to wing root section leading edge, a length of ΔL is added to wing CG x which refers to the aircraft nose datum. Wing position relative to the fuselage is thus determined by the value of ΔL .

5.1.2.3 Joining Wing and Fuselage

The design procedure is to add a pair of elastic wings to D-47B fuselage. Wing mounting position is therefore liable to its effect on both AC and CG location of the aircraft in order to satisfy a determined static margin.

$$\text{Static margin is defined as } K_n = -\left(\frac{dC_{MCG}}{dC_L}\right) = \frac{Gx_{AC} - Gx_{CG}}{MAC}.$$

Equation 5-1

Where

$$Gx_{AC} = \frac{F_f \times L_{AC-f} + F_w \times L_{AC-w}}{F_f + F_w} \qquad Gx_{CG} = \frac{M_f \times L_{CG-f} + M_w \times L_{CG-w}}{M_f + M_w}$$

Equation 5-2

All variables were known except

$$L_{AC-w} = \Delta L + \frac{1}{4} MAC \qquad L_{CG-w} = \Delta L + 961mm$$

If it starts from $\Delta L = 0$, $Gx_{AC} = 657mm$, $Gx_{CG} = 1161mm$, $Kn < 0$, aircraft is statically unstable. However, increase value of ΔL will push the wing towards the tail of the aircraft. This would shift the total CG of the aircraft backwards (increase Gx_{CG}). Meanwhile, increase of ΔL also moves the AC backwards but at a faster speed (increase Gx_{AC}).

Hence it is possible to find out a ΔL value that satisfies a preset positive static margin.

Equation 5-1 and Equation 5-2 were put into Excel spreadsheet; goal seeking function is used to find ΔL value which would give 5% static margin.

In this way, ΔL was found to be 1242mm. Final geometry of the D-47B SAW aircraft can now be fixed as shown in Figure 5-2.

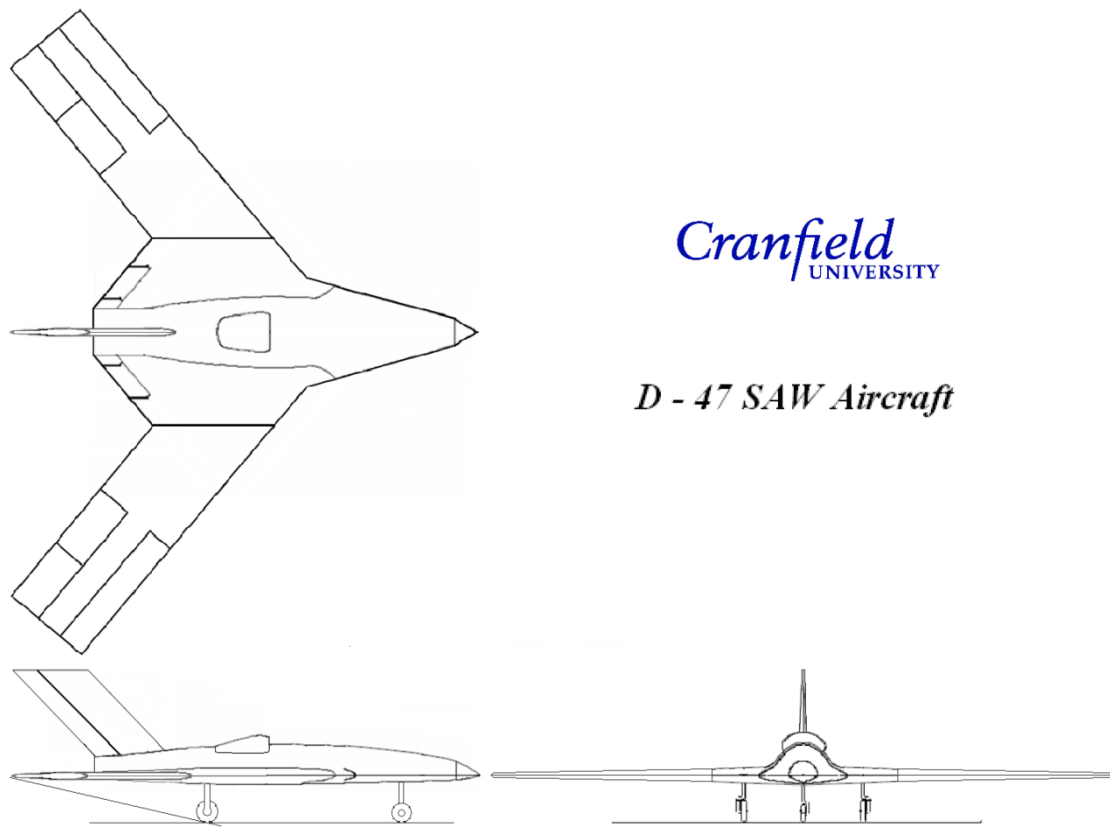


Figure 5-2 Schematics of D-47 example aircraft

5.1.2.4 Inertia properties

The 5 percent static margin fixed the relative positions between the SAW wing and fuselage. Final mass and inertia properties of D-47B are show in Table 5-6, CG reference point is the aircraft nose.

Detailed calculation of inertia properties, please see appendix.

Table 5-6 D-47B CG position, mass and inertia properties

<i>Mass & Inertia Properties</i>		
<i>MTOW</i>	88.00	kg
<i>CG_x</i>	1.74	m
<i>CG_y</i>	0.00	m
<i>CG_z</i>	0.03	m
<i>I_x</i>	70.01	kgm ²
<i>I_y</i>	109.27	kgm ²
<i>I_z</i>	113.66	kgm ²
<i>I_{xy}</i>	0.00	kgm ²
<i>I_{yz}</i>	0.00	kgm ²
<i>I_{xz}</i>	1.57	kgm ²

5.1.3 D-47B Aerodynamics properties

No flight test or scaled model wind tunnel tests were performed for this freshly designed SAW aircraft concept due to budget limit. Therefore aerodynamic coefficients and derivatives were estimated using empirical methods with the aid of existing computer software Tornado VLM.

5.1.3.1 Tornado Vortex Lattice Method

Tornado is a 3D-vortex lattice program with flexible wake, which outputs 3D forces acting on each panel, aerodynamic coefficients in both body and wind axis. Stability derivatives with respect to angle of attack, angle of sideslip, angular rates and rudder deflections. (Section 3.4.4)

Tornado axis system is different from flight dynamic axis system. In Tornado axis, X backward, Y right and Z up are positive. Roll left, pitch up and yaw left are positive. Hence, lift, drag, roll and yaw moment are in opposite sign.

5.1.3.2 Construct SAW aircraft model in the Tornado environment

Aircraft modelling in the Tornado environment is straight forward. But certain barriers need to get around with certain assumptions.

Since the fuselage of the SAW aircraft is the same as the Demon, it is assumed that the nose cone would not produce any lift, therefore wing partitions only comprises inboard tapered wing with flap, outboard fuel tank wing, and outboard elevon wing.

As initial assessment showed yawing moment coefficient due to sideslip angle C_{n_β} is positive, i.e. unstable. It was decided that a fin was required to provide sufficient directional stability. Subsequent design modification was performed in the Tornado environment as shown in Figure 5-3. It is assumed that CG position calculated in the previous section remains. Further aerodynamic complication would be demonstrated with fixed CG.

As conceptual design phase doesn't need highly accurate aerodynamic results, grid size chosen was approximately $0.1 \times 0.1\text{m}$, so that mesh density was kept reasonably low to save computational time.

Figure 5-4 shows vortex lattice covers each tornado wing element, which resembles AAW aircraft's aerodynamic feature. Pressure distribution is calculated and shown in Figure 5-5.

The full set of aerodynamic stability and control derivatives were obtained. From the span-wise lift and non-dimensional C_L plot, classic elliptical lift distribution can be observed, sweep back wing inboard lift lose due to span-wise flow is also visible. Abrupt C_L drop at centre wing indicated fuselage presence and its effect. C_L increase dramatically at wing tip due to area reduction.

These plot will be particularly useful when wing twist along the span start to affect lift generation.

Detailed derivatives and their definitions are in Appendix B.

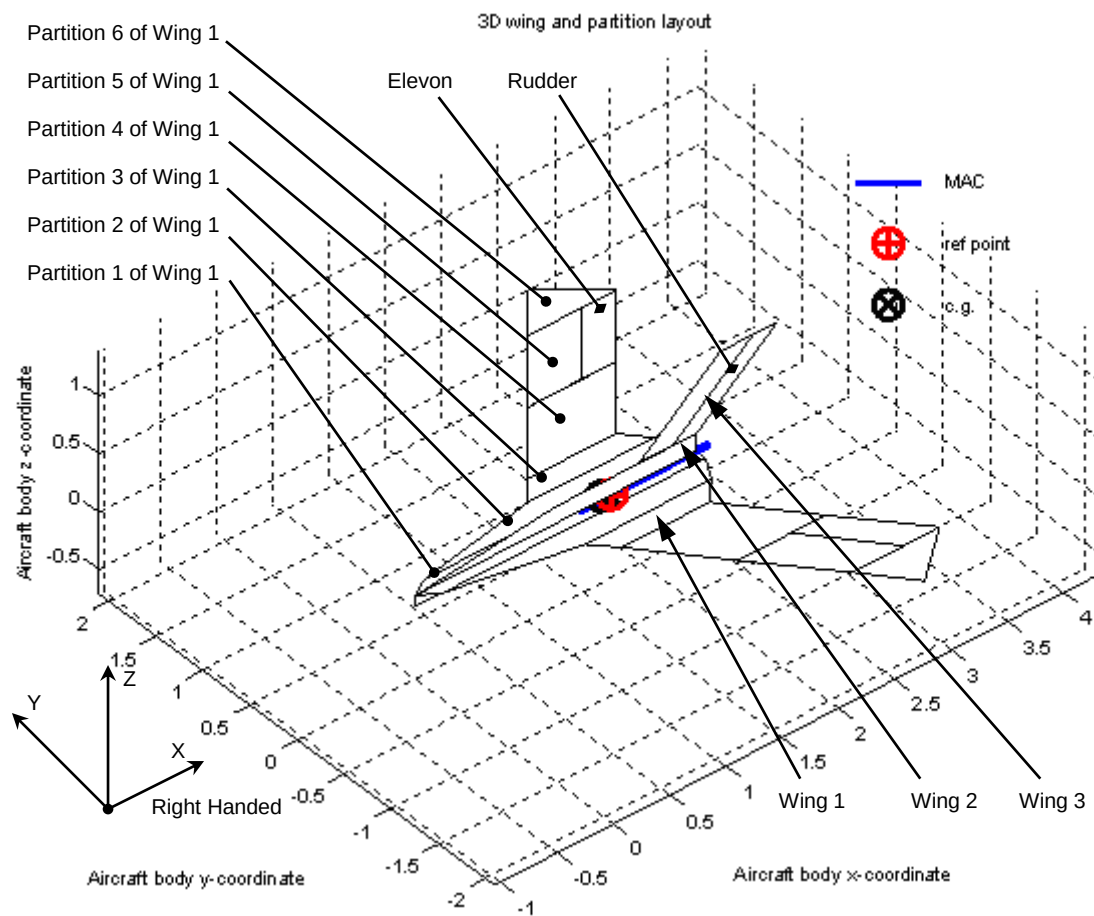


Figure 5-3 Tornado 3D aircraft element definition

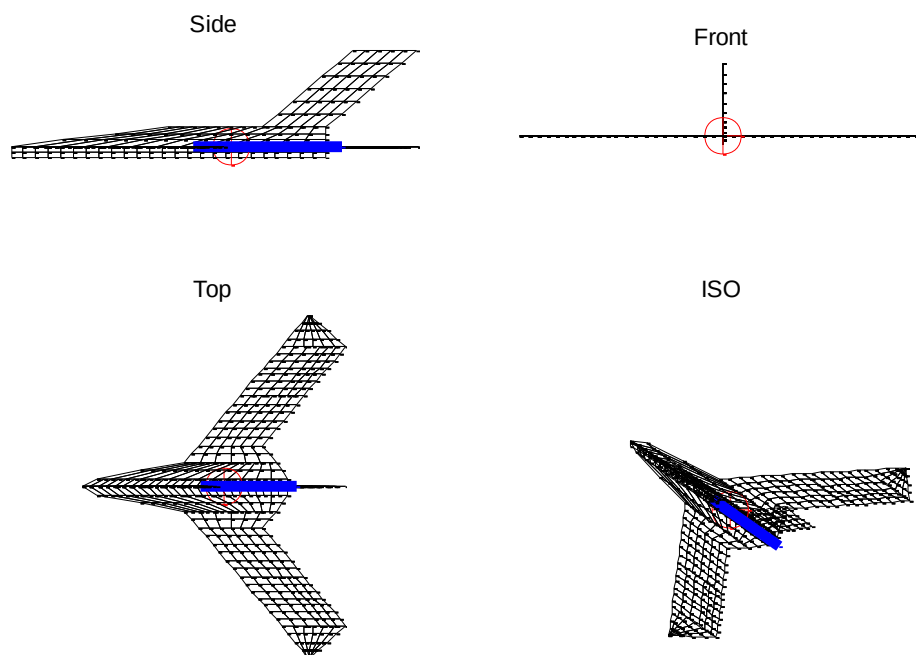


Figure 5-4 Tornado D-47 vortex panels

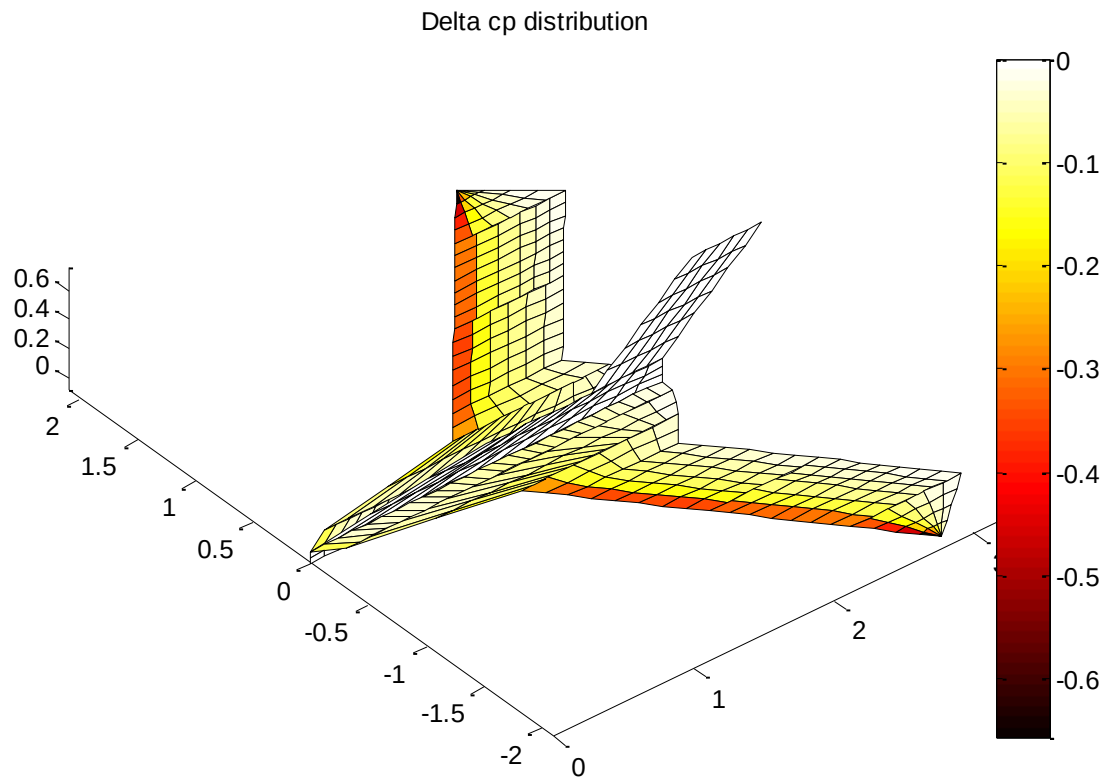
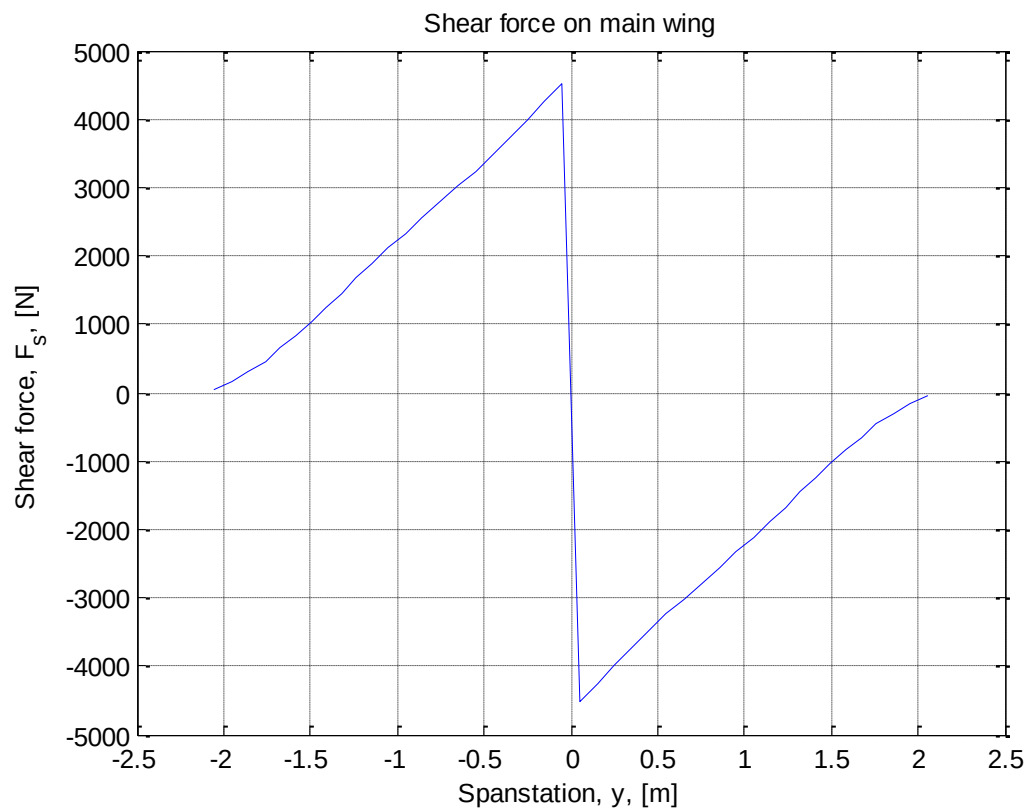


Figure 5-5 D-47 pressure coefficient distribution at 60m/s



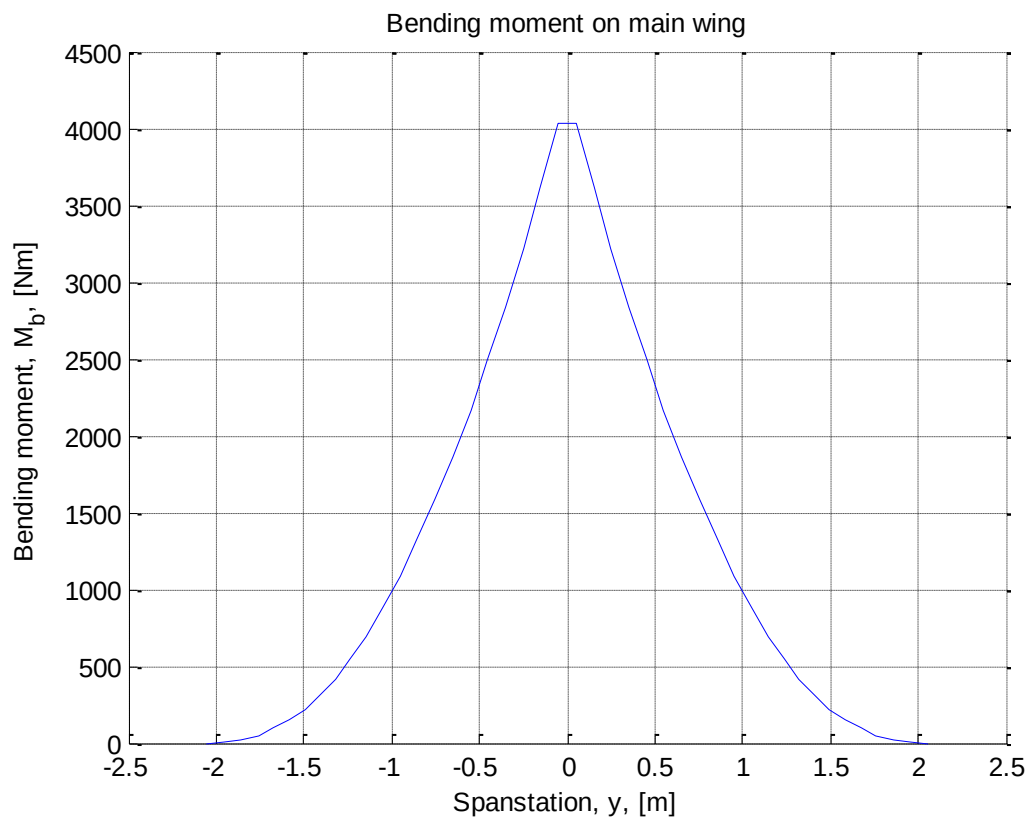
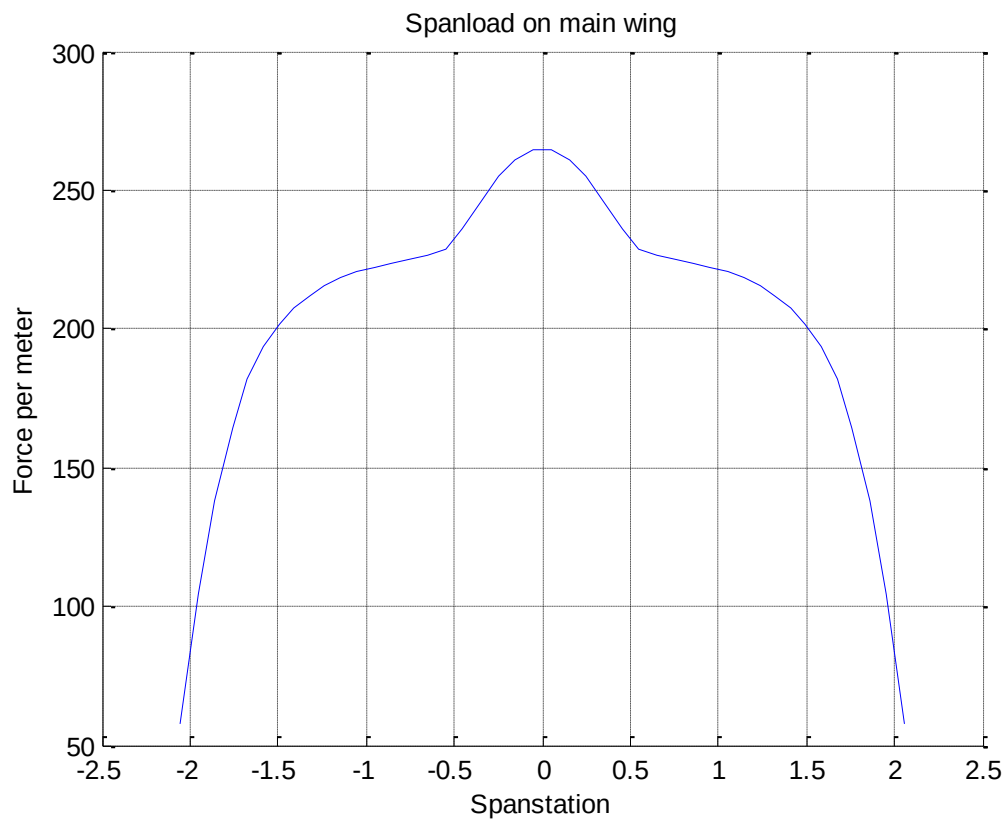


Figure 5-6 Span wise shear force and bending moment by Tornado VLM



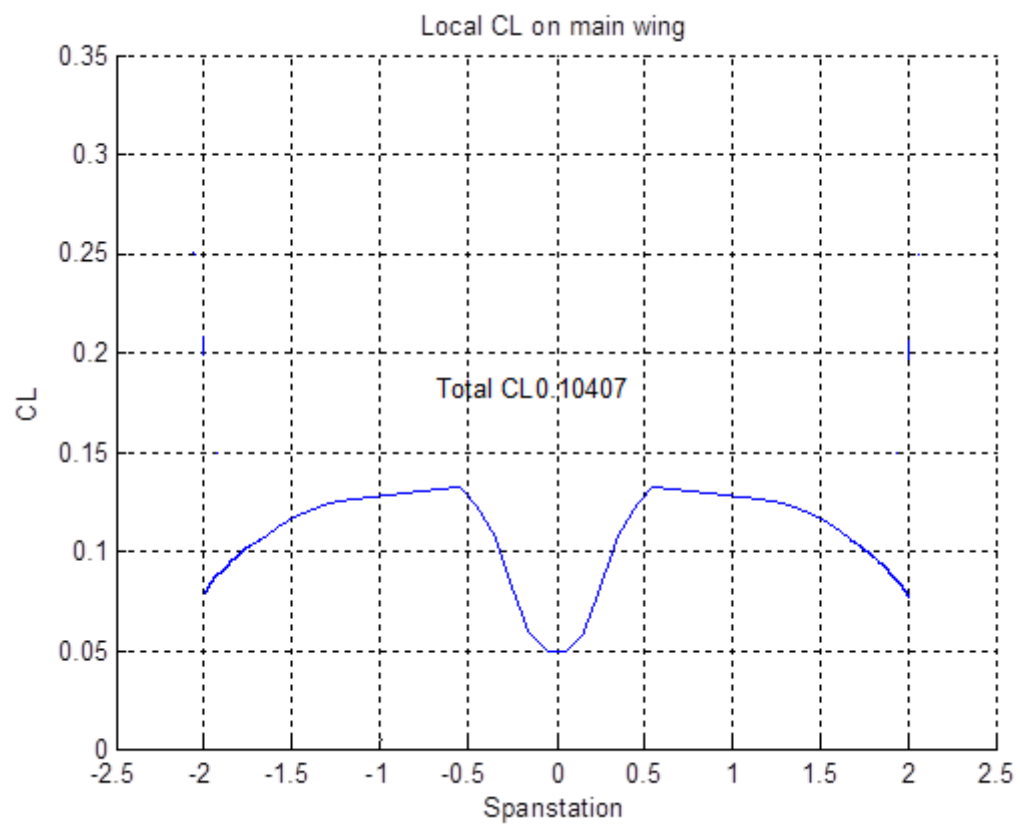


Figure 5-7 Span wise lift and CL plot by Tornado VLM

5.2 D-47 Modelling

Mathematical modelling of the D-47 aircraft was carried out using the Matlab Simulink Aerospace block set.

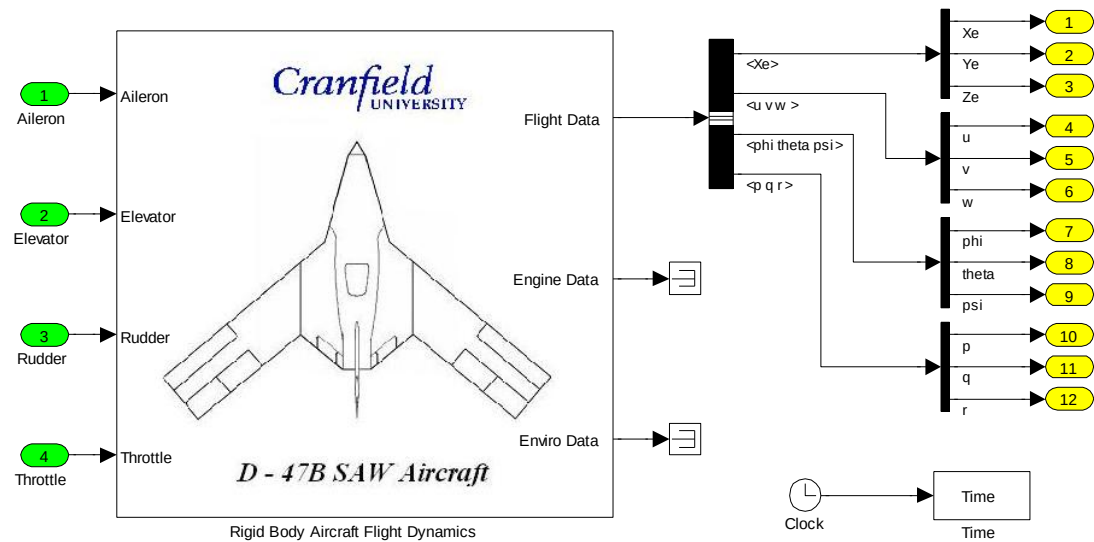


Figure 5-8 Simulink block diagram for baseline aircraft

Inputs to the AAW aircraft are coming from the pilot (or autopilot). Flight control commands are roll, pitch, yaw and throttle position.

Output of the AAW aircraft are position and attitude in space (X_e , Y_e , Z_e , ϕ , θ , ψ), plus translational and angular velocity about three body axis (u , v , w , p , q , r).

D-47 aircraft model comprises:

- Flight control computer model,
- Aerodynamics model,
- Propulsion model,
- Environment model and
- 6 DOF Euler angle equations of motion model.

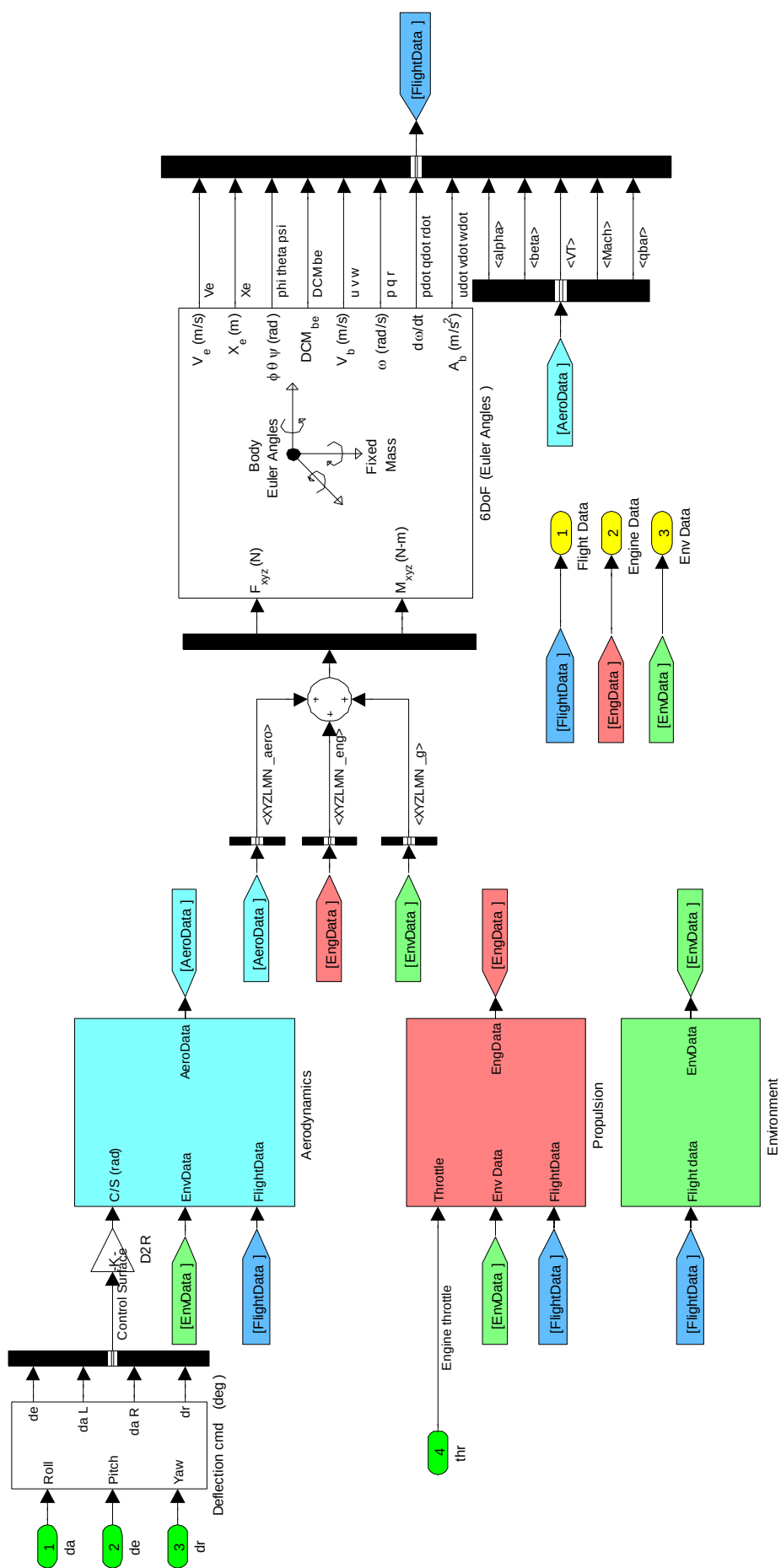


Figure 5-9 D-47 aircraft Matlab Simulink model

5.2.1 Flying control model

The reason for flight control command block to be added is that D-47 aircraft design was a blended wing body configuration, no separated elevators exist on the airframe; ailerons are therefore actuated independently to provide both pitch and roll control. In another word, they are elevons.

When a flight command is given, a roll, pitch and yaw response from the aircraft would be expected. Flight computer has to translate pilot command into co-ordinated control surface deflection command. Control derivatives were calculated by Tornado (Appendix B). Simulink model for this translation is shown in Figure 5-10. Pitch and roll control command were mixed into separated elevon actuator command.

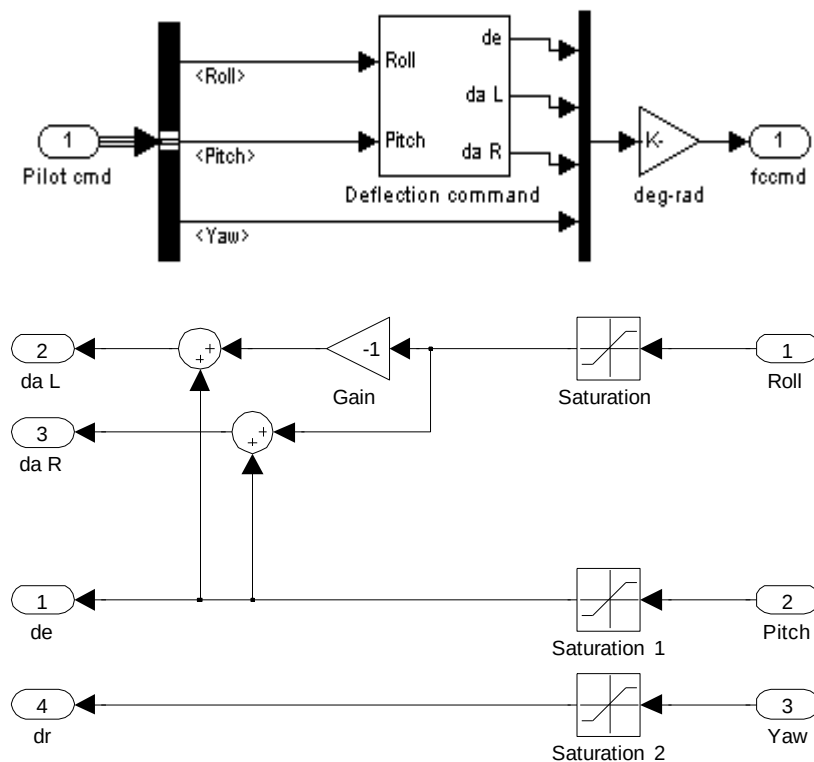


Figure 5-10 Flight computer command for mixed control input

5.2.2 Aerodynamics Model

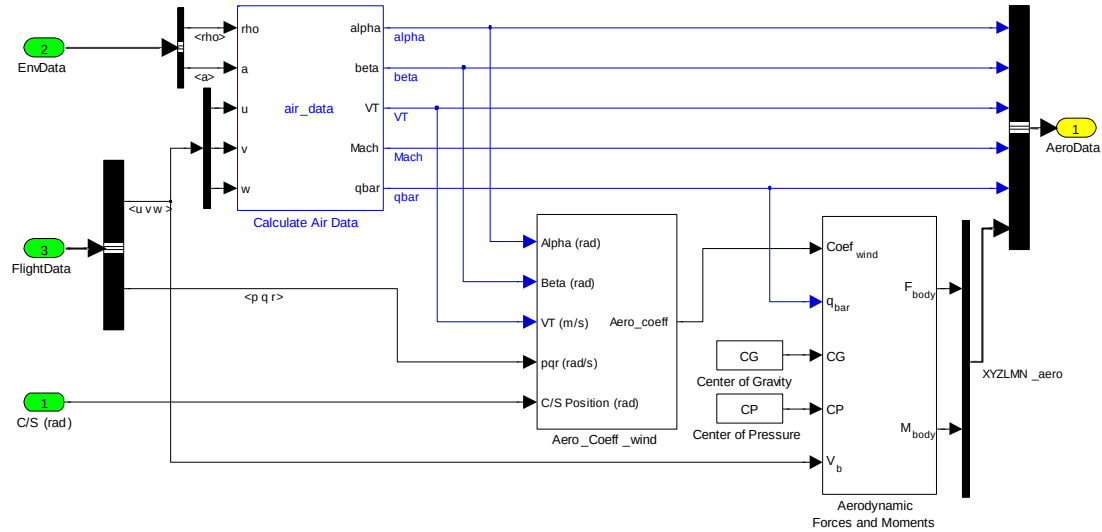


Figure 5-11 Aerodynamics block

Aerodynamic block has 3 subsystems:

- Aerodynamic coefficients in wind axis
- Air data calculation, and
- Aero forces and moments

5.2.2.1 Aerodynamic coefficients in wind axis

Aerodynamic coefficients and derivatives were obtained from Tornado VLM, They were categorised into three types.

- **Datum coefficients:** Plain C_D , C_Y , C_L , C_l , C_m , C_n aerodynamic coefficients and their derivatives with respect to angle of attack (α) and side slip angle (β). These data define the aircraft static stabilities.
- **Damping derivatives:** These data represent effects of rotational speed on forces and moments. Generally damping terms produce negative forces and moments to stop the aircraft rotational motion. These data define the aircraft dynamic stabilities, together with datum coefficients; they were known as stability derivatives.

- Actuation increments: Also known as control derivatives, conventional control input would include aileron, elevator and rudder. In D-47's case, there is no elevator; input comes from elevon δ_{aL} and δ_{aR} for pitch and roll control, rudder δ_r for yaw control.

These data were then put into force and moment equations to calculate aerodynamic forces and moments applied on the aircraft.

5.2.2.2 Aero forces and moments

This block sums up all coefficients and multiplies by different reference length/area and dynamic pressure to calculate total forces and moments, then transforms the forces and moments from wind axes to body axes before feed them to the EOM. (Appendix B4)

$$F_{wind} \equiv \begin{bmatrix} -D \\ -Y \\ -L \end{bmatrix} = DCM_{wb} \cdot \begin{bmatrix} X \\ Y \\ Z \end{bmatrix} \equiv DCM_{wb} \cdot F_{body}$$

5.2.3 Propulsion model

Propulsion model simulate the JetCat P200SX micro jet engine that powers the D-47 (Figure 5-12). The model was simplified assuming no thrust delay, no compressor surge, thrust line coincides with airplane x-axis through CG, mass flow rate increase due to fuel injection is negligible and fuel consumption does not shift aircraft CG position.

The specification of JetCat P200SX micro jet engine is shown in Table 5-7

Maximum static thrust is 230N. Thrust available in flight is given as

$$\text{Thrust in flight} = 230 \times \left(1 - \frac{V_t}{V_{jet}} + \frac{V_t^2}{R^2}\right) \times \sigma$$

Where V_t is flight true air speed, V_{jet} is engine jet velocity, R is compressor ram effect factor, σ is relative air density ratio at altitude. Simulink model of this mathematical jet engine expression is shown in Figure 5-13.



Figure 5-12 JetCat P200SX micro jet engine

Table 5-7 JetCat P200SX micro jet engine specification

<i>JetCat P200SX Micro Jet Engine Spec</i>	
Thrust	230N at 112,000 1/min
Weight	2370g including launch control
Diameter	130mm
Overall length	350mm
RPM	33000 - 112,000 1/min.
Exhaust gas temperature	max. 750°C
Fuel Consumption	730ml/min (full power)
Jet flow velocity	458.33 m/s
Mass flow rate	0.5018 kg/s
Fuel	Jet A1, kerosene, petroleum
Lubrication	5% fully synthetic. Turbine Oil
Maintenance interval	50 hours
Price	3861 Euros

This simplified jet engine model would provide sufficient data for flight performance assessment, in the mean time saving computing power for more complex aeroelastic calculation in the future.

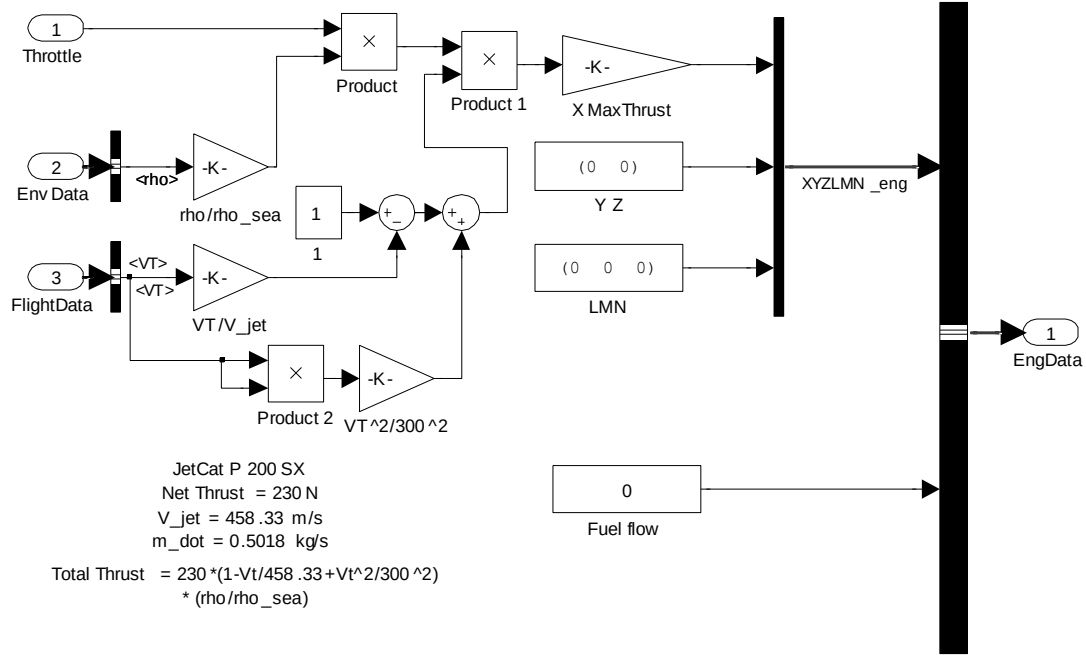


Figure 5-13 Micro jet engine Simulink implementation

5.2.4 Environment model

Environmental effects are atmospheric property, gravity and wind turbulences.

Atmospheric property is based on the ESDU data 77022, table 11.1 and 11.3.

Gravitational acceleration is g calculated by

$$g = \frac{G \times M_{earth}}{r^2}$$

Where G is Universal gravitational constant, M_{earth} is mass of the earth; r is earth average radius plus flight altitude.

The force and moment due to gravity is calculated and transformed from earth axis to aircraft body axis via DCM.

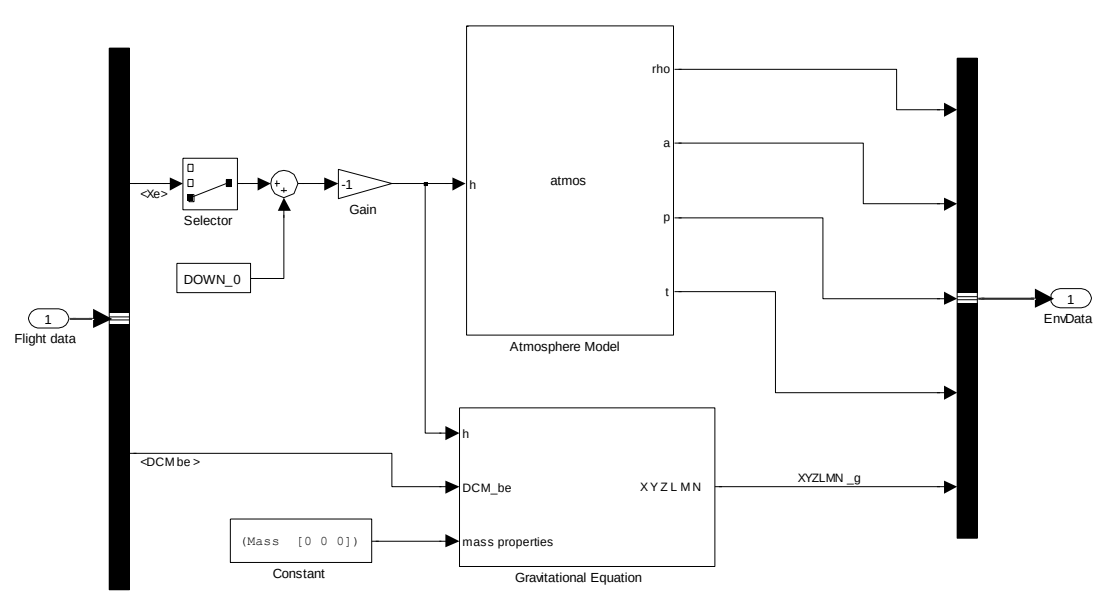
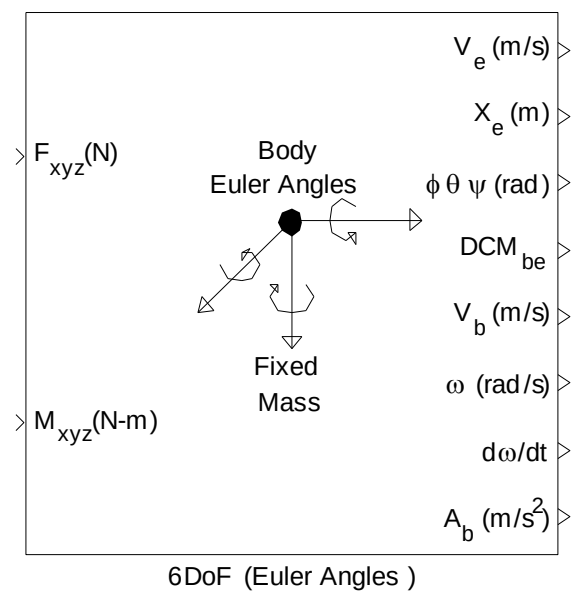


Figure 5-14 Environment input

5.2.5 6 DOF Euler angle EOM model



The 6DoF (Euler Angles) equation of motion block considers the rotation of a body-fixed coordinate frame about an Earth-fixed reference frame. The origin of the body-fixed coordinate frame is the centre of gravity of the body, and the body is assumed to be rigid, an assumption that eliminates the need to consider the forces acting between individual elements of mass.

Details of calculation from force and moment input to linear and angular motion output refer to section 3.2.1.

5.3 Performance and stability results of D-47

D-47 is a baseline aircraft for SAW technology demonstration, in which, performance analysis does not emphasize range, endurance, takeoff and landing characteristics. The seamless control surface and flexible wing feature would have much bigger impact on trimmed flight attitude, turn rate and roll rate of the aircraft. For comparative study, the rigid D-47A with conventional hinged control surfaces was analyzed first. Flight envelope, thrust map, V-n diagram, turn rate and roll rate were calculated in this case to provide the performance benchmark.

5.3.1 Trimmed Level Flight Envelope

Trimmed level flight was attempted over a wide speed and altitude range using Matlab trim routine, trim variables are AOA, engine thrust and elevon deflection angle. Results of trim input over speed range 30m/s to 120m/s and altitude of 0km, 2.5km and 5km were show in Figure 5-15.

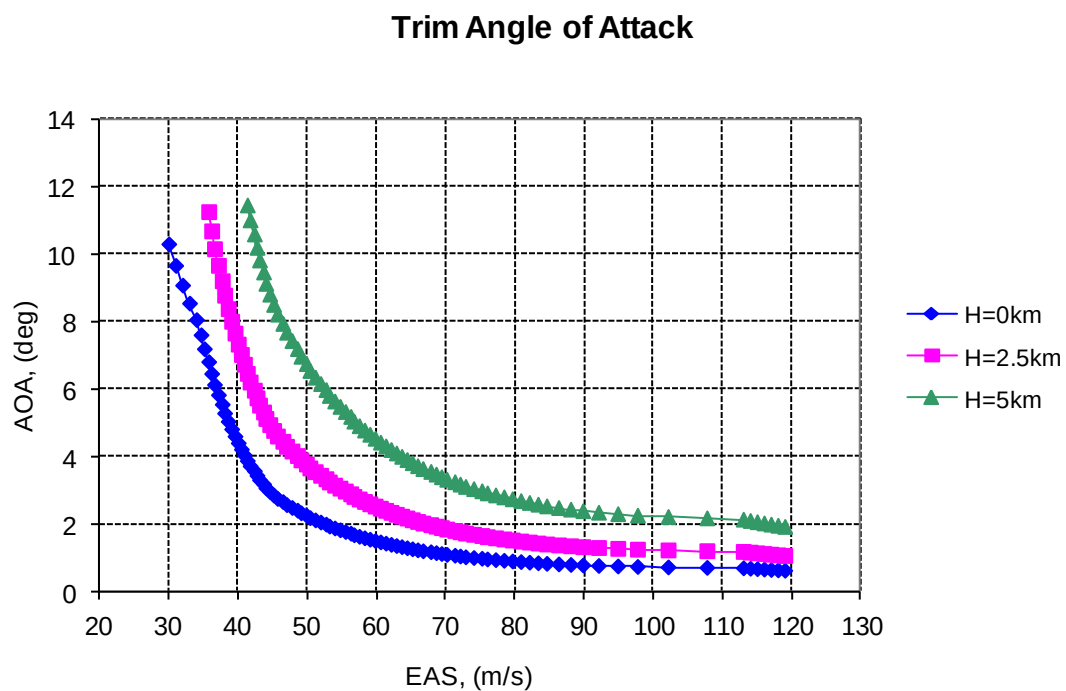
Engine net thrust, maximum AOA and elevon deflection were the limit to draw the boundary of the trimmed level flight envelope (Figure 5-16). Aircraft can reach flight conditions out of this envelope but can not maintain in there.

A very interesting feature on the flight envelop is that the low speed limit is not the wing stall limit but elevator deflection limit, i.e. full lift potential was not exploited. This is due to the seamless trailing edge control surface design has a max deflection of 20 degree which can not balance the pitching moment at higher AOA. For rigid D-47A aircraft, the solution would be to increase elevon area or deflection angle to provide more pitch

control moment. However, for the seamless control surface it is difficult to do due to its design. Also if the wing is flexible, coupled bending-twist deformation would move the pressure centre forward and result a smaller pitch down moment. This problem may not even exist.

A debate was raised at this point, whether pitch control should be powerful enough to achieve very high AOA to fully exploit aircraft's lift potential and extend flight envelop. Or should it be less powerful so that the wing can never stall at high speed and give extra safety margin. Perhaps aircraft's mission requirement will settle this argument one thing for all.

Eight flight conditions were selected at the boundaries of the flight envelope for aircraft stability and control assessments in section 4.3.6 and section 4.3.7.



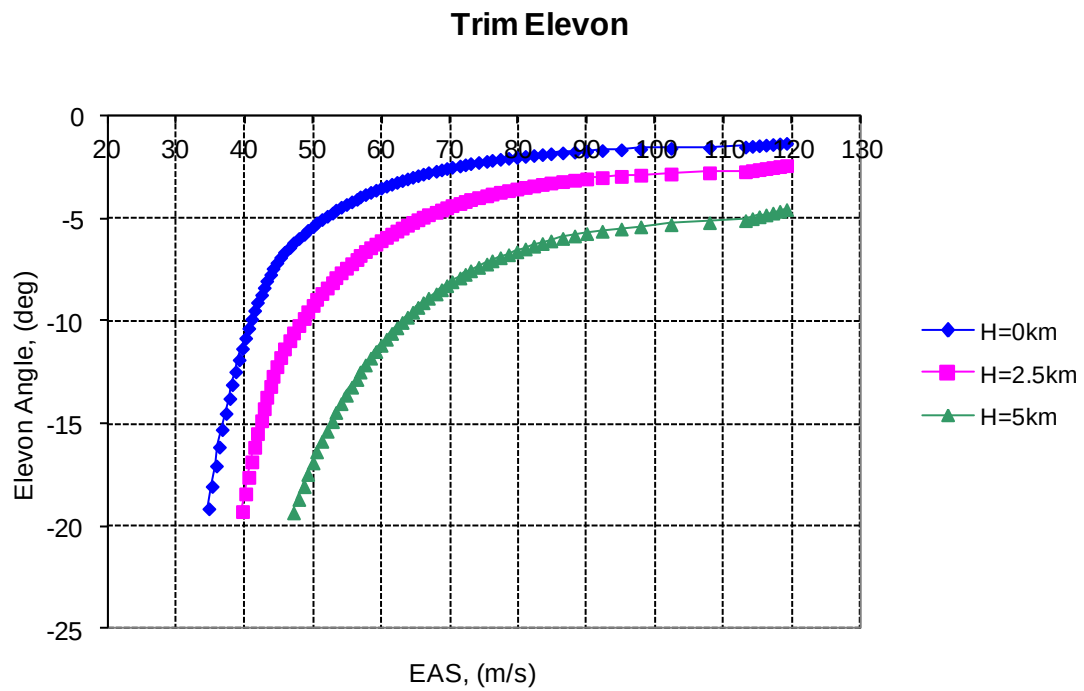
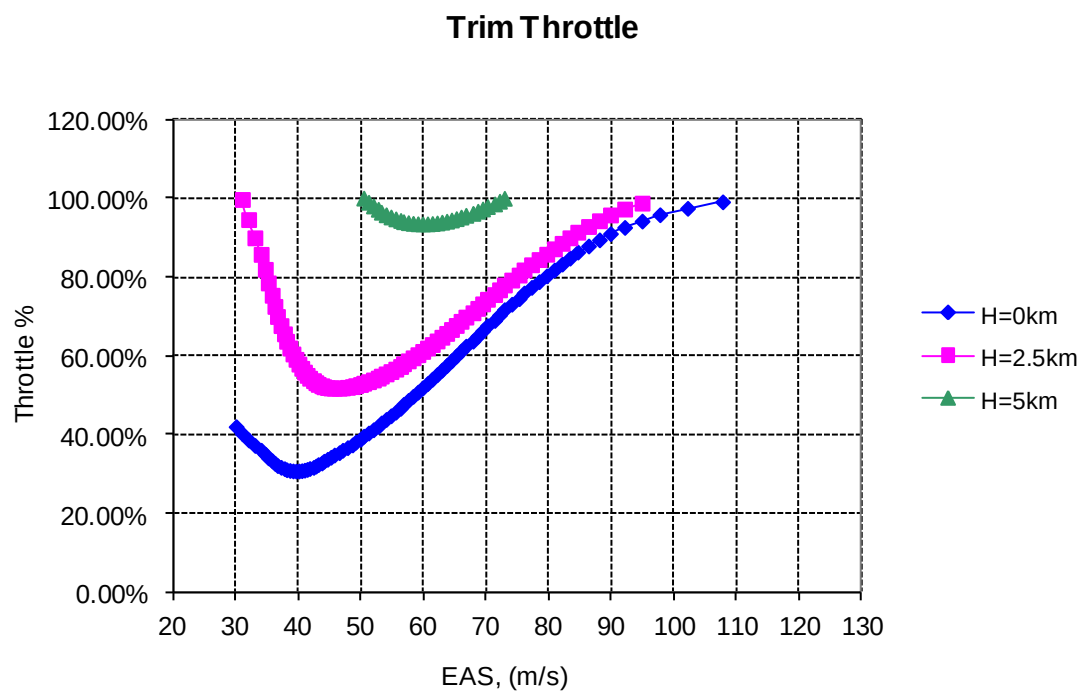


Figure 5-15 Trimmed level flight inputs for three different velocities and altitudes of interest

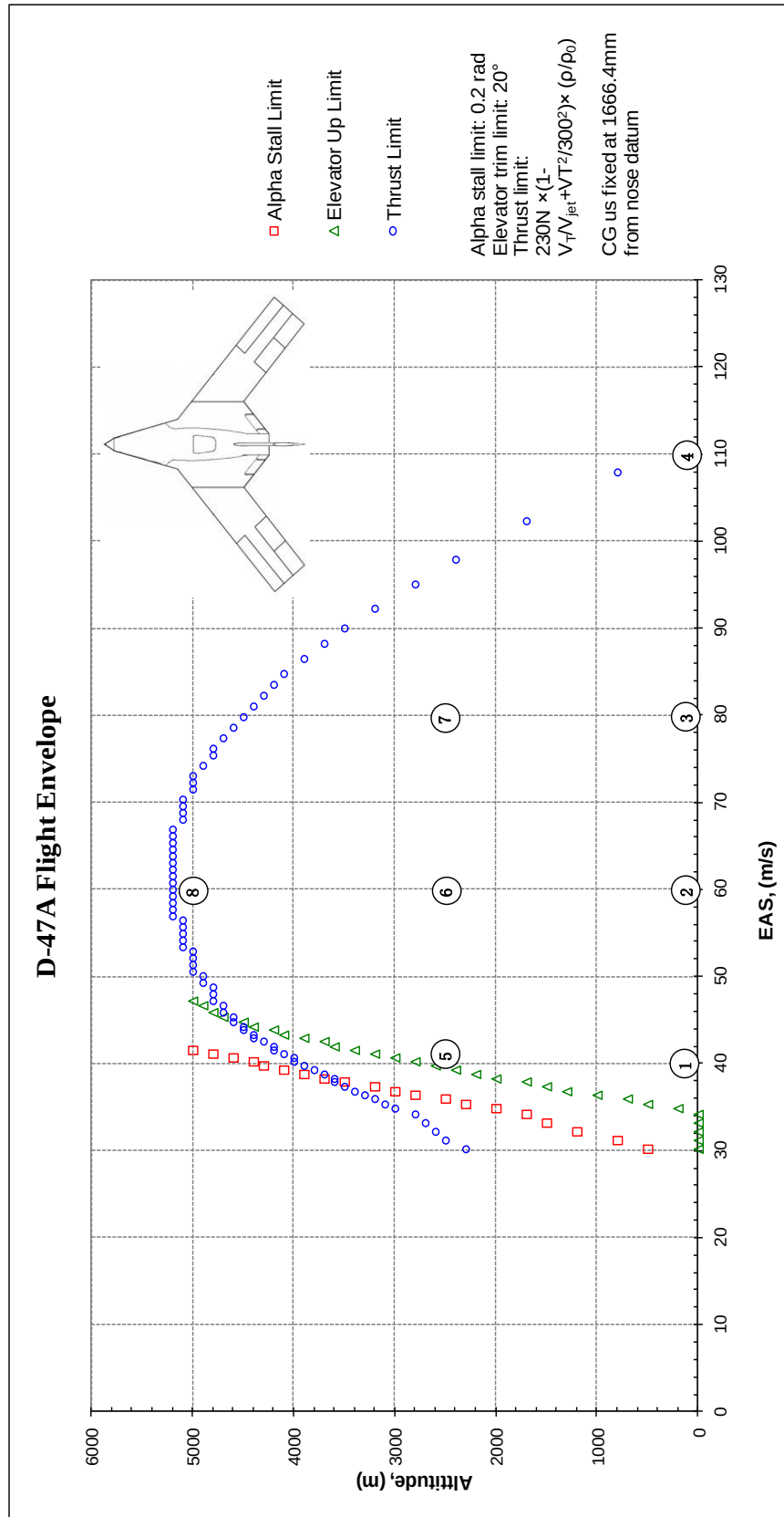


Figure 5-16 Rigid D-47A Flight Envelope

5.3.2 Thrust Curve

Required thrust (TR) was calculated from level flight drag. Tornado VLM calculation provides:

Zero-lift drag $CD_0 = 0.00672$

Induced drag $CD_\alpha = 0.77 \times \alpha^2 + 0.00014 \times \alpha + 0.00018$

Trim drag due to elevon deflection $CD_{\delta_e} = 0.051 \times \delta_e^2 + 0.00011$.

Available thrust (TA) is defined by the JetCat P200SX micro jet engine; net thrust was calculated with consideration of density height and intake ram effect (Figure 5-17).

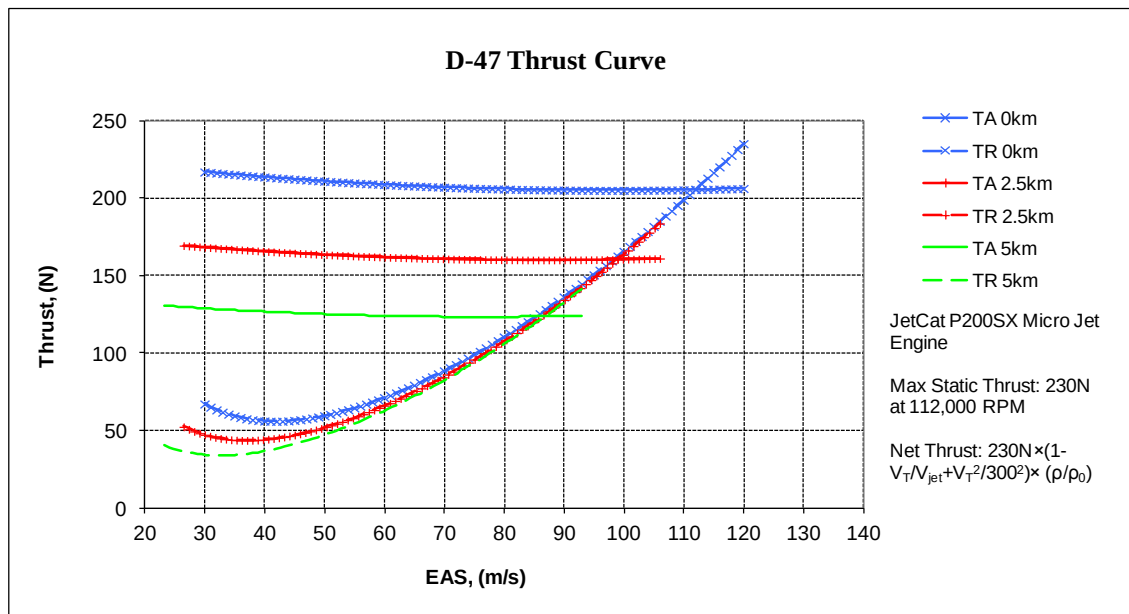


Figure 5-17 D-47 Thrust map

A side note is that, from empirical equation $CD = CD_0 + \frac{CL^2}{\pi e AR}$, Oswald efficiency factor e is found to be 0.95 for D-47 aircraft plan form design.

5.3.3 V-n Diagram

Load factor limits on the airframe were set in the range of -2.3 to +4.2 due to gust consideration. Speed is limited by the engine thrust and corner velocity V_c was the intersection of stall limit and load limit.

Again elevator limit caused reduction of envelop and changes corner velocity.

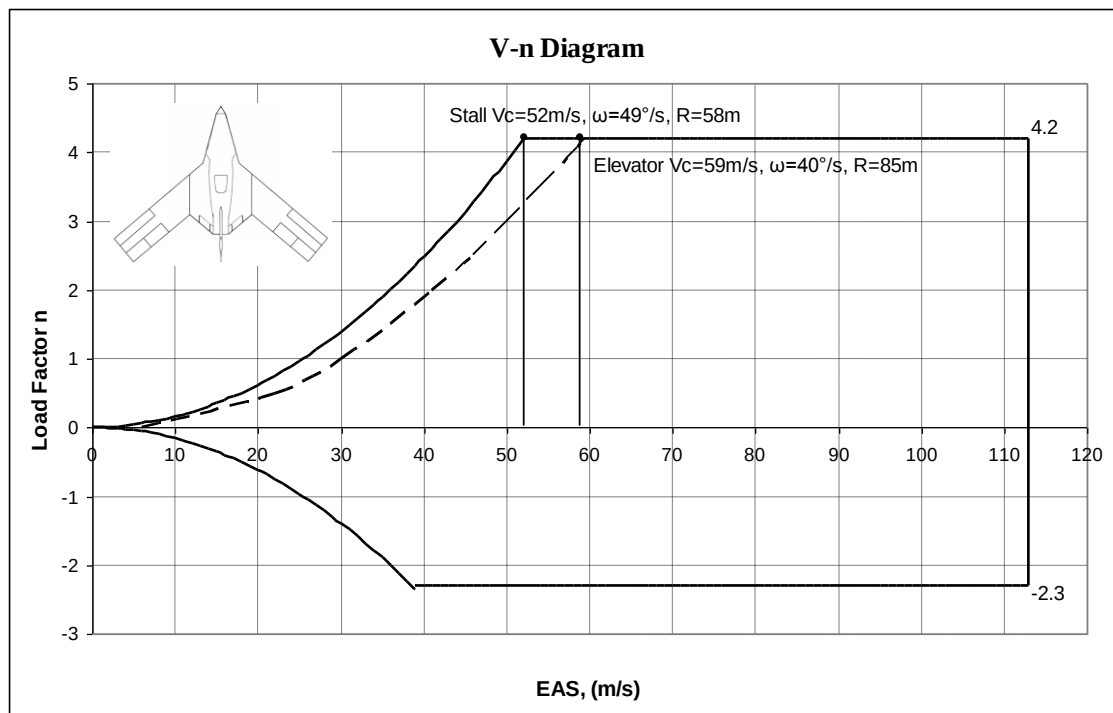


Figure 5-18 D-47 V-n diagram

5.3.4 Turn Rate

Turn rate calculation covers velocity range 0 to 130 m/s and load factor 1 to 9.

Thrust; stall lift (AOA), and airframe strength together shape the boundary of the plot. Elevator deflection once more decreases the achievable rate of turn. Interestingly, the engine thrust is more than enough to maintain maximum turn rate, therefore sustained turn rate (STR) is the same as instantaneous turn rate (ITR).

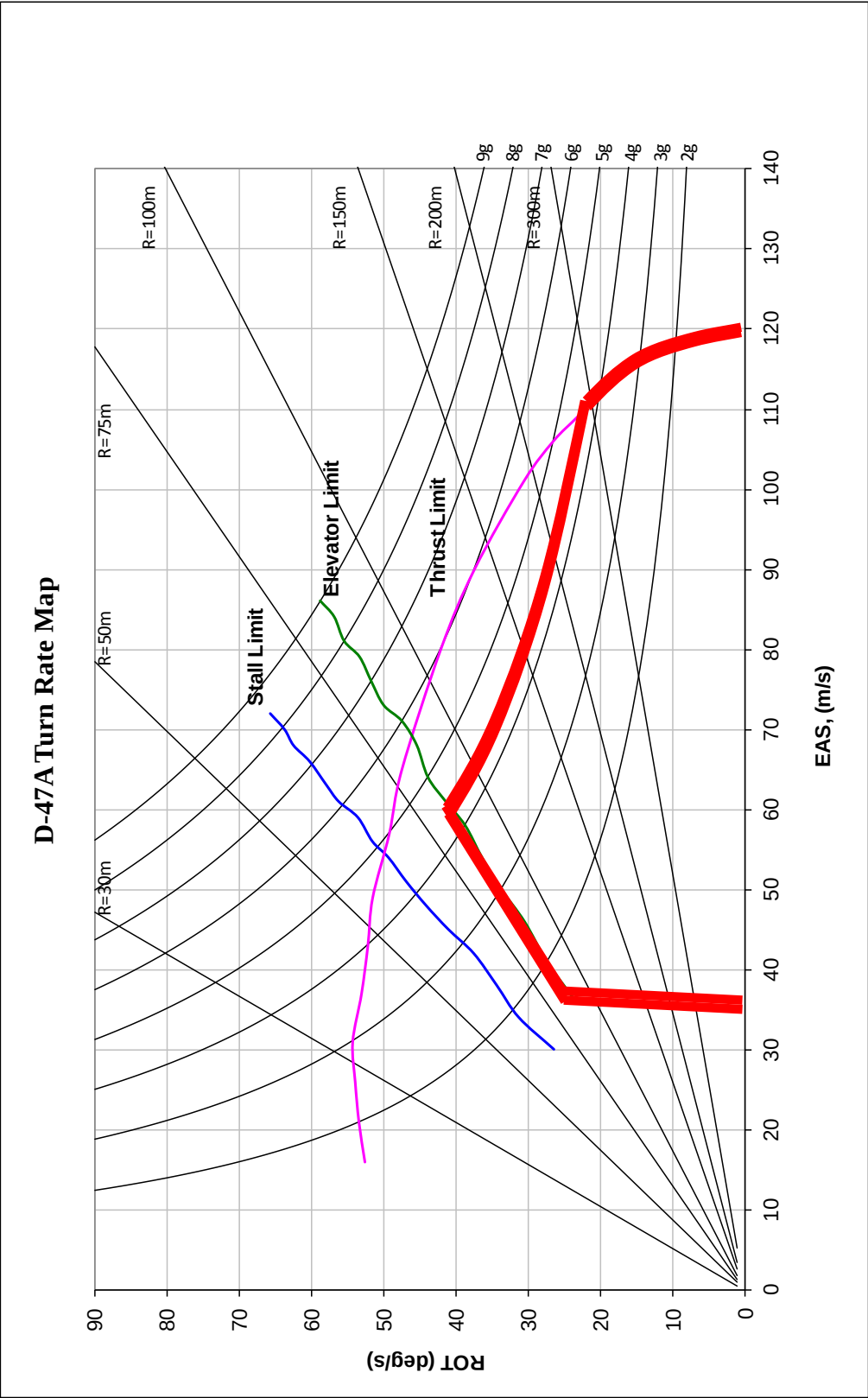


Figure 5-19 Turn rate map

5.3.5 Roll Rate

Roll rate calculation covers velocity range 0 to 130 m/s. Roll rate achievable increase linearly with velocity, meanwhile adverse yaw effect caused by aileron input and roll motion needs to be alleviated by rudder input. A fixed ratio of rudder compensation with regards to aileron input is calculated by the following equations.

$$\begin{aligned} L_p \times p + L_r \times r + L_\xi \times \xi + L_\zeta \times \zeta &= 0 \\ N_p \times p + N_r \times r + N_\xi \times \xi + N_\zeta \times \zeta &= 0 \end{aligned}$$

Equation 5-3

Where L and N were dimensional roll and yaw moment respectively, p is roll rate, r is yaw rate, ζ is rudder deflection, ξ is aileron deflection. In pure roll motion, yaw rate is zero. Roll rate can be found by:

$$p = - \frac{\left(L_\xi - \frac{L_\zeta \times N_\xi}{N_\zeta} \right)}{\left(L_p - \frac{L_\zeta \times N_p}{N_\zeta} \right)} \times \xi$$

Equation 5-4

Rudder deflection required is:

$$\zeta = - \frac{N_p \times p + N_\xi \times \xi}{N_\zeta}$$

Equation 5-5

A ratio between rudder and aileron deflection to maintain pure roll is aileron rudder interlink (ARI). ARI gain calculated to be 0.04 for D-47A.

$$ARI = \frac{\zeta}{\xi}$$

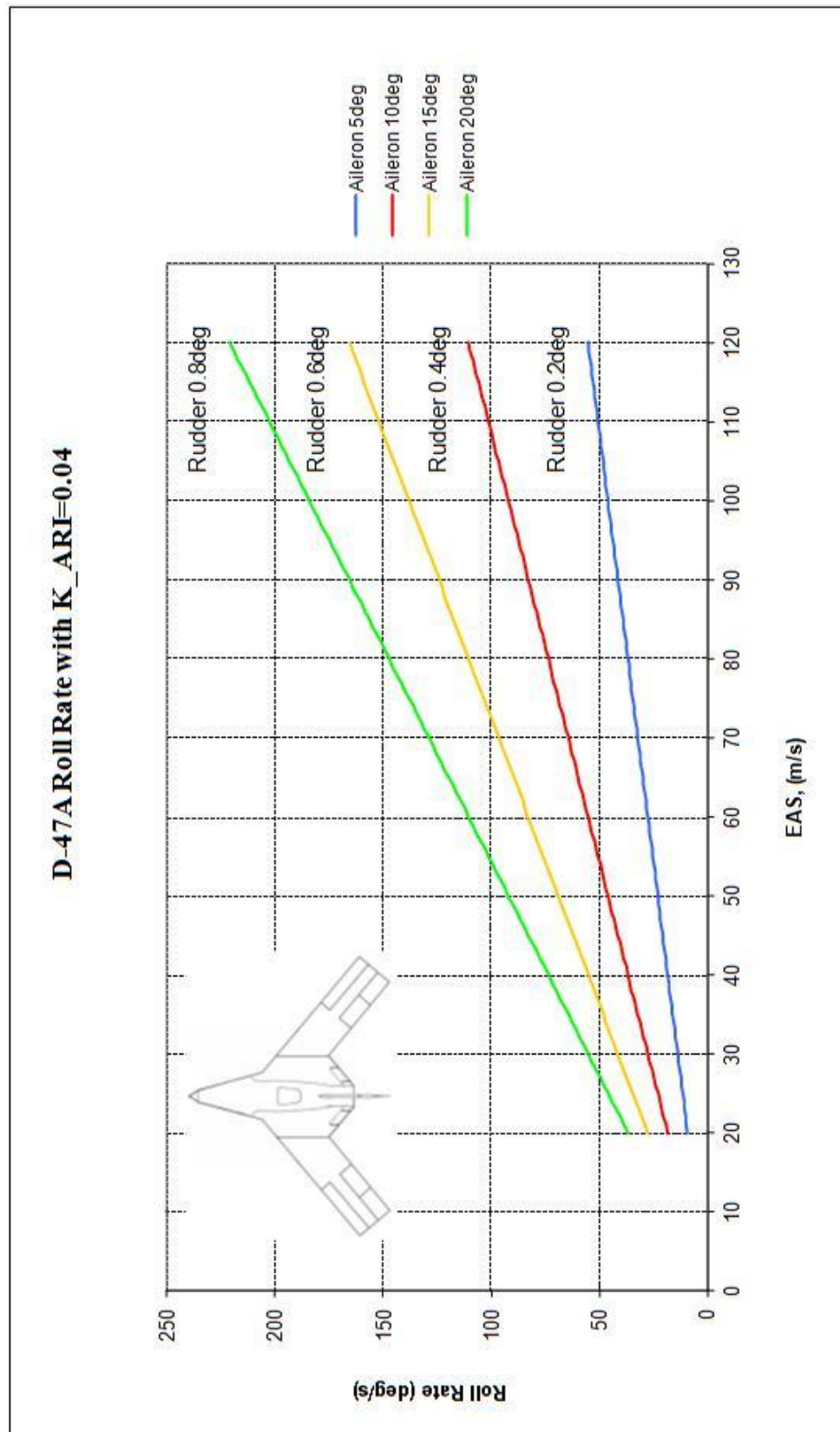


Figure 5-20 Roll rate map

5.3.6 Longitudinal stability

D-47A is longitudinal statically stable since $Cm_{\alpha} = -0.6335 < 0$. Dynamically, refer to flight envelope in Figure 5-16, eight flight conditions were tested for longitudinal stability assessment. Results were given in Table 5-8.

Table 5-8 D-47A longitudinal modes

H (m)	EAS (m/s)	<i>Logitudinal modes</i>			
		<i>SPPO</i>		<i>Phugoid</i>	
		ζ_s	ω_s (rad/s)	ζ_p	ω_p (rad/s)
0	40	0.349	8.64	0.11	0.104
0	60	0.356	12.6	0.185	0.0651
0	80	0.359	16.6	0.268	0.0556
0	112	0.361	23	0.389	0.0523
2500	40	0.271	7.57	0.0932	0.1
2500	60	0.28	12.8	0.189	0.0498
2500	80	0.282	17.1	0.278	0.0431
5000	80	0.209	9.67	0.122	0.0502

An elevon step input was given to trigger short period pitching oscillation (SPPO) and phugoid mode, time history of the D-47A response is shown in strip charts Figure 5-21 and Figure 5-22. Response to a step thrust increment is also tested and shown in Figure 5-23.

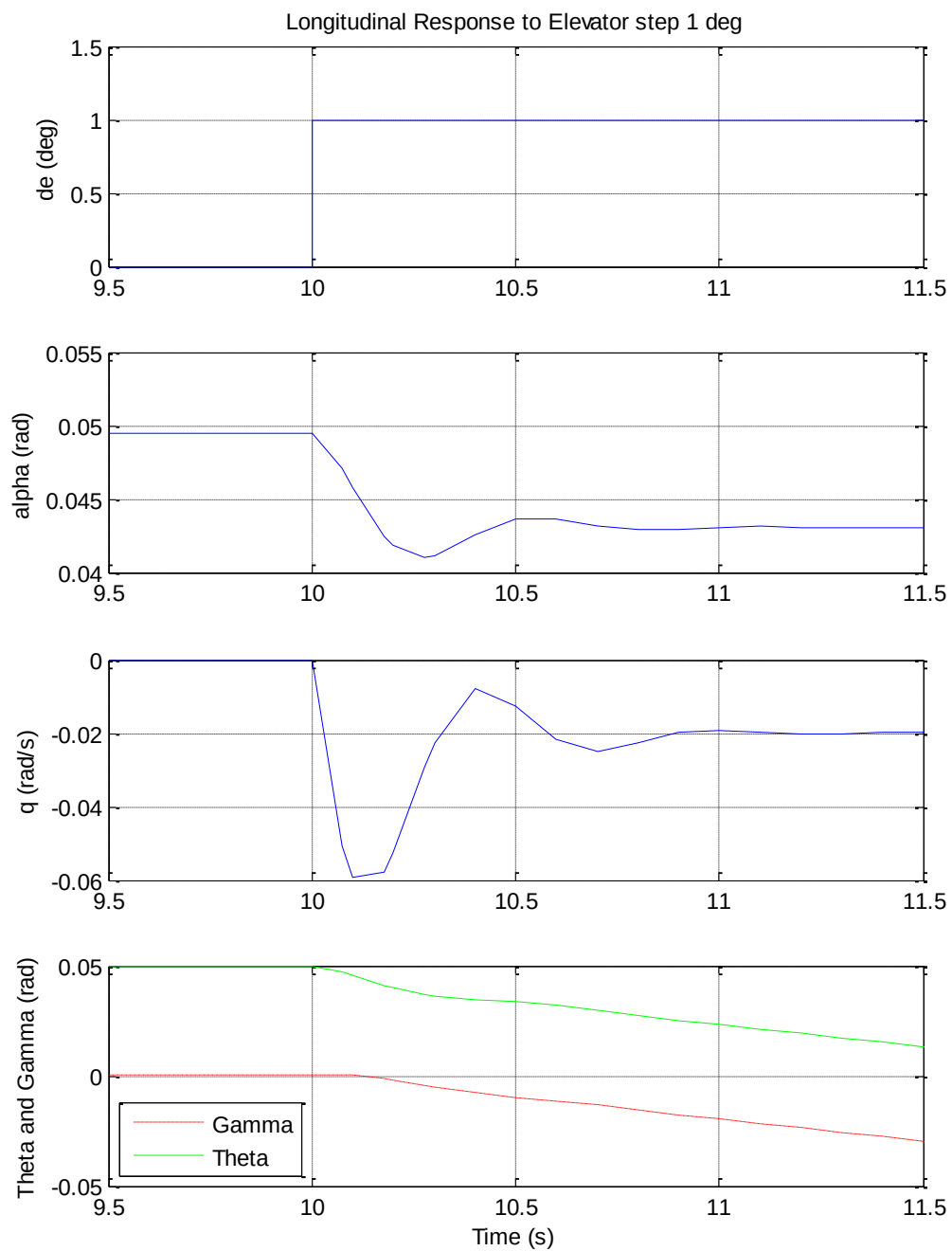


Figure 5-21 D-47A elevon step response SPPO mode in short time period

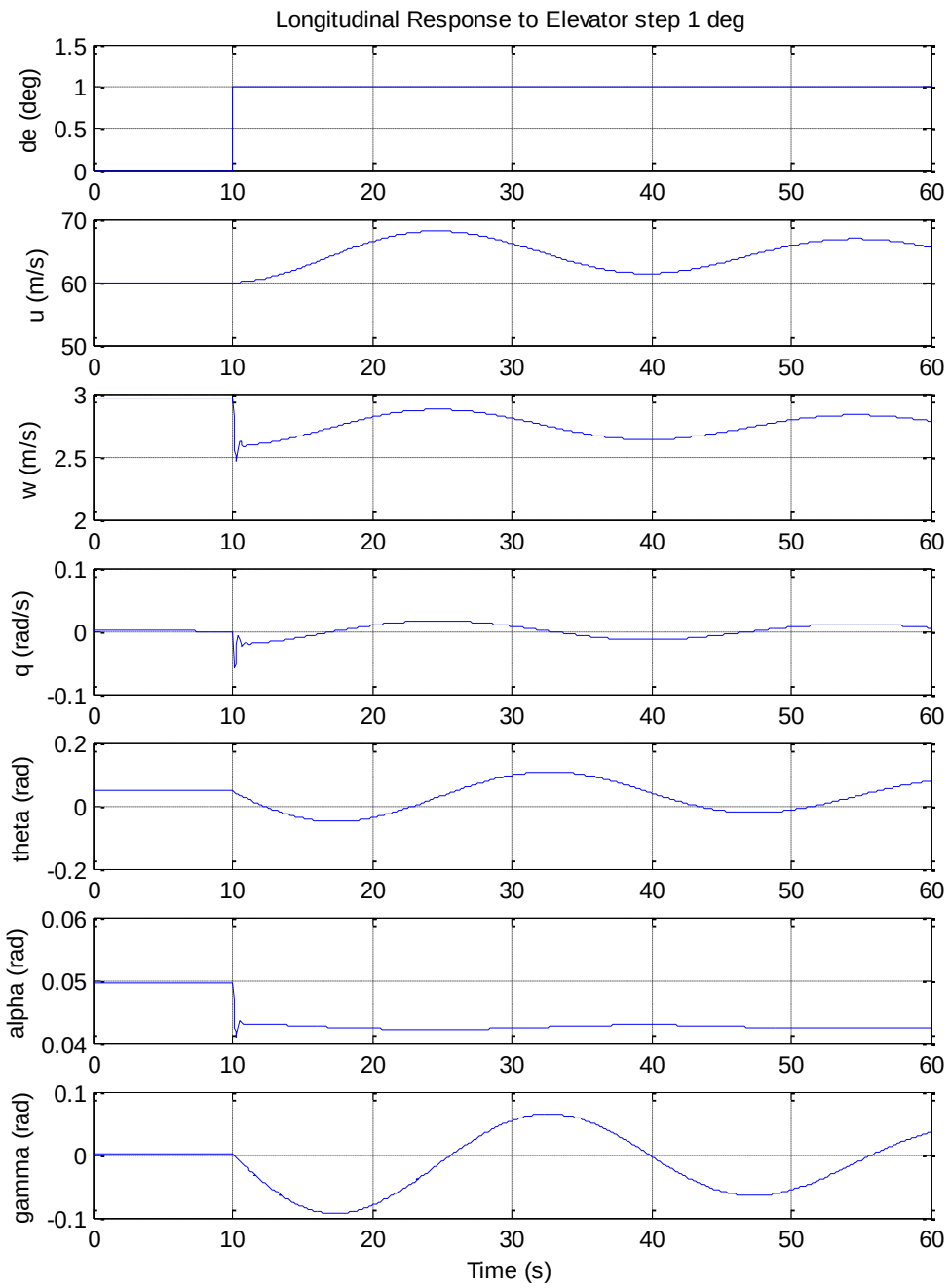


Figure 5-22 D-47A elevon step response phugoid mode in long time period

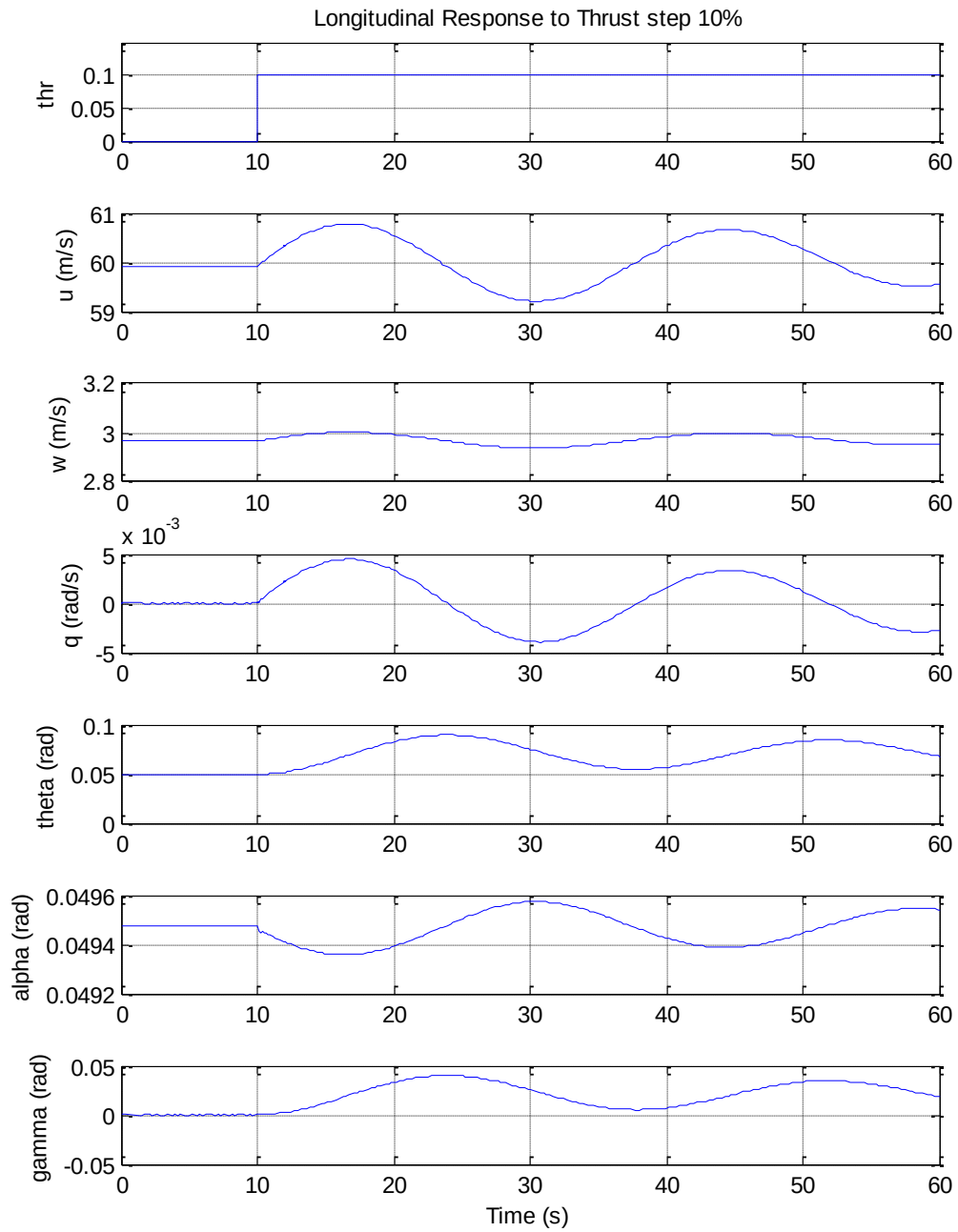


Figure 5-23 D-47A thrust increment response

5.3.7 Lateral and directional stabilities

D-47A is lateral and directional statically stable since $Cl_\beta = -0.00604 < 0$ and $Cn_\beta = 0.030903 > 0$.

Dynamically, refer to flight envelope in Figure 5-16, eight flight conditions were tested for lateral and directional stabilities assessment. Results were given in Table 5-9.

Table 5-9 D-47A lateral and directional modes

<i>Lateral-Directional modes</i>					
H (m)	EAS (m/s)	Roll T_r (s)	Dutch Roll ζ_d ω_d (rad/s)		Spiral T_2 (s)
0	40	0.267	0.088	2.47	47.48
0	60	0.179	0.093	3.62	957.39
0	80	0.135	0.096	4.79	364.81
0	112	0.096	0.097	6.68	280.63
2500	40	0.392	0.062	2.16	35.55
2500	60	0.223	0.070	3.72	3767.10
2500	80	0.166	0.073	4.99	427.87
5000	80	0.394	0.047	2.8	164.64

An elevon differential 2 second pulse input for roll was given to trigger roll subsidise mode, time history of the D-47A response is shown in strip charts Figure 5-24.

Response to a 1 degree rudder step input was also given to trigger Dutch roll and spiral mode, time history strip chart is shown in Figure 5-25.

In summary, D-47A rigid aircraft model simulation results showed the aircraft satisfied all level 1 flying quality requirements, except Dutch roll damping was too small. Suitability of flying quality requirements for a UAV is still under discussion. Never the less, D-47A provides a suitable foundation to build up the SAW D-47 aircraft series.

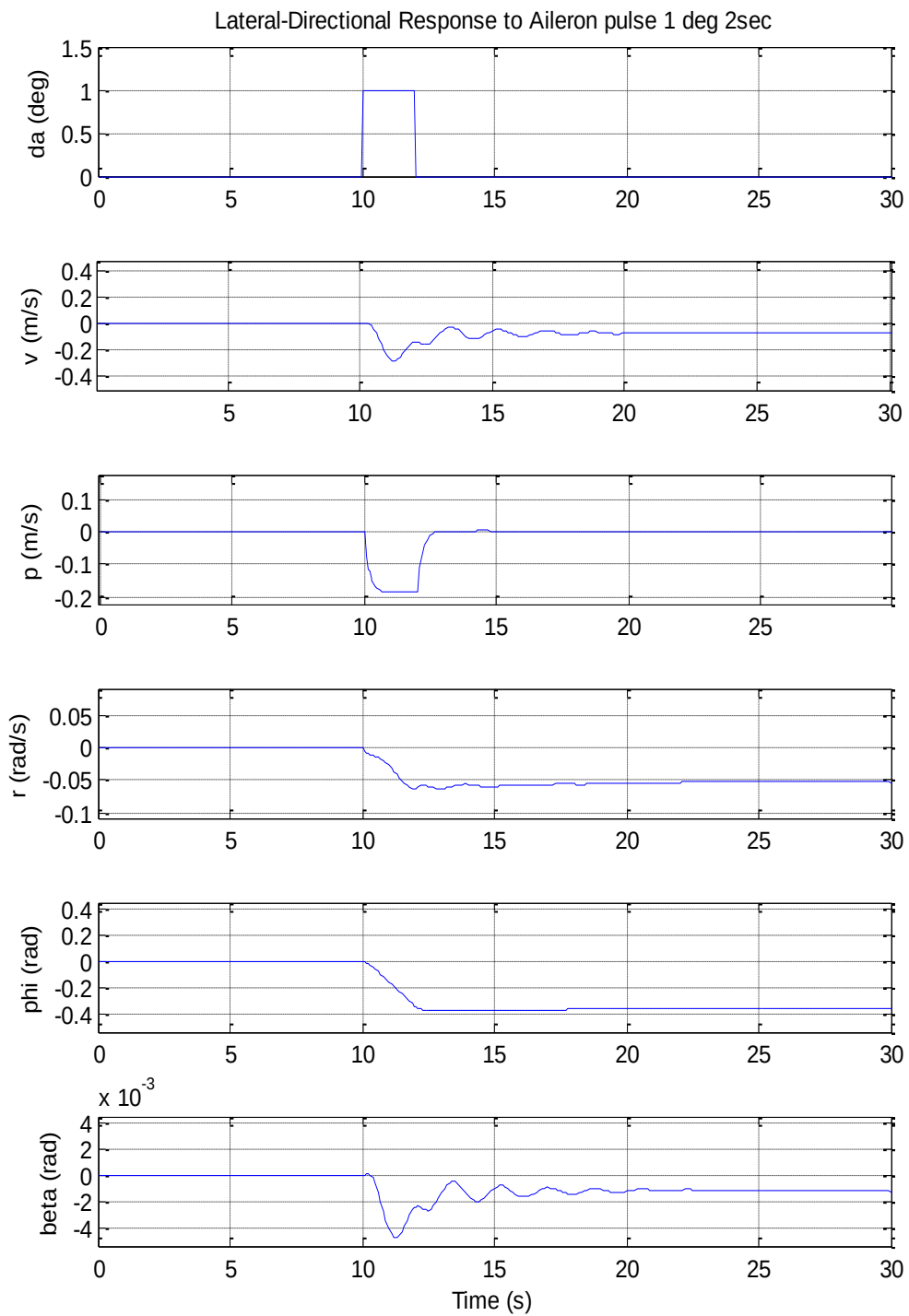


Figure 5-24 D-47A Elevon differential pulse input for roll response

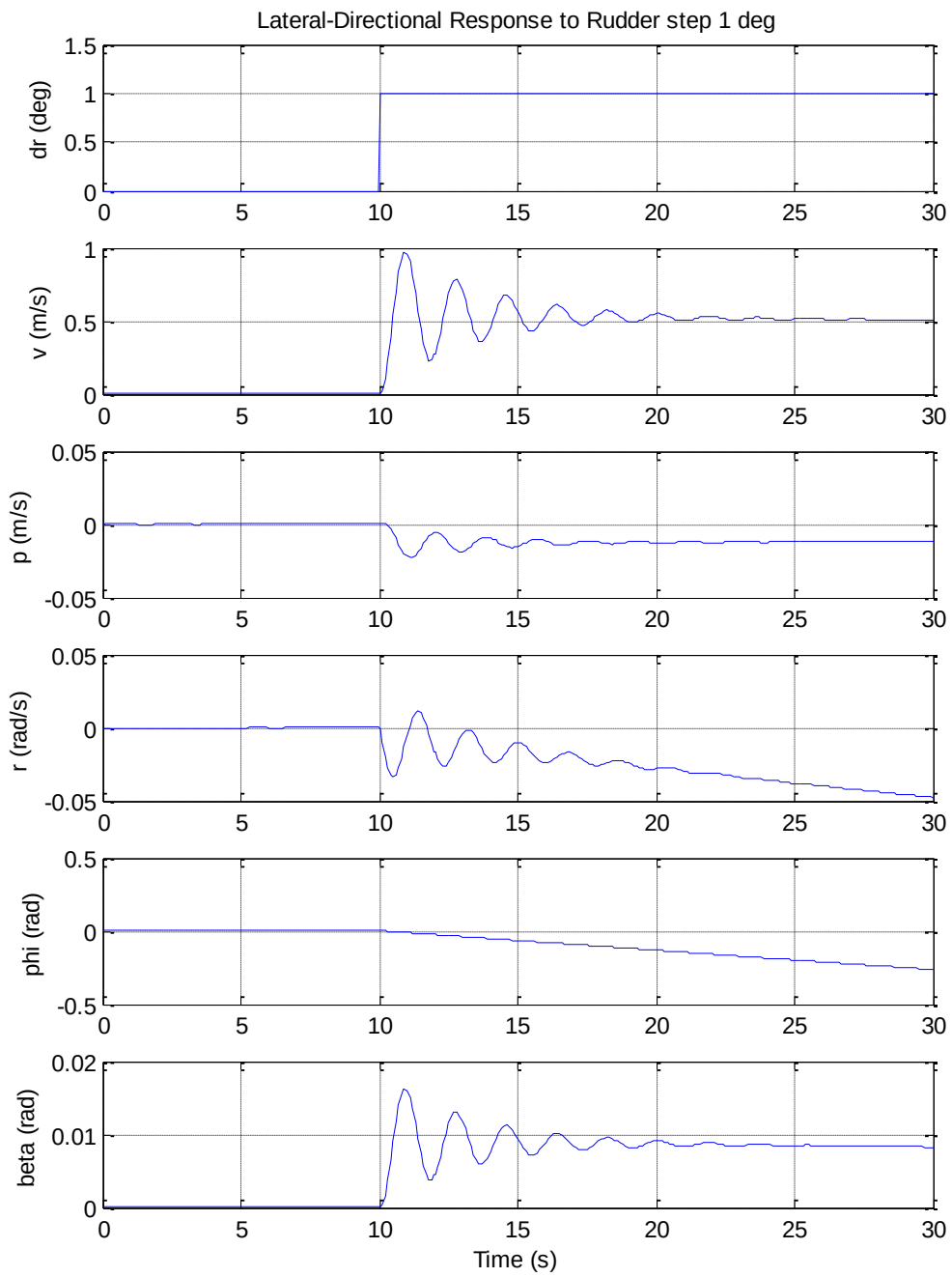


Figure 5-25 D-47A rudder step input response

Chapter 6 SAW Aircraft Modelling and Analysis

6.1 D-47SAW Aircraft Modelling

D-47SAW an aircraft model is created based on the SAW model and rigid aircraft model. Figure 6-1 shows the structure of the SAW model as part of the D-47SAW whole aircraft model. The two SAW modules representing the left and right wing are joined together with the fuselage model to produce lift, pitching moment and rolling moment. When an excitation load is applied on the wing, the structural dynamic response in terms of vertical and rotational accelerations on the fuselage node is also considered. The internal elastic forces and moments within the structure are included together with the aerodynamic forces and moments as the total input to the EOM. This would enable the elastic behaviour of the airframe structure be counted in the SAW aircraft flight response measured by the IMU sensors. The centre of pressure which varies with the wing deformation is also calculated.

To provide a complete set of input data into the equation of motion (EOM) of the model, any force and moment component that is not available or cannot be calculated by SAW model will be supplemented by D-47B rigid aircraft data (Figure 6-2). Multiple control surfaces were designed to give D-47SAW more authorities to overcome control efficiency drop at higher dynamic pressure. A flight control system (FCS) is design to manage the actuation of all these control surfaces to achieve this goal. Inherent stability of the D-47SAW is likely to be unacceptable, a stability augmentation system (SAS) is built in to solve potential problems (Figure 6-3). The compressibility and aeroelastic effects would have significant influence on the aerodynamic derivatives. The change of aerodynamic derivatives means the aircraft dynamic model would be completely different at flight conditions. Flight control gain is thus scheduled to suit every flight condition. Since the D-47SAW aircraft flight speed is limited at low subsonic ($M < 0.3$), compressibility is neglected and the aeroelastic effect on the flight control is the main focus in the following study.

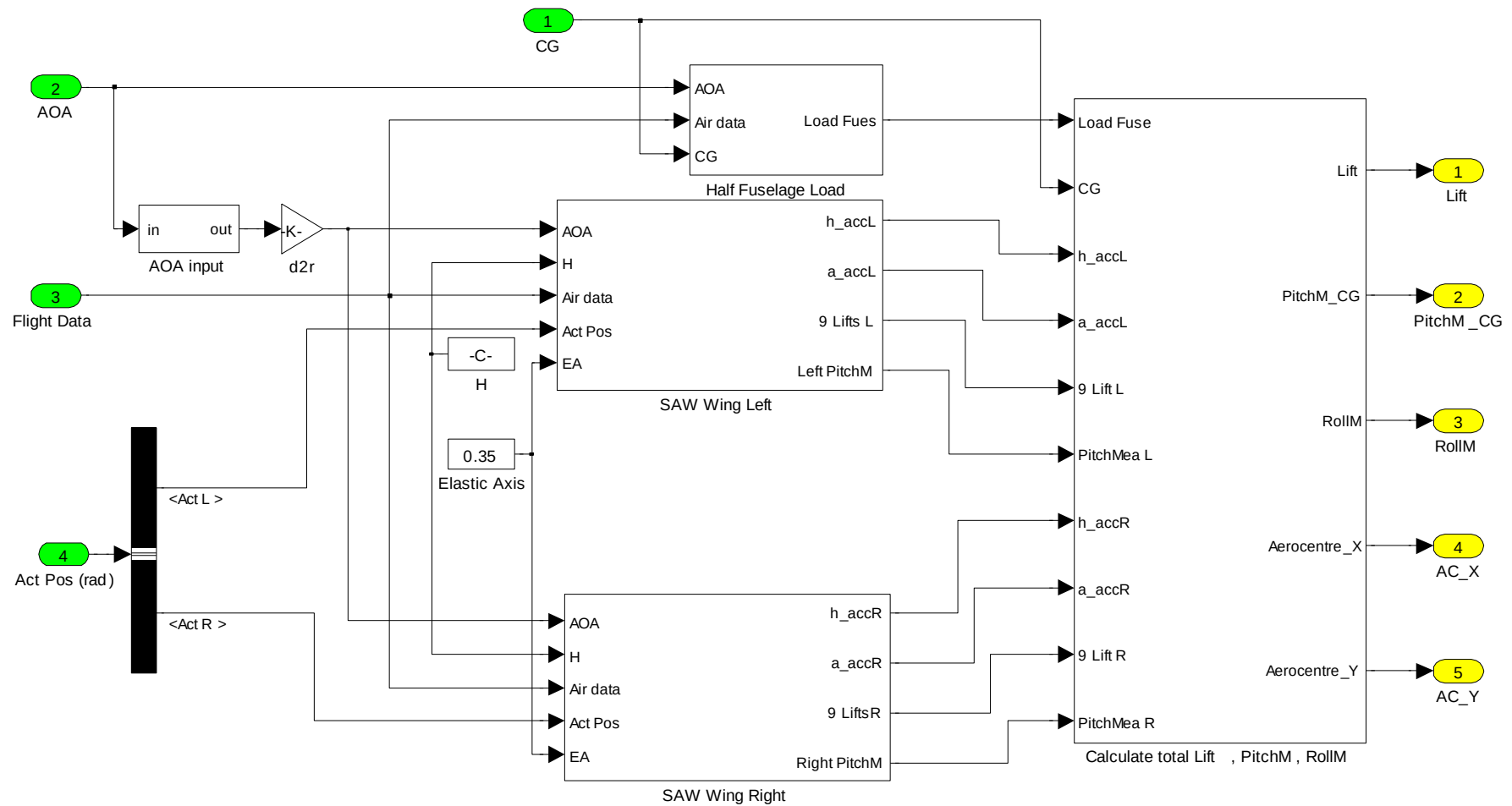


Figure 6-1 SAW model forces and moments generation

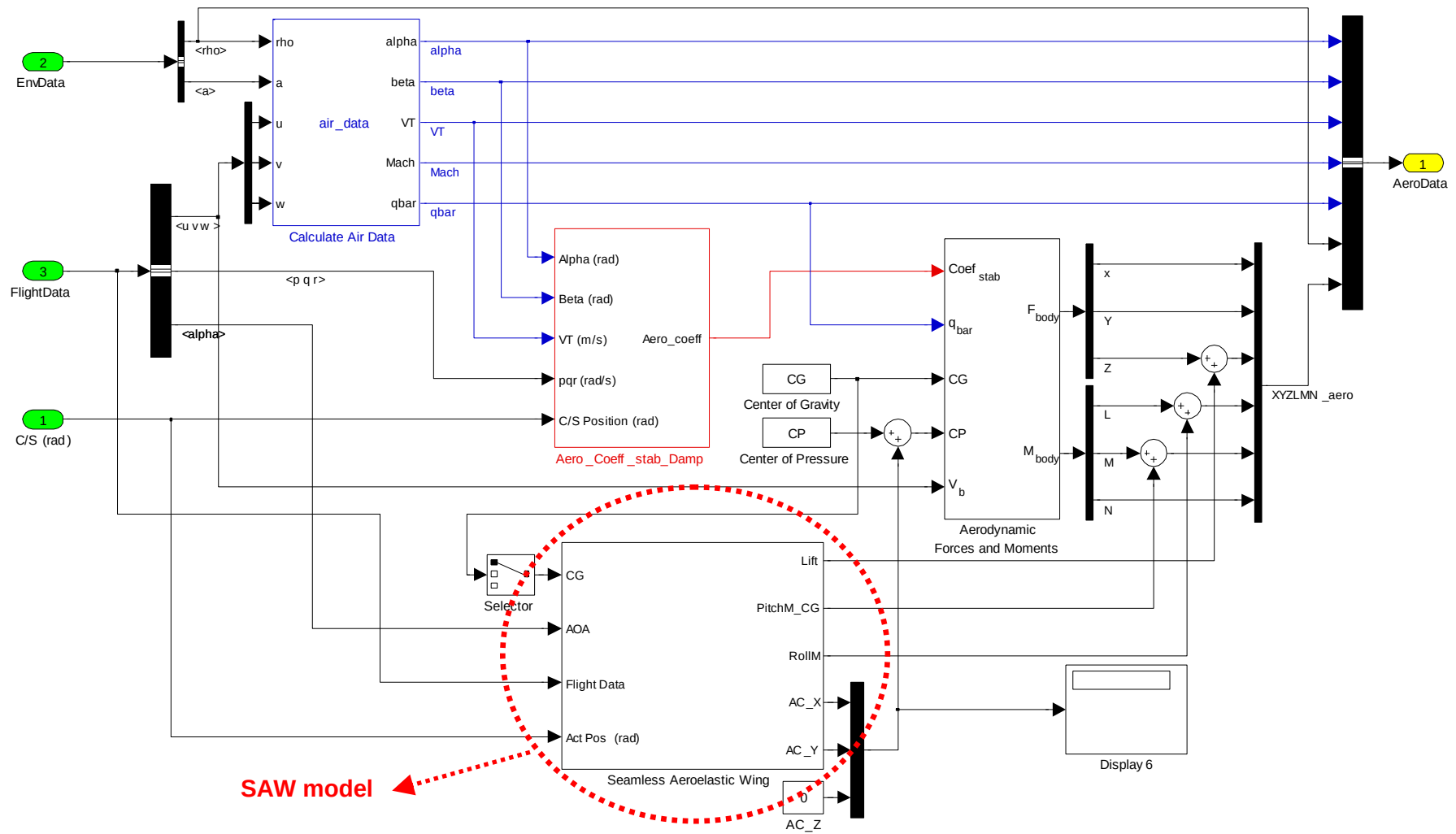


Figure 6-2 D-47SAW aerodynamic block

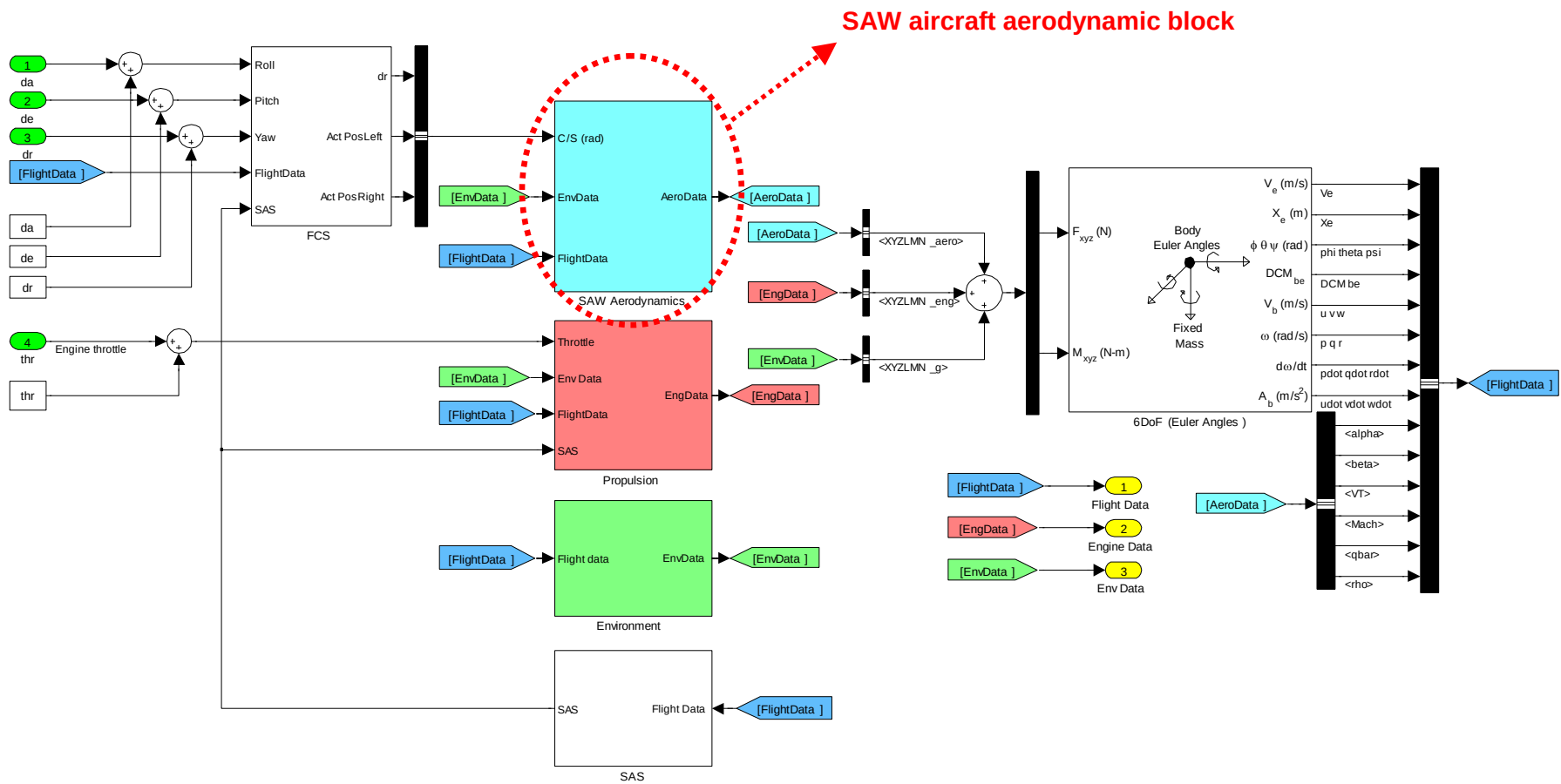


Figure 6-3 D-47SAW with FCS and SAS

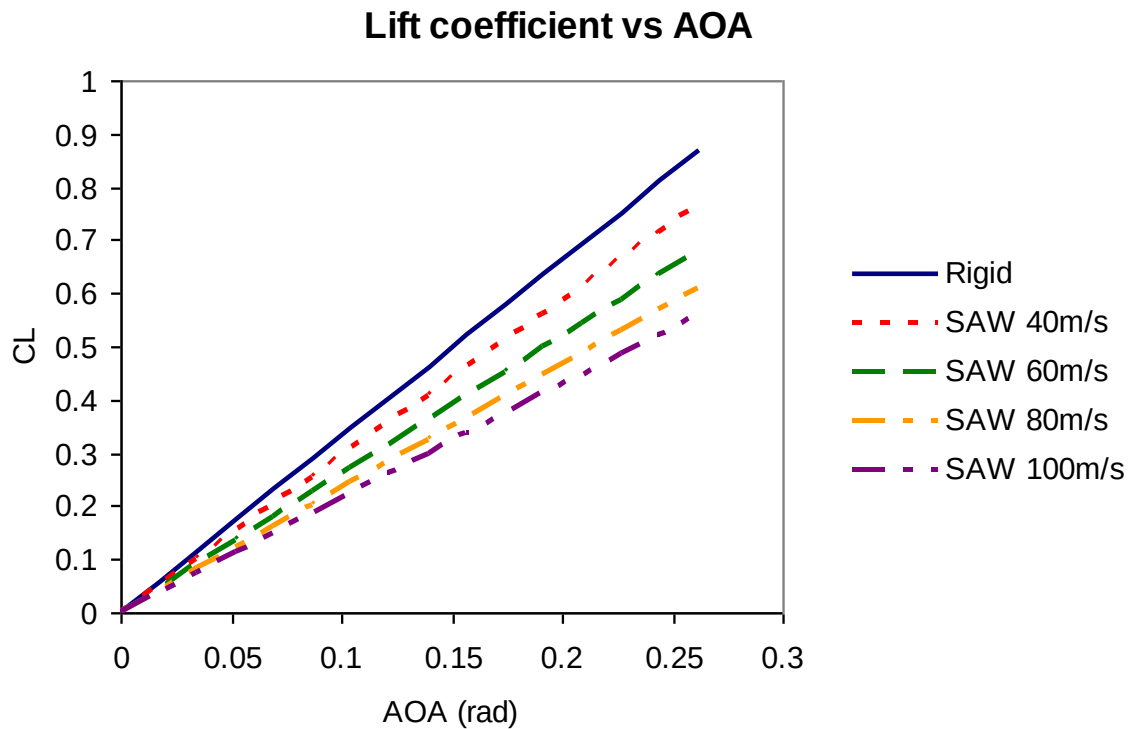
6.2 Flight Stability and Control Analysis

When examining the lift coefficients and pitching moment coefficients of the SAW aircraft model, many differences from the rigid wing version were observed. The lift curve slope of the SAW is reduced with increase of flight velocity and pitch moment curve slope changes from negative to positive as shown in Figure 6-4. Note that the pitching moment curve slope of D-47SAW aircraft is different from SAW wing only because of the addition of fuselage lift and reference centre moved from wing root to aircraft CG.

The resulting static margin $\frac{C_{m_\alpha}}{C_{L_\alpha}}$ indicates that the SAW aircraft is statically unstable

and the instability increase with velocity as presented in Table 6-1. Since stability of the D-47SAW aircraft is critical, a stability augmentation system (SAS) is necessary to maintain flight.

The design of SAS requires D-47SAW stability and control derivatives to compose linearized longitudinal, lateral and directional flight dynamic model. These derivatives were calculated and presented in the next two sections.



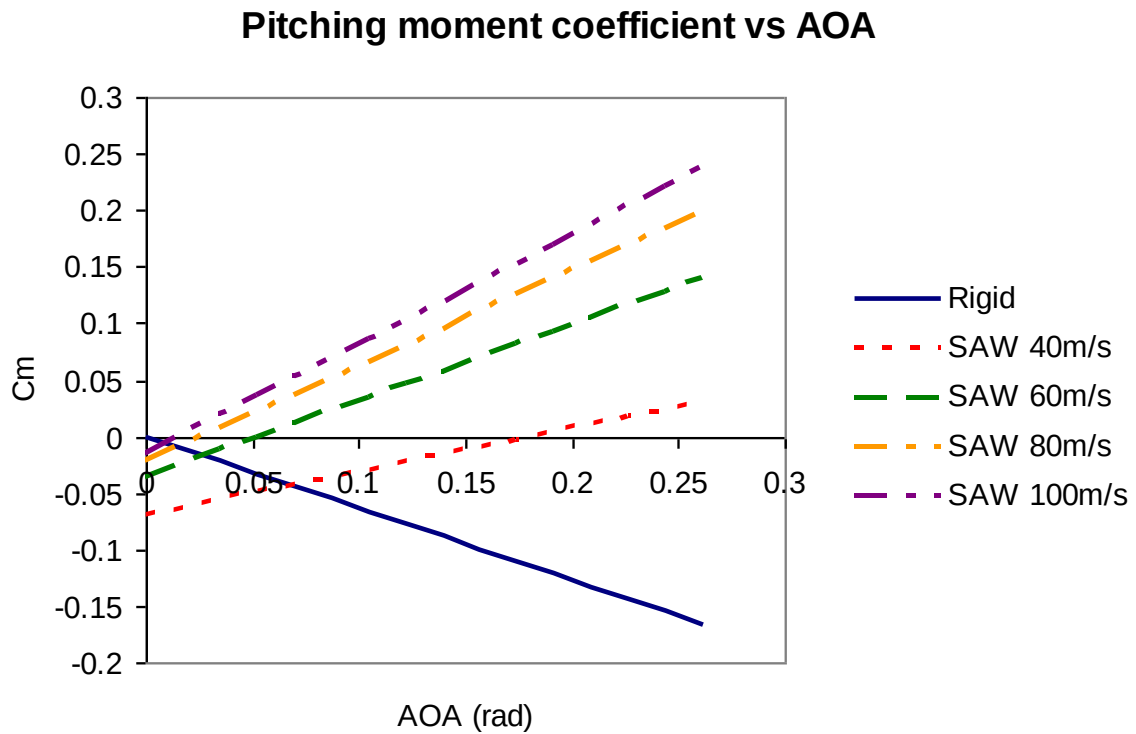


Figure 6-4 Lift and pitching moment coefficient of D-47SAW aircraft compared to rigid D-47B

Table 6-1 Static Margin of D-47SAW at different velocity

Rigid	Static Margin C_m/CL			
	SAW 40m/s	SAW 60m/s	SAW 80m/s	SAW 100m/s
-19.11%	13.10%	25.76%	36.24%	44.90%

6.2.1 Stability derivatives

The effect of the SAW on the static and dynamic stability derivatives is primarily in the longitudinal axis. C_{m_α} becomes positive which leads to pitch divergence and pitch damping C_{m_q} is reduced due to swept wing bending deformation. The lateral static stability is increased due to SAW deformation induced dihedral effect (Cl_β is increased from 0.006 to 0.019), but roll damping Cl_p is reduced due to wing deformation under manoeuvre load. Directional stability is mainly influenced by the fin, which is not flexible in the current study. The derivatives interested here are unique to flexible wing that

change with aerodynamic pressure, namely CL_α , Cm_α , Cm_q and Cl_p as shown in Figure 6-5 and Figure 6-6 against equivalent air speed (EAS). Lateral stability derivative Cl_β does not change with EAS but load factor, which is not concerned in this study.

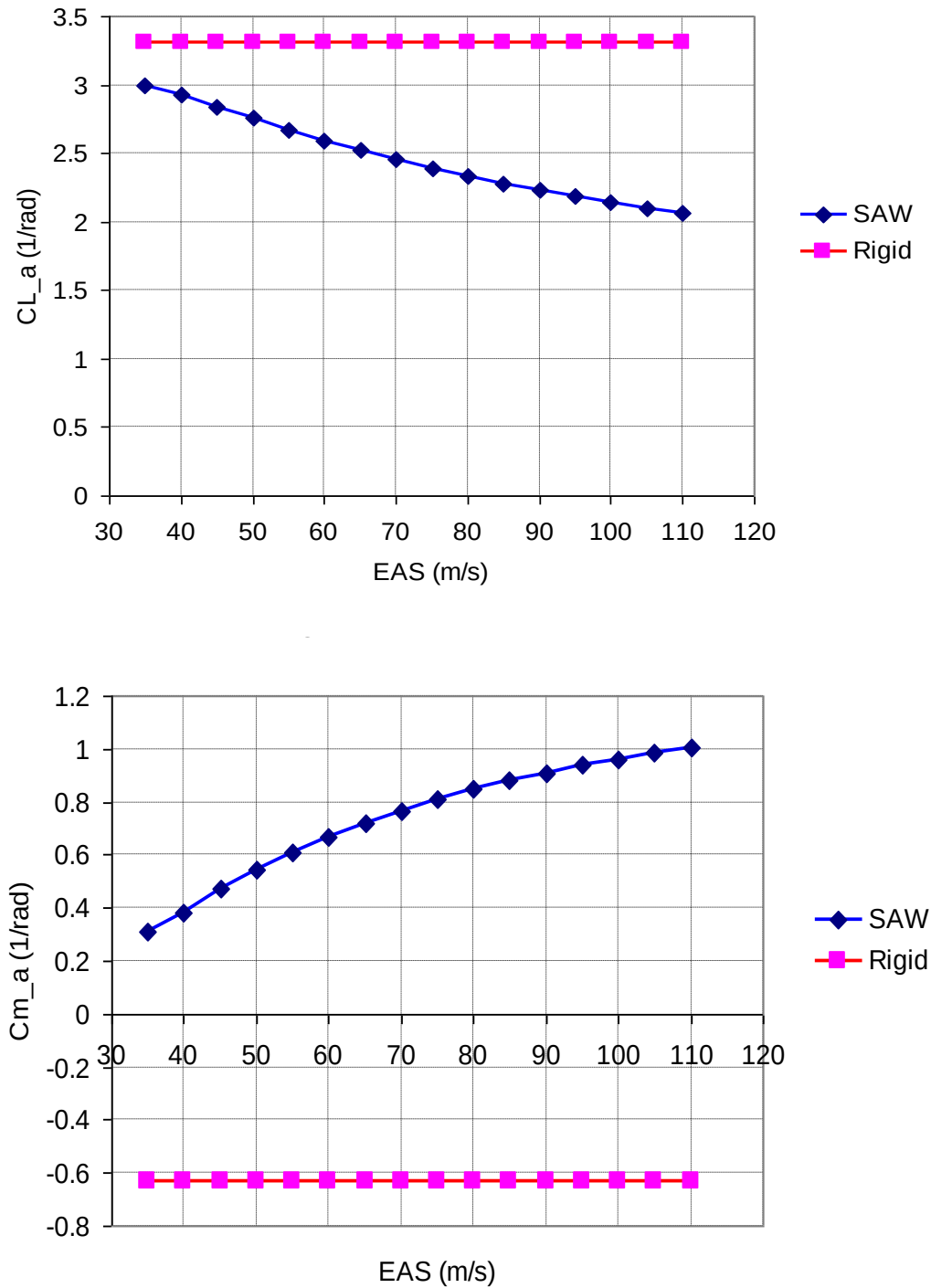


Figure 6-5 Stability derivatives CL_a and Cm_a

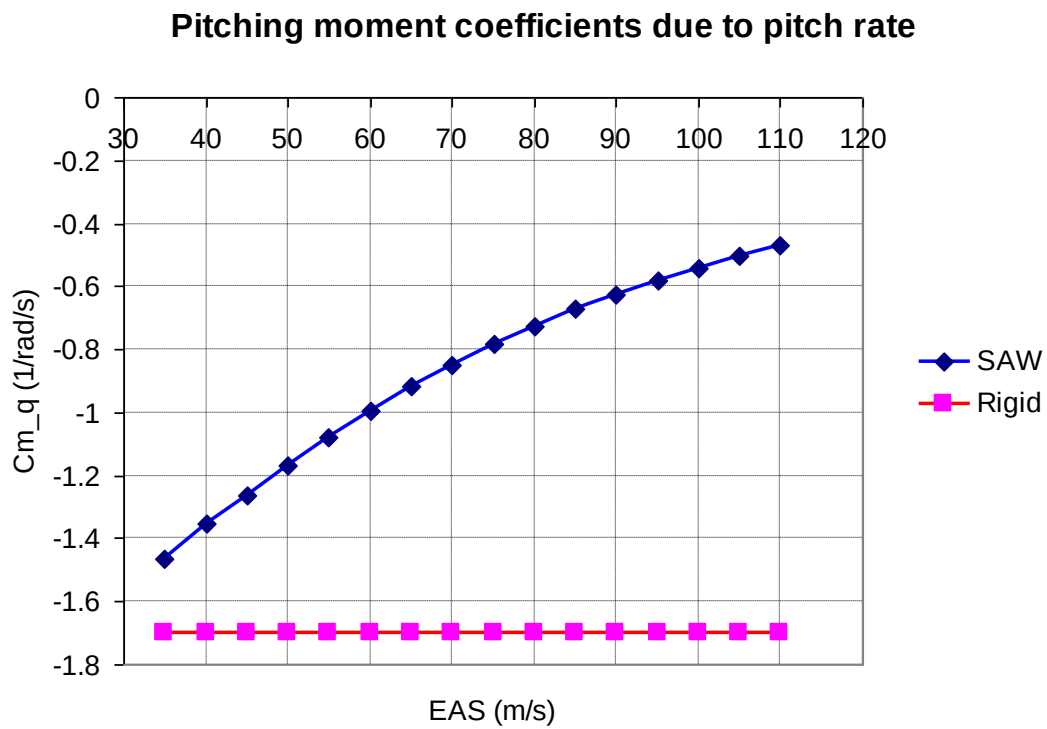
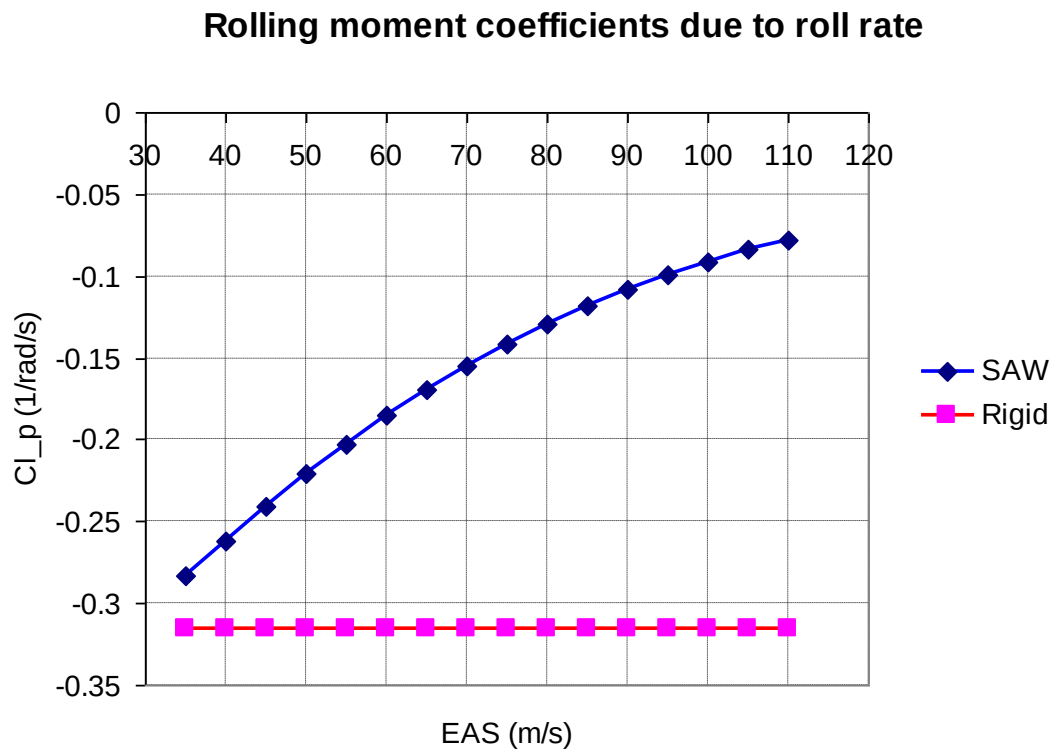


Figure 6-6 Damping derivatives C_{m_q} and C_{l_p}

6.2.2 Control derivatives

Control derivatives of the D-47SAW will be reduced due to wing twist at high dynamic pressure compared to its rigid counterpart D-47, whose derivatives were constant. Crucially these SAW derivatives even change sign above a critical velocity where control reversal occurs. For the D-47SAW aircraft, a solution to this problem is to use distributed inboard and outboard TE control surfaces, plus a large LE control surface as illustrated in Figure 6-7. The lift, pitching and roll moments coefficients due to these control surface deflections in comparison with a rigid wing elevon (RW Elevon) are presented in Figure 6-8, Figure 6-9 and Figure 6-10.

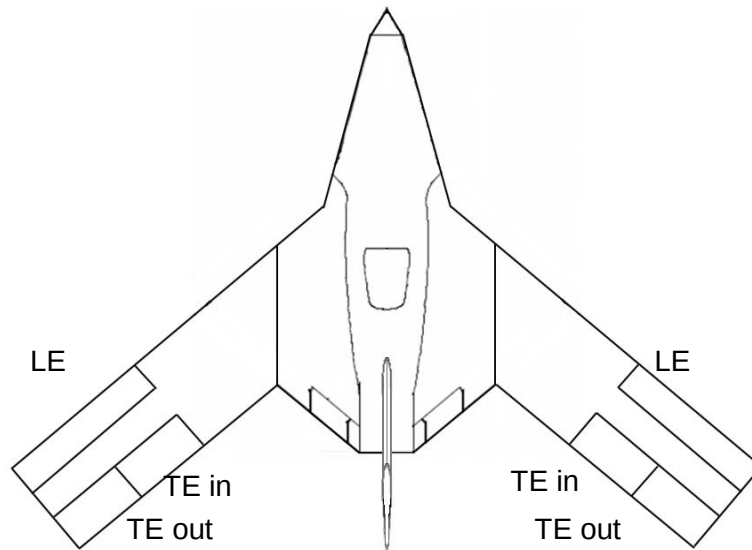


Figure 6-7 Distributed control surfaces on the D-47SAW aircraft

It can be observed that reversal velocity of lift and roll moment is at 56m/s for outboard TE, at 110m/s for inboard TE and 85m/s for LE. Reversal velocity for pitch moment is at 49m/s for outboard TE, at 67m/s for inboard TE, and 100m/s for LE.

These derivatives would be used for estimation of control surface angle for trim, turn rate and roll rate of the SAW aircraft.

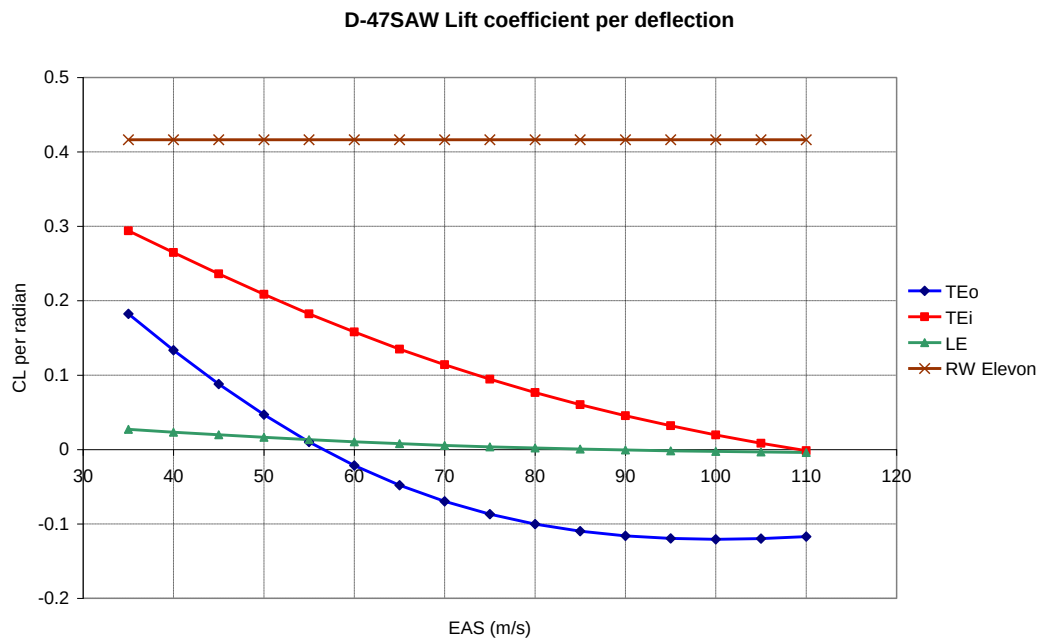


Figure 6-8 D-47SAW Lift coefficient due to control surface deflection.

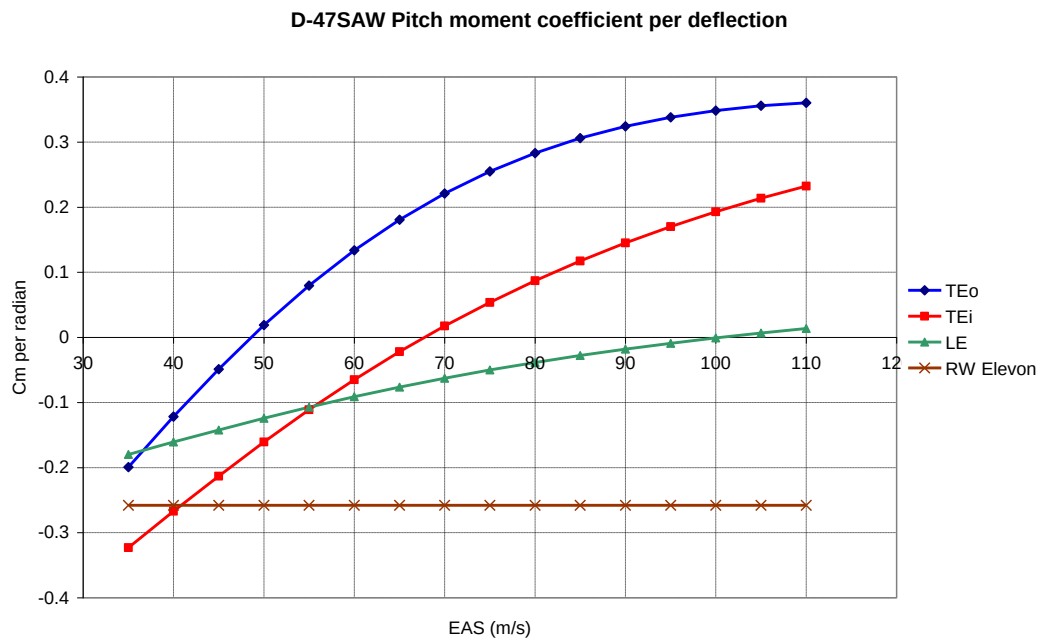


Figure 6-9 D-47SAW pitch moment coefficient due to control surface deflection.

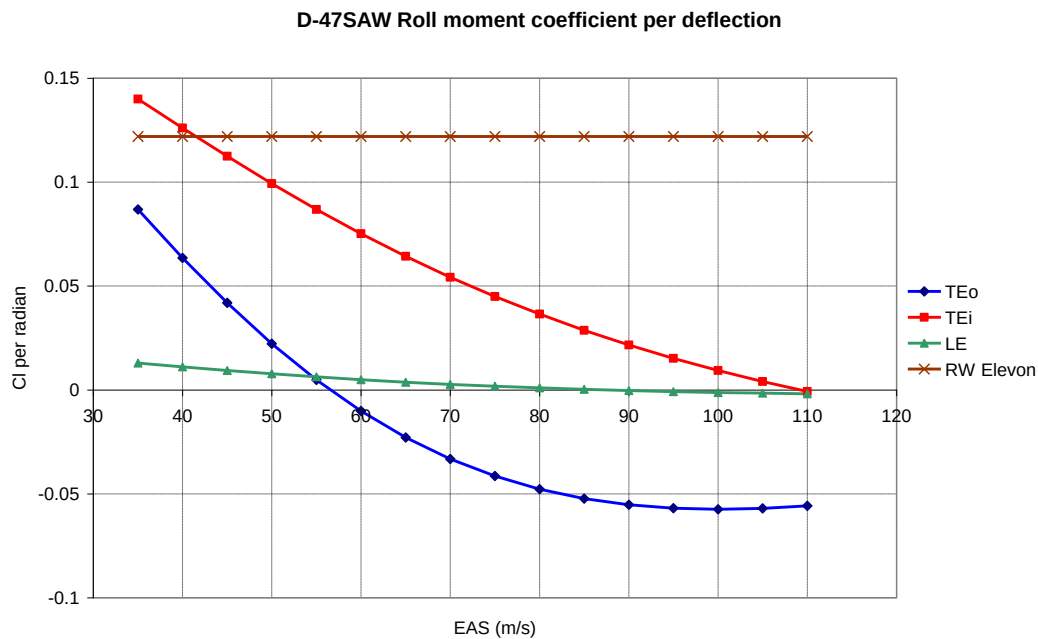


Figure 6-10 D-47SAW roll coefficient due to control surface deflection.

6.2.3 Trim, pitch rate and roll rate

Control surface deflection required for trimmed level flight is complicated. Figure 6-11 shows the requirement of control surface deflections for trim. It is discovered that the LE control surface, though not very effective, can be used for trim at majority of the low velocity flight. After 80m/s, outboard TE control surface is more effective for trim.

The pitch rate of the D-47SAW aircraft was assessed without gravity components so it does not vary with attitude. Similar to level turn rate, at 10 degree of control surface deflection, the pitch rate of the rigid D-47 model increases linearly with velocity as expected. The D-47SAW model on the other hand exhibits classic control reversal behaviour. The outboard TE can match the elevon in terms of pitch efficiency at low velocity, which dramatically decreased to zero at 56m/s. The inboard TE and LE control is less effective in pitch control. However when combined in operation, they can be means of suitable pitch control in the speed range of 50m/s to 65m/s. After 65m/s, reversed TE outboard deflection can provide exceptionally high pitch rate as shown in

Figure 6-12 due to the reduced pitch damping. However, reduced pitch damping may cause dynamic problem that jeopardise D-47SAW handling quality.

Figure 6-13 shows the D-47SAW aircraft roll rate at 10 degree deflection. Although roll control effectiveness is reduced for all control surfaces of the SAW aircraft, the roll damping is also reduced. Hence the resulting roll rate is higher than the rigid D-47, but quickly decreased to zero when velocity increases.

The inboard TE control surface in this case becomes much more effective than the outboard TE at majority of the speed range. At high velocity the outboard TE reversed deflection can be used if large roll rate is needed. The LE control surface provides little roll control authority. It is better used for trim purpose.

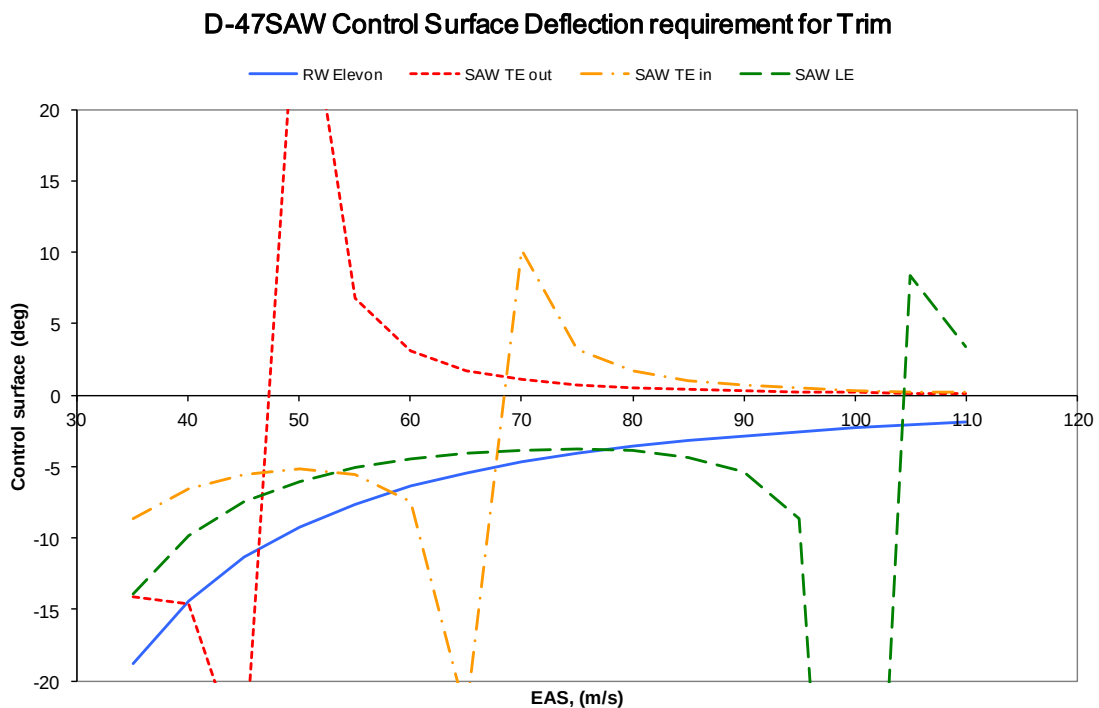


Figure 6-11 D-47SAW required control surface deflection for trim

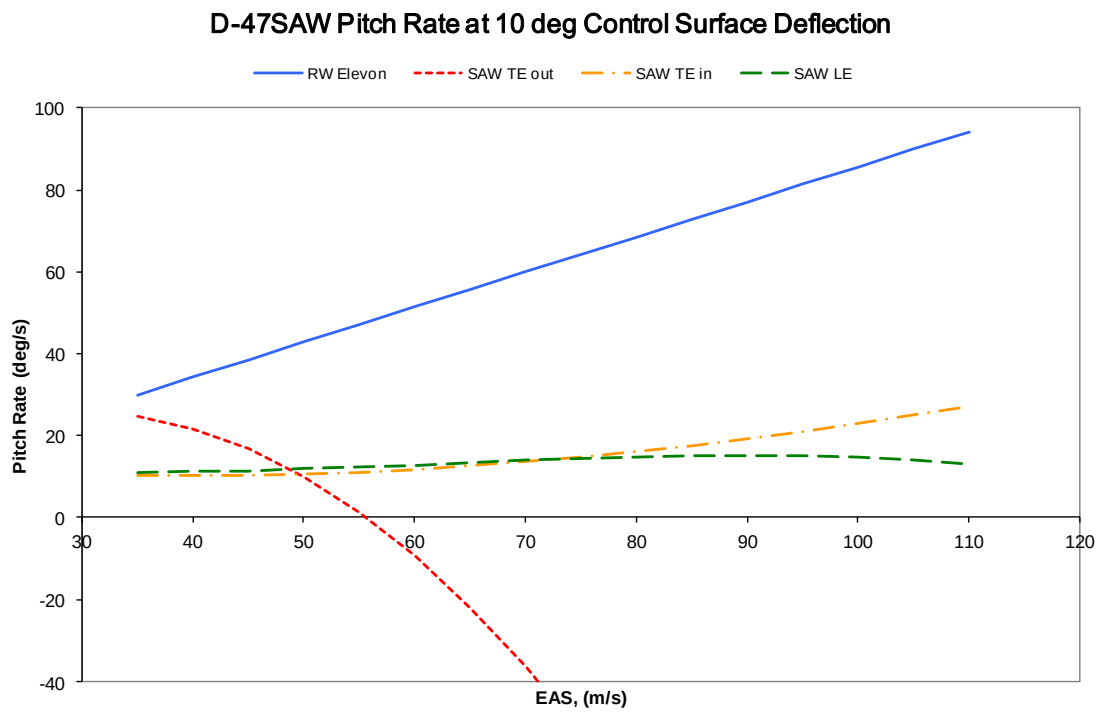


Figure 6-12 D-47SAW Pitch rate at 10deg deflection

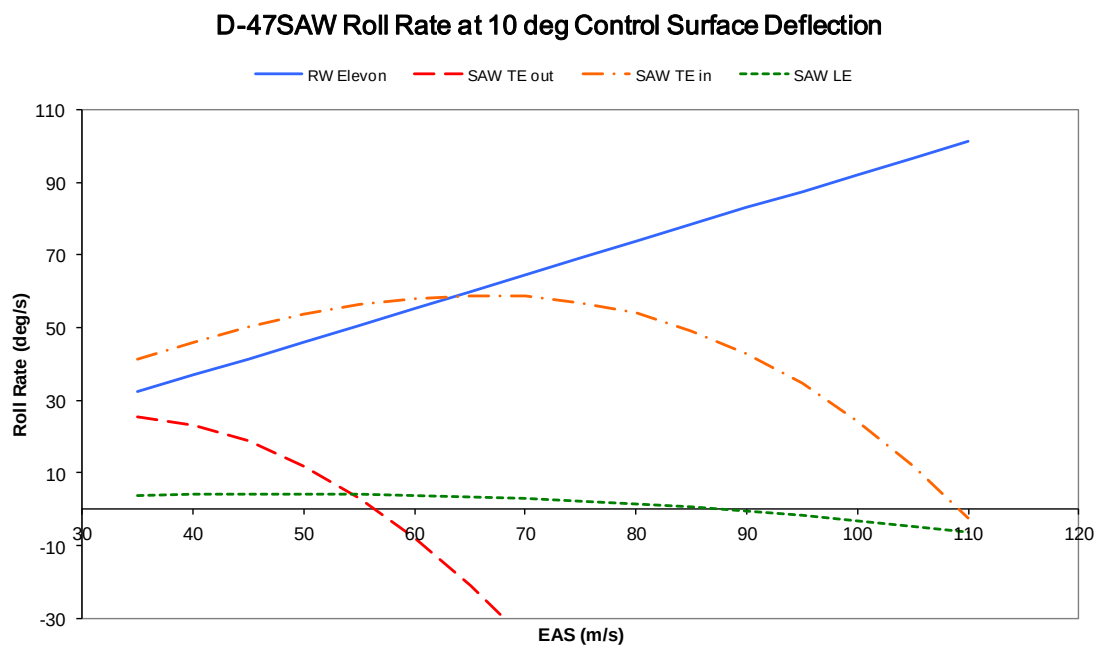


Figure 6-13 D-47SAW Roll rate at 10deg deflection

6.2.4 Summary and Flight Control Plan

In summary, compared to rigid aircraft D-47B, the aircraft D-47SAW of flexible SAW is statically unstable in longitudinal direction. Its lateral static stability is improved. Directional stability is not affected unless a flexible fin is added. An active stability augmentation system needs to be designed to maintain stable flight.

The control of the D-47SAW is complex due to multiple TE and LE control surfaces design. Each control surface has different flight control efficiency at different velocity range as shown in Table 6-2. All the aeroelastic effect, either positive or negative, can be utilised to maintain or even enhance the controllability of the flexible D-47SAW. The results indicate that the outboard TE is most prone to aeroelastic control reversal. This also made it very effective for control at high velocity if inversely actuated. The inboard TE is less influenced by aeroelastic problem. At low speed it is similar to elevon on rigid wing but with shorter moment arm. The LE control surface is the least effective in terms of flight control, but sufficient for flight trim.

Table 6-2 Flight control effectiveness of TE out, TE in and LE

<i>D-47SAW Integrated Flight Control Plans (D-47B effectiveness as 100%)</i>									
EAS (m/s)	TE out ($\pm 20^\circ$)			TE in ($\pm 20^\circ$)			LE ($\pm 20^\circ$)		
	Trim	Pitch	Roll	Trim	Pitch	Roll	Trim	Pitch	Roll
35	No	83%	79%	Yes	34%	128%	Yes	37%	12%
40	No	64%	63%	Yes	30%	125%	Yes	33%	11%
45	No	43%	45%	Yes	26%	121%	Yes	30%	10%
50	No	23%	26%	Yes	25%	116%	Yes	28%	9%
55	No	3%	6%	No	23%	111%	Yes	26%	8%
60	No	-18%	-14%	No	23%	105%	Yes	25%	7%
65	No	-39%	-35%	No	23%	98%	Yes	24%	6%
70	No	-61%	-56%	No	23%	91%	Yes	23%	4%
75	Yes	-82%	-76%	No	23%	82%	No	22%	3%
80	Yes	-104%	-96%	No	23%	73%	No	22%	2%
85	Yes	-125%	-114%	No	24%	63%	No	21%	1%
90	Yes	-147%	-132%	Yes	25%	52%	No	20%	-1%
95	Yes	-167%	-148%	Yes	26%	40%	No	18%	-2%
100	Yes	-188%	-163%	Yes	27%	26%	No	17%	-3%
105	Yes	-208%	-175%	Yes	28%	12%	No	15%	-5%
110	Yes	-227%	-186%	Yes	29%	-3%	No	14%	-6%

The flight control actuation plan is summarized in Figure 6-14. The outboard TE can be used at very low speed, but it is much more effective when reversely actuated at high

speed. The inboard TE is mainly used for roll but can also help pitch. LE is primarily used for trim but can help pitch control if needed.

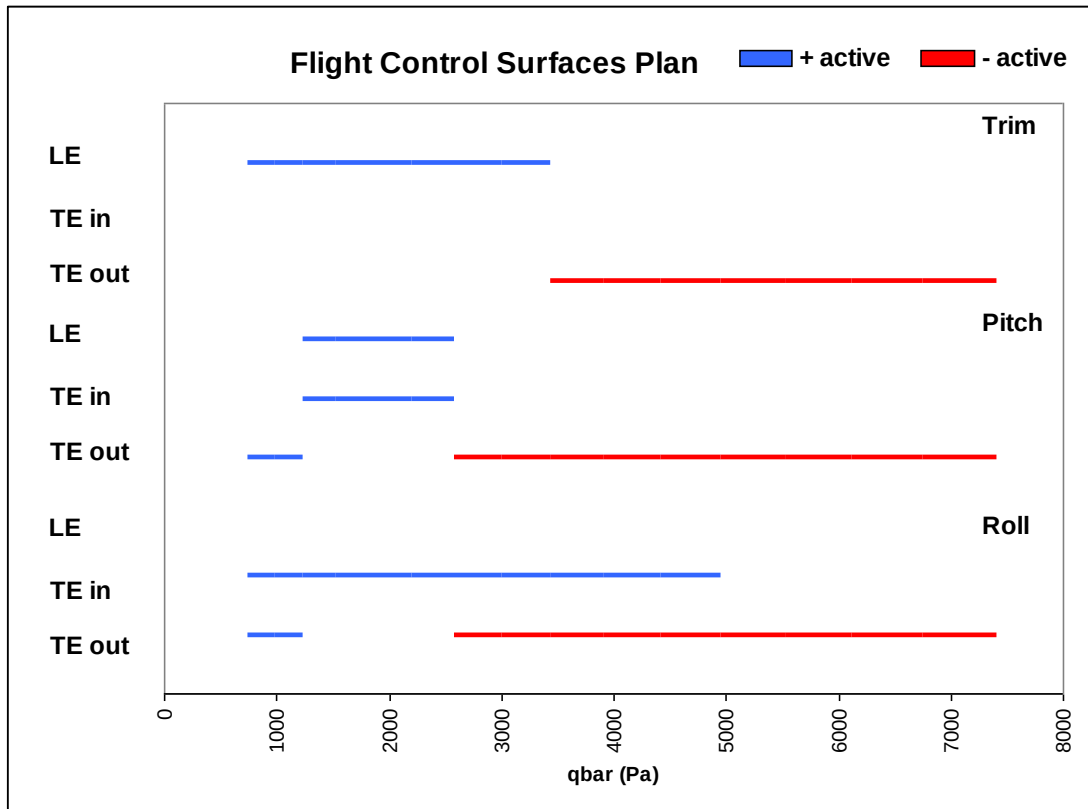


Figure 6-14 Control surface planning for trim, pitch and roll

Figure 6-15 shows implementation of flight control surface plan in the Matlab Simulink FCS, reverse deflection gain is scheduled with respect to dynamic pressure. Saturation block is added to represent the 20 degree deflection limit of seamless control surfaces. Aileron-rudder-interlink (ARI) gain of 0.04 from rigid aircraft analysis is not changed and added to counter adverse yaw.

The effect of generally reduced damping derivatives is positive on manoeuvrability. But it has negative effect on dynamic stability. The lack of restoring forces and moments are adding more oscillations to the aircraft dynamic mode. This will be analyzed in the next section.

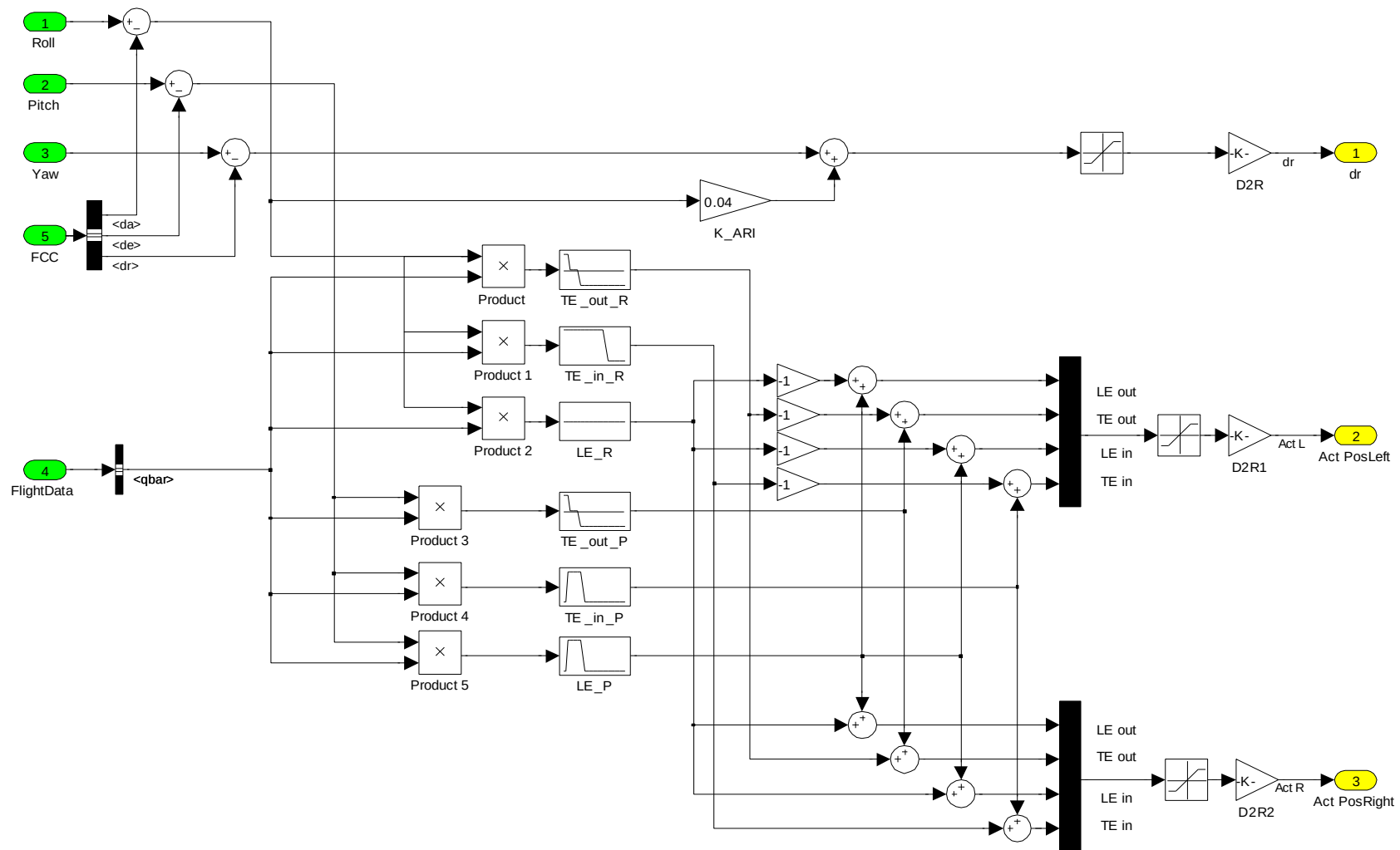


Figure 6-15 D-47SAW Simulink FCS on roll, pitch and yaw

6.3 D-47SAW Dynamic Analysis

Since the D-47SAW aircraft is longitudinally unstable, a stability augmentation system (SAS) was designed to actively stabilize the aircraft. The longitudinal dynamic modes were thus modified.

6.3.1 Stability Augmentation System

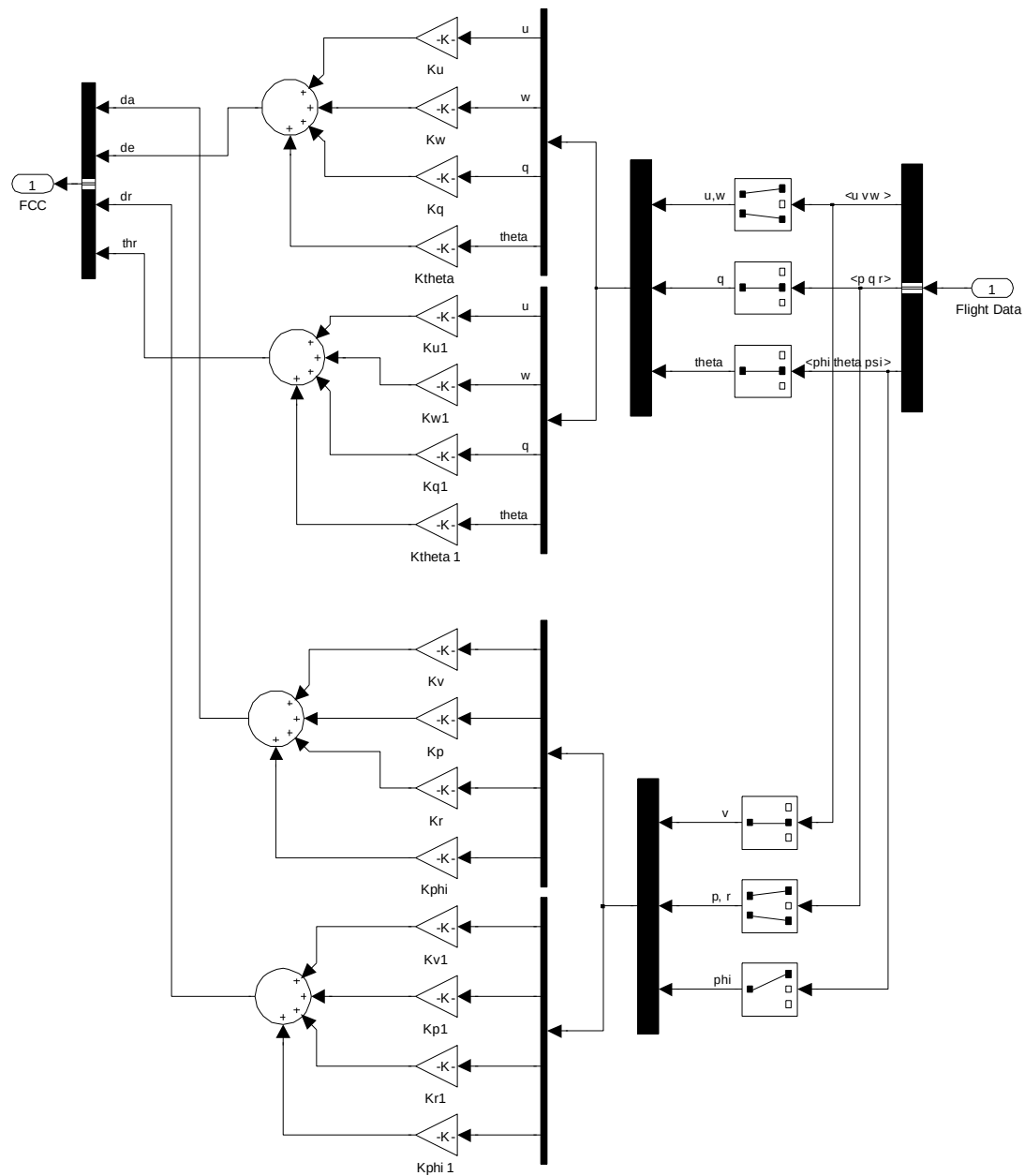


Figure 6-16 D-47SAW Simulink SAS for longitudinal motion

The SAS is designed using pole-placement method and implemented in D-47SAW Simulink model, Figure 6-16 shows the SAS controller in the feedback loop for both longitudinal and lateral-directional modes. Control gains were calculated using pole placement technique. The poles' positions were selected to meet MIL-F-8785C requirements of level 1 flying qualities. The longitudinal poles are $-3.5 \pm 7i$ and $-0.12 \pm 0.64i$ which gives:

Phugoid damping ratio $\zeta_p = 0.184$

Short period damping ratio $\zeta_s = 0.447$

Short period frequency $\omega_s = 7.825 \text{ rad / s}$

Lateral directional poles are -5.5 , $-1.8 \pm 3.67i$ and -0.0364 which gives:

Roll subsidise time constant $T_R = 0.182 \text{ s}$

Dutch roll damping ratio $\zeta_d = 0.44$

Dutch roll frequency $\omega_d = 4.09 \text{ rad / s}$

Spiral time to double amplitude $T_{S2} = 190.43 \text{ s}$

Control gains calculated to shift the closed loop poles were:

$$K_{long} = \begin{bmatrix} Ku_{\delta e} & Kw_{\delta e} & Kq_{\delta e} & K\theta_{\delta e} \\ Ku_{\tau} & Kw_{\tau} & Kq_{\tau} & K\theta_{\tau} \end{bmatrix} \quad K_{lat} = \begin{bmatrix} Kv_{\delta a} & Kp_{\delta a} & Kr_{\delta a} & K\phi_{\delta a} \\ Kv_{\delta r} & Kp_{\delta r} & Kr_{\delta r} & K\phi_{\delta r} \end{bmatrix}$$

At flight speed of 40, 60, 80 and 100m/s, scheduled control gains were:

40m/s

$$K_{long} = \begin{bmatrix} -10.901 & 62.231 & 119.718 & -11.172 \\ -0.6466 & 2.7659 & 9.8672 & -5.2156 \end{bmatrix} K_{lat} = \begin{bmatrix} -7.720 & -0.993 & 30.275 & -6.306 \\ -8.557 & 14.681 & -108.98 & 28.398 \end{bmatrix}$$

60m/s

$$K_{long} = \begin{bmatrix} -1.4351 & 34.695 & 59.482 & -1.2604 \\ -0.0151 & 3.31 & 12.8027 & -5.3324 \end{bmatrix} K_{lat} = \begin{bmatrix} -3.216 & -1.696 & 27.977 & -4.077 \\ -13.57 & 0.018 & 48.386 & -5.849 \end{bmatrix}$$

80m/s

$$K_{long} = \begin{bmatrix} -0.6241 & 24.024 & 39.1962 & -0.525 \\ 0.0732 & 4.0064 & 16.1055 & -5.288 \end{bmatrix} K_{lat} = \begin{bmatrix} -0.297 & -0.612 & 13.697 & -1.641 \\ -2.382 & -1.501 & 65.126 & -7.433 \end{bmatrix}$$

100m/s

$$K_{long} = \begin{bmatrix} -0.6241 & 24.024 & 39.1962 & -0.525 \\ 0.0732 & 4.0064 & 16.1055 & -5.288 \end{bmatrix} K_{lat} = \begin{bmatrix} -0.204 & -0.571 & 11.585 & -1.125 \\ -2.707 & -3.041 & 103.443 & -9.824 \end{bmatrix}$$

Take flight speed at 60m/s for example, the linearized D-47SAW longitudinal model in state-space expression is:

$$M_{mlong} \dot{x} = M_{along} x + M_{clong} ui$$

Where:

$$M_{mlong}^{-1} M_{along} = \begin{bmatrix} -0.0217 & -0.0002 & 0 & -9.8203 \\ 0 & 4.0943 & 56.2834 & 0 \\ 0.0032 & 8.0540 & -2.0461 & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix}, M_{mlong}^{-1} M_{clong} = \begin{bmatrix} 0 & 2.3756 \\ -0.5455 & 0 \\ 0.4746 & 0 \\ 0 & 0 \end{bmatrix}$$

If let $M_{mlong}^{-1} M_{along} = A$, $M_{mlong}^{-1} M_{clong} = B$, K is the feedback control gain and vi is input to the closed-loop system, then this augmented system is:

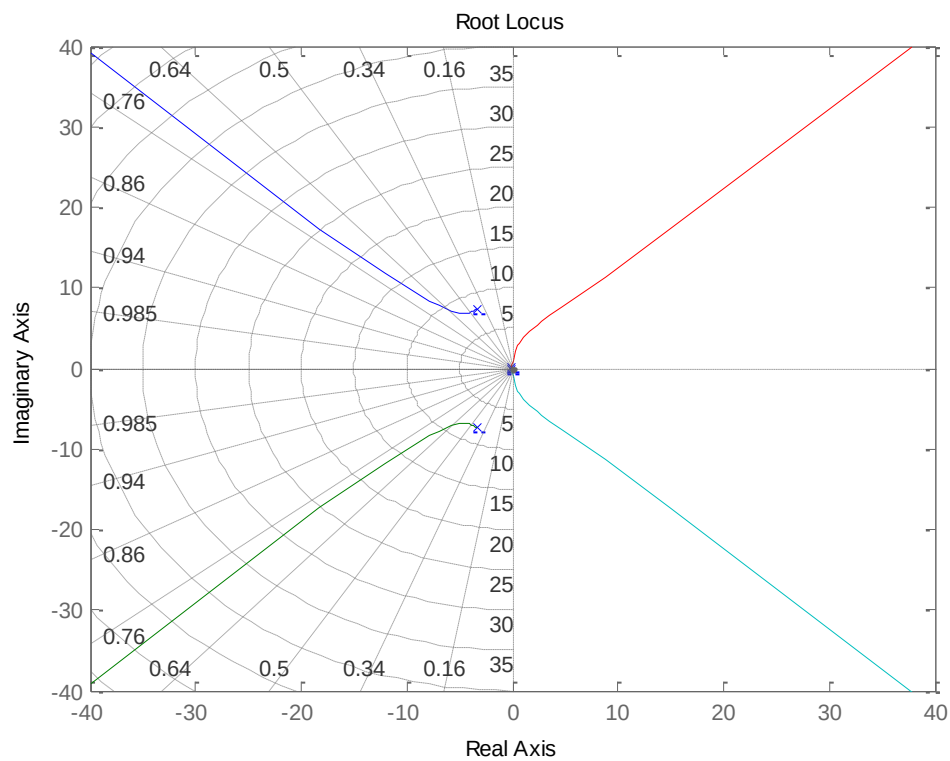
$$\dot{x} = (A - BK)x + Bvi$$

Where:

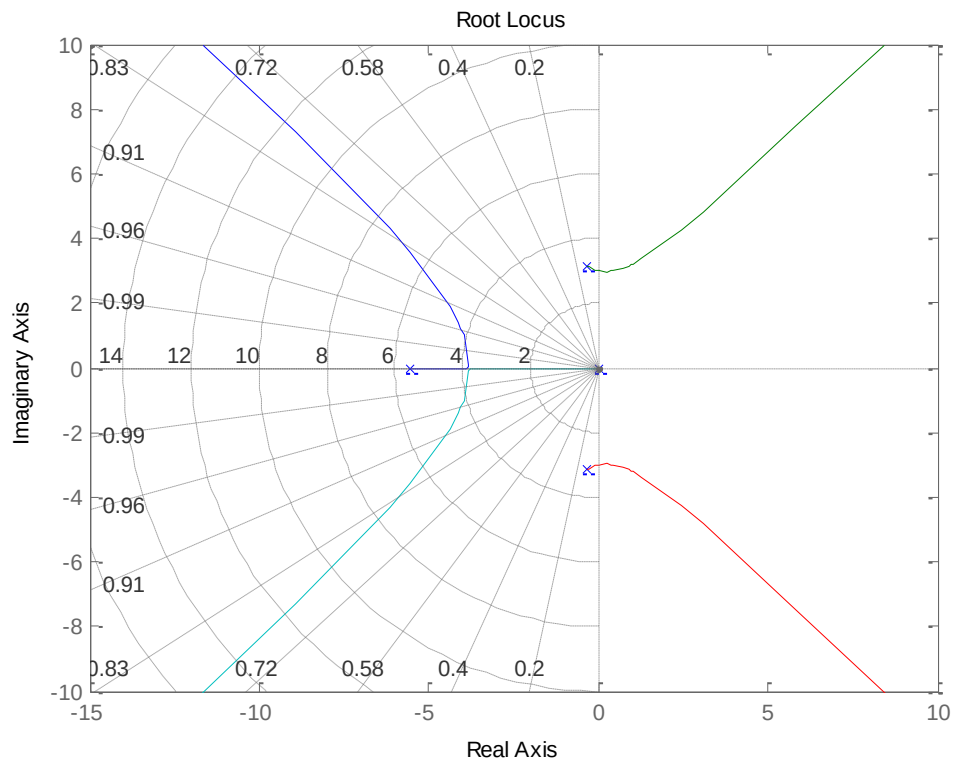
$$(A - BK) = \begin{bmatrix} 0.0142 & -7.8655 & -30.4145 & 2.8475 \\ -0.7828 & 23.0204 & 88.7308 & -0.6875 \\ 0.6842 & -8.4113 & -30.2746 & 0.5981 \\ 0 & 0 & 1 & 0 \end{bmatrix}$$

Now the augmented D-47SAW aircraft should be dynamically similar to the rigid D-47B aircraft. The root locus plot for rigid D-47B aircraft is shown in Figure 6-17.

Rigid D-47B



Longitudinal



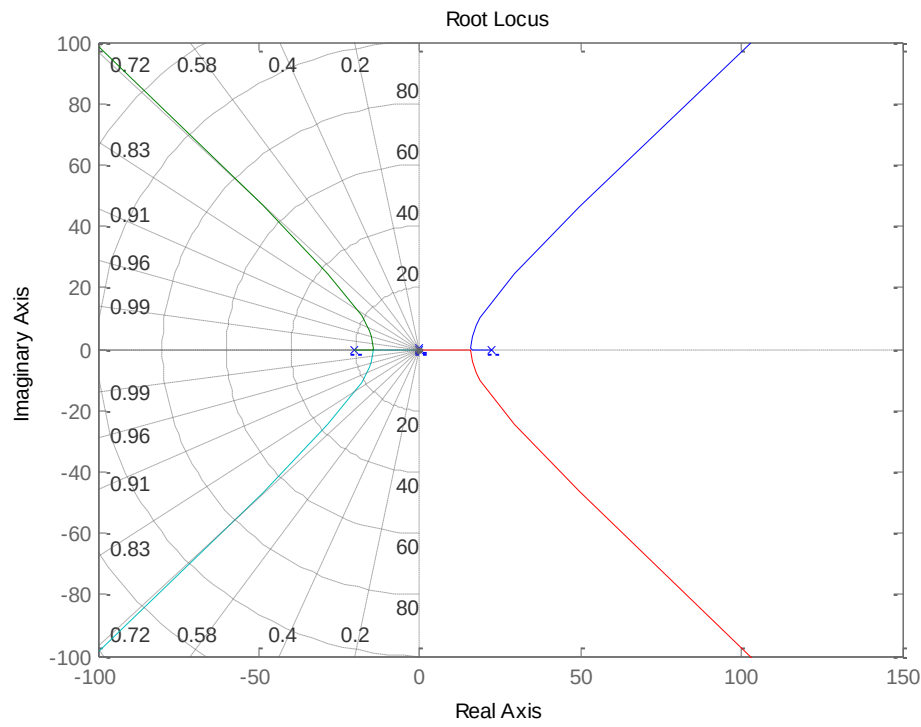
Lateral-directional

Figure 6-17 D-47B, D-47SAW and augmented D-47SAW root locus plot

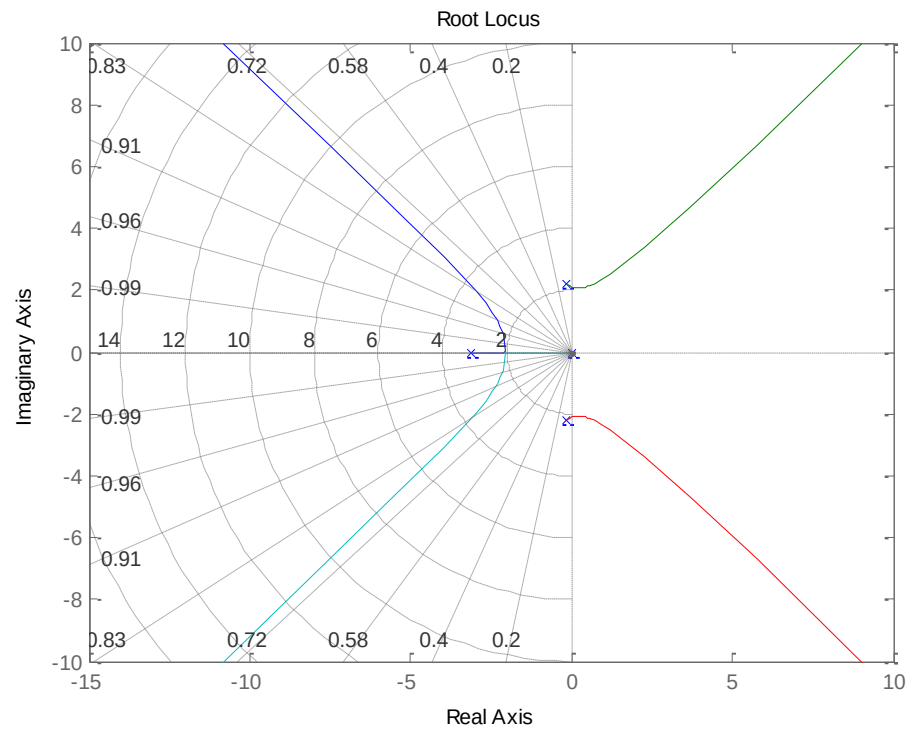
Original and relocated pole positions for D-47SAW aircraft are shown in the root locus plot in Figure 6-18.

The root locus plot above confirmed that D-47SAW aircraft is unstable, SAS can stabilized the D-47SAW aircraft and modify its dynamic modes to match that of the rigid D-47B aircraft. Since the SAS is designed on the foundation of linearized D-47SAW model, it is necessary to check if the SAS is still applicable in the non-linear D-47SAW model.

D-47SAW

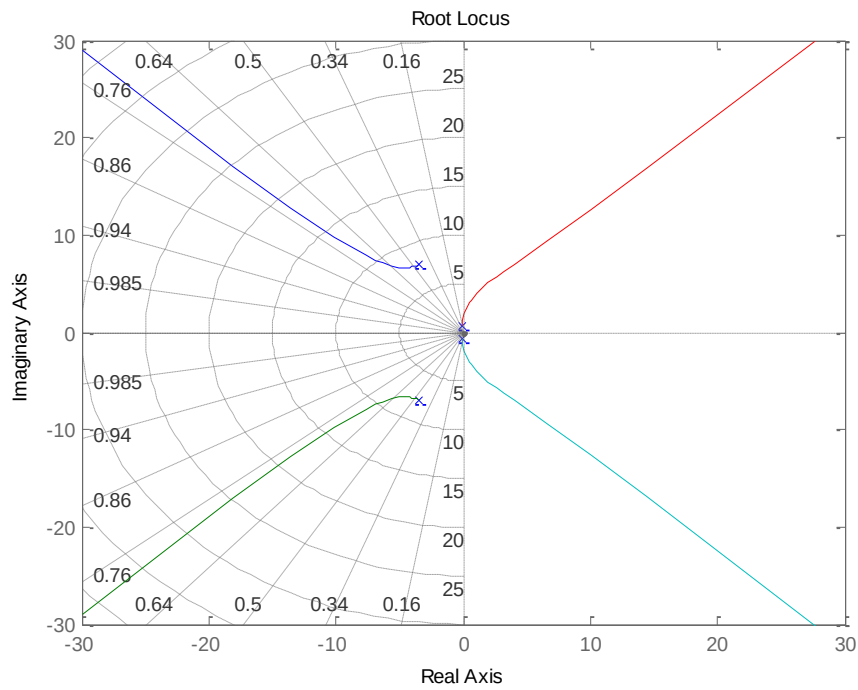


Longitudinal

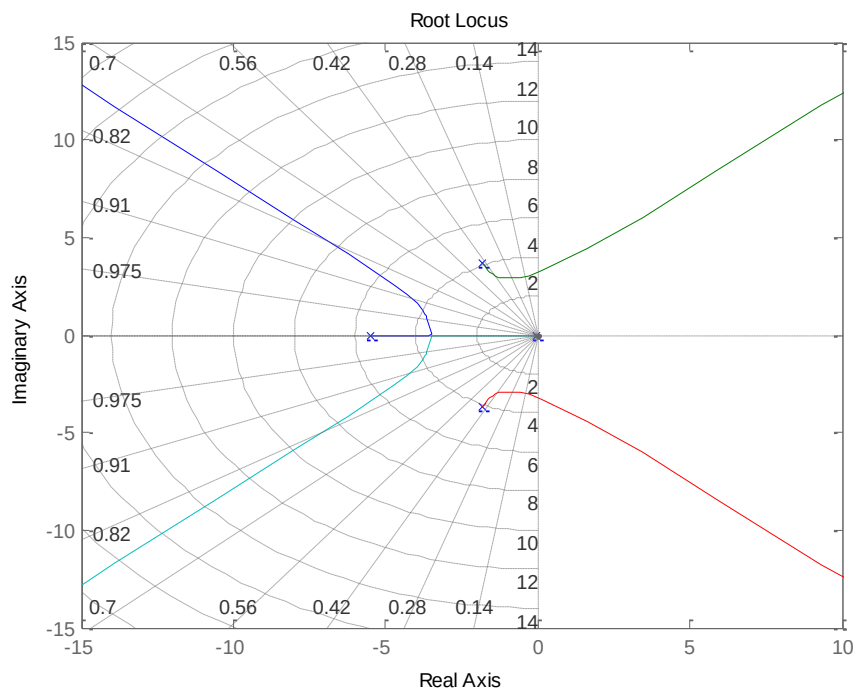


Lateral-directional

D-47SAW with SAS



Longitudinal



Lateral-directional

Figure 6-18 D-47B, D-47SAW and augmented D-47SAW root locus plot

6.3.2 Flight Response to Excitation

The validation of applicability of SAS in the non-linear D-47SAW model is carried out by acquiring time history plot of the state variables. The state variables interested were u , w , q and θ for longitudinal motion; v , p , r and ϕ for lateral directional motion.

Longitudinal response to elevator step input at 1 degree is shown in

Figure 6-19. It can be observed that reduced aerodynamic damping caused by airframe flexibility increased natural frequency of the SPPO and phugoid mode. However, SAS can introduce the damping force via control surfaces actively and quickly stabilize the aircraft to steady state attitude.

Lateral-directional response to 1 degree aileron pulse for 5 seconds is shown in Figure 6-20. Steady roll rate p of the D-47SAW aircraft is less than that of the rigid wing D-47B, increase of roll rate is also slower than D-47B. Roll subsidize is now overshooting due to inertia force of the flexible wing. Bending deformation of the wing improves the lateral stability, therefore sideslip reduces bank angle equivalent to dihedral effect.

These results demonstrated SAW aircraft flight dynamic characteristics and proved that the SAS is working effectively on the non-linear model.

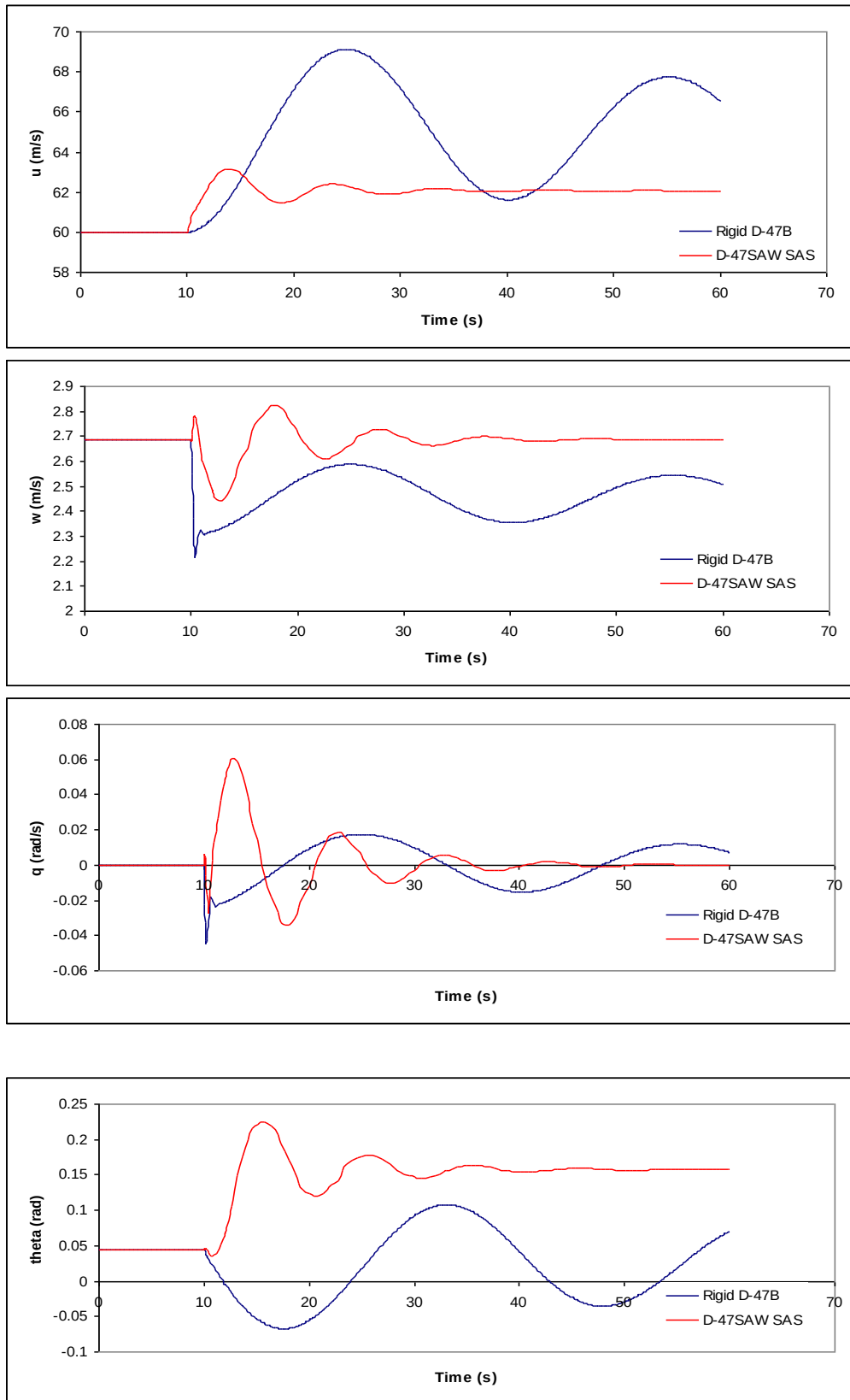


Figure 6-19 Longitudinal time history of D-47B and D-47SAW with SAS

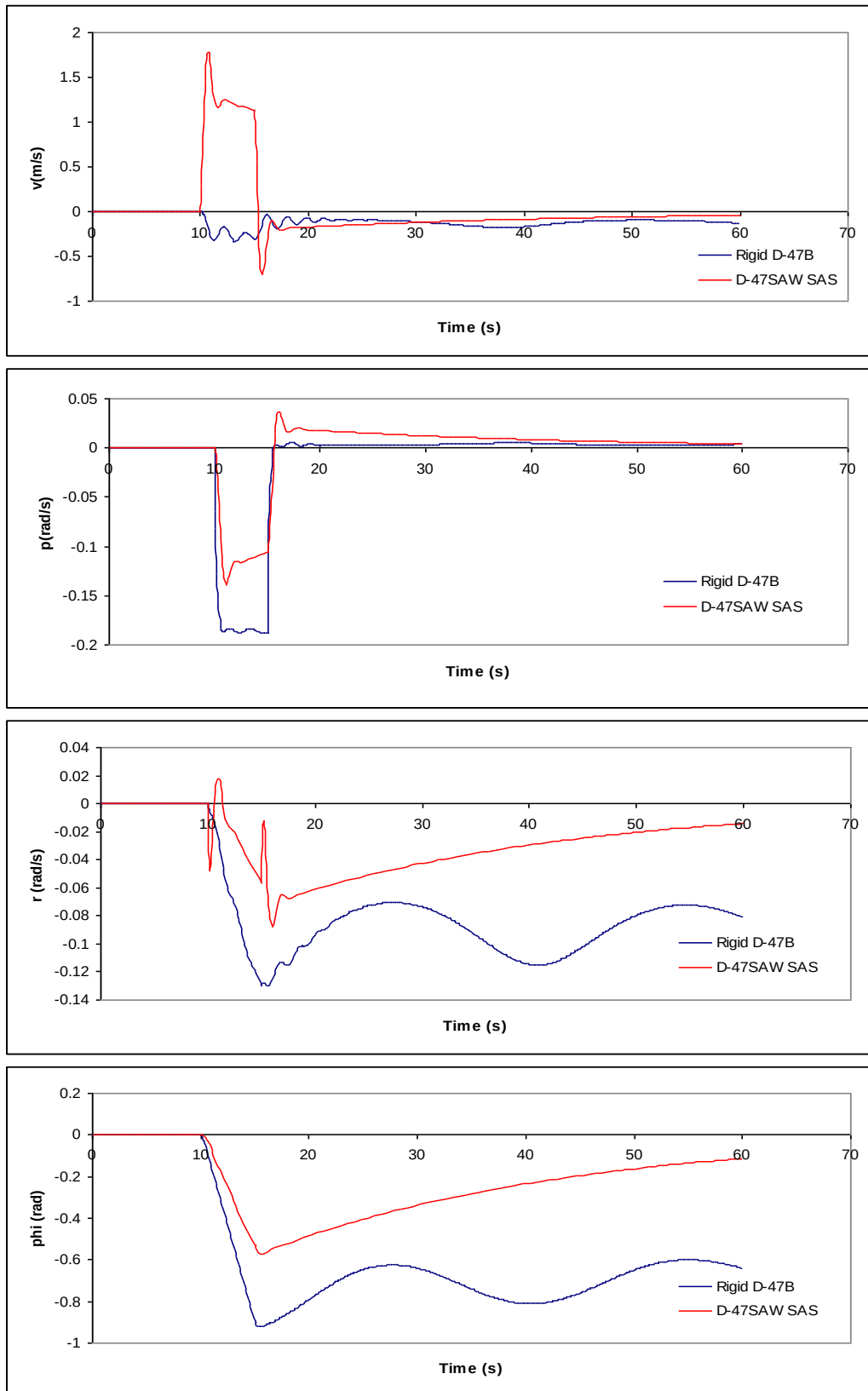


Figure 6-20 Lateral-directional time history of D-47B and D-47SAW with SAS

6.3.3 Control surface actuator effect

The control surface actuator is modelled for the D-47SAW model. Each second order nonlinear actuator follows the flight computer's command and controls deflection of one control surface. The Simulink actuator is modelled as a second order differential equation with saturation nonlinearity.

The default characteristics are:

Natural frequency	150 rad/s
Damping ratio	0.7
Maximum deflection	20 deg
Minimum deflection	-20 deg
Maximum rate	500 deg/s

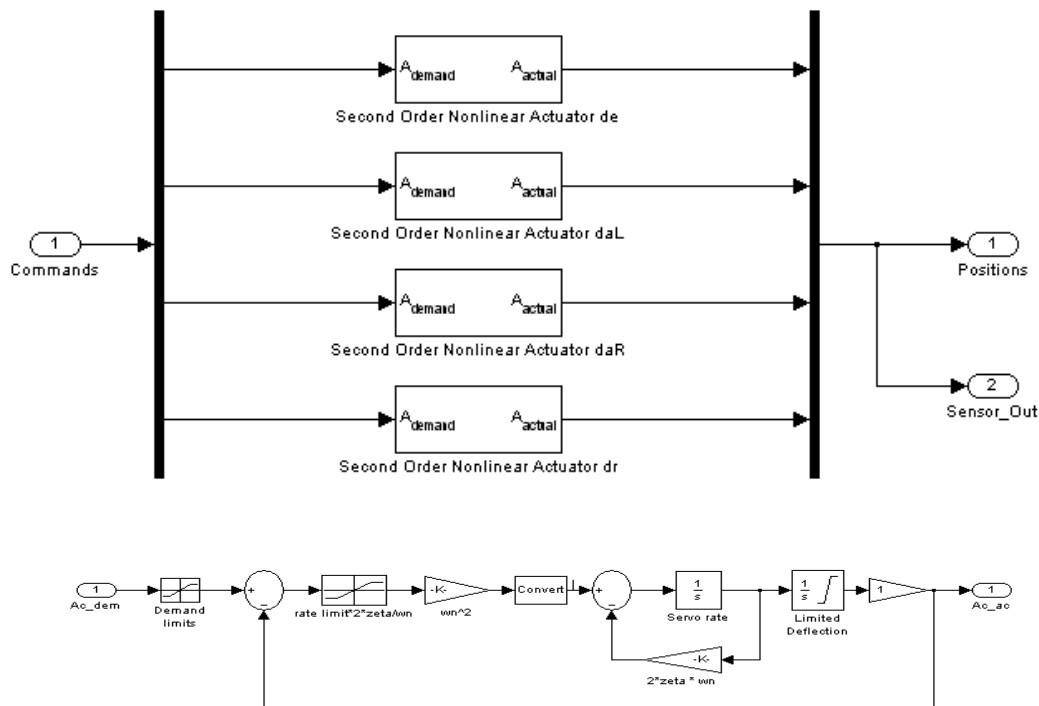


Figure 6-21 Airframe actuator model

The impact of actuator dynamics on flight response is in the study of aero-servo-elasticity, which is not covered in this analysis.

Chapter 7 Conclusion and Recommendation

This thesis presents an investigation into the design modelling and control of a seamless aeroelastic wing (SAW) aircraft. Through the study, the objectives of this research have all been achieved, which include

- A generic rigid flying wing aircraft numerical model was first constructed in Matlab Simulink environment. It provides a base and necessary data for a seamless flexible wing aircraft model. The mass, inertia and aerodynamic data can also be replaced to model a different flying wing aircraft.
- For the seamless flexible wing, a novel morphing wing actuation mechanism was designed. A demonstration prototype is manufactured and tested. Aerodynamic features of the seamless morphing airfoil are extracted.
- Based on the design, a numerical model of SAW was built. It was then integrated into the rigid body aircraft model to produce the SAW aircraft model. This allows the dynamic behaviour of the SAW structure to be taken into account in the flight dynamic performance and response.
- The study was then focused on the performance, stability and control characteristics of the SAW aircraft. The flight control compensations were made to satisfy relevant stability and flying quality requirements.

7.1 Conclusion

The aeroelastic problems of this type of flying wing SAW aircraft are mainly longitudinal instability, pitch-roll control reversal and degraded flying qualities. The first two problems are related in static deformation. The design feature of a large swept back wing of the D-47SAW has made the problem more critical due to the bending-twist coupling. Firstly, the centre of pressure of the deformed swept back wing moves forward, causing reduced static margin at low speed and pitch divergence at high speed. Secondly, spanwise accumulated negative pitching moment exacerbates the control reversal problem on both pitch and roll motion.

Without additional weight, one solution was made to change the fibre orientation of the composite skin to increase the wing rigidity in torsion or bending. This can marginally mitigate the problem. Another solution is to design distributed control surfaces. By making a closely orchestrated control deflection plan, the problem can be partially

solved to further improve the performance. Undeniably, control reversal phenomenon is physically unavoidable on swept back wing when dynamic pressure increases sufficiently beyond the safe limit. However, the creative idea of using reversed control surface input at high speed fundamentally changed the approach to this problem. Control surfaces are, in this case, manipulating wing twist to maintain desired flight control authority. This study contributes to the knowledge in this type of aircraft design and performance evaluation.

The third problem is related to flight dynamic damping, which is a typical challenge to flexible wing aircraft design. Two factors are contributing to the problem in this case. First one is the inertia force from the airframe mass itself. Second one is the velocity terms (linear and angular) in the unsteady aerodynamic force. If the wing vibrates during a flight manoeuvre, velocity dependant unsteady aerodynamic force will damp out the oscillation on the wing, but in the meantime it is acting as both positive and negative damping force on the aircraft in alternating cycles.

Reduction of flight dynamic damping can be compensated by stability augmentation system (SAS). But SAS design introduces two other problems related to sensor and actuator.

SAS input is the feedback signals from the inertia measurement unit (IMU) sensors. These accelerometers and gyros measure airframe motion as well as vibration. High frequency vibration can be filtered but low frequency vibration interferes with intended aircraft motion signal. In order to minimise low frequency vibration, the location of the IMU is preferred to be on the neutral point of the airframe vibration mode. But this location may not coincide with aircraft CG.

SAS output is executed by electrical or hydraulic actuators. Actuator dynamics introduce additional state variables to the system, causing delayed input to aircraft. Due to airframe flexibility, aircraft response is delayed further, if signal digitizing delay and data processing delay were all added together, they would undermine the close loop system stability. Wing vibration suppression is not what SAS is designed for, so actuator bandwidth does not need to cover the structure vibration frequency. This study leads to a forward way to approach a solution to this problem from system level.

7.2 Future works

- SAW aircraft flight dynamic model validation by unmanned air vehicle demonstrator.
- Leading edge seamless control surface actuation mechanism design.
- Swept forward SAW aerodynamic characteristics study and active divergence suppression system development.
- Advanced control algorithm design for D-47SAW aircraft flight control.
- Aero-servo-elastic study on the D-47SAW model with inclusion of actuator dynamics, sensor noise and digitized controller delays.

7.3 Dissemination of works

Conference paper

D. Yang, S. Guo, *Rolling Active Control for an Aircraft of Seamless Aeroelastic Wing*, 50th AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, Palm spring, CA, USA. 4-5 May 2009

D. Yang, S. Guo, *Active Control System Design for a Seamless Aeroelastic Wing*, CEAS2009 European Air and Space Conference, Manchester, UK. 2009

Daqing Yang, Melanie Perera, Shijun Guo, *Active Control of an Optimised Seamless Aeroelastic Wing*, IFASD 2009 Seattle, USA. 2009

D. Li, S. Guo, N. Di Matteo, D. Yang, *Design, Experiment and Aerodynamic Calculation of a Flapping Wing Rotor Micro Aerial Vehicle*, 52nd AIAA/SME/ ASEC/AHS/ ASC Structures, Structural Dynamics and Materials Conference, Denver, Colorado, 4-7 April 2011, AIAA 2011-1988, pp1-9

Journal paper

Published

ZJ Wang, S Guo, D Yang, Rolling active control for an aircraft of seamless aeroelastic wing, Int. Journal of Modelling, Identification and Control (IJMIC), Vol.7, No.3, 2009

Submitted

D. Yang, S. Guo, Design Modelling and Control of Seamless Aeroelastic Wing Aircraft Institution of Mechanical Engineers, Part G Journal of Aerospace Engineering, 2011.

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ESDU 71017, Effect of wing on yawing moment due to yawing

ESDU 72021, Effect of wing on rolling moment due to yawing.

ESDU 73006, Effects of isolated body and wing-body interference on rolling moment due to sideslip: $L_{\text{sub } v}$ (with Addendum A for nacelle effects).

ESDU 77021, Properties of a standard atmosphere

ESDU 77022, Equations for calculation of ISA

ESDU 79006, Wing-body yawing moment and side-force derivatives due to sideslip: N_v and Y_v (with Addendum A for nacelle effects).

ESDU 80033, Contribution of wing planform to rolling moment derivative due to sideslip, $(L_v)_w$, at subsonic speeds.

ESDU 80034, Effect of trailing-edge flaps on rolling moment derivative due to sideslip, $(L_{\text{sub } v})_{\text{sub } f}$.

ESDU 81013, Effect of trailing-edge flaps on side-force and yawing moment derivatives due to sideslip, $(Y_v)_f$ and $(N_v)_f$.

ESDU 82010, Contribution of fin to side-force, yawing moment and rolling moment derivatives due to sideslip, $(Y_v)_F$, $(N_v)_F$, $(L_v)_F$, in the presence of body, wing and tailplane.

ESDU 82017, Contribution of fin to side-force, yawing moment and rolling moment derivatives due to rate of yaw, $(Y_r)_F$, $(N_r)_F$, $(L_r)_F$.

ESDU 83006, Contribution of fin to side-force, yawing moment and rolling moment derivatives due to rate of roll, $(Y_p)_F$, $(N_p)_F$, $(L_p)_F$, in the presence of body, wing and tailplane.

ESDU 83026, Contribution of body to yawing moment and side-force derivatives due to rate of yaw, $(N_r)_B$ and $(Y_r)_B$.

ESDU 85006, Contribution of wing dihedral to side-force, rolling moment and yawing moment

ESDU 86021, Introduction to aerodynamic derivatives, equations of motion and stability,

ESDU 87008, Rudder side-force, yawing moment and rolling moment control derivatives at low speeds: $Y_{\text{sub } \zeta}$, $N_{\text{sub } \zeta}$ and $L_{\text{sub } \zeta}$.

ESDU 88013, Rolling moment derivative for plain aileron

ESDU 88029, Yawing moment coefficient for plain ailerons at subsonic speeds.

ESDU 88040, Program for the calculation of aileron rolling moment and yawing moment coefficients at subsonic speeds

ESDU 89034, The maximum lift coefficient of plain wings at subsonic speeds

ESDU 92006, A background to the handling qualities of aircraft.

ESDU 92029, Contribution of ventral fins to side-force and yawing moment derivatives due to sideslip at low angle of attack.

ESDU 96003, Lift curve of wings with high-lift devices deployed at low speeds

ESDU 96037, A qualitative introduction to static aeroelasticity

ESDU 97032, Static aeroelasticity: a formal analysis using assumed modes

ESDU 98024, Quaternion representation of aeroplane attitude and motion characteristics

ESDU 99033, Static aeroelasticity: a formal analysis using normal modes

ESDU A.06.01.01, Stability derivative L_p , rolling moment due to rolling for swept wing

ESDU A.06.01.03, Stability derivative (L_v) sub gamma. Contribution of full-span dihedral to rolling moment due to sideslip.

APPENDICES

Appendix A SAW structure properties

A.1 SAW structure Beam3D.exe input file

9 0 1 2 24 2

1

0 0 0 0 0 0

23.0913 0.087872 0 0 0.01 0.01 0.01 0 0 0

0.0 0.0 0.0 0.0 0.0 0.0

0 0 0

2

0 2 3 0 5 0

3.32780 0.174965 0 0 0.01 0.01 0.01 0 0 0

0.0 0.0 0.0 0.0 0.0 0.0

0.1678 0.1999 0

3

0 2 3 0 5 0

3.26840 0.171842 0 0 0.01 0.01 0.01 0 0 0

0.0 0.0 0.0 0.0 0.0 0.0

0.3355 0.3999 0

4

0 2 3 0 5 0

3.14690 0.165454 0 0 0.01 0.01 0.01 0 0 0

0.0 0.0 0.0 0.0 0.0 0.0

0.5033 0.5998 0

5

0 2 3 0 5 0

2.95540 0.155386 0 0 0.01 0.01 0.01 0 0 0

0.0 0 0.0 0.0 0.0 0.0

0.6711 0.7998 0

6

0 2 3 0 5 0

2.68120 0.140969 0 0 0.01 0.01 0.01 0 0 0

0.0 0 0.0 0.0 0.0 0.0

0.8388 0.9997 0

7

0 2 3 0 5 0

2.29880 0.120864 0 0 0.01 0.01 0.01 0 0 0

0.0 0 0.0 0.0 0.0 0.0

1.0066 1.1996 0

8

0 2 3 0 5 0

1.94400 0.102209 0 0 0.01 0.01 0.01 0 0 0

0.0 0 0.0 0.0 0.0 0.0

1.1744 1.3996 0

9

0 2 3 0 5 0

0.9060 0.047635 0 0 0.01 0.01 0.01 0 0 0

0.0 0 0.0 0.0 0.0 0.0

1.3421 1.5995 0

2 1

1.60E+10	2.36E-06	1.53E-06	5.62E-04	2.31E-04	4.00E+09
3.43E-05	2.34E-05	5.02E-04	4.73E-04	5.02E-04	
4.73E-04	5.02E-04	4.73E-04	-4.26E+03		

1 2 0 0

2 1

1.60E+10	1.53E-06	1.12E-06	2.31E-04	2.10E-04	4.00E+09
2.34E-05	1.89E-05	4.73E-04	4.57E-04	4.73E-04	
4.57E-04	4.73E-04	4.57E-04	-3.07E+03		

2 3 0 0

2 1

1.60E+10	1.12E-06	7.69E-07	2.10E-04	1.88E-04	4.00E+09
1.89E-05	1.47E-05	4.57E-04	4.41E-04	4.57E-04	
4.41E-04	4.57E-04	4.41E-04	-2.47E+03		

3 4 0 0

2 1

1.60E+10	7.69E-07	4.98E-07	1.88E-04	1.67E-04	4.00E+09
1.47E-05	1.09E-05	4.41E-04	4.25E-04	4.41E-04	
4.25E-04	4.41E-04	4.25E-04	-1.86E+03		

4 5 0 0

2 1

1.60E+10	4.98E-07	2.99E-07	1.67E-04	1.45E-04	4.00E+09
1.09E-05	7.75E-06	4.25E-04	4.09E-04	4.25E-04	
4.09E-04	4.25E-04	4.09E-04	-1.34E+03		

5 6 0 0

2 1

1.60E+10	2.99E-07	1.71E-07	1.45E-04	1.24E-04	4.00E+09
7.75E-06	5.23E-06	4.09E-04	3.93E-04	4.09E-04	
3.93E-04	4.09E-04	3.93E-04	-9.02E+02		

6 7 0 0

2 1

1.60E+10	1.71E-07	9.25E-08	1.24E-04	1.02E-04	4.00E+09
5.23E-06	3.40E-06	3.93E-04	3.77E-04	3.93E-04	
3.77E-04	3.93E-04	3.77E-04	-5.74E+02		

7 8 0 0

2 1

1.60E+10	9.25E-08	8.69E-08	1.02E-04	8.10E-05	4.00E+09
3.40E-06	1.78E-07	3.77E-04	3.61E-04	3.77E-04	
3.61E-04	3.77E-04	3.61E-04	-3.46E+02		

8 9 0 0

-1

A.2 Beam3D.exe instruction

DATA FILE BEAM3D.IN FOR PROGRAM BEAM3D.EXE

NN, ICK, IUNIT, ISMA, IPR

NN – Total Number of Node/Stations ($1 \leq NN \leq 40$ for the current version);

ICK -- Control Marker for Stiffness Coupling (ICK=1 to include coupling K_{35} =

SL/PAK(NK,15); =0 for no stiffness coupling);

IUNIT -- Control Marker for UNIT System (=1 for SI unit; the input data will be

converted to IMP. Unit for calculation within the program; =2 IMP. Unit.

In either Unit, the [K], [M] within the code is in IMP Unit);

ISMA -- Control Marker for Analysis (=1 for static; =2 for vibration analysis);

IPR -- Control Marker for Print Results (=0 for end results only; =1, 2 to print the system and element stiffness/mass matrices);

NI -- The Nth Node/Station;

NW(NI, J), (J=1-6) – Freedom indicator for the displacements u, w, Φ , v, θ , ψ at the node

(=0 for constrained; >0 for free DOF);

ALOAD(NI, J), (J=1-6) – Load and moments applied at the node: S_x , S_z , T_y , P_y , M_x , M_z

PAM(NI, J) – Mass, mass moments and inertias at the node

M_I , MX_I , MY_I , MZ_I , I_{XXI} , I_{YYI} , I_{ZZI} , I_{XYI} , I_{YZI} , I_{XZI}

(e.g. $MX_I = M_I(X_m - X_I)$, $I_{XXI} = M_I[(Y_m - Y_I)^2 + (Z_m - Z_I)^2]$)

COR(NI,J), J=1-3 – X, Y, Z-coordinates of the node NI;

(the above data will repeat NN times)

ITY, NE – constants 2, 1

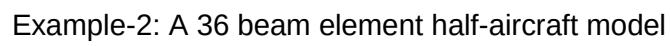
PAK(NI, J) – 15 Properties E , I_{XX1} , I_{XX2} , I_{ZZ1} , I_{ZZ2} , G , J_1 , J_2 , A_1 , A_2 , A_{SX1} , A_{SX2} , A_{SZ1} , A_{SZ2} , CK

N12(NI,J) – The node No. at the ends of the element, and ICY and $\tan(a)$ to indicate the

orientation and incidence of the beam respectively;

(the above data will repeat NN-1 times)

MEND=-1 – End of data file marker;



A.3 SAW mass and stiffness matrix

Mass matrix of the SAW

0.228025	-0.03933	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-0.03933	0.007376	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0.007376	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	0.223955	-0.03863	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	-0.03863	0.007376	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0.007376	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	0.21563	-0.0372	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	-0.0372	0.007376	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0.007376	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	0.202508	-0.03493	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	-0.03493	0.007376	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0.007376	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	0.183719	-0.03169	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	-0.03169	0.007376	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0.007376	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0.157517	-0.02717	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	-0.02717	0.007376	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0.007376	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0.133205	-0.02298	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	-0.02298	0.007376	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0.007376	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0.06208	-0.01071	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	-0.01071	0.007376	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0.007376

Stiffness matrix of the SAW

2395784	81271.3	-96450.1	-974106	-254066	303000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
81271.3	617428.7	41463.45	281884.4	-106320	-188648	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-96450.1	41463.45	602830.4	-336177	-188648	-39518.9	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-974106	281884.4	-336177	1666572	51538.88	-61730.2	-692466	-178693	212877.3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
-254066	-106320	-188648	51538.88	448121.2	35795.43	202526.6	-90518.8	-146682	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
303000	-188648	-39518.9	-61730.2	35795.43	435476.6	-241270	-146682	-38904.5	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	-692466	202526.6	-241270	1155081	42160.55	-50144.5	-462614	-118148	140819.9	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	-178693	-90518.8	-146682	42160.55	338329.9	49373.91	136532.8	-74271.2	-109475	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	212877.3	-146682	-38904.5	-50144.5	49373.91	320936	-162733	-109475	-35638.1	0	0	0	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	-462614	136532.8	-162733	752747.3	31575.62	-37624.9	-290133	-73058	87085.86	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	-118148	-74271.2	-109475	31575.62	244769.7	53154.93	86572.24	-57823.4	-77982.4	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	140819.9	-109475	-35638.1	-37624.9	53154.93	226007.5	-103195	-77982.4	-30288.6	0	0	0	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	-290133	86572.24	-103195	460532.6	21807.74	-26031.4	-170400	-42559	50700.44	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	-73058	-57823.4	-77982.4	21807.74	169108.8	49745.05	51250.28	-42874.5	-53289	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	87085.86	-77982.4	-30288.6	-26031.4	49745.05	151568.1	-61054.4	-53289	-24123.2	0	0	0	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	-170400	51250.28	-61054.4	265452.3	13731.74	-16341.4	-95052.7	-23501.7	28011.54	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	-42559	-42874.5	-53289	13731.74	112118.9	41215.15	28827.21	-29944.6	-34753.5	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	50700.44	-53289	-24123.2	-16341.4	41215.15	97600.08	-34359	-34753.5	-17680.2	0	0	0
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	-95052.7	28827.21	-34359	161465.4	5041.515	-6006.84	-66412.6	-18079.9	21551.37
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	-23501.7	-29944.6	-34753.5	5041.515	72252.68	30148	18460.16	-19213.1	-22922.4
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	28011.54	-34753.5	-17680.2	-6006.84	30148	61611.5	-22004.7	-22922.4	-11119.5
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	-66412.6	18460.16	-22004.7	66412.64	18079.86	-21551.4
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	-18079.9	-19213.1	-22922.4	18079.86	29160.62	11064.93
0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	21551.37	-22922.4	-11119.5	-21551.4	11064.93	25253.7

Appendix B Tornado Aerodynamic Coefficients and Derivatives for D-47B Aircraft

B.1 Zero-lift-drag coefficients

TORNADO Zero lift drag estimation.

JID:	D-47B	Re:	4830017
Reference area:	3.7607	CD ₀ :	0.0067206
Reference chord:	1.1756	Swet:	9.2245
Reference span:	4.2	Volume:	0.30167

Zero lift drag per part:	0.00060829	0.0007178	0.00063833	0.0015545	0.001545	0.0005947
	0.00030585	0.00025019	0	0	0	0
	0.00050593	0	0	0	0	0

Wetted area per part:	0.97763	1.4025	0.77913	1.9305	1.9188	0.61352
	0.43052	0.56318	0	0	0	0
	0.60871	0	0	0	0	0

Internal volume per part:	0.072003	0.089404	0.034602	0.053363	0.038011	0.0065458
	0	0	0	0	0	0
	0.0077441	0	0	0	0	0

B.2 Angle and Rate derivatives

TORNADO CALCULATION RESULTS, Derivatives

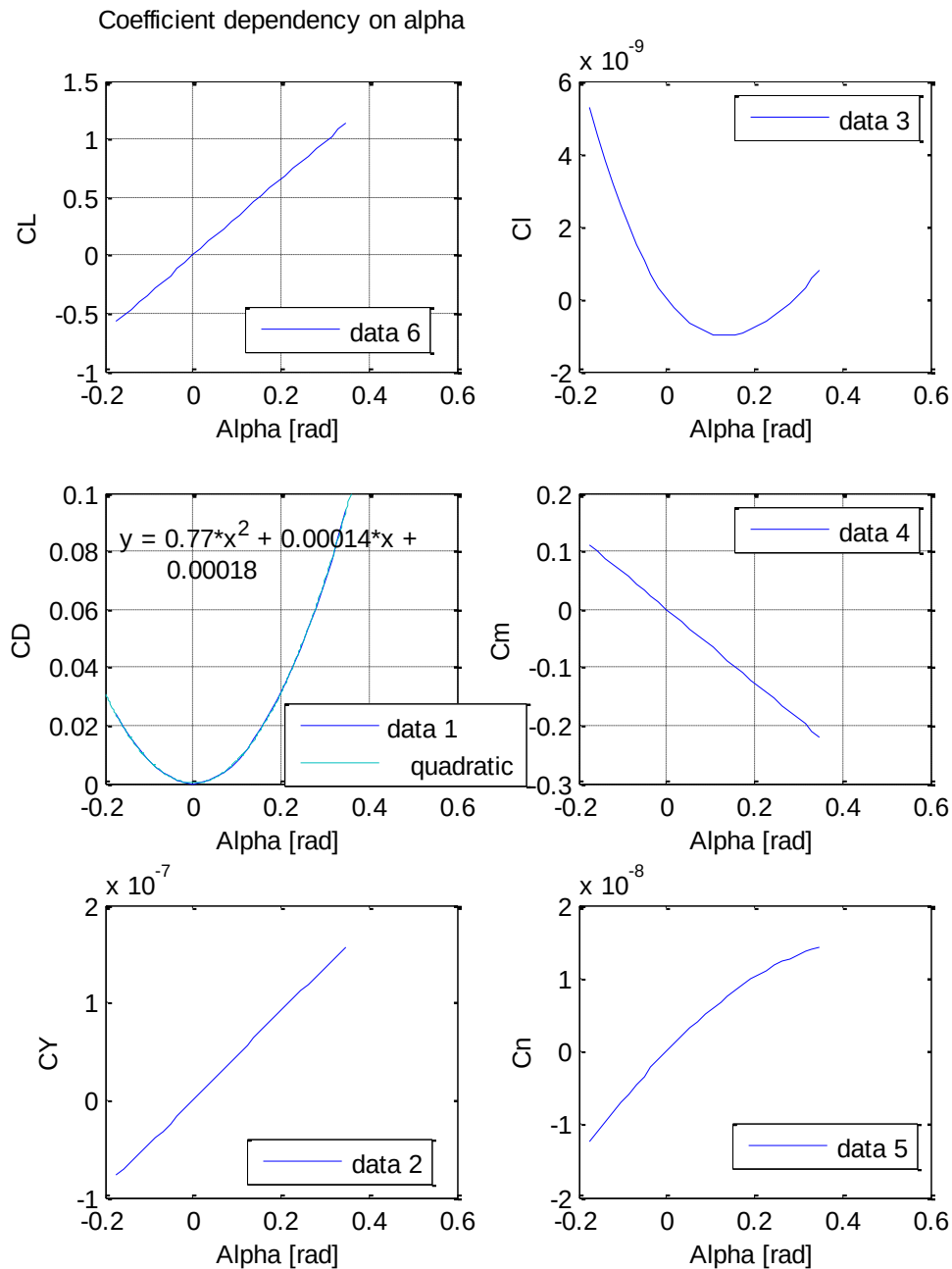
JID:	D-47B		
Reference area:	3.7607	α :	0
Reference chord:	1.1756	β :	0
Reference span:	4.2	Airspeed:	60
		P:	0
		Q:	0
		R:	0

CL derivatives :	CD derivatives :	CY derivatives :
CL _{α} 3.314	CD _{α} 7.9065e-005	CY _{α} 4.5862e-007
CL _{β} -1.3895e-005	CD _{β} 9.3499e-006	CY _{β} -0.27953
CL _P 8.1271e-007	CD _P 7.4953e-007	CY _P -0.014994
CL _Q 4.0267	CD _Q 1.5121e-006	CY _Q 1.0714e-006
CL _R 2.328e-007	CD _R 6.9148e-008	CY _R -0.12287

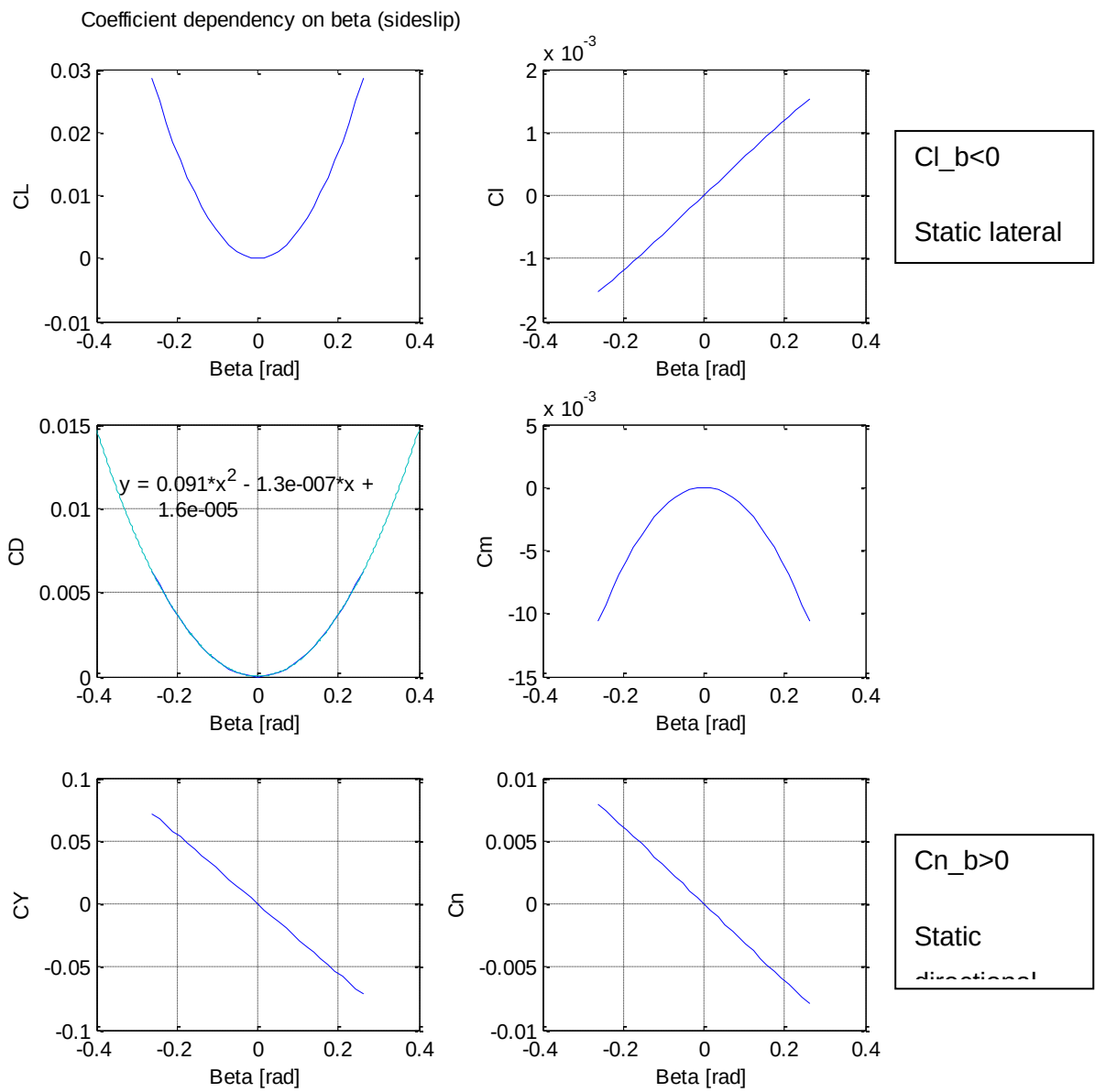
Roll derivatives :	Pitch derivatives :	Yaw derivatives :
Cl _{α} -1.6311e-008	Cm _{α} -0.63335	Cn _{α} 6.2199e-008
Cl _{β} 0.0060375	Cm _{β} 3.1498e-006	Cn _{β} -0.030903
Cl _P -0.31584	Cm _P -2.266e-008	Cn _P -0.00075701
Cl _Q -5.1634e-008	Cm _Q -1.7006	Cn _Q 1.8649e-007
Cl _R 0.0046033	Cm _R 2.0563e-007	Cn _R -0.021227

Curve fitting to find induced drag coefficient C_D with the 2nd order polynomial for angle of attack (AOA) and side slip.

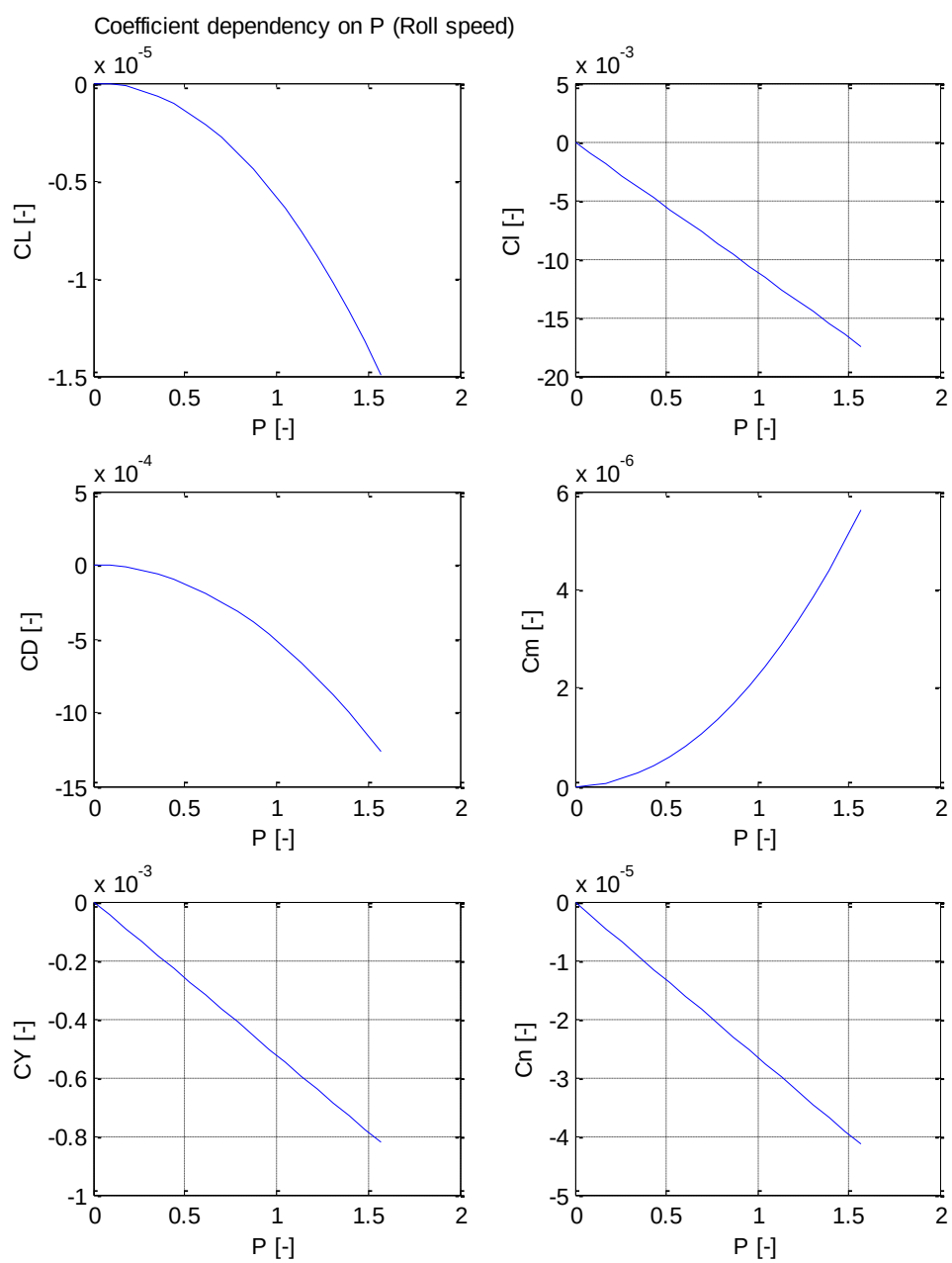
Derivatives due to AOA α



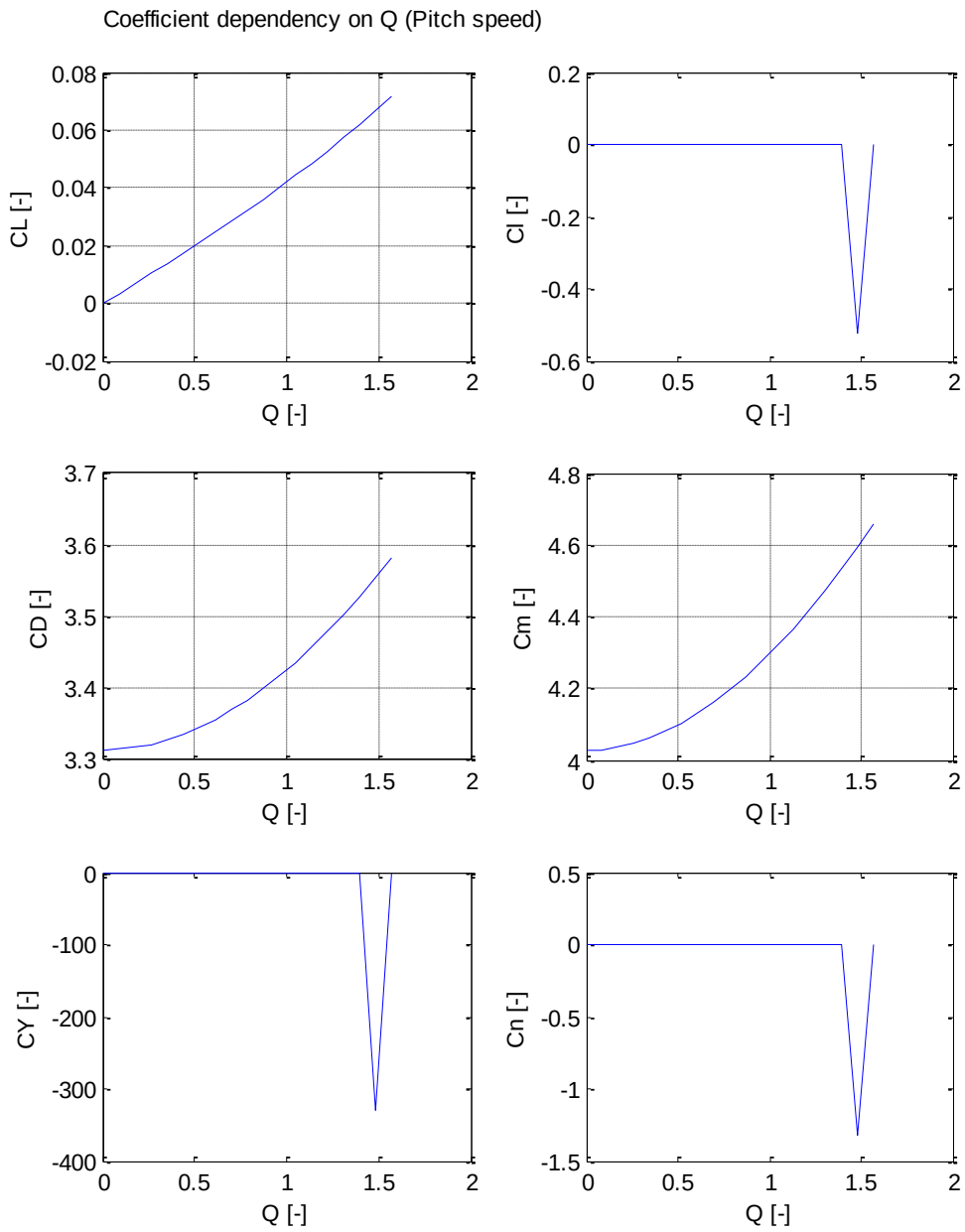
Derivatives due to Side slip angle β



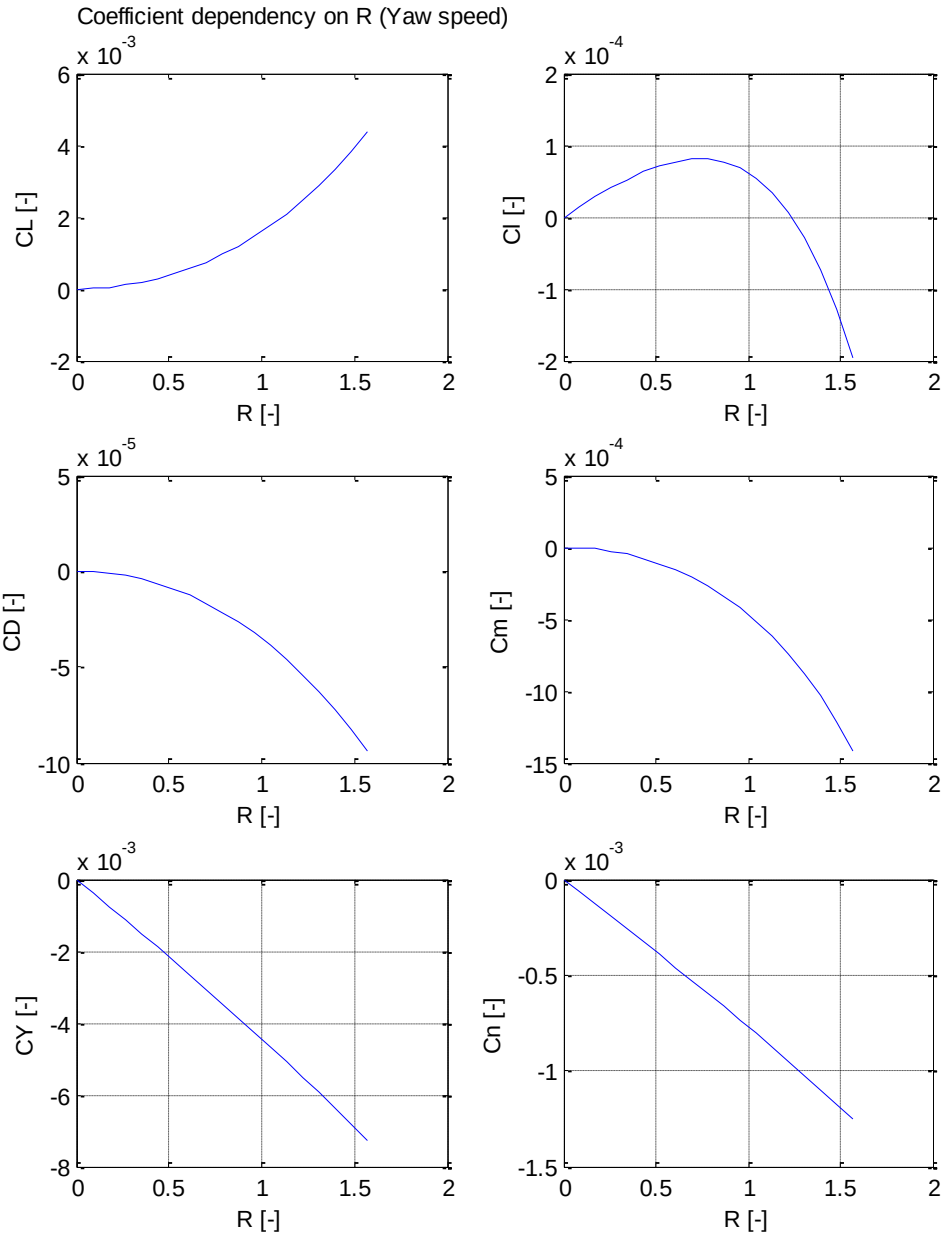
Derivatives due to Roll rate P



Derivatives due to Pitch rate Q



Derivatives due to Yaw rate R



B.3 Control derivatives

TORNADO CALCULATION RESULTS, Central difference, RUDDER DERIVS

JID: D-47B

Reference area: 3.7607 α : 0 P: 0

Reference chord: 1.1756 β : 0 Q: 0

Reference span: 4.2 Airspeed: 60 R: 0

CL_δ	-1.9349e-008	CD_δ	6.6949e-006	CY_δ	0.0064771
	-1.5929e-006		1.3736e-006		-0.091234

Cl_δ	0.12192	Cm_δ	-3.2243e-008	Cn_δ	0.00099669
	0.0051276		1.9812e-006		-0.017396

TORNADO CALCULATION RESULTS, Central difference, RUDDER DERIVS

JID: D-47Be

Reference area: 3.7607 α : 0 P: 0

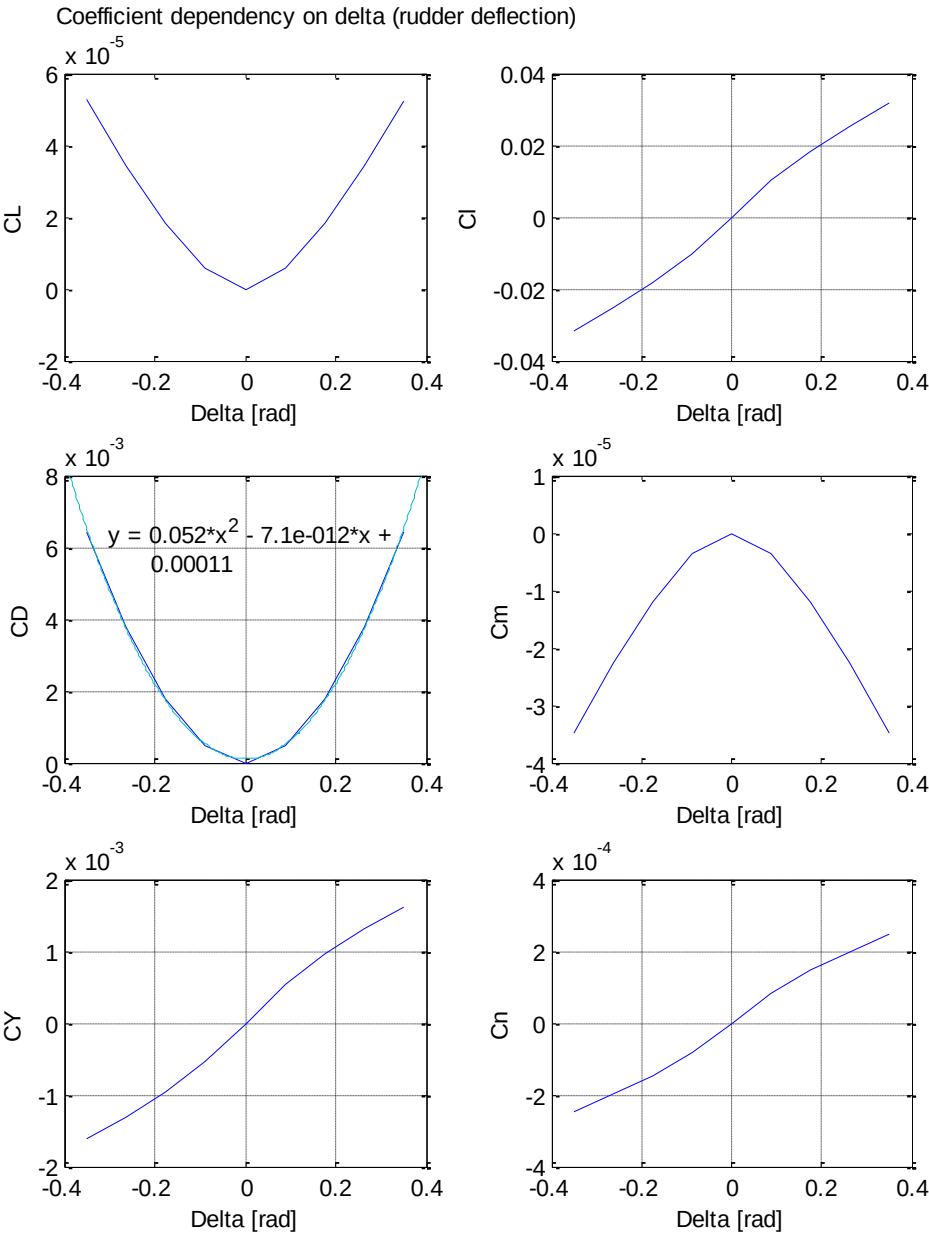
Reference chord: 1.1756 β : 0 Q: 0

Reference span: 4.2 Airspeed: 60 R: 0

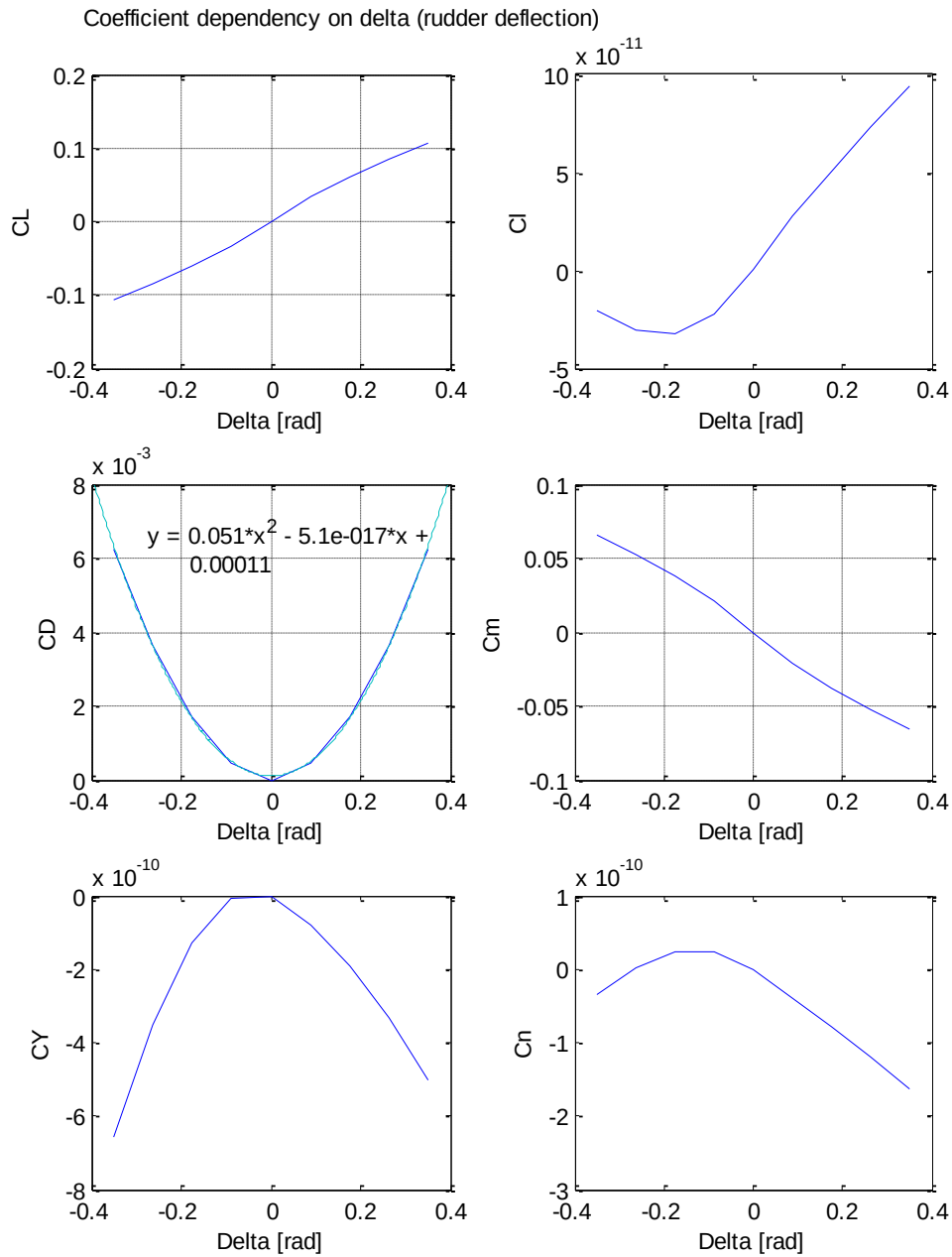
CL_δ	0.41622	CD_δ	6.5622e-006	CY_δ	-5.33e-010
	-1.5929e-006		1.3736e-006		-0.091234

Cl_δ	3.1543e-010	Cm_δ	-0.25794	Cn_δ	-3.9921e-010
	0.0051276		1.9812e-006		-0.017396

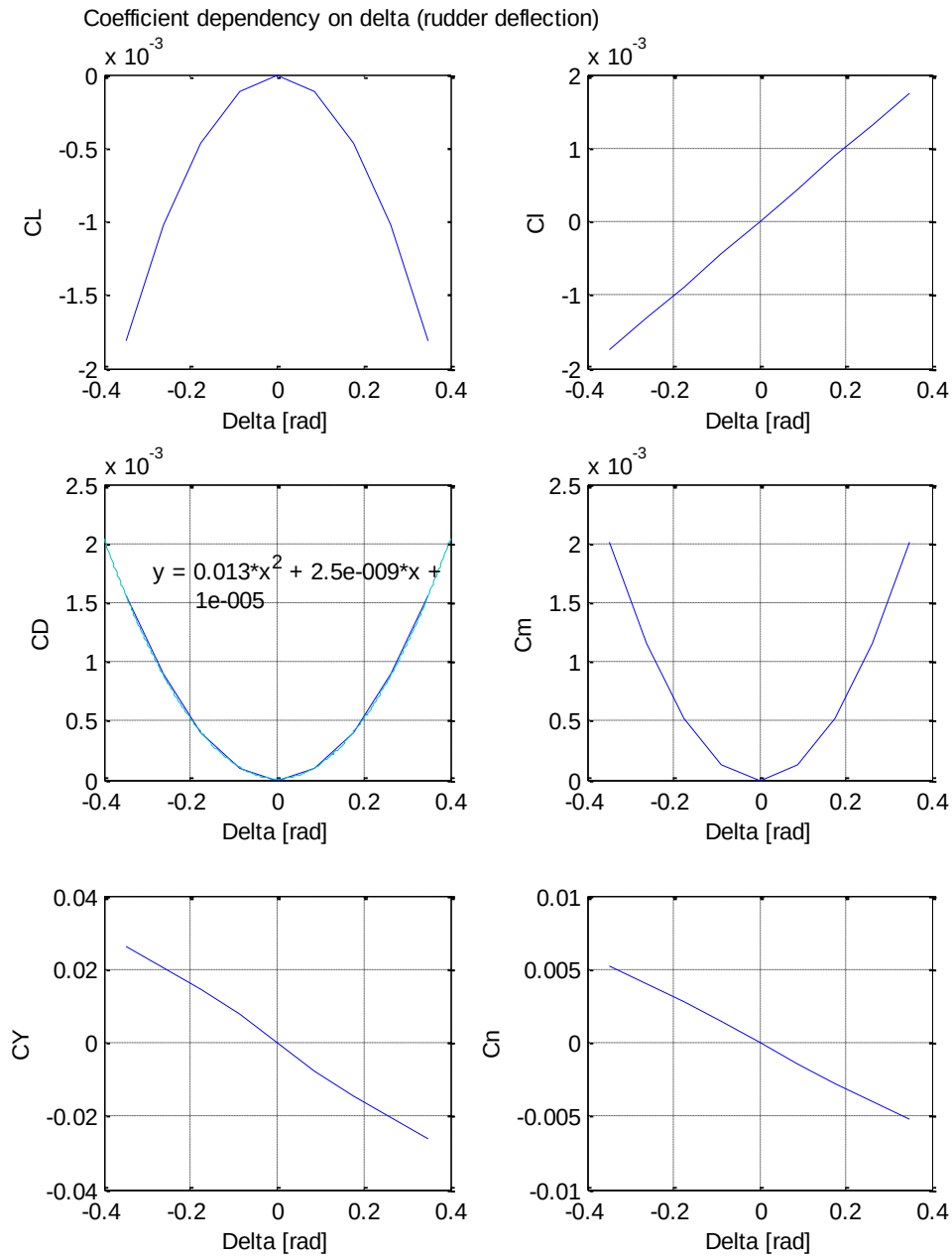
Derivatives when Elevon is used as Aileron ($\pm 20^\circ$)



Derivatives when Elevon is used as Elevator ($\pm 20^\circ$)



Derivatives due to Rudder ($\pm 20^\circ$)



B.4 Calculation of air data

This block converts body velocity and atmospheric data into flight parameter.

Total velocity - true air speed

$$V_T = \sqrt{u^2 + v^2 + w^2}$$

Flight Mach number

$$M = \frac{V_T}{a}$$

Body angle of attack

$$\alpha = \sin^{-1}\left(\frac{w}{\sqrt{u^2 + v^2}}\right)$$

Body sideslip angle

$$\beta = \sin^{-1}\left(\frac{v}{V_T}\right)$$

Dynamic pressure

$$\bar{q} = \frac{1}{2} \rho V_T^2$$

B.5 Calculation of aerodynamic force and moments

Total force coefficients

$$\begin{aligned}
 C_{D_total} &= C_D + \alpha \times C_{D_a} + \beta \times C_{D_b} + p \times C_{D_p} \times \frac{b_{ref}}{2V_t} + q \times C_{D_q} \times \frac{c_{ref}}{2V_t} + r \times C_{D_r} \times \frac{b_{ref}}{2V_t} + \\
 &\delta_{aL} \times C_{D_aL} + \delta_{aR} \times C_{D_aR} + \delta_r \times C_{D_r} \\
 C_{Y_total} &= C_Y + \alpha \times C_{Y_a} + \beta \times C_{Y_b} + p \times C_{Y_p} \times \frac{b_{ref}}{2V_t} + q \times C_{Y_q} \times \frac{c_{ref}}{2V_t} + r \times C_{Y_r} \times \frac{b_{ref}}{2V_t} + \\
 &\delta_{aL} \times C_{Y_aL} + \delta_{aR} \times C_{Y_aR} + \delta_r \times C_{Y_r} \\
 C_{L_total} &= C_L + \alpha \times C_{L_a} + \beta \times C_{L_b} + p \times C_{L_p} \times \frac{b_{ref}}{2V_t} + q \times C_{L_q} \times \frac{c_{ref}}{2V_t} + r \times C_{L_r} \times \frac{b_{ref}}{2V_t} + \\
 &\delta_{aL} \times C_{L_aL} + \delta_{aR} \times C_{L_aR} + \delta_r \times C_{L_r}
 \end{aligned}$$

Total moment coefficients

$$\begin{aligned}
 C_{l_total} &= C_l + \alpha \times C_{l_a} + \beta \times C_{l_b} + p \times C_{l_p} \times \frac{b_{ref}}{2V_t} + q \times C_{l_q} \times \frac{c_{ref}}{2V_t} + r \times C_{l_r} \times \frac{b_{ref}}{2V_t} + \\
 &\delta_{aL} \times C_{l_aL} + \delta_{aR} \times C_{l_aR} + \delta_r \times C_{l_r} \\
 C_{m_total} &= C_m + \alpha \times C_{m_a} + \beta \times C_{m_b} + p \times C_{m_p} \times \frac{b_{ref}}{2V_t} + q \times C_{m_q} \times \frac{c_{ref}}{2V_t} + r \times C_{m_r} \times \frac{b_{ref}}{2V_t} + \\
 &\delta_{aL} \times C_{m_aL} + \delta_{aR} \times C_{m_aR} + \delta_r \times C_{m_r} \\
 C_{n_total} &= C_n + \alpha \times C_{n_a} + \beta \times C_{n_b} + p \times C_{n_p} \times \frac{b_{ref}}{2V_t} + q \times C_{n_q} \times \frac{c_{ref}}{2V_t} + r \times C_{n_r} \times \frac{b_{ref}}{2V_t} + \\
 &\delta_{aL} \times C_{n_aL} + \delta_{aR} \times C_{n_aR} + \delta_r \times C_{n_r}
 \end{aligned}$$

Force equation

$$D = \bar{q} S_{ref} C_{D_total}, \quad Y = \bar{q} S_{ref} C_{Y_total}, \quad L = \bar{q} S_{ref} C_{L_total}$$

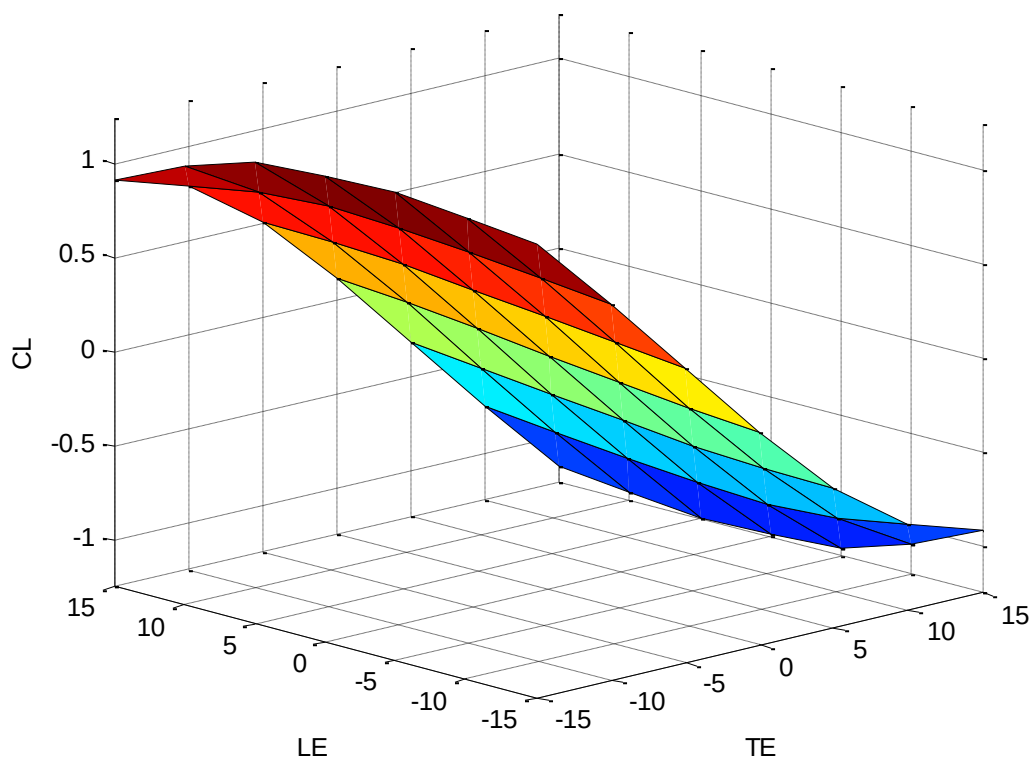
Moment equation

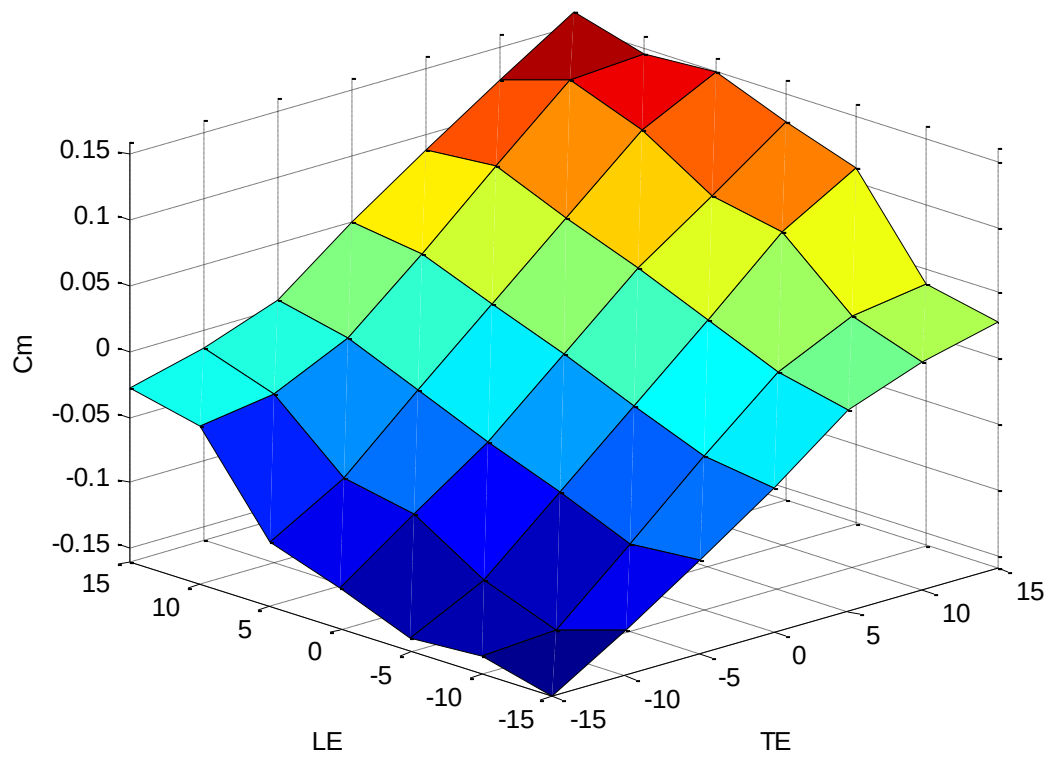
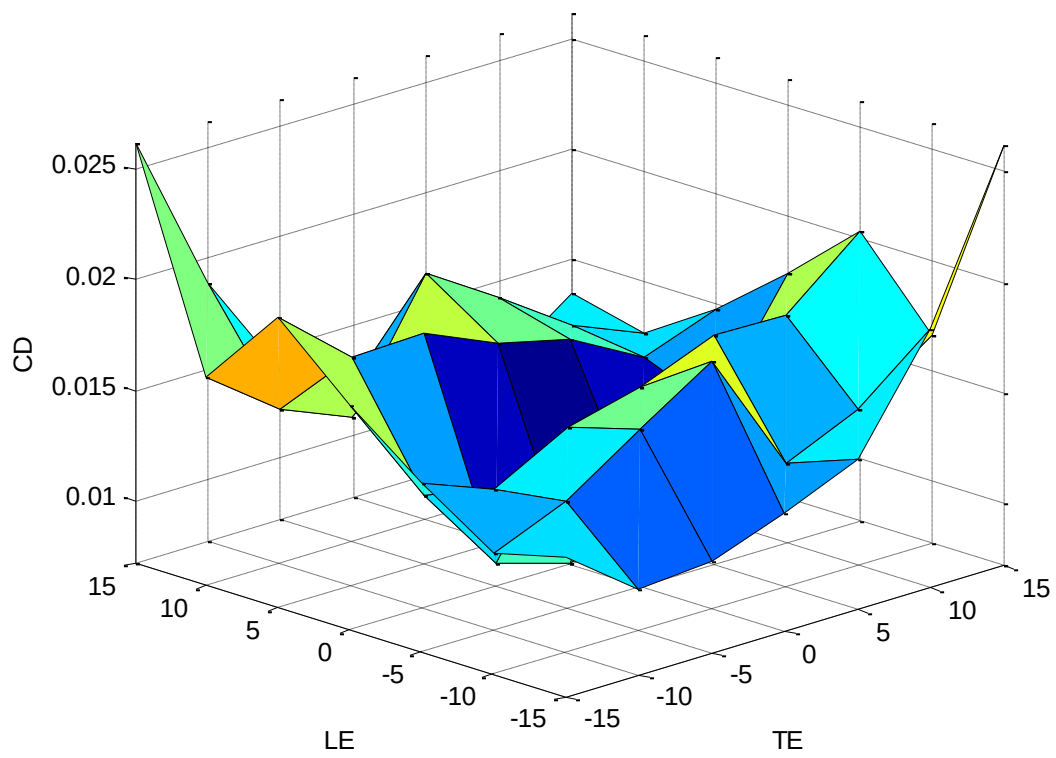
$$L = \bar{q} S_{ref} b_{ref} C_{l_total}, \quad M = \bar{q} S_{ref} c_{ref} C_{m_total}, \quad N = \bar{q} S_{ref} b_{ref} C_{n_total}$$

Appendix C Aerodynamic data of SAW LE and TE combination

Surface plot of NACA0009 aerodynamic property with **hinged** LE and TE.

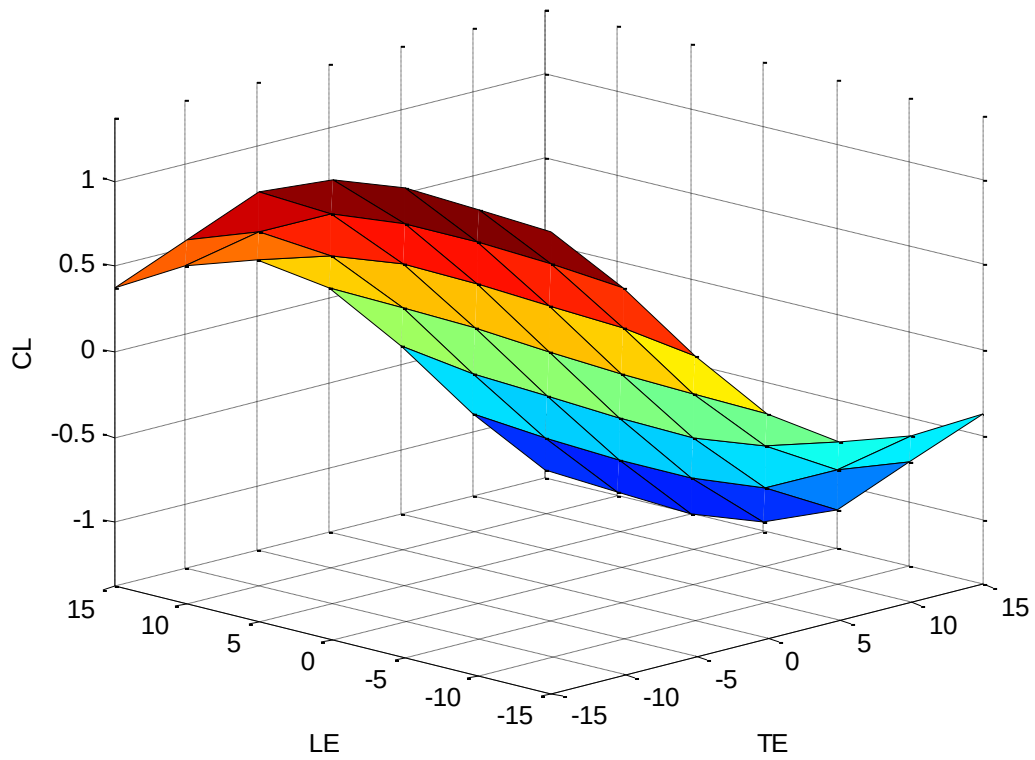
Deflection range is -15 to +15 degree with 5 degree interval.

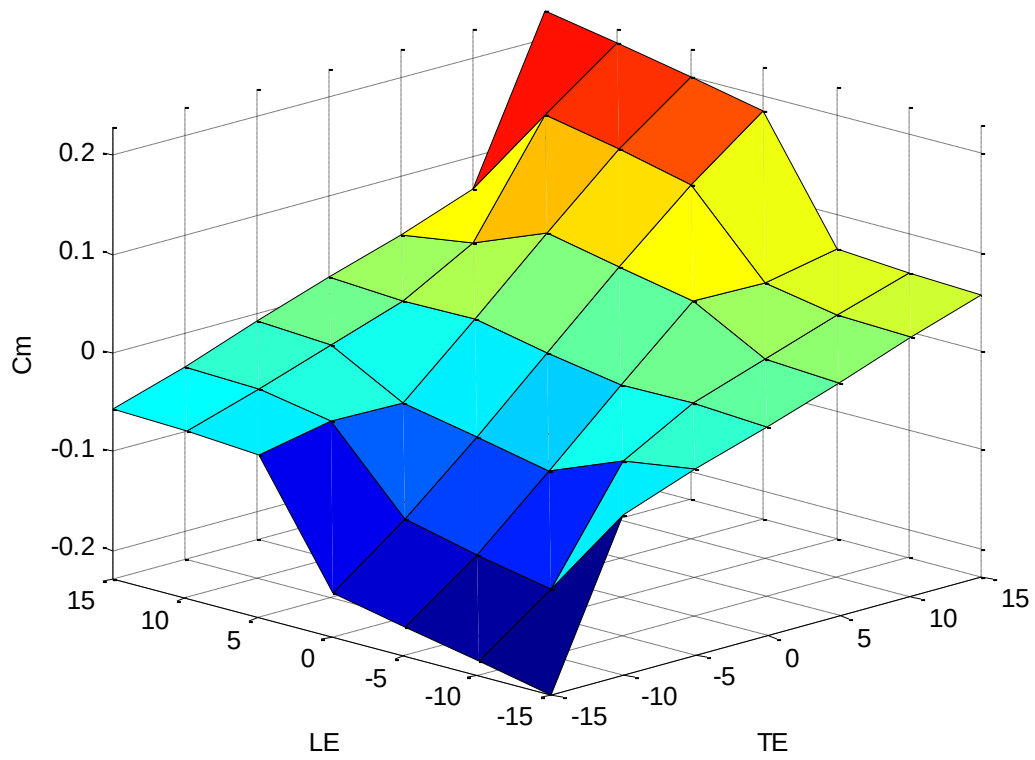
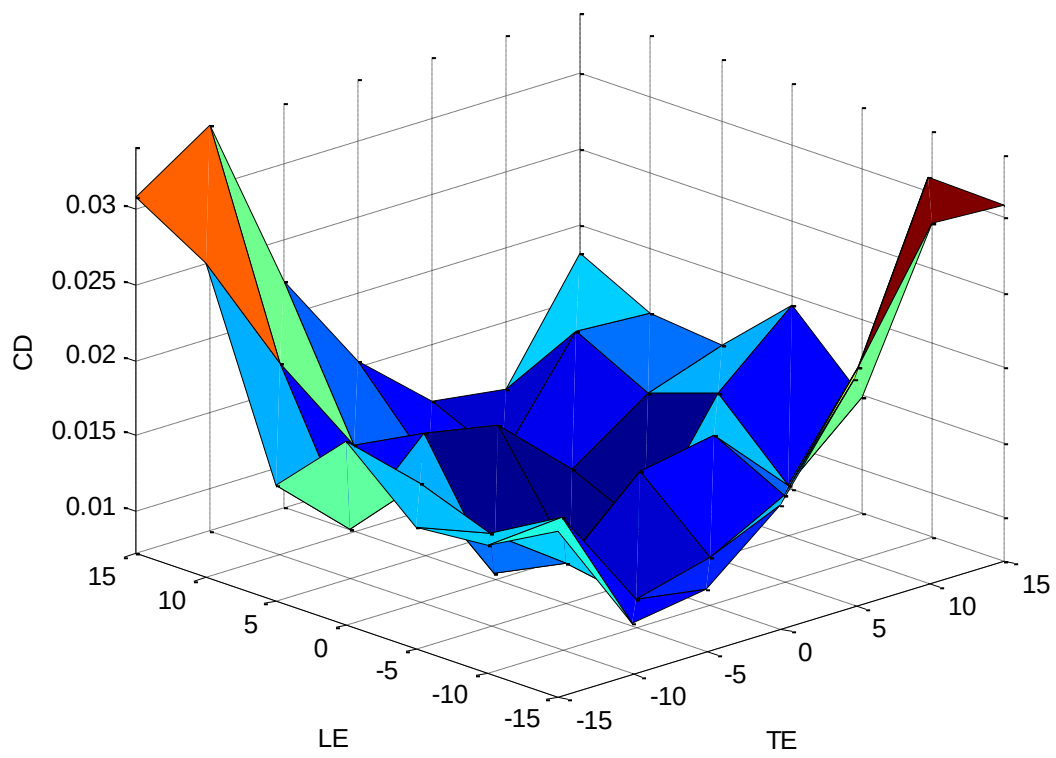




Surface plot of NACA0009 aerodynamic property with **Seamless** LE and TE.

Deflection range is -15 to +15 degree with 5 degree interval.





Appendix D Mass and inertia properties for D-47 Aircraft

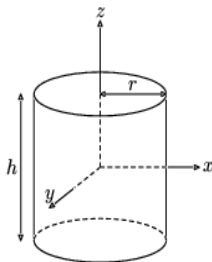
Mass Breakdown	Original (kg)	Double (kg)	Modified (kg)
Structural items	11.94	23.88	36.38
Power supply, Sensors & Actuators	3.7	7.4	10
Main and nose landing gear	1.7	3.4	3.6
Avionics	3.6	7.2	7.2
Pneumatics system	4.3	8.6	0
Fuel system	1.2	2.4	2.4
Engine	2.5	5	5
Engine air intake	0.46	0.92	0.92
FTV system	2	4	0
Circulation Control system	1.3	2.6	0
Ballast (dead weight)	0.2	0.4	0.5
Total Empty Mass	32.9	65.8	66
Fuel	6.8	13.6	22
Maximum Take-Off Mass	39.7	79.4	88
Maximum Landing Mass	37.9	75.8	80

Mass & Inertia Properties

MTOW	88.00	kg
CG _x	1.74	m
CG _y	0.00	m
CG _z	0.03	m
I _x	70.01	kgm ²
I _y	109.27	kgm ²
I _z	113.66	kgm ²
I _{xy}	0.00	kgm ²
I _{yz}	0.00	kgm ²
I _{xz}	1.57	kgm ²

I_{xx}

When looking for the moment of inertia in roll motion.
Fuselage can be assumed as a solid cylinder.



Fuselage moment of inertia I_{xx} about fuselage centre is

$$I_z = \frac{mr^2}{2} = 5.10 \text{ kgm}^2$$

Fuselage moment of inertia I_{xx} about aircraft CG is

$$= 5.30 \text{ kgm}^2$$

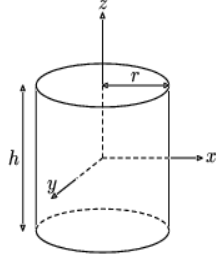
Wing moment of inertia I_{xx} about aircraft CG is

$$= 64.72 \text{ kgm}^2$$

Total I_{xx} is the summation

I _{xx} total=	70.01 kgm ²
------------------------	------------------------

lyy



Fuselage moment of inertia lyy about nose datum is

$$I_z = \frac{mr^2}{2} = 75.17 \text{ kgm}^2$$

Fuselage moment of inertia lyy about aircraft CG is

$$= 48.95 \text{ kgm}^2$$

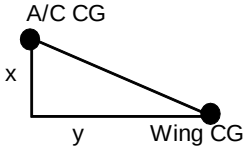
Wing moment of inertia lyy about nose datum is 204.12 kgm²

Wing moment of inertia lyy about aircraft CG is 60.32 kgm²

Total lyy about aircraft CG is

lyy total = 109.27 kgm²

lzz



Fuselage lzz about nose datum is = 75.17 kgm²

Fuselage lzz about CG is = 48.95 kgm²

Wing moment of inertia lzz about aircraft CG is 64.72 kgm²

Total lzz about aircraft CG is

lzz total = 113.66 kgm²

lxz

Fuselage lxx about nose datum is = 3.88 kgm²

Fuselage lxx about CG is = 1.57 kgm²

Wing lxx about nose datum is = 0 kgm²

Wing lxx about CG is = 0 kgm²

Total lxx about aircraft CG is

lxx total = 1.57 kgm²

Appendix E Matlab codes

E.1 D47SAW Aircraft Input

```
%% D-47SAW Aircraft data 25/03/2011
clear all
clc
% Mass & Inertia
% Inertia = [Ixx -Ixy -Ixz; -Ixy Iyy -Iyz; -Ixz -Iyz Izz];
Mass = 88.0146;
MassFuse = 40.775;
m0 = 3.0913;
m1 = 3.3278;
m2 = 3.2684;
m3 = 3.1469;
m4 = 2.9554;
m5 = 2.6812;
m6 = 2.2988;
m7 = 1.944;
m8 = 0.906;
Ixx = 70.01190169;
Iyy = 109.2716194;
Izz = 113.6640375;
Ixy = 0;
Iyz = 0;
Ixz = 1.568377947;
% Centre of Gravity and Centre of Aero Pressure (Add CP in Z direction)
% +ve is forward right and down
CG=[-1.7447 0 0];
CP=[0 0 -0.18096];
% Geometric Reference Values
Sref = 3.7607;
cref = 1.1756;
bref = 4.2;

% Engine data
MaxThrust = 230;

%% Aerodynamic derivatives come from 'Tornado Vortex Lattice' version
134 Beta
%due to inaccurate drag estimation, CD were expressed in 2nd order
%polynomial to compensate.

% Datum coeff
CL = 0.000000;
CD = 0.006721;
CY = 0.000000;
Cl = 0.000000;
Cm = 0.000000;
Cn = 0.000000;

CL_a = 3.314;
CD_a1 = 0.77; CD_a2 = 0.00014; CD_a3 = 0.00018; % 2nd poly
CY_a = 0.000000;
```

```

Cl_a = 0.000000;
Cm_a = -0.63335;
Cn_a = 0.000000;

CL_b = 0.000000;
CD_b = 0.091;          % 2nd poly
CY_b = -0.27953;
Cl_b = -0.00604;       %change +ve from tornado to -ve
Cm_b = 0.000000;
Cn_b = 0.030903;       %change -ve from tornado to +ve

% Damping coeff
CL_P = 0.000000;
CD_P = 0.000001;
CY_P = -0.014994;
Cl_P = -0.31584;
Cm_P = 0.000000;
Cn_P = -0.000757;

CL_Q = 4.0267;
CD_Q = 0.000000;
CY_Q = 0.000000;
Cl_Q = 0.000000;
Cm_Q = -1.7006;
Cn_Q = 0.000000;

CL_R = 0.000000;
CD_R = 0.000000;
CY_R = -0.12287;
Cl_R = 0.0046033;      %very complicated
Cm_R = 0.000000;
Cn_R = -0.021227;

% Control increment coeff
% Elevator (removed)
CL_de = 0;
CD_de = 0;
CY_de = 0;
Cl_de = 0;
Cm_de = 0;
Cn_de = 0;
% Aileron (Elevon)
CL_da = 0.41622;
CD_da = 0.052;          % 2nd poly
CY_da = -0.006477;      %change +ve from tornado to -ve
Cl_da = -0.12192;       %change +ve from tornado to -ve
Cm_da = -0.25794;
Cn_da = -0.000997;      %change +ve from tornado to -ve
% Rudder
CL_dr = 0 ;
CD_dr = 0 ;
CY_dr = -0.091234 ;
Cl_dr = 0.0051276 ;
Cm_dr = 0 ;
Cn_dr = -0.017396 ;

%% Initialize flight condition

```

```
%Set initial value SI units
NORTH_0= 0;
EAST_0 = 0;
DOWN_0 = 0;
U_0 = 40;
V_0 = 0;
W_0 = 0;
PHI_0=0;
THETA_0=0;
PSI_0=0;
P_0=0;
Q_0=0;
R_0=0;
```

```
de=0;          % deg
da=0;          % deg
dr=0;          % deg
thr=0;         % 0~1
```

```
NORTH_dot=0;      %x(1)
EAST_dot=0 ;      %x(2)
DOWN_dot=0 ;      %x(3)
U_dot=0 ;         %x(4)
V_dot =0 ;        %x(5)
W_dot=0 ;         %x(6)
PHI_dot=0 ;       %x(7)
THETA_dot=0;      %x(8)
PSI_dot =0 ;      %x(9)
P_dot =0 ;        %x(10)
Q_dot =0 ;        %x(11)
R_dot=0 ;         %x(12)
```

```
VT=sqrt(U_0^2+W_0^2+V_0^2);
gamma=0;
```

```
%% Distributed Control Command
```

```
qbar_sched = [750  980 1240  1531  1853  2205  2588  3001
3445  3920  4425  4961  5528  6125  6753  7411
];
```

```
TE_out_R = [1  1  1  0  0  0  -1  -1  -1  -1  -1  -1  -1  -1  -1
-1];
TE_in_R = [1  1  1  1  1  1  1  1  1  1  1  1  0  0  0
0];
LE_R = [0  0  0  0  0  0  0  0  0  0  0  0  0  0  0 0];

TE_out_P = [1  1  1  0  0  0  -1  -1  -1  -1  -1  -1  -1  -1  -1
-1];
TE_in_P = [0  0  1  1  1  1  1  0  0  0  0  0  0  0  0
0];
LE_P = [0  0  1  1  1  1  1  0  0  0  0  0  0  0  0 0];
```

E.2 Direct_MK09

Direct solution to mass and stiffness matrix of 9 nodes

```
%BEAM3D_READ
%Read full aircraft model .out file
%07/11/2008, 10:17

%% Open .out Output File

% Define output file name
Beam3D_name_output = 'beam3d.out';
% Open .out file
fid = fopen(Beam3D_name_output);
% Scan text in .out file to variable
Beam3D_output = textscan(fid,'%s');

%% Identify Stiffness, [K], and Mass, [M], Matrices

% Find number of degrees of freedom in model, i.e. size of M and K
DOF = str2double(Beam3D_output{1,1}(7));

% Initialise mass and stiffness matrices
K = zeros(DOF,DOF);
M = zeros(DOF,DOF);

% Loop through .out file, and find mass and stiffness matrices
for i = 1:size(Beam3D_output{1,1},1)
    switch cell2mat(Beam3D_output{1,1}(i))
        case 'STIFFNESS'
            if strcmp( cell2mat(Beam3D_output{1,1}(i - 1)) ,
'THE' ) %Stiffness Matrix
                if strcmp( cell2mat(Beam3D_output{1,1}(i + 1)) ,
'MATRIX' )
                    l = 1;
                    m = 4;
                    for j = 1:DOF
                        for k = 1:DOF
                            K(j,k) =
str2double(Beam3D_output{1,1}(i + l + m));
                            l = l + 1;
                        end
                    end
                end
            end
        case 'MASS'
            if strcmp( cell2mat(Beam3D_output{1,1}(i + 1)) ,
'MATRIX' ) %Mass Matrix
                l = 1;
                m = 5;
                for j = 1:DOF
                    for k = 1:DOF
                        M(j,k) = str2double(Beam3D_output{1,1}(i +
l + m));
                        l = l + 1;
                    end
                end
            end
        end
    end
end
```

```

                                break %Iteration completed, found [K] and [M], end
loop
                                end
                                end
                                end
%% Set damping C and excitation force F

    % Assume zeta=2% structure damping, then C=2*zeta*K/w
% f1=35.987 f2=135.647 f3=315.271
C=2*0.02*K/35.987; % 1st mode freq
% F=[12.3859722 0 0, 12.4048234 0 0, 12.7915846 0 0, 13.4338332 0 0,
14.0082648 0 0, 12.09747 0 0, 5.1773562 0 0, -3.0890416 0 0]';

```

E.3 Trim and Linearization

```

%% Trim the aircraft in flight
% Trim the aircraft in flight using the command '[x,u,y,dx]'
=trim('sys',x0,u0,y0,ix,iu,iy,dx0,idx) '
% Define variables & derivatives, then freeze selected variables &
% derivatives for desired trim condition (Level flight or steady climb
&turn).
% Daqing Yang 10/2010

```

```

%% Define Variables & Derivatives
% Find state variables x0 in the Simulink model
% x0=[NORTH_0;          %x(1)
%     EAST_0 ;          %x(2)
%     DOWN_0 ;          %x(3)
%     U_0 ;             %x(4)
%     V_0 ;             %x(5)
%     W_0 ;             %x(6)
%     PHI_0 ;           %x(7)
%     THETA_0;          %x(8)
%     PSI_0 ;           %x(9)
%     P_0 ;             %x(10)
%     Q_0 ;             %x(11)
%     R_0] ;            %x(12)
[sizes,x0,xstr] = SAW_Aircraft_trim([],[],[],0);

```

```

%Define input variables u0
u0=[da;                %u(1)
    de;                %u(2)
    dr;                %u(3)
    thr];              %u(4)

```

```

%Define output variables y0
y0=[VT; gamma];

```

```

% Define state derivative dx0
dx0=[NORTH_dot;        %x(1)
     EAST_dot ;        %x(2)
     DOWN_dot ;        %x(3)
     U_dot ;           %x(4)
     V_dot ;           %x(5)

```

```

W_dot      ;          %x(6)
PHI_dot    ;          %x(7)
THETA_dot  ;          %x(8)
PSI_dot    ;          %x(9)
P_dot      ;          %x(10)
Q_dot      ;          %x(11)
R_dot]     ;          %x(12)

% Freeze Variables & Derivatives

%Define freeze state variables
ix=[3;5;6;7;8;9;10;11;12];
% ix=[3;5;7;10;11;12];
%Define freeze input variables
iu=[];

%Define freeze output variables
iy=[1;2];

% Define freeze state derivative
idx=[3;4;5;6;7;8;9;10;11;12];

% Run Trim Funtion
[xe,ue,ye,dx] = trim('SAW_Aircraft_trim',x0,u0,y0,ix,iu,iy,dx0,idx)

% Apply trim results to aircraft model input

da=ue(1,1);
de=ue(2,1);
dr=ue(3,1);
thr=ue(4,1);
U_0=xe(4,1);
W_0=xe(6,1);

%% Linearization

[A,B,Cx,Dx]=linmod('SAW_Aircraft_lin')

A_long=[A(4,4),A(4,6),A(4,11),A(4,8);A(6,4),A(6,6),A(6,11),A(6,8);A(11,4),A(11,6),A(11,11),A(11,8);A(8,4),A(8,6),A(8,11),A(8,8)];
B_long=[B(4,2),B(4,4);B(6,2),B(6,4);B(11,2),B(11,4);B(8,2),B(8,4)];
C_long=eye(4);
D_long=zeros(4,2);

A_lat=[A(5,5),A(5,10),A(5,12),A(5,7);A(10,5),A(10,10),A(10,12),A(10,7);A(12,5),A(12,10),A(12,12),A(12,7);A(7,5),A(7,10),A(7,12),A(7,7)];
B_lat=[B(5,1),B(5,3);B(10,1),B(10,3);B(12,1),B(12,3);B(7,1),B(7,3)];
C_lat=eye(4);
D_lat=zeros(4,2);

[num_long1,den_long1] = ss2tf(A_long,B_long,C_long,D_long,1);    %
elevator
[num_long2,den_long2] = ss2tf(A_long,B_long,C_long,D_long,2);    %
thrust

```

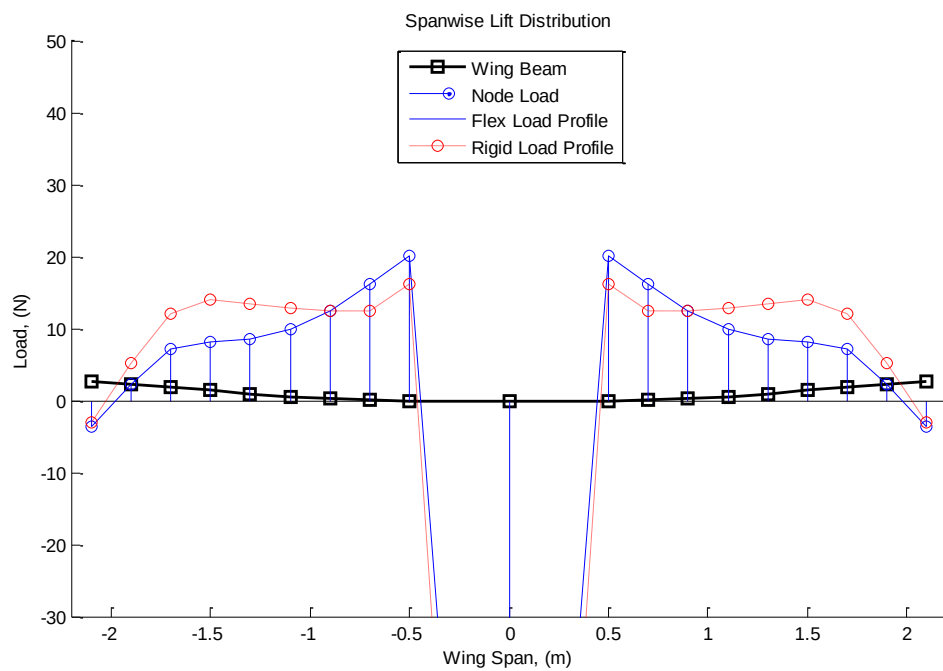
```

[num_lat1,den_lat1] = ss2tf(A_lat,B_lat,C_lat,D_lat,1);      %
aileron
[num_lat2,den_lat2] = ss2tf(A_lat,B_lat,C_lat,D_lat,2);      %
rudder
h1 = tf(1,den_long1);
h2 = tf(1,den_lat1);
damp(A_long)
damp(A_lat)

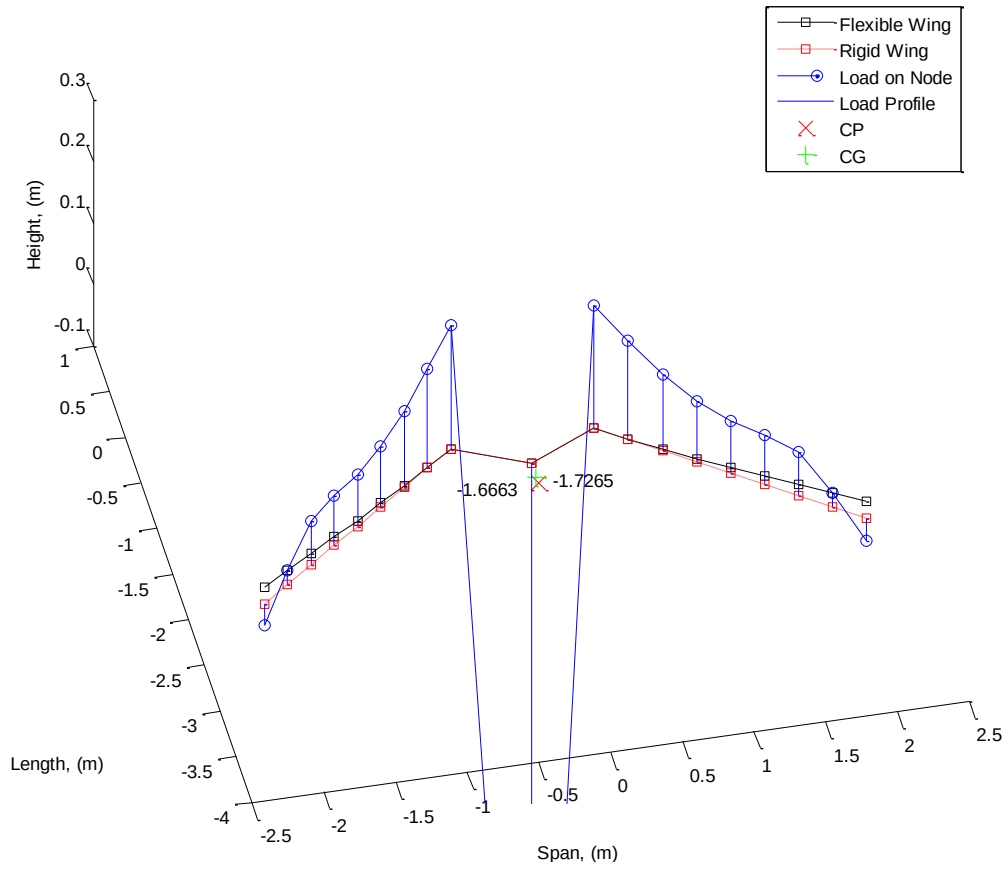
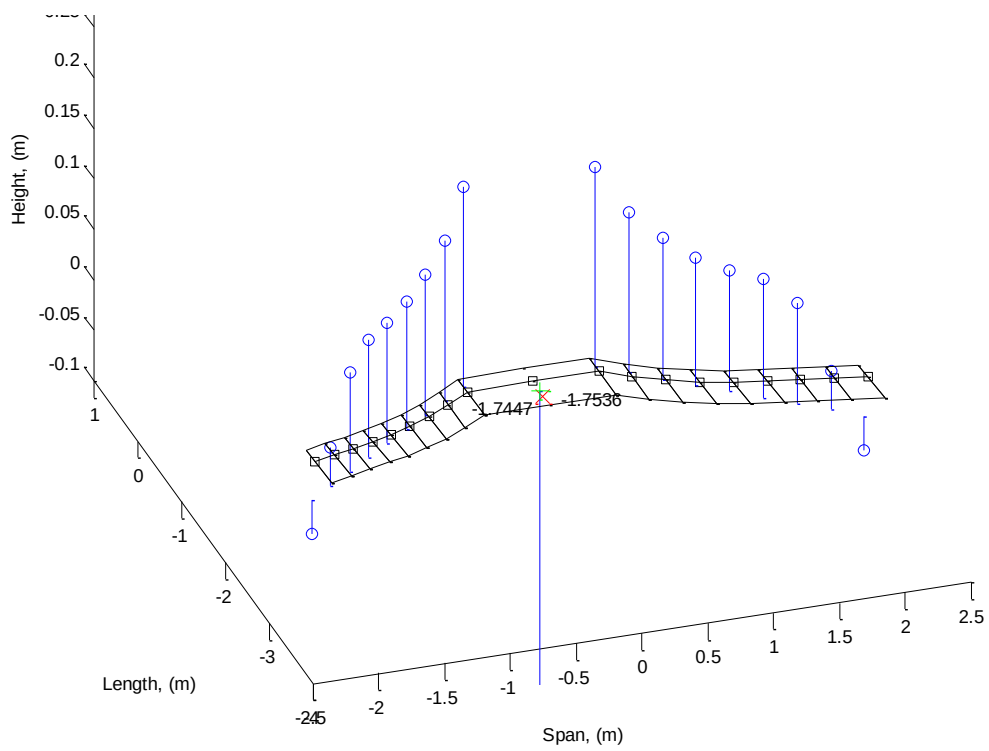
% Plot root locus
figure(1)
rlocus(h1)
grid on
figure(2)
rlocus(h2)
grid on

```

E.4 SAW 3D loading plot and movie



Loading, CP-CG Location on the Flexible Wing



```

%% Lift distribution + Wing bending movie
figure(1)
for f=1:size(LoadsL,1);
set(gcf,'Position',[10 300 800 500])
set(gca,'nextplot','replacechildren');
axis([-2.2 2.2,-30 50])
Y_span = [-2.1 -1.9 -1.7 -1.5 -1.3 -1.1 -0.9 -0.7 -0.5 0 0.5 0.7 0.9
1.1 1.3 1.5 1.7 1.9 2.1];
Z_span = [LoadsL(f,9) LoadsL(f,8) LoadsL(f,7) LoadsL(f,6) LoadsL(f,5)
LoadsL(f,4) LoadsL(f,3) LoadsL(f,2) LoadsL(f,1) LoadFuse(f) LoadsR(f,1)
LoadsR(f,2) LoadsR(f,3) LoadsR(f,4) LoadsR(f,5) LoadsR(f,6) LoadsR(f,7)
LoadsR(f,8) LoadsR(f,9)];
X = [-2.5062356 -2.3384156 -2.1705956 -2.0027756 -1.8349556 -
1.6671356 -1.4993156 -1.3314956 -1.1636756 -1.8605 -1.1636756 -
1.3314956 -1.4993156 -1.6671356 -1.8349556 -2.0027756 -2.1705956
-2.3384156 -2.5062356];
Z = [hL(f,8) hL(f,7) hL(f,6) hL(f,5) hL(f,4) hL(f,3) hL(f,2) hL(f,1) 0
0 0 hR(f,8) hR(f,7) hR(f,6) hR(f,5) hR(f,4) hR(f,3) hR(f,2) hR(f,1)];
plot(Y_span,Z*100,'-sk','linewidth',2)
hold on
stem(Y_span,Z_span,'b');
plot(Y_span,Z_span,'b');
xlabel('Wing Span, (m)')
ylabel('Load, (N)')
Frame1 = getframe;
end
Load_0 = [-3.0890416 5.1773562 12.09747 14.0082648 13.4338332
12.7915846 12.4048234 12.3859722 16.1215546 -191.2705444
16.1215546 12.3859722 12.4048234 12.7915846 13.4338332 14.0082648
12.09747 5.1773562 -3.0890416
];
plot(Y_span,Load_0,'o:r')
legend('Wing Beam','Node Load','Flex Load Profile','Rigid Load
Profile','Location','North')
title('Spanwise Lift Distribution')

%% Wing motion, CP CG movie
figure(2)
for v=1:size(LoadsL,1);
set(gcf,'Position',[210 50 800 700])
set(gca,'nextplot','replacechildren');
axis([-2.5 2.5,-4 1,-0.1 0.3])
campos([-1 -6 0.8])

Y = [-2.1 -1.9 -1.7 -1.5 -1.3 -1.1 -0.9 -0.7 -0.5 0 0.5 0.7 0.9 1.1
1.3 1.5 1.7 1.9 2.1];
X = [-2.58456 -2.41674 -2.24892 -2.0811 -1.91328 -1.74546
-1.57764 -1.40982 -1.242 -1.506 -1.242 -1.40982 -1.57764
-1.74546 -1.91328 -2.0811 -2.24892 -2.41674 -2.58456];
Z = [hL(v,8) hL(v,7) hL(v,6) hL(v,5) hL(v,4) hL(v,3) hL(v,2) hL(v,1) 0
0 0 hR(v,8) hR(v,7) hR(v,6) hR(v,5) hR(v,4) hR(v,3) hR(v,2) hR(v,1)];

X_span = [-2.5062356 -2.3384156 -2.1705956 -2.0027756 -1.8349556
-1.6671356 -1.4993156 -1.3314956 -1.1636756 -1.8605 -1.1636756 -
1.3314956 -1.4993156 -1.6671356 -1.8349556 -2.0027756 -2.1705956
-2.3384156 -2.5062356];
Z_span = [LoadsL(v,9) LoadsL(v,8) LoadsL(v,7) LoadsL(v,6) LoadsL(v,5)
LoadsL(v,4) LoadsL(v,3) LoadsL(v,2) LoadsL(v,1) LoadFuse(v) LoadsR(v,1)

```

```

LoadsR(v,2) LoadsR(v,3) LoadsR(v,4) LoadsR(v,5) LoadsR(v,6) LoadsR(v,7)
LoadsR(v,8) LoadsR(v,9)];

%Plot Wing beam
plot3(Y,X,Z,'-sk','linewidth',1,'MarkerSize',5)
hold on
plot3(Y,X,zeros(size(X)),':sr','linewidth',1,'MarkerSize',5)
%Plot Loading
stem3(Y,X,Z_span/100,'b');
plot3(Y,X,Z_span/100,'b');

%Plot AC, CG
plot(AC_Y(v),AC_X(v),'xr','MarkerSize',12,'linewidth',1)
plot(0,-1.6795,'+g','MarkerSize',10,'linewidth',1)
% grid on

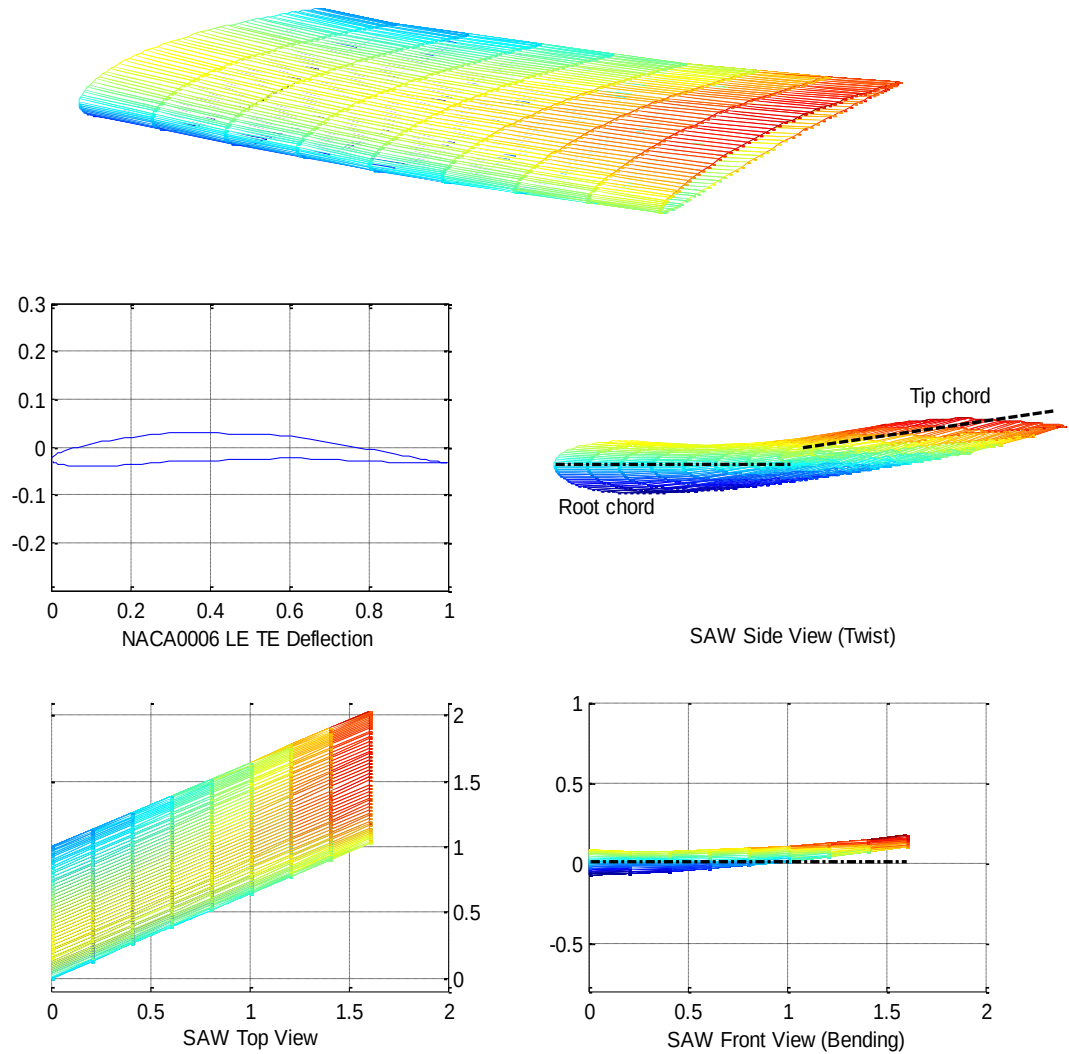
xlabel('Span, (m)')
ylabel('Length, (m)')
Zlabel('Height, (m)')
text(-0.55, -1.6795, '-1.6795')
text(AC_Y(v)+0.1, AC_X(v), num2str(AC_X(v)))
Frame2 = getframe;
end

title('Loading, CP-CG Location on the Flexible Wing')
legend('Flexible Wing ','Rigid Wing', 'Load on Node','Load
Profile','CP','CG','Location','NorthEast')

```

E.5 SAW 3D deformation plot and movie

SAW 3D Deformation Plot



```
% SAW_Wing_3D input
% Define Aircraft AoA and Wing sweep angle, Leading Edge(LE)and
Trailing
% Edge(TE) deflection angles
% Outside program input bend and twist is calculated by AeroBeamSaw

le = [0 0 0 5 10 10];    % degree
te = [0 -5 -5 0 5 5];    % degree
aoa = [1 1 1 1 1 1];    % degree
```

```

BEND=[0.0  0.0005  0.0018  0.0040  0.0068  0.0103  0.0142
0.0184  0.0227
0.0  0.0005  0.0018  0.0040  0.0068  0.0103  0.0142
0.0184  0.0227
0.0  0.0012  0.0047  0.0105  0.0186  0.0289  0.0412
0.0550  0.0699
0.0  0.0012  0.0047  0.0105  0.0186  0.0289  0.0412
0.0550  0.0699
0.0  0.0009  0.0036  0.0081  0.0141  0.0214  0.0299
0.0391  0.0487
0.0  0.0009  0.0036  0.0081  0.0141  0.0214  0.0299
0.0391  0.0487
];
TWIST=[1.0  0.9326  0.8743  0.8252  0.7872  0.7625  0.7534
0.7577  0.7894
1.0  0.9326  0.8743  0.8252  0.7872  0.7625  0.7534
0.7577  0.7894
1.0  0.9413  0.5406  0.0744  0.0833  0.1128  0.1299
0.1294  0.1223
1.0  0.9413  0.5406  0.0744  0.0833  0.1128  0.1299
0.1294  0.1223
1.0  0.8603  0.7320  0.8980  0.8738  0.6847  0.6632
0.6572  0.6802
1.0  0.8603  0.7320  0.8980  0.8738  0.6847  0.6632
0.6572  0.6802
];
% movie frame setting
figure('Position',[50 50 800 600])
for f=1:size(le,2);
LE = le(f); % degree
TE = te(f); % degree
AoA = aoa(f); % degree
SW = 40; % degree
bend= BEND(f,:);
twist= TWIST(f,:);

%% Separate LE and TE coordinates for NACA 0006

% X Upper coordinates
X06_LE_up=[0 0.00068523 0.00273905 0.00615583 0.0109262
0.01703709 0.02447174 0.03320979 0.04322727 0.05449674 0.0669873
0.08066472 0.0954915 0.11142702 0.12842759 0.14644661 0.1654347
0.1853398 0.20610737 0.22768048 0.25 0.27300475 0.29663168
];
X06_CS_up=[0.32081603 0.3454915 0.37059048 0.39604415 0.42178277
0.44773577 0.47383202 0.5 0.52616798 0.55226423 0.57821723
0.60395585
];
X06_TE_up=[0.62940952 0.6545085 0.67918397 0.70336832 0.72699525
0.75 0.77231952 0.79389263 0.8146602 0.8345653 0.85355339
0.87157241 0.88857298 0.9045085 0.91933528 0.9330127 0.94550326
0.95677273 0.96679021 0.97552826 0.98296291 0.9890738 0.99384417
0.99726095 0.99931477 1
];
% X Lower coordinates
X06_LE_low=[0.29663168 0.27300475 0.25 0.22768048 0.20610737
0.1853398 0.1654347 0.14644661 0.12842759 0.11142702 0.0954915
0.08066472 0.0669873 0.05449674 0.04322727 0.03320979 0.02447174
0.01703709 0.0109262 0.00615583 0.00273905 0.00068523 0
];

```

```

];
X06_CS_low=[0.60395585 0.57821723 0.55226423 0.52616798 0.5
0.47383202 0.44773577 0.42178277 0.39604415 0.37059048 0.3454915
0.32081603
];
X06_TE_low=[1 0.99931477 0.99726095 0.99384417 0.9890738
0.98296291 0.97552826 0.96679021 0.95677273 0.94550326 0.9330127
0.91933528 0.9045085 0.88857298 0.87157241 0.85355339 0.8345653
0.8146602 0.79389263 0.77231952 0.75 0.72699525 0.70336832
0.67918397 0.6545085 0.62940952
];
% Z Upper coordinates
Z06_LE_up=[0 -0.002305625 -0.004557231 -0.00675166 -
0.008884783 -0.010951586 -0.012946279 -0.014862448 -
0.016693209 -0.018431402 -0.02006977 -0.021601162 -
0.023018722 -0.024316079 -0.025487511 -0.026528104 -
0.027433876 -0.028201886 -0.028830302 -0.029318453 -
0.029666836 -0.029877108 -0.029952043
];
Z06_CS_up=[-0.029895464 -0.029712159 -0.029407777 -0.028988711
-0.028461977 -0.027835088 -0.027115925 -0.026312626 -
0.02543347 -0.024486786 -0.023480865 -0.022423899
];
Z06_TE_up=[-0.021323933 -0.020188823 -0.019026232 -0.017843613
-0.016648227 -0.015447156 -0.014247322 -0.013055515 -
0.011878418 -0.010722623 -0.009594647 -0.00850094 -
0.007447875 -0.006441735 -0.005488688 -0.004594741 -
0.003765698 -0.0030071 -0.002324161 -0.001721696 -
0.001204061 -0.000775073 -0.000437956 -0.000195279 -
4.8915E-05 8.32667E-18
];
% Z Lower coordinates
Z06_LE_low=[0.029952043 0.029877108 0.029666836 0.029318453
0.028830302 0.028201886 0.027433876 0.026528104 0.025487511
0.024316079 0.023018722 0.021601162 0.02006977 0.018431402
0.016693209 0.014862448 0.012946279 0.010951586 0.008884783 0.00675166
0.004557231 0.002305625 0
];
Z06_CS_low=[0.022423899 0.023480865 0.024486786 0.02543347
0.026312626 0.027115925 0.027835088 0.028461977 0.028988711
0.029407777 0.029712159 0.029895464
];
Z06_TE_low=[-8.32667E-18 4.8915E-05 0.000195279 0.000437956
0.000775073 0.001204061 0.001721696 0.002324161 0.0030071
0.003765698 0.004594741 0.005488688 0.006441735 0.007447875 0.00850094
0.009594647 0.010722623 0.011878418 0.013055515 0.014247322
0.015447156 0.016648227 0.017843613 0.019026232 0.020188823
0.021323933
];
%% Separate LE and TE coordinates for NACA 0015

% X Upper coordinates
X15_LE_up=[0 0.00068523 0.00273905 0.00615583 0.0109262
0.01703709 0.02447174 0.03320979 0.04322727 0.05449674 0.0669873
0.08066472 0.0954915 0.11142702 0.12842759 0.14644661 0.1654347
0.1853398 0.20610737 0.22768048 0.25 0.27300475 0.29663168
];

```

```

X15_CS_up=[0.32081603    0.3454915    0.37059048    0.39604415    0.42178277
0.44773577    0.47383202    0.5    0.52616798    0.55226423    0.57821723
0.60395585
];
X15_TE_up=[0.62940952    0.6545085    0.67918397    0.70336832    0.72699525
0.75    0.77231952    0.79389263    0.8146602    0.8345653    0.85355339
0.87157241    0.88857298    0.9045085    0.91933528    0.9330127    0.94550326
0.95677273    0.96679021    0.97552826    0.98296291    0.9890738    0.99384417
0.99726095    0.99931477    1
];
% X Lower coordinates
X15_LE_low=[0.29663168    0.27300475    0.25    0.22768048    0.20610737
0.1853398    0.1654347    0.14644661    0.12842759    0.11142702    0.0954915
0.08066472    0.0669873    0.05449674    0.04322727    0.03320979    0.02447174
0.01703709    0.0109262    0.00615583    0.00273905    0.00068523    0
];
X15_CS_low=[0.60395585    0.57821723    0.55226423    0.52616798    0.5
0.47383202    0.44773577    0.42178277    0.39604415    0.37059048    0.3454915
0.32081603
];
X15_TE_low=[1    0.99931477    0.99726095    0.99384417    0.9890738
0.98296291    0.97552826    0.96679021    0.95677273    0.94550326    0.9330127
0.91933528    0.9045085    0.88857298    0.87157241    0.85355339    0.8345653
0.8146602    0.79389263    0.77231952    0.75    0.72699525    0.70336832
0.67918397    0.6545085    0.62940952
];
% Z Upper coordinates
Z15_LE_up=[0    -0.005764062    -0.011393078    -0.016879151    -
0.022211958    -0.027378966    -0.032365697    -0.03715612    -
0.041733023    -0.046078506    -0.050174425    -0.054002904    -
0.057546804    -0.060790196    -0.063718777    -0.066320259    -
0.068584691    -0.070504714    -0.072075756    -0.073296132    -
0.07416709    -0.074692771    -0.074880107
];
Z15_CS_up=[-0.074738659    -0.074280397    -0.073519441    -0.072471777
-0.071154943    -0.069587719    -0.067789813    -0.065781565    -
0.063583676    -0.061216964    -0.058702162    -0.056059749
];
Z15_TE_up=[-0.053309832    -0.050472058    -0.047565579    -0.044609032
-0.041620569    -0.03861789    -0.035618305    -0.032638788    -
0.029696045    -0.026806558    -0.023986618    -0.02125235    -
0.018619687    -0.016104338    -0.013721719    -0.011486851    -
0.009414245    -0.00751775    -0.005810402    -0.004304241    -
0.003010152    -0.001937682    -0.001094889    -0.000488198    -
0.000122288    2.08167E-17
];
% Z Lower coordinates
Z15_LE_low=[0.074880107    0.074692771    0.07416709    0.073296132
0.072075756    0.070504714    0.068584691    0.066320259    0.063718777
0.060790196    0.057546804    0.054002904    0.050174425    0.046078506
0.041733023    0.03715612    0.032365697    0.027378966    0.022211958
0.016879151    0.011393078    0.005764062    0
];
Z15_CS_low=[0.056059749    0.058702162    0.061216964    0.063583676
0.065781565    0.067789813    0.069587719    0.071154943    0.072471777
0.073519441    0.074280397    0.074738659
];

```

```

Z15_TE_low=[-2.08167E-17    0.000122288 0.000488198 0.001094889
0.001937682 0.003010152 0.004304241 0.005810402 0.00751775
0.009414245 0.011486851 0.013721719 0.016104338 0.018619687 0.02125235
0.023986618 0.026806558 0.029696045 0.032638788 0.035618305 0.03861789
0.041620569 0.044609032 0.047565579 0.050472058 0.053309832
];

%% Sub-Plot Airfoils Clock-wise drawing (LE_up, TE_up, TE_low, LE_low)
X06_init=horzcat(X06_LE_up,X06_CS_up,X06_TE_up,X06_TE_low,X06_CS_low,X
06_LE_low);
Z06_init=horzcat(Z06_LE_up,Z06_CS_up,Z06_TE_up,Z06_TE_low,Z06_CS_low,Z
06_LE_low);

X15_init=horzcat(X15_LE_up,X15_CS_up,X15_TE_up,X15_TE_low,X15_CS_low,X
15_LE_low);
Z15_init=horzcat(Z15_LE_up,Z15_CS_up,Z15_TE_up,Z15_TE_low,Z15_CS_low,Z
15_LE_low);

% subplot(2,1,1); plot(X06_init,Z06_init)
% xlabel('NACA0006')
% axis([0 1, -0.1 0.1])
% subplot(2,1,2); plot(X15_init,Z15_init)
% xlabel('NACA0015')
% axis([0 1, -0.1 0.1])

%% Deflected LE and TE
% LE has 6 segments, TE has 8 segments
% LE=6*a, TE=8*b
% Control surface covers 4 sections, which means 5~9 airfoil are
deflected.
% How smooth is the wing depends on the number of deflected airfoils
chosen.

[X06_def9,Z06_def9] =
LETE_def(LE*5/5,TE*5/5,X06_LE_up,X06_CS_up,X06_TE_up,X06_TE_low,X06_CS
_low,X06_LE_low,Z06_LE_up,Z06_CS_up,Z06_TE_up,Z06_TE_low,Z06_CS_low,Z0
6_LE_low);
[X15_def9,Z15_def9] =
LETE_def(LE*5/5,TE*5/5,X15_LE_up,X15_CS_up,X15_TE_up,X15_TE_low,X15_CS
_low,X15_LE_low,Z15_LE_up,Z15_CS_up,Z15_TE_up,Z15_TE_low,Z15_CS_low,Z1
5_LE_low);

[X06_def8,Z06_def8] =
LETE_def(LE*4/5,TE*4/5,X06_LE_up,X06_CS_up,X06_TE_up,X06_TE_low,X06_CS
_low,X06_LE_low,Z06_LE_up,Z06_CS_up,Z06_TE_up,Z06_TE_low,Z06_CS_low,Z0
6_LE_low);
[X15_def8,Z15_def8] =
LETE_def(LE*4/5,TE*4/5,X15_LE_up,X15_CS_up,X15_TE_up,X15_TE_low,X15_CS
_low,X15_LE_low,Z15_LE_up,Z15_CS_up,Z15_TE_up,Z15_TE_low,Z15_CS_low,Z1
5_LE_low);

[X06_def7,Z06_def7] =
LETE_def(LE*3/5,TE*3/5,X06_LE_up,X06_CS_up,X06_TE_up,X06_TE_low,X06_CS
_low,X06_LE_low,Z06_LE_up,Z06_CS_up,Z06_TE_up,Z06_TE_low,Z06_CS_low,Z0
6_LE_low);
[X15_def7,Z15_def7] =
LETE_def(LE*3/5,TE*3/5,X15_LE_up,X15_CS_up,X15_TE_up,X15_TE_low,X15_CS

```

```

_low,X15_LE_low,Z15_LE_up,Z15_CS_up,Z15_TE_up,Z15_TE_low,Z15_CS_low,Z1
5_LE_low);

[X06_def6,Z06_def6] =
LETE_def(LE*2/5,TE*2/5,X06_LE_up,X06_CS_up,X06_TE_up,X06_TE_low,X06_CS
_low,X06_LE_low,Z06_LE_up,Z06_CS_up,Z06_TE_up,Z06_TE_low,Z06_CS_low,Z0
6_LE_low);
[X15_def6,Z15_def6] =
LETE_def(LE*2/5,TE*2/5,X15_LE_up,X15_CS_up,X15_TE_up,X15_TE_low,X15_CS
_low,X15_LE_low,Z15_LE_up,Z15_CS_up,Z15_TE_up,Z15_TE_low,Z15_CS_low,Z1
5_LE_low);

[X06_def5,Z06_def5] =
LETE_def(LE*1/5,TE*1/5,X06_LE_up,X06_CS_up,X06_TE_up,X06_TE_low,X06_CS
_low,X06_LE_low,Z06_LE_up,Z06_CS_up,Z06_TE_up,Z06_TE_low,Z06_CS_low,Z0
6_LE_low);
[X15_def5,Z15_def5] =
LETE_def(LE*1/5,TE*1/5,X15_LE_up,X15_CS_up,X15_TE_up,X15_TE_low,X15_CS
_low,X15_LE_low,Z15_LE_up,Z15_CS_up,Z15_TE_up,Z15_TE_low,Z15_CS_low,Z1
5_LE_low);

% figure(1)=subplot
% subplot(2,1,1); plot(X06_def,Z06_def)
% xlabel('NACA0006')
% axis([0 1, -0.2 0.2])
% subplot(2,1,2); plot(X15_def,Z15_def)
% xlabel('NACA0015')
% axis([0 1, -0.2 0.2])

%% Introduce Y make 3D wing

% Y is 1.6m wing semi-span.
Y = 0:0.2:1.6;

%% Interpolate for 8 airfoil sections plus deflection cases

% Because taper ratio is 1, X_interp would all be the same.
% Z_interp would have 9 sets of airfoil values, from NACA0006 to
NACA0015.
% Control surface covers 4 sections, which means 5 airfoil are
deflected.

% Initial airfoil
X_interp_i=zeros(9,size(X06_init,2));
Z_interp_i=zeros(9,size(Z06_init,2));
for i=1:size(X06_init,2);
    x=[0 8];
    y1=[X06_init(i),X15_init(i)];
    xi=0:8;
    X_interp_i(:,i)=interp1(x,y1,xi,'linear');
end
for j=1:size(Z06_init,2);
    z=[0 8];
    y2=[Z15_init(j),Z06_init(j)];
    zi=0:8;
    Z_interp_i(:,j)=interp1(z,y2,zi,'linear');
end

```

```

% Deflected airfoil 9
X_interp_d9=zeros(9,size(X06_def9,2));
Z_interp_d9=zeros(9,size(Z06_def9,2));
for i=1:size(X06_def9,2);
    x=[0 8];
    y1=[X06_def9(i),X15_def9(i)];
    xi=0:8;
    X_interp_d9(:,i)=interp1(x,y1,xi,'linear');
end
for j=1:size(Z06_def9,2);
    z=[0 8];
    y2=[Z15_def9(j),Z06_def9(j)];
    zi=0:8;
    Z_interp_d9(:,j)=interp1(z,y2,zi,'linear');
end

% Deflected airfoil 8
X_interp_d8=zeros(9,size(X06_def8,2));
Z_interp_d8=zeros(9,size(Z06_def8,2));
for i=1:size(X06_def8,2);
    x=[0 8];
    y1=[X06_def8(i),X15_def8(i)];
    xi=0:8;
    X_interp_d8(:,i)=interp1(x,y1,xi,'linear');
end
for j=1:size(Z06_def8,2);
    z=[0 8];
    y2=[Z15_def8(j),Z06_def8(j)];
    zi=0:8;
    Z_interp_d8(:,j)=interp1(z,y2,zi,'linear');
end

% Deflected airfoil 7
X_interp_d7=zeros(9,size(X06_def7,2));
Z_interp_d7=zeros(9,size(Z06_def7,2));
for i=1:size(X06_def7,2);
    x=[0 8];
    y1=[X06_def7(i),X15_def7(i)];
    xi=0:8;
    X_interp_d7(:,i)=interp1(x,y1,xi,'linear');
end
for j=1:size(Z06_def7,2);
    z=[0 8];
    y2=[Z15_def7(j),Z06_def7(j)];
    zi=0:8;
    Z_interp_d7(:,j)=interp1(z,y2,zi,'linear');
end

% Deflected airfoil 6
X_interp_d6=zeros(9,size(X06_def6,2));
Z_interp_d6=zeros(9,size(Z06_def6,2));
for i=1:size(X06_def6,2);
    x=[0 8];
    y1=[X06_def6(i),X15_def6(i)];
    xi=0:8;
    X_interp_d6(:,i)=interp1(x,y1,xi,'linear');
end
for j=1:size(Z06_def6,2);
    z=[0 8];

```

```

        y2=[Z15_def6(j),Z06_def6(j)];
        zi=0:8;
        Z_interp_d6(:,j)=interp1(z,y2,zi,'linear');
    end

    % Deflected airfoil 5
    X_interp_d5=zeros(9,size(X06_def5,2));
    Z_interp_d5=zeros(9,size(Z06_def5,2));
    for i=1:size(X06_def5,2);
        x=[0 8];
        y1=[X06_def5(i),X15_def5(i)];
        xi=0:8;
        X_interp_d5(:,i)=interp1(x,y1,xi,'linear');
    end
    for j=1:size(Z06_def5,2);
        z=[0 8];
        y2=[Z15_def5(j),Z06_def5(j)];
        zi=0:8;
        Z_interp_d5(:,j)=interp1(z,y2,zi,'linear');
    end
    %% LE TE spanwise proportion
    % Inboard wing is clean airfoil 1~4.
    % Outboard wing has control surfaces airfoil 5~9.

    X_interp=vertcat(X_interp_i(1:4,:), X_interp_d5(5,:), X_interp_d6(6,:),
        X_interp_d7(7,:), X_interp_d8(8,:), X_interp_d9(9,:));
    Z_interp=vertcat(Z_interp_i(1:4,:), Z_interp_d5(5,:), Z_interp_d6(6,:),
        Z_interp_d7(7,:), Z_interp_d8(8,:), Z_interp_d9(9,:));

    %% Define Wing Sweep angle (SW=40 degree)
    % SW=40;

    X_sweep=zeros(size(X_interp));
    for k=1:size(Y,2)
        X_sweep(k,:) =
        Y(k)*sin(SW*pi/180)*ones(size(X_interp(k,:)))+X_interp(k,:);
    end

    %% Wing Bending deformation
    % bend=[0 0.0012 0.0046 0.0101 0.0175 0.0266 0.0371
    0.0485 0.0605];

    Z_bend=zeros(size(Z_interp));
    for b=1: numel(bend)
        Z_bend(b,:)=Bending(Z_interp(b,:),bend(1,b));
    end

    %% Wing Twist deformation
    % twist=[AoA 1.3983 1.2110 1.0385 1.3402 1.2993 1.2742
    1.2628 1.2762];

    twist_diff = twist-AoA*ones(size(twist));

    X_twist=zeros(size(X_sweep));
    Z_twist=zeros(size(Z_bend));
    for t=1: numel(twist_diff)
        [X_twist(t,:),Z_twist(t,:)] =
        Twisting(X_sweep(t,:),Z_bend(t,:),twist_diff(1,t));
    end

```

```

end

% %% Mesh plot
%
% % figure(2) = mesh(X_twist,Y,Z_twist);
% figure = mesh(X_twist*0.6,Y,Z_twist*0.6);
% set(gca,'YDir','reverse')
% % axis([-0.1 2.1, 0 2, -0.8 1.0])
% axis([-0.1 1.7, 0 2, -0.8 1.0])

%% Make a movie

set(gca,'nextplot','replacechildren');

% Record the movie
    mesh(X_twist*0.6,Y,Z_twist*0.6)
    axis([-0.1 1.7, 0 2, -0.8 1.0]);
    campos([2 3 0.5]);
    camtarget([1 1 0]);
    set(gca,'YDir','reverse')
    % colorbar
    % axis off
    F = getframe;
end

% Play the movie 10 times
movie(F,10)

```

E.6 Unsteady aerodynamic force

```

unction [L,Mea] =
Unsteadyaero_abs(rho,Chord,EA,VT,alpha,alpha_dot,alpha_ddot,h_dot,h_dd
ot)

% Assume CG coincide with EA.

b = Chord/2;          %b is semi-chord
a = (EA-0.5)*2;       %a is the pitch axis location relative to
the mid-chord of the airfoil and is measured in terms of semi-chord
                        %Pitch axis is the EA which is 35% of chord, or 70%
of b.

span=0.2;             %segment span in meters

Ck=1;

% Lift L
L = span* (pi*rho*b*(b*h_ddot+b*alpha_dot*VT-
b^2*a*alpha_ddot)+2*pi*rho*b*(h_dot*VT+alpha*VT^2+b*alpha_dot*(0.5-
a)*VT)*Ck);

% Pitching moment M is relative to 0.5 chord
M50 = span* (-rho*b^2*(pi*(0.5-
a)*VT*b*alpha_dot+pi*b^2*(1/8+a^2)*alpha_ddot-

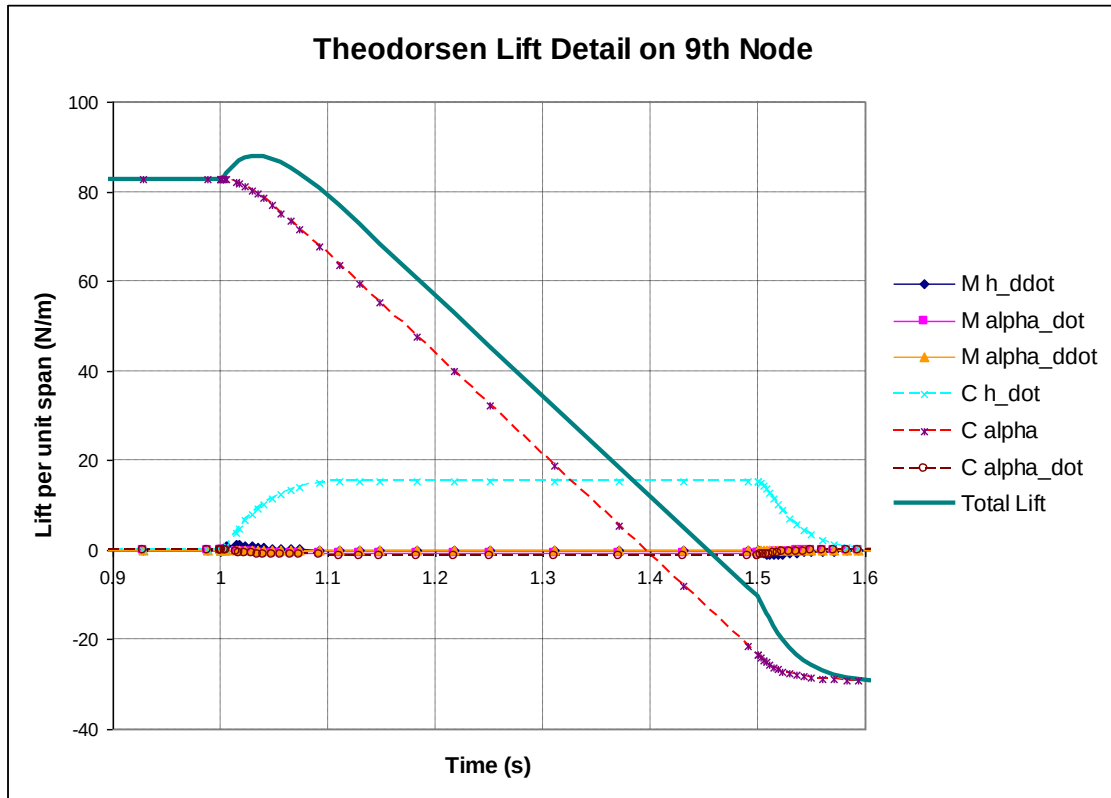
```

```

a*pi*b*h_ddot)+(2*rho*VT*b^2*pi*(a+0.5)*(VT*alpha+h_dot+b*(0.5-
a)*alpha_dot)*Ck));

% Offset to EA plus moment by circulatory term F(c)*(0.5-EA)*chord
Mea = M50 - Chord*(0.5-
EA)*span*(2*pi*rho*b*(h_dot*VT+alpha*VT^2+b*alpha_dot*(0.5-a)*VT)*Ck);

```



E.7 Air data

```

function [alpha,beta,VT,Mach,qbar] = air_data(rho,a,u,v,w)
% This block supports an embeddable subset of the MATLAB language.
% See the help menu for details.

```

```

VT          = sqrt(u^2 + v^2 + w^2);    % total velocity - true air
speed
Mach        = VT/a;                    % flight Mach number
alpha       = asin(w/sqrt(u^2 + w^2)); % body angle of attack
beta        = asin(v/VT);              % body sideslip angle
qbar        = 0.5 * rho * VT^2;        % dynamic pressure

```

E.8 Gravity

```

function XYZLMN_g = Gravity(h,m,cx,cy,cz,DCM_be)

```

```

% This function determines the forces and moments due to gravity

```

```

G    = 6.6726e-11;    % Universal gravitational constant
Me   = 5.9737e24;     % Earth mass
r    = 6.371e6 + h;   % Earth average radius
g    = G*Me/r^2;      % Acceleration due to gravity

offset = [0,-cz,cy;cz,0,-cx;-cy,cx,0]; % set up matrix of cg offsets

XYZ_g   = DCM_be * [0;0;m*g];    % determine body forces due to weight
LMN_g   = offset * XYZ_g;        % determine body moments due to cg
offset

XYZLMN_g = [XYZ_g;LMN_g];        % set up row vector

```

E.9 International Standard Atmosphere (ISA)

```

function [rho,a,p,t] = atmos(h)

% This function determines the atmospheric properties based on the
% altitude(AKA pressure height)
% Ref: ESDU 77022 table 11.1 and table 11.3
% Yang Da Qing 28th May 2008

if h < 11000
    t = 288.15 - 0.0065 * h;
    p = (8.9619638 - 0.00020216125 * h)^5.2558797;
elseif h < 20000
    t = 216.65;
    p = 128244.5 * exp(-0.00015768852 * h);
elseif h < 32000
    t = 196.65 + 0.0010 * h;
    p = (0.70551848 + 0.0000035876861 * h)^-34.163218;
elseif h < 47000
    t = 139.05 + 0.0028 * h;
    p = (0.34926867 + 0.0000070330980 * h)^-12.201149;
else %h < 50000
    t = 270.65;
    p = 41828.420 * exp(-0.00012622656 * h);
end

rho    = p/(287.05287 * t);    % determine air density
a      = sqrt(1.4 * 287.05287 * t); % determine local speed of sound

```

Appendix F Flying Quality Requirements

F.1 Aircraft Classification

Class 1: Small, light airplanes, such as light utility, primary trainer, and light observation craft.

Class 2: Medium-weight, low to medium manoeuvrability airplanes, such as heavy utility/search and rescue, light or medium transport/cargo/tanker, reconnaissance, tactical bomber, heavy attack, and trainer for class 2.

Class 3: Large, heavy, low to medium manoeuvrability airplanes, such as transport/cargo/tanker, heavy bomber, and trainer for class 3.

Class 4: High-maneuvrability airplanes, such as fighter/interceptor attack, tactical reconnaissance, observation, and trainer for class 4.

F.2 Flight phases categories (A, B non-terminal. C terminal)

Category A: Non-terminal flight phases that require rapid manoeuvring, precision tracking, or precise flight-path control. Included in the category are air-to-air combat, ground attack, weapon delivery/launch, aerial recovery, reconnaissance, in-flight refuelling (receiver), terrain following, antisubmarine search, and close formation flying.

Category B: Non-terminal flight phases that are normally accomplished using gradual manoeuvres and without precision tracking, although accurate flight-path control may be required. Included in the category are climb, cruise, loiter, in-flight refuelling (tanker), descent, emergency descent, emergency deceleration, and aerial delivery.

Category C: Terminal flight phases are normally accomplished using gradual manoeuvres and usually require accurate flight-path control. Included in this category are takeoff, catapult takeoff, approach, wave-off/go-around, and landing.

F.3 Levels of flying qualities

Level 1. Flying qualities clearly adequate for the mission flight phase.

Level 2. Flying qualities adequate to accomplish the mission flight phase when some increase in pilot workload or degradation in mission effectiveness exists or both.

Level 3. Degraded flying qualities such that the mission can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate or both.

F.4 Handling Quality: Cooper-Harper scale:

The Cooper-Harper rating scale is a set of criteria used by test pilots and flight test engineers to evaluate the handling qualities of aircraft during flight test. The scale ranges from 1 to 10, with 1 indicating the best handling characteristics and 10 the worst. The criteria are evaluative and thus the scale is considered subjective.

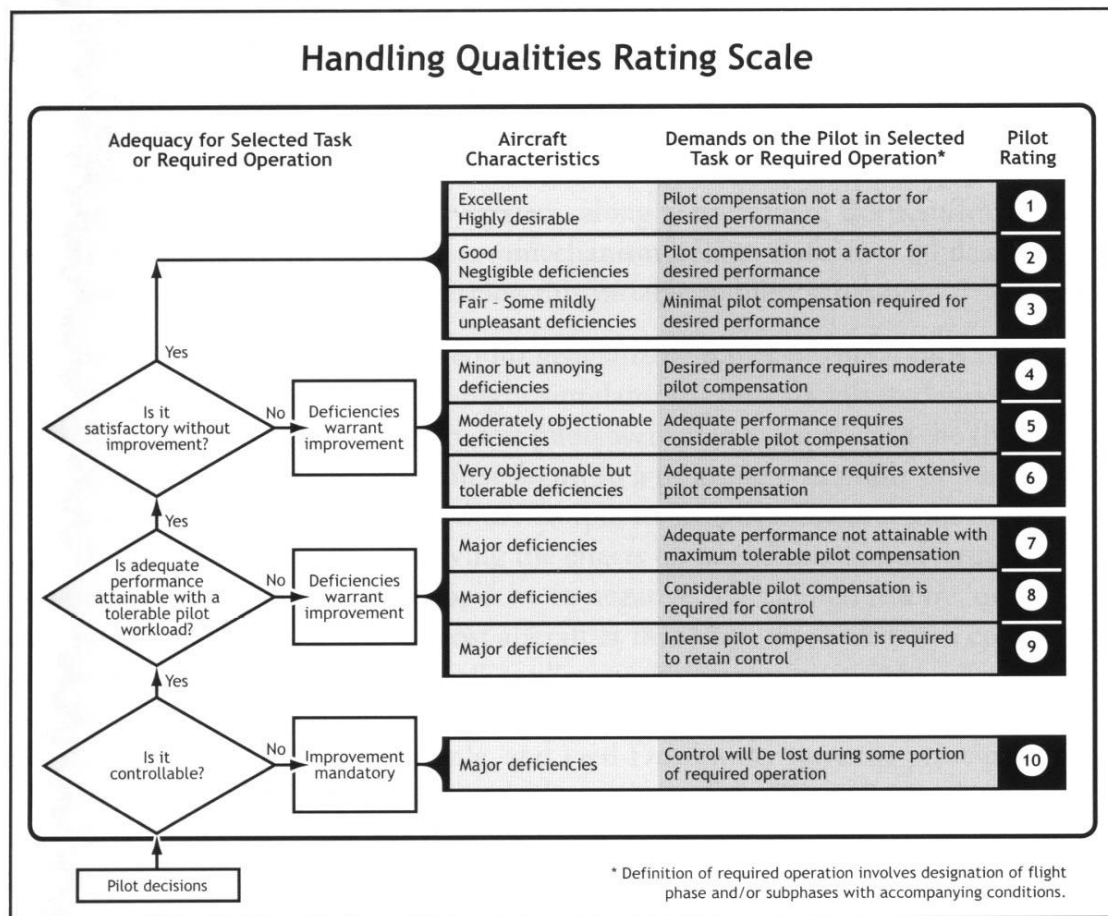


Figure 7-1 Cooper-Harper scale

F.5 Longitudinal Flying Qualities

Phugoid mode

Level 1	$\zeta > 0.04$
Level 2	$\zeta > 0$
Level 3	$T_2 > 55s$

SPPO damping ratios limits:

	Cat A	Cat A	Cat B	Cat B	Cat C	Cat C
	ζ min	ζ max	ζ min	ζ max	ζ min	ζ max
Level 1	0.35	1.30	0.30	2.00	0.50	1.30
Level 2	0.25	2.00	0.20	2.00	0.35	2.00
Level 3	0.10	-	0.15	-	0.25	-

SPPO limits on Control Anticipation Parameter $CAP = \frac{\omega_s^2}{n_\alpha} = \frac{mg\bar{c}}{I_y} H_m$

	Cat A	Cat A	Cat B	Cat B	Cat C	Cat C
	min	max	min	max	min	max
Level 1	0.28	3.6	0.085	3.6	0.16	3.6
Level 2	0.16	10.0	0.038	10	0.096	10.0
Level 3	0.16	-	0.038	-	0.096	-

$$n_\alpha = \frac{L}{W} = \frac{\frac{1}{2}\rho V^2 S C_L}{W \alpha}$$

F.6 Lateral-Directional Flying Qualities

Roll performance (Time to achieve stated change of bank angle)

Class	Category	Level 1	Level 2	Level 3
1	A	60° in 1.3s	60° in 1.7s	60° in 1.3s
	B	60° in 1.7s	60° in 2.5s	60° in 1.3s
	C	30° in 1.3s	30° in 1.8s	60° in 1.3s
2	A	45° in 1.4s	45° in 1.9s	45° in 2.8s
	B	45° in 1.9s	45° in 2.8s	45° in 3.8s
	C	30° in 2.5s	30° in 3.5s	30° in 5.0s
3	A	30° in 1.5s	30° in 2.0s	30° in 3.0s
	B	30° in 2.0s	30° in 3.0s	30° in 4.0s
	C	30° in 3.0s	30° in 4.5s	30° in 6.0s
4	A	90° in 1.3s	90° in 1.7s	90° in 6.0s
	B	60° in 1.7s	60° in 2.5s	60° in 3.4s
	C	30° in 1.0s	30° in 1.3s	30° in 2.0s

Roll mode (max allowable roll-subsidence mode time constant)

Class	Category	Level 1	Level 2	Level 3
1, 4	A	1.0	1.4	10.0
2, 3	A	1.4	3.0	10.0
All	B	1.4	3.0	10.0
1, 4	C	1.0	1.4	10.0
2, 3	C	1.4	3.0	10.0

Dutch roll mode

Level	Category	Class	Min ζ	Min $\zeta \omega_n^*$ (rad/s)	Min ω_n (rad/s)
1	A	1, 4	0.19	0.35	1.0
1	A	2, 3	0.19	0.35	0.4
1	B	All	0.08	0.15	1.0
1	C	1, 2-C**, 4	0.08	0.15	1.0
1	C	2-L**, 3	0.08	0.15	0.4
2	All	All	0.02	0.05	0.4
3	All	All	0.02	-	0.4

* The governing damping requirement is that yielding the larger value of ζ .

**The letters C and L denote the carrier-based and land-based aircraft.

Spiral mode (Time to double bank angle)

Class	Category	Level 1	Level 2	Level 3
1, 4	A and C	12s	8s	4s
1, 4	B	20s	8s	5s
2, 3	All	20s	12s	5s

Appendix G Demon Aircraft Background

In 2005 BAE System and EPSRC jointly funded a 5 years research project known as FLAVIIR (Flapless Air Vehicle Integrated Industrial Research). The project aimed: "To develop technologies for a maintenance free, low cost UAV without conventional control surfaces and without performance penalty over conventional craft."

The project, managed from Cranfield University and including 9 other university partners, covers all the key aspects of the next generation UAV through technical development in the following 7 themed areas: Aerodynamics, Control systems, Electromagnetics, Manufacturing, Materials/Structures, Numerical simulation and integration.

After 3 years intensive research phase, FLAVIIR project enters a 2-year demonstration phase. The demonstration vehicle “Demon”, which is 15% larger than its successful predecessor “Eclipse”, is a diamond shape flying wing Unmanned Air Vehicle (UAV) powered by a single micro jet engine and fitted with a single fin and fixed undercarriage.

Flight control scheme explores FTV (fluidic thrust vectoring) for pitch and CC (circulation control) actuators for roll and lift increase. FTV introduced a secondary jet flow and coanda effect on a two dimensional jet nozzle to divert thrust direction. CC replaces conventional flaps by blowing controlled air from the trailing edge of the wing. Both devices operated on compressed air generated by an onboard Auxiliary Power Unit (APU). Yawing stability and control were still achieved by conventional fin and rudder,

On 27th of September 2010, Walney Island airfield, Cumbria, UK. Demon successfully demonstrate the first unmanned “Flapless Flight”.



Figure 1-2 Flapless flight demonstration UAV “Demon”

Along with the development of the Demon, an alternative design concept to achieve flapless flight was originated in Cranfield University research group. Rather than using

pneumatic pipelines and actuators, the new design concept endeavour to employ controlled morphing airfoil to induce sufficient wing deformation to realize flight control.

This new concept exploits the full potential of composite material and aeroelastic tailoring technique, combines cutting edge morphing aerofoil design and relaxed wing rigidity together with advanced control technology, and formulates a new solution for flapless flight known as Seamless Aeroelastic Wing (SAW) technology.