

**Numerical optimisation of ply
orientation for aeroelastic tailoring**

**A Thesis Submitted for the
Degree of Master of
Philosophy**

By

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I certify that this thesis, and the research to which it refers, are the product of my own work and that any ideas or quotations from the work of other people, published or otherwise, are fully acknowledged in accordance with the standard referencing practices of the discipline.

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Abstract

This research explores the aeroelastic performance of a composite cantilever wing. The aim is clear: design a structure that resists flutter while staying light. We connect aerodynamic analysis with structural dynamics to reach this goal.

First, we run CFD simulations to measure lift and drag coefficients at different flow speeds. We then convert these coefficients into forces. Next, we apply these forces as uniform pressure loads on the wing surfaces in ANSYS Workbench. This step directly links fluid flow results to structural response.

We model the structure in both shell and solid forms. Each model captures stiffness and mass differently. The modal analysis reveals the natural frequencies and mode shapes. We track changes in torsional and bending modes as the ply count increases. More plies make the structure stiffer, but the gain slows after a certain thickness. The 8-ply configuration offers the best balance between stiffness, weight, and mode separation.

Harmonic response simulations test the wing under realistic oscillating loads. We focus on vibration amplitudes near resonance. We compare multiple stacking sequences, especially for the 12-ply case. Some layups reduce torsion–bending coupling and shift resonance away from critical flight frequencies. One configuration shows the lowest displacement peaks at both 60 m/s and 80 m/s, making it the top candidate.

These results give engineers a direct method to link aerodynamic loads with structural vibration data. The workflow is simple, fast, and repeatable. Future work will extend the method to high-speed compressible flows and validate results with wind tunnel experiments.

Nomenclature

A : Wing planform area (m^2)

A1, A2, A3, A4, A5 : Laminate stacking sequence configurations

ACP : ANSYS Composite Prepost

AERO : Aerodynamic

BEM : Blade Element Momentum

C_D : Drag coefficient (-)

C_L : Lift coefficient (-)

CFD : Computational Fluid Dynamics

E : Young's modulus (Pa)

f_n : Natural frequency of mode n (Hz)

FEM : Finite Element Method

Hz : Hertz (unit of frequency)

k : Turbulent kinetic energy (m^2/s^2)

k_g : Shear correction factor (-)

L : Lift force (N)

m : Mass (kg)

Mach : Mach number (-)

MPa : Megapascal (10^6 Pa)

N : Newton (unit of force)

p : Pressure (Pa)

Pa : Pascal (N/m^2)

Re : Reynolds number (-)

SIMPLE : Semi-Implicit Method for Pressure Linked Equations

t : Laminate thickness (m)

U : Flow velocity (m/s)

ρ : Air density (kg/m^3)

ω : Specific dissipation rate (1/s)

CHAPTER 1: INTRODUCTION

The search for safer, more efficient and more advanced technology in the field of aerospace engineering is a continuous process. As we travel through the skies of the twenty-first century, aircraft design and performance are increasingly intertwined with the goal of achieving optimum aerodynamic stability. One of the most important problems in this search is the reduction of oscillatory forces resulting from the interaction of aerodynamic, structural and inertial forces.

This MPhil research aims to improve aeroelastic performance by using numerical optimization techniques to solve challenge problems related to aircraft flutter phenomenon. Flutter, characterized by self-excited oscillations, brings serious problems to aircraft construction, requiring good research and new solutions to ensure the safe design and operation of aircraft.

The main purpose of this article is the use of numerical optimization to analyse, model and finally optimize the aeroelastic properties of the aircraft to maintain or reduce flutter. Through a combination of computational fluid dynamics (CFD), finite element analysis (FEA) and optimization techniques, this research aims to reveal the interactions of aerodynamics, structural dynamics and control systems that together control the initial flow and evolution of the dynamics.

The importance of this research is that it can change current practices in the aerospace industry. By integrating state-of-the-art mathematical techniques, not only understanding the theoretical understanding of flutter is sought, but also design changes to improve aircraft performance, safety and stability is proposed. The results of this research will benefit the continued improvement of aircraft design and aeroelasticity as the aviation industry seeks to improve performance and sustainability.

The next chapters of this research unfold as a journey that delves into the theoretical foundations, methodological complexities and empirical findings that lead to our understanding of mathematical optimization in the context of flutter. Through this research, uncovering new horizons in aeroelastic design and, in doing so, making a positive contribution to the continued development of aviation technology are hoped.

1.1 Background

Aviation industry advancements have been a focal point of technological innovation, contributing to the efficiency and safety of air travel. Despite notable progress, the occurrence of aerodynamic instabilities, particularly flutter, remains a critical concern. Flutter poses a threat to aircraft structural integrity and flight stability. Understanding and mitigating flutter is, therefore, imperative for the continued progress of aerospace engineering.

1.2 Problem Statement

Despite significant advances in aerospace engineering, precise prediction and control of aircraft crash patterns still face significant challenges. Current methods are limited in accurately modelling the interaction between aerodynamics and model performance, especially in different conditions. This research aims to fill this important gap by using advanced mathematical techniques to improve aircraft flutter prediction and mitigation, ultimately contributing to increased degrees of aerodynamic design and flight safety.

1.3 Scope of the Research

This work aims to provide a comprehensive study of the numerical optimization of aircraft flutter, with a focus on improving the understanding of aeroelastic phenomena. The resource involves the use of advanced computational techniques, including computational fluid dynamics (CFD), finite element analysis (FEA) and optimization algorithms. The research aims to uncover new insights that can be translated into designs by revealing the interaction between aerodynamics, performance models and control.

The expansion continues with the development of technology and the use of numerical optimization methods to analyze and reduce interactions, thus contributing to the expansion of aeroelasticity. This research is not only of theoretical importance, but also a practical research that aims to propose new ideas to improve the safety, occupational safety and security of aircraft.

1.4 Motivation of Work

Today's increasing demand for electricity affects many sectors, including aviation. Airlines are looking for solutions that include reducing fuel consumption, reducing environmental impact, improving safety and more. One of the best ways to make these improvements is to increase the wing ratio of the aircraft. But a significant difficulty arises, because the higher ratio easily leads to the wing, causing a vibration

effect. This requires in-depth investigation of flutter, a generally imperceptible force that is difficult to relate to the vibration effects of aircraft wings. Understanding and reducing flutter becomes important in the quest to find fuel-efficient designs for next-generation aircraft.

Aircraft design and performance optimization are important aspects of the aerospace industry and constantly push the boundaries of technological advancement. In this case, the aeroelastic behaviour of the wings combined in dynamic material plays an important role in ensuring the integrity and safety of the flight as a whole. The motivation behind this work stems from the need to understand the complexity of composite wings based on aerodynamic effects. Through in-depth research on vibration analysis and fluctuation rate analysis, we aim to provide recommendations to guide the design and construction of aircraft components. This research is not only an interesting study, but also has implications for improving the reliability and performance of air. Through this research, we want to combine different knowledge in aeroelasticity and provide solutions that will shape the future of aircraft design.

CHAPTER 2: LITERATURE REVIEW

Researchers study composite wing structures for better performance. They focus on light weight and high strength. Engineers design laminates with specific fiber orientations. These orientations control stiffness and vibration. CFD predicts aerodynamic loads with precision. Finite element models show structural deformation. Experiments confirm the numerical results. Many studies explore flutter resistance at different speeds. They also examine the effect of stacking sequences. This research builds on those studies. It targets both low-speed and high-speed conditions. The goal is clear: improve aeroelastic stability and efficiency.

2.1. Pre-World War II Period

People became acquainted with these vehicles with the invention of the airplane by the White brothers in 1903. From this date on, airplanes began to develop. Especially when a war was likely to break out in Europe, countries wanted to make a difference in this potential war by using this invention to their advantage. As airplanes began to develop, various problems began to arise, as is the case with the development of every new invention. One of the most important of these problems was flutter, and this is defined as “an unstable, self-excited structural oscillation. That's why there were many accidents or planes became unusable even if they didn't crash. Thereupon, researchers and various teams began to work on this subject and achieved various improvements from different cases.

Flutter is a self-excited aeroelastic instability phenomenon that occurs due to the interaction between aerodynamic forces, structural elasticity, and inertial effects within an aircraft or similar structure. This dynamic instability leads to rapid oscillatory motion, which can grow uncontrollably if not properly damped. Flutter typically involves a coupling between different structural modes, such as bending and torsion, leading to sustained and sometimes catastrophic oscillations. It is a significant concern in aerospace engineering due to its potential to cause severe structural damage or even complete structural failure within seconds. The study and prevention of flutter have therefore become crucial elements in the design and testing of flight vehicles, especially with the increasing demands for higher speeds, lighter structures, and more flexible materials. Throughout the history of aviation, numerous flutter incidents have led to critical safety measures and advancements in aeroelastic design, including innovations in control systems, structural reinforcements, and material applications. Understanding and mitigating flutter is not only essential for the safe operation of aircraft but also plays a pivotal role in the development of modern high-speed aerospace vehicles.

2.1.1. S.P. Langley and His Aerodrome

The December 1903 failure of Professor Samuel P. Langley's powered flying machine during its second launch attempt is often linked to aeroelastic effects. Structural failures, including the collapse of the rear wing and tail, suggested

inadequate torsional rigidity. Hill's hypothesis indicated wing torsional divergence, a non-oscillatory aeroelastic instability akin to zero-frequency flutter, as a probable cause due to insufficient wing-tip stiffness. Griffith Brewer further highlighted the susceptibility of monoplane wings to twisting, exacerbated by the movement of the centre of pressure during speed variations.

Subsequent modifications of Langley's machine, including significant reinforcement of the wing structure and trussing, markedly improved stiffness, reducing the likelihood of aeroelastic failure. This highlights the importance of structural rigidity in mitigating flutter risks. However, modern evaluations of the restored Langley machine revealed that the elastic axis location near the quarter-chord line likely precluded static aeroelastic divergence (which is the uncontrolled bending and twisting of a wing depending on the aircraft speed due to the disruption of the balance between aerodynamic and elastic forces. This situation usually occurs as a result of the interaction between the wing elastic axis and the aerodynamic centre (centre of pressure) and is a critical structure-instability problem at high speeds.). Despite this, the cambered wing's downward twisting moment and overall lack of torsional rigidity likely contributed to the failure during launch (Cambered wings, due to their aerodynamic structure, create different pressure distributions on the air flow. This creates a downward twisting moment that tends to bend the wing downward, producing a negative moment on the wing in particular. This effect becomes more pronounced when the wing structure has low torsional rigidity.).

This historical case underscores the critical need for torsional stiffness and proper load distribution to mitigate aeroelastic instabilities in early aircraft designs, influencing the shift towards biplanes for improved structural stability [1].

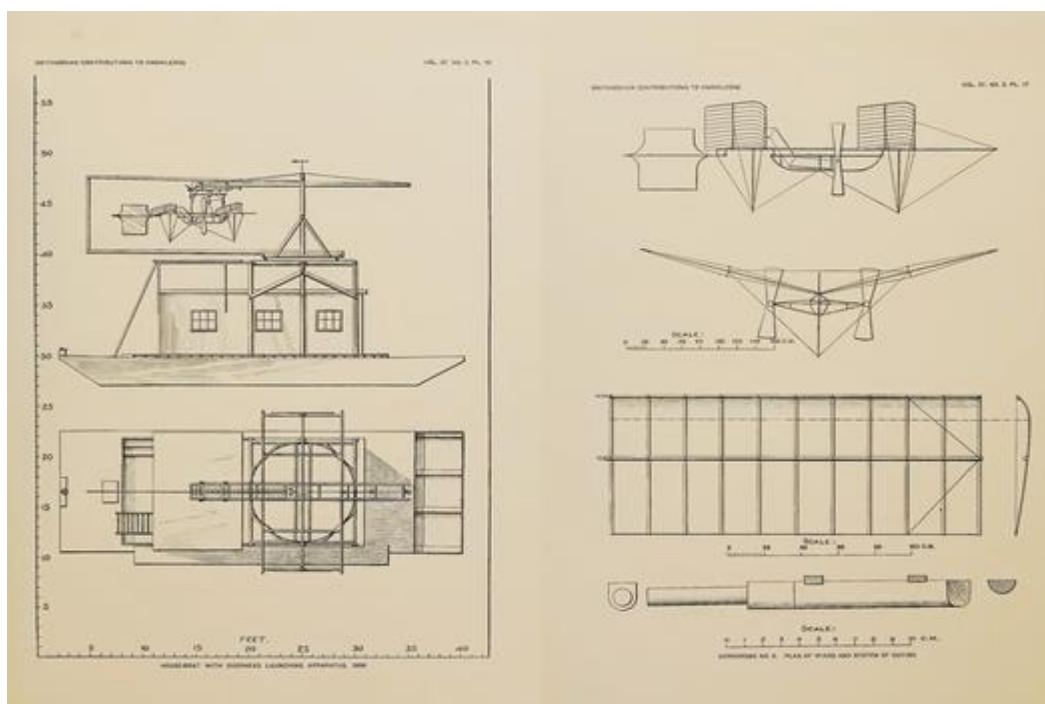


Figure Langley Technical Schematic [2]

"Technical schematics of Langley's Aerodrome, illustrating its wing structure, launching system, and truss-based support mechanisms. The aircraft experienced structural failure due to insufficient torsional rigidity, leading to aeroelastic instability during its attempted flight in December 1903. The right-side diagrams highlight the wing support system, a critical element that later proved inadequate for mitigating torsional divergence."

2.1.2. The Handley Page O/400 Bomber Flutter Incident of 1916

The first recorded crash occurred in 1916 on a Handley Page O/400 (twin-engine biplane bomber). This significant event highlighted dangerous aerodynamic instabilities that were not fully understood at the time. The aircraft experienced large tail oscillations due to the coupling mechanism of the fuselage torsion mode and the antisymmetric elevator rotation mode. This interaction caused a significant impact on the rear fuselage. Self-propelled elevators moved asymmetrically, increasing instability and making the situation dangerous. Later Lanchester's study provided important insights into the flutter phenomenon. One of his pivotal recommendations was the rigid connection of the left and right elevators using a stiff shaft, which subsequently became a standard design requirement in aircraft engineering. This solution aimed to mitigate the asymmetric elevator movements that contributed to the flutter. It was determined that the conflict occurred because the individual elevators were not moving in unison, causing further vibration. This modification allows the elevator to move synchronously, reducing the potential for asymmetric movement and disrupting the interference pattern [3].

This phenomenon and its solutions are important for improving the understanding of flutter and influencing future aircraft designs, and highlight the importance of synchronized control surfaces to ensure stability.

different from the prototype in that it had a reinforced rear spar. The purpose of this change was to proportionally increase the strength of the spar, but it inadvertently made the wing vulnerable to problems related to aeroelastic divergence. The shift of the elastic axis as a result of the rear spar's strengthening led to this fragility [1].

This torsional divergence occurs as a result of the rigid structure of the wing being exposed to aerodynamic moments, which causes loss of structural integrity. It refers to the importance of the aerodynamic centre of pressure and the elastic axis (also known as the flexion axis). The flexural axis is where the loading of the wing can be used without competition. Understanding this concept is important for the design of beam-like wings with larger aspect ratio because it helps reduce aeroelastic instabilities [4].

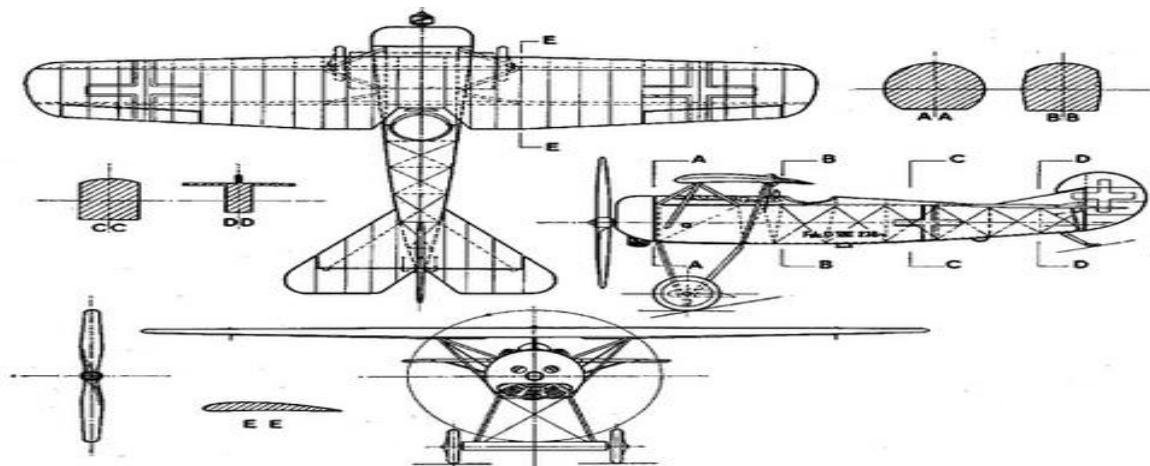


Figure Fokker D-VIII Technical Schematic [5]

"Technical schematics of the Fokker D-VIII, a cantilever parasol monoplane introduced near the end of World War I. Despite its innovative design, early flight tests revealed severe static divergence issues. Modifications to the rear spar inadvertently shifted the elastic axis, making the wing susceptible to torsional divergence under high-speed flight conditions. This case highlighted the importance of elastic axis placement in preventing structural instabilities."

2.1.4. The De Havilland DH 80 Flutter Incident

The De Havilland DH 80 Kitty Moth's flutter problem was actually caused by a combination of unbalanced ailerons and inadequate wing strut stiffness. These poor designs led to wing flutter. Unbalanced winglets made them more susceptible to oscillatory forces, allowing aerodynamically induced vibrations to resonate with the wing's natural frequency and cause flutter. In addition, the lack of wing struts meant they couldn't withstand aerodynamic forces, causing the wing to bend and twist excessively in turbulent conditions. The lack of stiffness made it easier for the wing to enter the flutter state. The whole wing structure, together with weak ailerons and poor struts, created an environment in which flutter could be easily initiated and sustained, with dangerous and sometimes fatal effects. Changes were made,

including the addition of a winglet mass meter, which made the winglets less susceptible to aerodynamic forces and more stable. The addition of temporary supports increases the stiffness and strength of the wing structure, reducing the potential for excessive bending and twisting. Additionally, the overall supportive wing structure helped better to resist the aerodynamic loads encountered during flight. These corrective measures effectively reduced the flutter problem and made it safer for the aircraft in different conditions [5].

2.1.5. MO-1 Flutter Incident

The flutter case on the tail wing of MO-1 aircraft happened due to different deflection values of the two supporting spars. This caused a strong combination of bending and torsion coupling, which led to strong oscillations. It was initially thought that rolling vibrations on the tail surfaces were caused by the impact of disturbed airflow coming from the wing roots. However, wind tunnel tests acknowledged that the airflow over the tail plane was not disturbed enough to create significant amount of oscillations. Flexibility of the tail was surface was then thought to be the main reason of the possibility of vibration.

After analysis, the main problem was unequal stiffness of the supporting spars differential deflections, which then resulted in torsional deformation. This coupling effect between bending and torsion was more pronounced at small, high-speed angles of attack and caused the surface to be subjected to continuous oscillations. To reduce the flutter, it was recommended that the system's centre of mass to be placed in front of the main supporting spar, with the centre of pressure behind the member. It was also recommended to make it rigid enough to avoid the effect of high pressure and torsional deformation at high speed loads. Calculations indicated that the rear spar must have been approximately 73 to 75 percent of the stiffness of the front spar to reduce differential deflection. This would help offset the deflections and reduce the tendency for oscillations. Therefore, the main cause of the flutter in the MO-1 tail plane could be traced to the differential deflections of its spars, and the suggested solutions focused on balanced and improved adjustment of the torsional stiffness by increasing the torsional rigidity of the structure [6].

2.1.6. Tacoma Bridge Flutter Incident

The bridge collapsed just four months after its opening. The bridge's design led to its vulnerability to wind-induced vibrations. A single study was conducted on the properties of the former Tacoma Narrows Bridge, which was destroyed by wind in 1940, and flutter analysis was performed to understand the bridge's eigenvalues and natural frequencies and torsional flutter response characteristics. From this numerical simulation, it was found that although the initial asymmetric fluctuation occurred in the bridge, the first symmetric torsional flutter would have occurred if the middle beam pair hadn't been not broken. A pair of centrelines were broken and the

wind speed is less than 19 m/s, half the span on each side, the first asymmetric torsional flutter, which is the same type of fluctuation. The flutter mechanism involved a coupling between the bridge's torsional oscillations and the wind's aerodynamic forces. As winds reached around 19 m/s, the bridge began to twist and oscillate violently. These oscillations increased in amplitude due to a feedback loop between the bridge's natural frequency and the aerodynamic forces, leading to a dramatic increase in motion. The bridge eventually tore itself apart due to these oscillations, resulting in its collapse [7].

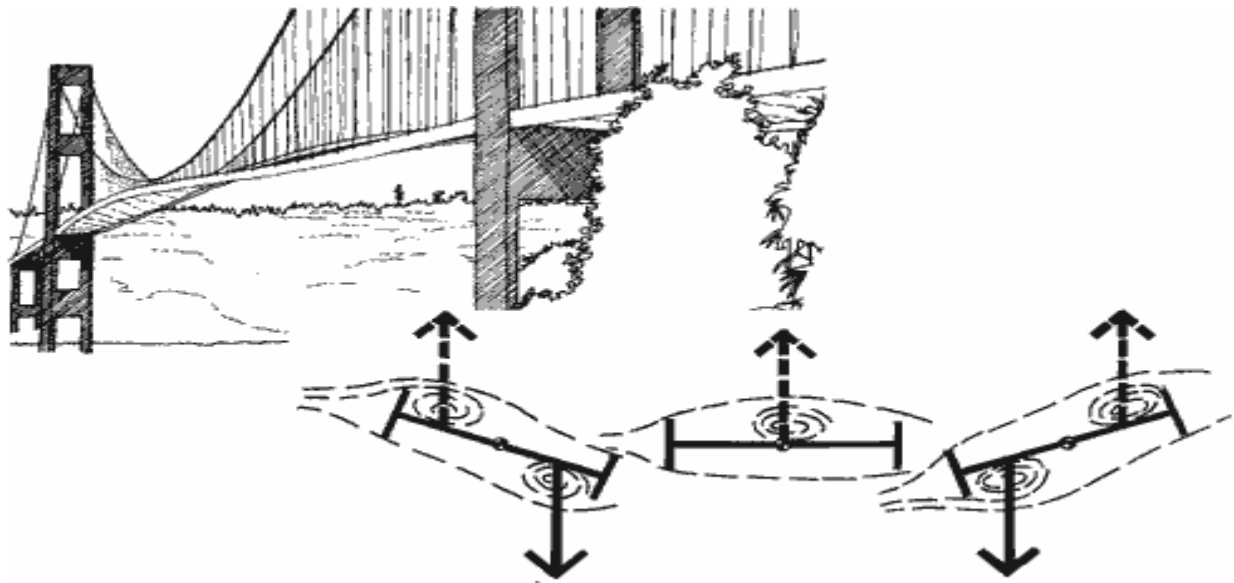


Figure Tacoma Bridge destruction [8]

"Illustration of the Tacoma Narrows Bridge collapse mechanism, showing how wind-induced vortex shedding led to self-excited torsional oscillations. The bottom diagram represents the coupling effect between aerodynamic forces and the bridge's structural response, amplifying oscillatory motion until structural failure occurred."

2.1.7. Theoretical Studies

2.1.7.1. British Contributions to Flutter Research and Mitigation Strategies

The significance of flutter as a critical aeroelastic instability was formally recognized in the 1924-1925 yearbook of the British Aeronautical Research Committee (ARC), where Chairman R.T. Glazebrook emphasized the necessity of theoretical and experimental investigations to fully understand and mitigate this phenomenon. Early efforts to collect data on wing rigidity and analyse a series of flutter-induced accidents led to the establishment of a Flutter Subcommittee, which assigned

dedicated research responsibilities to the National Physical Laboratory (NPL) and the Royal Aeronautical Establishment (RAE).

One of the earliest documented cases involved wing-aileron flutter in biplane configurations, notably affecting the Gloster Grebe and Gloster Gamecock aircraft. Observed at an estimated 15 Hz flutter frequency, pilots described the phenomenon as a visual "blur," while ground observers likened it to a "hovering hawk." To counteract these effects, engineers implemented mass balancing techniques by relocating the aileron interplane connecting strut closer to the centre of mass and reducing unbalanced surface area near the aileron tips. The investigation further revealed that similar flutter incidents had been reported in the Netherlands and the United States, underscoring the global nature of the issue [9].

A significant theoretical advancement emerged in 1929 with the publication of a comprehensive monograph by Frazer and Duncan of the NPL, colloquially referred to as "The Flutter Bible." This work established foundational methodologies for flutter analysis by employing simplified wind tunnel models to identify key parameters affecting aeroelastic stability. One of the notable contributions was the introduction of "semirigid modes," a concept that simplified flutter dynamics by approximating oscillatory behaviour as a set of discrete degrees of freedom. This method enabled the formulation of ordinary differential equations (ODEs) to model aeroelastic instabilities, avoiding the complexity of partial differential equations (PDEs). The approach aligned with Rayleigh's principles of assumed modes in vibration analysis, particularly in conservative mechanical systems with symmetrical cross-coupling coefficients. However, in contrast to classical vibration theory, flutter involves nonconservative, non-self-adjoint systems, making theoretical treatment significantly more complex. Duncan later acknowledged that early theoretical models lacked accurate aerodynamic representations, relying on empirical aerodynamic constants that failed to account for the interaction effects of wake vortices, a limitation explicitly mentioned in the AGARD Manual on Aeroelasticity [10].

Concurrently, Perring (1928) pioneered the use of scaled models for experimental determination of critical flutter speeds, a practice that would later become standard in aeroelastic testing. The research focused on scaled-down biplane configurations, where flutter incidents had been well-documented. The development of scaling laws by McKinnon Wood and Horace Lamb (1927) established the geometric, mass, and elastic similarity conditions necessary for dynamic equivalence between a scaled model and a full-scale prototype. Notably, their analysis excluded compressibility (Mach number), viscosity (Reynolds number), and gravity (Froude number) effects, deeming them insignificant for the problem at hand. A one-third scale semi-span wind tunnel model was constructed, maintaining the same mass density while reducing structural stiffness to one-ninth of the full-scale aircraft. This model was subjected to experimental testing in the RAE 7-ft wind tunnel, where its flutter speeds and oscillation frequencies closely correlated with real-world observations. This study demonstrated, for the first time, the efficacy of aeroelastically scaled wind-tunnel models in predicting critical flutter speeds, setting the stage for future experimental validation methods in aeroelasticity [11].

The early British research on flutter laid the groundwork for subsequent theoretical and experimental methodologies in aeroelasticity. The semi-rigid mode approach, the use of scaled wind-tunnel models, and the implementation of mass balancing techniques were crucial in refining flutter mitigation strategies. These findings not only informed the design modifications of contemporary aircraft but also emphasized the necessity of accurate aerodynamic modelling and dynamic scaling laws in ensuring structural integrity and flight safety. The lessons drawn from these pioneering efforts continue to influence modern flutter analysis and control techniques in both fixed-wing and rotary-wing aircraft.

2.1.7.2. Theodorsen's Two-Dimensional Flutter Theory

In 1934, Theodore Theodorsen introduced a pivotal approach to flutter analysis, significantly shaping American methodologies in aeroelasticity. His work, presented in NACA Report No. 496, provided a fundamental framework for evaluating two-dimensional oscillatory behaviour of flat plate airfoils subjected to translational, torsional, and aileron-type motions. Theodorsen's key contribution was the separation of non-circulatory and circulatory components in the velocity potential, with the wake effects influencing the latter. By enforcing the trailing-edge flow condition, he derived an analytical solution involving Bessel (Hankel) functions, leading to the formulation of Theodorsen's circulation function, $C(k)$, where $k = wb/V$ is the reduced frequency, analogous to the Strouhal number. This function characterizes the phase lag between airfoil motion and the aerodynamic forces, which is critical in determining whether energy extraction from the airflow can induce flutter.

Theodorsen's approach diverged from previous methods by not relying on Routh's discriminants. Instead, he treated the flutter determinant as a complex function, requiring both its real and imaginary components to vanish simultaneously to determine the onset of flutter. This provided a parametric framework applicable to both binary and ternary flutter modes. His contributions enabled the practical implementation of strip theory, where sectional representations of wings could be analysed for flutter stability, particularly in moderately high aspect ratio wings.

Further refinement of Theodorsen's theory was made through his collaboration with Garrick, where they systematically investigated the impact of mass distribution, elastic axis position, moment of inertia, mass ratio, hinge location, and bending-to-torsion frequency ratio on flutter characteristics. They also introduced hysteresis damping, represented by a complex elastic restoring force factor, to account for material damping effects. Experimental validation was conducted using cantilever models with high aspect ratios, confirming the predictive accuracy of the theoretical framework.

Parallel research efforts included Silverstein and Joyner's direct experimental confirmation of oscillating lift in pitching motion, as well as independent solutions to the flutter problem by Küssner (1936), who expanded the analysis to include gust responses via the gust function $k(s)$. Theodorsen's work remains foundational in flutter analysis, bridging theoretical insights with experimental validation, and

influencing subsequent developments in aeroelastic stability, wing design, and control surface dynamics.

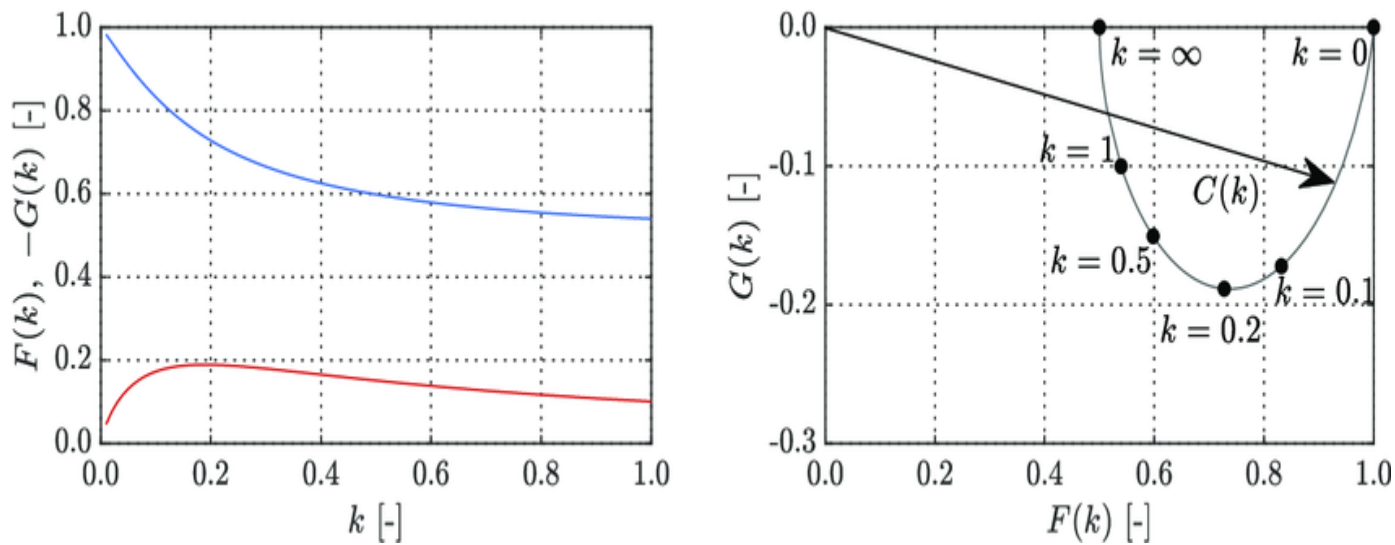


Figure Theodorsen Function [12]

"Theodorsen's Circulation Function $C(k)$ illustrates the aerodynamic phase lag between oscillating motion and resulting aerodynamic forces. The left graph represents the real ($F(k)$) and imaginary ($G(k)$) components as a function of reduced frequency k , while the right graph shows the trajectory of $C(k)$ in the complex plane, demonstrating the frequency-dependent aerodynamic damping effect on flutter stability."

1.1.7.3. Other Developments [1] [3] [11] [13]

1. Early Theoretical Foundations (1920s)

- a) Bryan's Stability Theory: Extended to nine degrees of freedom (Baird & Nayler), incorporating control surface motions in flutter analysis.
- b) Aerodynamic Derivatives: Introduced to quantify unsteady aerodynamic forces in flutter conditions.
- c) Strip Theory Validation: Initially developed for propellers, later adapted for wing flutter analysis.
- d) Rayleigh's Vibration Theory & Elasticity Theory: Provided mathematical tools for structural response predictions.

2. Initial Flutter Cases and Observations

a) Von Baumhauer & Koning (1923):

- First recorded wing-aileron flutter study.
- Proposed mass balancing to mitigate flutter effects.

b) Gloster Grebe & Gloster Gamecock Cases (1925, UK):

- Prompted dedicated flutter research at RAE & NPL.
- Led to systematic investigations and early theoretical models.

c) Junkers F.13 Tail Buffeting Case (1930, UK):

- Investigations led by Frazer & Duncan identified wake-induced tail oscillations as a probable cause.

3. Advancements in Aeroelastic Modelling

a) Gates (RAE, 1925):

- Attempted solving flexure-torsion oscillations using integro-differential equations.
- Highly complex, but helped classify instability types and parametric importance.

b) McKinnon Wood (1925):

- Proposed low-speed wind tunnel testing using scaled models with reduced elasticity.
- Frazer & Duncan (1928, NPL, "The Flutter Bible"):
- Introduced semi-rigid mode approximation to simplify aeroelastic calculations.
- Developed test determinants as alternatives to Routh criteria.
- Studied energy balance & dissipation functions for flutter stability analysis.
- Introduced graphical methods to evaluate parametric effects on flutter onset.
- Conducted wind tunnel tests to validate aerodynamic derivatives.

4. Structural & Control Enhancements

a) Irreversible Aileron Controls (Southwell's Idea, 1920s):

- Designed to prevent unintended aileron deflections due to aeroelastic effects.
- Evolved into powered control systems in later aircraft designs.

b) Aileron Reversal Investigations (1930s, RAE, Roxbee Cox & Pugsley):

- Found that aileron downwash produced a rearward force, causing nose-down twisting.
- Identified torsional stiffness as the key parameter in aileron reversal.
- Developed statistical wing stiffness analysis, leading to flutter-resistant design criteria.

c) Mass-Balanced Aileron Failure Case (1930s, UK):

- Despite mass-balancing, chordwise movements induced by staggered V-struts resulted in aileron flutter.

- Demonstrated that structural interactions can override theoretical flutter prevention methods.
- Highlighted the necessity for comprehensive structural-aerodynamic integration in flutter mitigation.

5. Experimental & Wind Tunnel Testing

a) Pugsley (1930s, RAE):

- Conducted parametric flutter analysis and nonlinear Frise aileron behavior investigations.
- Demonstrated the importance of control circuit stiffness.

b) Flight Flutter Testing (von Schlippe, Germany; Frazer & W.P. Jones, UK):

- Developed practical in-flight flutter testing methods.
- Led to long-term research on flutter aerodynamics by W.P. Jones.

6. Airscrew (Propeller) Flutter Studies

a) Duncan & Collar (1930s, UK):

- Investigated metal-bladed airscrew flutter.

a1) Experimental setup:

- Step-down gearbox rotating a mirror to capture high-speed oscillations.
- Stroboscopic imaging for high-frequency flutter analysis.

a2) Theoretical analysis:

- Adapted wing flexure-torsion flutter models to airscrew blades.
- Faced challenges due to high aspect ratio & spanwise mass variations.

7. Contributions to Aerodynamic Theory

a) Glauert's Frequency Parameter (1929):

- Defined reduced frequency dependence of aerodynamic forces.
- Extended by Duncan & Collar to pitch-heave coupled motion.

b) Theodorsen's Circulation Function (1935, US):

- Developed oscillating control surface models.
- Recognized Hankel functions as solutions for unsteady aerodynamics.

2.1.8. Summary of this Period

The early studies on aeroelasticity and flutter during the first half of the 20th century played a crucial role in shaping modern aircraft design methodologies. From the failure of Langley's Aerodrome in 1903 to the structural challenges faced by German fighters in World War I, the limitations of early aircraft structures revealed the critical need for improved torsional stiffness, mass balancing, and synchronized control surfaces. These incidents emphasized the fundamental importance of aerodynamic load distribution and structural rigidity in preventing destructive flutter effects.

One of the earliest recorded flutter incidents was the Handley Page O/400 crash in 1916, where the interaction between fuselage torsion and antisymmetric elevator motion resulted in large oscillations, leading to instability. This event ultimately led to Lanchester's recommendation of a rigid elevator connection, which became a standard feature in aircraft design. Similarly, the Albatros D-III and Fokker D-VIII fighters experienced static divergence failures due to structural weaknesses and unintended modifications that shifted the elastic axis, demonstrating the necessity of precise load distribution and flutter-resistant wing structures.

Further cases, such as the De Havilland DH 80 and the MO-1 tail flutter incident, highlighted the role of aileron balancing, wing strut stiffness, and torsional rigidity in mitigating flutter risks. The DH 80's instability, caused by unbalanced ailerons and weak wing struts, underscored the significance of structural reinforcement to prevent aeroelastic instabilities. Similarly, the MO-1's tail flutter, driven by differential spar stiffness and bending-torsion coupling, demonstrated the necessity of proper stiffness distribution to avoid excessive deflections. These studies led to improved mass distribution techniques and modifications in control surface designs to mitigate flutter effects.

Beyond aviation, the Tacoma Narrows Bridge collapse in 1940 provided a striking example of flutter-induced structural failure in civil engineering. The bridge's torsional oscillations, amplified by aerodynamic forces, resulted in self-excited vibrations that ultimately led to collapse. This event reinforced the need for aeroelastic simulations in structural engineering and influenced modern approaches to bridge design and wind resistance analysis.

The theoretical advancements in this period, particularly the British contributions to flutter mitigation and Theodorsen's two-dimensional flutter theory, established rigorous analytical frameworks for flutter prediction. The British Royal Aeronautical Establishment (RAE) and National Physical Laboratory (NPL) developed fundamental semi-rigid mode approximations, enabling simplified differential equation models to assess flutter stability. The introduction of wind tunnel testing and mass balancing techniques allowed for experimental validation of theoretical models, leading to enhanced aircraft safety. Meanwhile, Theodorsen's circulation function ($C(k)$) and his studies on frequency-dependent aerodynamic derivatives provided key insights into the energy extraction mechanisms driving flutter, influencing strip theory applications in wing flutter analysis.

This period marked the foundation of modern aeroelasticity research, integrating theoretical advancements, experimental methodologies, and structural modifications

to mitigate flutter effects. The transition from biplanes to monoplanes, the introduction of stressed-skin structures, and the implementation of mass balancing and structural stiffening techniques directly influenced the design of next-generation aircraft and civil engineering structures. These early studies remain cornerstones of aeroelastic stability analysis, continuing to shape the safety and performance of modern flight vehicles and structures.

These initial flutter incidents and their corresponding technical solutions laid the groundwork for a more comprehensive understanding of aeroelastic stability in aircraft design. The lessons learned during this period not only shaped future approaches to structural reinforcement and control system synchronization but also highlighted the importance of advanced testing methods, such as wind tunnel simulations. As the aviation industry transitioned into the World War II era and beyond, these foundational principles would continue to evolve in response to new technological challenges and higher performance demands.

2.2. After World War II Period

Since it was still the first years of aircrafts during World War I, aircrafts were generally used for reconnaissance purposes and crashed for various reasons, thus proving that they were not yet fully suitable for use as weapons. However, the advances made especially in flutter and the various systematic solutions developed have somehow paved the way for them to be used more actively in wars. Especially the unstable situation in Europe after World War I seemed to herald the arrival of a new big war. This made it an obligation, not an option, for countries to make a breakthrough in this regard. When the world war started, especially in 1941 Germany attacked Soviet Union and destroyed their most of aircrafts and this led to superiority in air during that year. With this superiority, when Germany invaded the interior of Soviet Union and took control of the most important strategic points, it was understood that this consideration was correct.

2.2.1. P-80 Flutter Incident

During the 1944 flight tests of the P-80 aircraft, a type of flutter known as aileron buzz was observed. This phenomenon is a single-degree-of-freedom flutter that occurs due to the interaction between aileron rotation and the chordwise movement of shock waves on the wing [14].

Aileron buzz is a type of flutter often affects aircraft control surfaces, such as ailerons, during high-speed flight, especially at transonic or low supersonic speeds. The P-80 was identified in 1944 when a new type of aileron vibration was detected in flight tests at high subsonic speeds. This flutter is caused by the coupling between the rotational motion of the aileron around its hinge and the chordwise movement of shock waves on the wing surface. Essentially, as the aileron rotates, it interacts with the shock waves generated in the wing, causing the shock waves to move back and forth along the chord of the wing. This interaction results in self-excited oscillations confined to a single degree of freedom: the rotation of the aileron.

The incidence of aileron buzz is related to the Mach number, rather than the equivalent airspeed. This leads to the idea of a critical buzz Mach number, below which this flutter doesn't occur. The actual mechanism behind buzz oscillations is complex and varies with different flow regimes. A theoretical explanation involves negative aerodynamic damping, where potential flow over a thin aerofoil can result in instability at low frequencies for subsonic speeds and practical frequencies at supersonic speeds. But, that doesn't fully explain the buzz at Mach numbers of 0.8 or below. This shows that buzz can take many forms depending on factors such as wing section, angle of incidence and Mach number [15].

Preventive measures for aileron buzz include modifications to the control surfaces and their operating mechanisms. For example, a splitter-plate rudder can fix the location of shock waves for small deflections, preventing their differential movement. Blunt trailing edges can also help by altering the aerodynamic characteristics of the control surface. Additionally, fully powered controls with actuators positioned directly

on the control surfaces can provide the necessary force and responsiveness to counteract and dampen the oscillations, thereby mitigating the buzz phenomenon [16].

2.2.2. Lockheed Electra Flutter Incident

After the 2 major Lockheed Electra crashes in 1959 and 1960, the Federal Aviation Agency (FAA) launched an intensive investigation and discovered that a critical phenomenon known as "whirl mode" was the reason of these crashes. This case shows up when the structural integrity of the powerplant installation, especially the engine nacelle or propeller mount, is compromised. In this case, the gyroscopic force of the propeller causes a large vibration that cannot be quickly damped with a decrease in structural stiffness. These oscillations, or whirl mode, combined with natural frequencies of the wings can cause flutter at lower speeds than normally it should be. This type of flutter makes a big difference and causes serious damage. After observing this, the FAA took several steps like; reducing operating speeds, adjusting significant maintenance inspections, installing flight recorders, and re-assessing the design of the aircraft. The program approved that despite having a robust original design, the weakness of powerplant led to whirl mode to become more hazardous. Therefore, changes should have been to stabilize the propeller, such as adding additional mounts, strengthening the wings with additional braces and thicker panels, reinforcing the nacelle structure. By making these adjustments, the aircraft's ability to dampen oscillations was notably improved, which eliminating the conditions leading to whirl mode [17].

2.2.3. Panel Flutter and Saturn V Apollo Flutter Incident

Panel flutter is a self-excited, dynamic-aeroelastic instability of thin plate or shell-like components of a vehicle. To prevent panel flutter in structural design, certain principles enhance stability and reduce the risk of flutter without the need for extensive testing or advanced calculations. First, panel thickness plays a crucial role: thinner panels are more vulnerable to flutter, especially in high-speed airflow, whereas thicker panels designed to endure significant static (such as compressive) or dynamic (like bending) forces inherently resist flutter. To further improve stability, aligning the shorter panel edges and stiffeners parallel to the airflow direction is recommended, which promotes aerodynamic stability. Additionally, preventing closely spaced natural frequencies is essential, as frequencies that are too close can lead to unwanted resonance. Small adjustments, like eliminating imperfections in panel geometry and adding curvature perpendicular to airflow, further reduce flutter susceptibility.

Defining flutter margins and using conservative assumptions in panel flutter assessment are key design strategies. Given the current limitations in analysing certain panel configurations and loading conditions, a typical safety margin of 50% on flutter dynamic pressure is often applied to ensure robustness. However, overly conservative assumptions can lead to excessively thick and heavy panels. To

counter this, designers may opt for more accurate prediction methods, such as wind tunnel or flight tests, to reduce uncertainties. If flutter is observed during testing, sometimes it is shown to be non-destructive, avoiding the need for redesign. This approach is usually reserved for short-lived, single-use vehicles.

Preliminary panel flutter predictions commonly use established design charts based on empirical and theoretical data for specific configurations and flow conditions. These charts offer theoretical frameworks for high-Mach-number situations, such as rectangular panels under compressive loads, though caution is recommended at lower Mach numbers or with pressurized panels. While these references are useful, they require careful consideration of panel design parameters, as their simplified assumptions may not fully address all conditions. As a result, panels with high flutter dynamic pressure margins are prioritized for detailed analysis to streamline design efforts effectively.

These steps underscore lessons learned from the Saturn V Apollo missions in the 1960s, where panel flutter led to significant structural issues and cost overruns. By integrating robust flutter margins and conservative early-phase testing, such structural challenges - as seen in the Saturn V Apollo - can be better anticipated and mitigated in similar high-speed aerospace projects [18].

2.2.4. Key Developments in Aeroelasticity and Flutter Research (After the start of WW II) [1] [14] [20] [21] [22] [23]

1. Changes in Aeroelastic Research During World War II

- Shift in focus due to war; key researchers like Frazer, Duncan, and Pugsley reassigned.
- Increased importance of compressibility effects due to the advent of jet engines.
- Research split between vibration analysis, quasi-static problems, flutter studies, and theoretical investigations.

2. Major Aeroelastic Problems and Their Solutions

a) Vibration Issues

- Airframe vibrations caused by engines and propellers.
- Early resonance tests used crude methods; improved with new sensors and actuators.
- Nose-wheel shimmy identified as a friction-induced oscillation, solved using the Marstrand twin-tread tyre.

b) Quasi-Static Aeroelastic Issues

b1) Loss and Reversal of Elevator Control

- Wooden aircraft in 1942 experienced control reversals due to fuselage flexibility.
- Investigation led to standardized structural stiffness requirements for tailplanes and elevators.

b2) Aileron Control Issues

- Aileron snatch and overbalance found on Spitfire due to varying cable tensions.

b3) Hinged Wing Proposal (Rouanet-Rey Design):

- An aeroelastic wing design that could flex to generate roll control.
- Deemed impractical due to hinge complexity and spring constraints.

c) Flutter Research and Case Studies

c1) Tab Flutter

- Common in trim and geared tabs; mass balance position critical.
- Fatal Meteor accident due to incorrect tab mass placement.

c2) Typhoon Tail Flutter Issue:

- Mass-balancing error in control system caused complete tail detachment.
- Corrected by redesigning mass placement near the fuselage node.

c3) Meteor Wing Flutter Investigation:

- Wind tunnel and theoretical analysis confirmed buried jet engines posed no major aeroelastic risk.

d) Theoretical Investigations

d1) Supersonic Aeroelastic Analysis

- Early 1943 work on a Miles Aircraft supersonic prototype.
- Extended Ackeret's theory to estimate stiffness requirements.

d2) Sweepback Wing Investigations

- Post-war research into aerodynamic and aeroelastic benefits of wing sweep for compressibility effects.

3. Unsteady Aerodynamics and Experimental Aeroelastic Models

a) Development of Aeroelastic Wind-Tunnel Testing

- Increased reliance on experimental validation of flutter predictions.
- Plastic-based scaled models (e.g., PBM-1 seaplane, Vultee XP-54, Junkers JU-288).

b) Shift from Replica Models to Simplified Beam Models

- Focused only on significant vibration modes.
- Used lightweight materials with adjustable mass distribution.

c) Use of Specialized Wind Tunnels

- 4-ft Langley Wind Tunnel designed for high-speed flutter research (1946).
- Use of freon gas to simulate subsonic and transonic conditions.

4. Transonic and Supersonic Flutter Problems

a) Transonic Flutter Issues

a1) Aileron Buzz (P-80 Aircraft, 1944)

- Caused by coupling between aileron rotation and shock wave motion.
- Traditional mass balancing was ineffective.
- Solved through increased stiffness, dampers, and airfoil profile changes.

a2) Empirical Flutter Criteria Developed:

- Regier's empirical relations for control-surface buzz avoidance and propeller stall flutter.

b) The Impact of Swept Wings on Aeroelasticity

- Boeing B-47 bomber used sweep and aeroelastic tailoring for flutter prevention.
- Wing flexibility (30-ft tip deflection) required precise jig shaping.

c) Early Transonic and Supersonic Flutter Testing Methods

c1) Alternative Experimental Techniques

- Free-flying models: Dropped from aircraft or launched via rockets.
- Wing-flow technique: Used transonic flow regions on aircraft wings.
- Rocket sleds: Used to investigate full-scale component flutter.

d) Supersonic Flutter Research

- Analytical studies expanded on Possio's theory and piston theory (Lighthill).

d1) Panel Flutter

- First observed in V-2 rockets (70+ failures due to panel vibration).
- Occurred in Saturn V Apollo requiring costly investigation.
- Common issue in fighter jets, causing cockpit noise and hydraulic failures.

5. Flutter Incident Reports and Military Design Considerations

e1) 1956 NACA Flutter Incident Report

- 54 documented flutter cases on U.S. military aircraft.
- 21 cases involved transonic control-surface buzz.
- 7 wing flutter cases linked to externally mounted stores (e.g., pylon engines).
- 4 incidents involved all-movable control surfaces at transonic/supersonic speeds.

e2) FAA & Military Design Regulations

- Guidelines for preventing flutter from ice accumulation, surface painting, free play in controls, balance weight failures.

6. The Rise of Computational Methods in Flutter Analysis

a) Early Computational Approaches

a1) Analog Computers (Differential Analyzers)

- Used mechanical means to solve aeroelastic problems.
- Applied in flutter analysis at Caltech (MacNeal, McCann, Wilts).

a2) Development of Digital Computers

- IBM symposium in early 1950s highlighted industry adoption for flutter analysis.

a3) Finite Element Analysis (FEA):

- Developed in the 1950s for aeroelasticity and structural analysis.
- Work by Argyris, Turner, and others established modern FEA frameworks.
- Numerical aerodynamic methods (e.g., panel methods, lattice methods) laid foundations for FEA-based flutter prediction.

2.2.5. Summary of this Period

Following World War I, aircraft technology advanced significantly, particularly in addressing flutter-related issues. These developments contributed to the increasing role of aircraft in warfare, as evidenced by their strategic use during World War II. The conflict demonstrated the necessity of air superiority, especially when Germany's 1941 invasion of the Soviet Union showcased the decisive impact of aerial dominance.

In the post-war era, flutter phenomena continued to challenge aircraft development, as seen in incidents like the P-80 aileron buzz in 1944. This type of single-degree-of-freedom flutter, caused by the interaction between aileron movement and shock waves on the wing, emphasized the importance of aerodynamic stability at high speeds. The investigation of aileron buzz led to critical design modifications, including improved control surface mechanisms and enhanced actuator systems to suppress oscillations.

Similarly, the Lockheed Electra whirl mode incidents of 1959 and 1960 highlighted the dangers of structural weaknesses in powerplant installations. The gyroscopic forces of the propeller induced severe oscillations that, when coupled with natural wing frequencies, resulted in catastrophic flutter. In response, regulatory agencies implemented strict operational limitations, enhanced maintenance protocols, and mandated structural reinforcements to mitigate these risks.

Another significant case was panel flutter, which became particularly evident during the Saturn V Apollo program in the 1960s. This self-excited instability, affecting thin structural panels in high-speed airflow, underscored the necessity of robust design principles. Engineers applied strategies such as increased panel thickness, optimized structural alignment, and defined flutter safety margins to enhance stability. While conservative assumptions helped prevent structural failures, excessive conservatism sometimes led to unnecessary weight increases, necessitating advanced prediction methods like wind tunnel testing.

World War II accelerated the need for deeper aeroelastic analysis, leading to better experimental and computational tools.

Flutter research expanded beyond aircraft to include missiles, rockets, and space vehicles (e.g., Saturn V panel flutter issues).

Wind tunnel and free-flight experiments were critical for validating flutter predictions in transonic and supersonic regimes.

The adoption of swept wings and thin airfoils required new aeroelastic considerations, influencing aircraft designs like the Boeing B-47 and B-52.

Computational methods (FEA, digital computers) revolutionized flutter analysis, enabling more precise and efficient predictions.

2.3. Ply Orientation

Flutter is a self-activated and potentially destructive phenomenon in aircraft structures and is a significant problem in aeroelastic operations. As aircraft design increasingly focuses on lighter and stronger composite materials, it is important to understand the impact of various aspects of waviness. One of the most striking features is the ply orientation in composite wings.

Composites created from different processes or layers have the advantage of cutting materials to specific specifications. The structure of the layers, especially the angle of the longitudinal wing, plays an important role in determining the overall skin and aeroelastic behaviour. Ply orientation affects the bending and torsional stiffness of the wing, which in turn affects flutter generation.

Researches have shown that the properties of composite wings depend on the direction of the plies. For example, maximum bending stiffness is achieved when the ply is oriented relative to the direction of flight (0 or 90 degrees). Conversely, orientations such as ± 45 degrees lead to increased torsional rigidity. The coupling of bending and torsional modes, influenced by ply orientation, has a direct impact on flutter speed.

The significance of ply orientation will become more evident in aeroelastic adaptation, where designers aim to optimize the wing configuration with respect to surge resistance. It has been found that certain orientations (e.g. ± 45 degrees) achieve a combination of high torsional stiffness and low joint stiffness, ultimately leading to higher flutter speeds.

It was also used to study layer-oriented fluctuation behaviour such as the velocity damping (V-g) method through wind tunnel measurements and observations. These studies provide a better understanding of the interaction between composite materials and aeroelastic conditions [24]

In summary, understanding the flutter phenomenon in composite wings requires research on ply orientation effects. As the aerospace industry continues to adopt advanced materials, optimizing ply orientation is becoming an important aspect of aeroelastic design, providing the ability to improve surge resistance and overall structural performance.

2.4. Discussion and Conclusion

The Literature Review section provides a comprehensive overview of the aeroelastic behaviour of cantilever wings. Reviews and analyses address various factors critical in wing design. These factors include wing flutter analysis, optimization of flight performance, material selection and aeroelastic control strategies.

In particular, the reviewed literature addresses in depth the current knowledge regarding the aeroelastic issues faced by cantilever wings. The most important of these problems are the elastic properties of the materials used in wing design and the wing geometry. In addition, how aerodynamic loads change over time and the effects of these changes on the wing are examined in detail.

However, certain limitations and knowledge gaps were also recognized during this comprehensive review process. In particular, it is noteworthy that there are no studies focused on a specific methodology and that specific parameters are not discussed in detail. At this point, it is recommended that future research focus on specific methodologies and parameters and conduct more comprehensive studies considering various technical details.

This literature review brings together existing knowledge on aeroelastic behaviour while highlighting important issues that can guide future researchers. Further experimental and numerical studies focusing on aeroelastic control strategies, material selection and wing geometry will expand and improve knowledge in this field. This review has the potential to be an important reference source regarding the aeroelastic behaviour of cantilever wings and aims to shed light on future research.

CHAPTER 3 – MODEL EQUATIONS

3.1 Introduction

In order to accurately simulate the dynamic and aeroelastic behaviour of composite wing-like structures, it is essential to establish a mathematical foundation that captures both the structural mechanics and the aerodynamic loading mechanisms. This chapter presents the governing equations, assumptions, and modelling approaches used in the modal, harmonic, and (where applicable) static structural analyses conducted in ANSYS Workbench. The derivations are provided in a step-by-step manner to ensure clarity and reproducibility. All variables are defined upon first use, and the equations are presented in a general form before being adapted to the specific parameters of this study.

3.2 Structural Dynamics: Governing Equations

The starting point for the vibration analysis of any structure is the general form of the equation of motion:

$$[M]\{\ddot{u}(t)\} + [C]\{\dot{u}(t)\} + [K]\{u(t)\} = \{F(t)\}$$

where:

[M] = global mass matrix,

[C] = damping matrix,

[K] = stiffness matrix,

{u(t)} = displacement vector,

{F(t)} = external force vector.

For undamped free vibration (modal analysis), damping and forcing are neglected:

$$[M]\{\ddot{u}(t)\} + [K]\{u(t)\} = 0$$

A harmonic trial solution of the form

$$\{u(t)\} = \{\phi\}e^{i\omega t}$$

yields the eigenvalue problem:

$$([K] - \omega^2[M])\{\phi\} = 0$$

Here, ω is the natural circular frequency (rad/s), and $\{\phi\}$ is the corresponding mode shape vector. The natural frequency in Hz is: $f_n = \omega / (2\pi)$.

3.3 Mode Superposition for Harmonic Response

For harmonic response analysis, the solution is expressed as a linear combination of mode shapes:

$$\{u(t)\} = \sum_{r=1}^m \{\phi_r\} q_r(t)$$

Substituting into the full equation of motion and applying modal orthogonality decouples the system into:

$$\ddot{q}_r(t) + 2\zeta_r\omega_r\dot{q}_r(t) + \omega_r^2 q_r(t) = \frac{F_r(t)}{M_r}$$

where

ζ_r = modal damping ratio,

M_r = generalised modal mass,

$F_r(t)$ = generalised modal force.

For a harmonic force, the steady-state amplitude of mode r is:

$$Q_r = \frac{F_r/M_r}{\omega_r^2 - \Omega^2 + 2i\zeta_r\omega_r\Omega}$$

The total displacement is reconstructed from the modal contributions.

3.4 Aerodynamic Load Modelling

The aerodynamic lift L and drag D are obtained from:

$$L = \frac{1}{2}\rho V^2 S C_L$$

$$D = \frac{1}{2}\rho V^2 S C_D$$

where ρ = air density (assumed 1.225 kg/m³),

V = free-stream velocity,

S = reference area ($1.0 \times 0.05 = 0.05$ m²), and C_L , C_D are lift and drag coefficients.

The corresponding uniform pressure load p applied in the finite element model is:

$$p_L = \frac{L}{S}, \quad p_D = \frac{D}{S}$$

In this study, drag-induced pressures are much smaller than lift-induced pressures

and are thus neglected for out-of-plane deformation analysis.

3.5 Laminate Modelling: Classical Laminate Theory (CLT)

The composite layups are modelled using Classical Laminate Theory. Each ply is treated as an orthotropic lamina with its own stiffness matrix $[Q]$ in the material coordinates, which is transformed into the laminate reference frame using the transformation matrix $[T(\theta)]$ based on the ply orientation angle θ .

The transformed reduced stiffness for a ply is:

$$[\bar{Q}] = [T]^{-1}[Q][T]^{-T}$$

The laminate extensional, coupling, and bending stiffness matrices ($[A]$, $[B]$, $[D]$) are obtained by integrating through the thickness:

$$[A] = \sum_{k=1}^n [\bar{Q}]_k (z_k - z_{k-1})$$

$$[B] = \frac{1}{2} \sum_{k=1}^n [\bar{Q}]_k (z_k^2 - z_{k-1}^2)$$

$$[D] = \frac{1}{3} \sum_{k=1}^n [\bar{Q}]_k (z_k^3 - z_{k-1}^3)$$

3.6 High-Speed Scenario as Future Work

For a future work scenario at $V = 200$ m/s ($Mach \approx 0.59$), compressibility effects become significant. In this regime, the lift and drag coefficients should be corrected using Prandtl–Glauert transformation:

$$C_{L,\beta} = \frac{C_L}{\sqrt{1 - M^2}}$$

$$C_{D,\beta} = \frac{C_D}{\sqrt{1 - M^2}}$$

where $M = V/a$ is the Mach number and a is the speed of sound (~ 340 m/s at sea level). The aerodynamic loads are then recalculated using these corrected coefficients and applied in a prestressed modal or harmonic response framework to capture potential flutter onset.

3.7 Summary

This chapter has provided the mathematical backbone for the simulations conducted in this study, bridging structural dynamics, composite laminate theory, and aerodynamic load estimation. The formulations here enable replication of the analysis for different velocities, ply configurations, and future high-speed cases.

CHAPTER 4: NUMERICAL SIMULATIONS AND ANALYSIS PROCEDURE

This chapter presents the results of the numerical simulations used to investigate the cantilevered composite wing's aeroelastic behaviour.

The entire procedure is explained in depth, including the steps of pre-processing (meshing and setup), solution strategy, and post-processing.

To evaluate the aerodynamic performance and structural integrity under various conditions, a series of assessments are carried out, including static, dynamic, and aeroelastic evaluations.

To guarantee accurate and dependable findings, each type of study is customised with the proper physical models, boundary conditions, and solution controls.

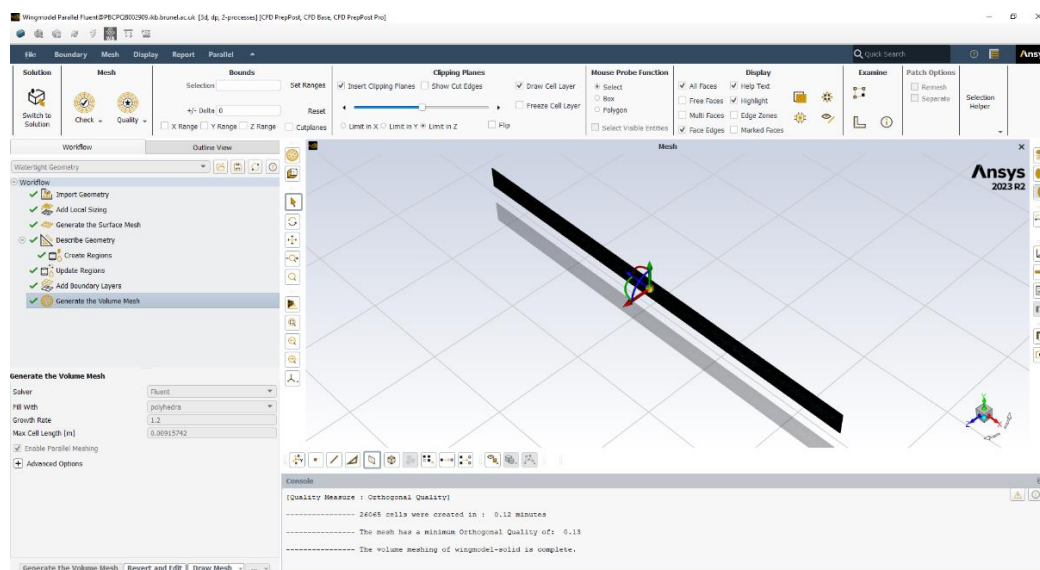
4.1 Static Analysis

The primary objective of the static analysis stage is to ascertain the aerodynamic force distribution and related surface pressure fields acting on the composite wing under steady-state flow conditions.

Before adding time-dependent or structural deformations in later research, this analysis is the first step towards comprehending the aerodynamic properties.

The choice of turbulence model, material definitions, boundary condition assignments, and solver settings are all covered in detail in this section on the computational setup for the static simulations.

4.1.1. Fluent Meshing



A solid-body based meshing technique was used with ANSYS Fluent Meshing as part of the Watertight Geometry workflow to generate the mesh for the composite cantilever wing's flutter simulation. Since external flow properties surrounding the wing are of greater importance than inside fluid domains, the geometry was classified as a solid region. This choice guarantees compatibility with further structural analyses, in which the definition of composite layups, ply orientations, and anisotropic material behaviours depend on solid elements.

The process of meshing involved first creating a surface mesh and then utilising a polyhedral filling approach to create the volume mesh. In order to ensure precise resolution at the leading and trailing edges of the wing, the surface mesh settings were used to capture both curvature and proximity-based characteristics. To ensure a seamless transition from the wall-adjacent cells to the far-field, a growth rate of 1.2 was used.

The requirement for precise resolution of near-wall flow gradients led to the inclusion of boundary layers in the meshing procedure. With three inflation layers, a transition ratio of 0.272, and a layer growth rate of 1.2, a smooth-transition offset approach was used. This guarantees that the mesh is fine-tuned close enough to the solid surface to capture pressure and velocity gradients that directly affect shear stress and, in turn, surface pressure distributions—essential for precise mapping of aeroelastic loads.

The Polyhedra fill option was used for volume meshing. Compared to conventional tetrahedral meshes, this meshing technique offers better convergence behaviour and a lower element count, making it beneficial for complex geometry. Because of its improved element aspect ratios and decreased skewness, the polyhedral mesh structure enables higher solution accuracy at a lower computational cost. A 0.00915742 m maximum cell size was mandated.

Upon mesh generation, ANSYS Fluent reported the following quality metrics:

- Orthogonal Quality (minimum): 0.13

- Average Skewness: ~0.073

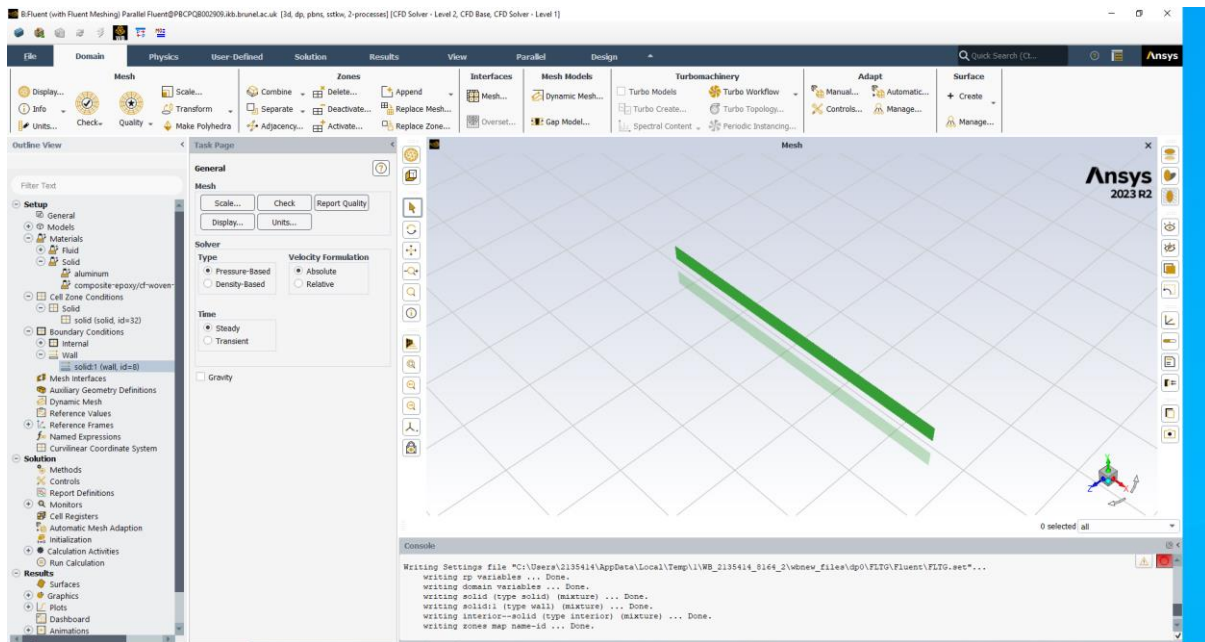
- Maximum Skewness: ~0.50

- Cell Count: 26,065 cells

These values are very excellent in terms of mesh quality. For Fluent simulations, an average skewness of less than 0.25 and a maximum skewness of less than 0.9 are typically acceptable; lower values denote better element alignment and shape. Even though it is a little lower, the orthogonal quality metric of 0.13 falls within acceptable boundaries for applications involving external flow, particularly when polyhedral meshing is used to account for local element deformation.

All things considered, the produced mesh ensures that the surface pressure distributions can be precisely translated to a structural solver and offers a solid basis for high-fidelity external aerodynamic analysis. This is crucial for flutter prediction in composite structures and for subsequent phases of the aeroelastic workflow.

4.1.2. Setup



To guarantee reliable simulation results, the CFD solver (ANSYS Fluent) was built up with certain model selections and material assignments during the setup phase of the static aerodynamic study for the cantilevered composite wing.

The turbulence model, which combines the benefits of the $k-\epsilon$ and $k-\omega$ models, was first chosen as the SST $k-\omega$ model. When applied to external aerodynamic flows over lifting surfaces like wings, SST $k-\omega$ offers better forecasts in areas with unfavourable pressure gradients and separation. To improve model stability and realism close to curved surfaces, curvature correction and production limiter options were made available.

The fluid domain in the Materials section was designated as air under typical atmospheric circumstances, with a constant density of 1.225 kg/m^3 and a dynamic viscosity of $1.7894\text{e-}05 \text{ kg/m-s}$.

Aluminium was initially designated as the solid material; however, a carbon fibre reinforced epoxy composite material chosen from the GRANTA MDS database was used to represent the anticipated composite construction. This modification guarantees that the true physical characteristics of the composite wing are reflected in aerodynamic interactions.

The selected composite was used in place of aluminium as the structural material for the solid domain in the Cell Zone Conditions. The solid domain refers to the newly chosen composite material, so even though Fluent technically kept the aluminium material "in use" because of default settings, it has no effect on the final aerodynamic simulation.

In terms of boundary conditions, Fluent immediately recognised the solid wing surface as a "wall" barrier. Additional structural boundary conditions (like fixed support or elastic restrictions) were not explicitly implemented in Fluent since the

analysis concentrates on aerodynamic force outputs (like lift and drag) rather than complete structural deformation. To accurately simulate the effects of surface friction, the wall's no-slip condition was preserved.

Since the simulation only looks at external flow forces in a steady-state environment, no special wall functions or temperature gradients were turned on. The typical correlation-based configuration suitable for SST $k-\omega$ was maintained for near-wall treatment.

All crucial CFD setup parameters, such as fluid characteristics, turbulence models, and boundary definitions, have been completed at this point, allowing the analysis to proceed to the initialization and solution run phases.

4.1.3. Solution

ANSYS After the mesh generation and geometry preparation steps, Fluent was utilised to start the computational fluid dynamics (CFD) solution process. To ensure numerical accuracy and convergence, the following approach was employed:

a) Solution Methods: The SIMPLE (Semi-Implicit Method for Pressure Linked Equations) pressure-velocity coupling system was selected due to its efficiency and numerical stability in steady-state incompressible flow scenarios.

-Second Order Upwind methods were used for pressure, momentum, turbulent kinetic energy (k), and particular dissipation rate (ω) to increase the accuracy of gradient approximations across the mesh.

b) Solution Controls: The default under-relaxation factors were used to stabilise the iterative solution. To make convergence happen faster and keep the numbers stable, certain changes were made. For example, the pressure relaxation factor was adjusted to 0.3, the momentum to 0.7, and the turbulent variables (k and ω) to 0.8.

c) Residual Monitors and Convergence Criteria: Residuals were used to keep an eye on convergence. The absolute convergence requirements were established at 10^{-3} for the particular dissipation rate, turbulent kinetic energy, continuity, and velocity components.

-When all of the residuals fell below the set thresholds, it was assumed that the solution would be stable and that a full convergence would happen.

d) Initialisation: A Hybrid Initialisation method was used, which combined parts of both standard and prospective flow initialisations to give a good first approximation for the flow field and make sure that it converged quickly.

e) Run Calculation: The simulation was done when the conditions were stable.

-After the computation was done, it was found that the lift, drag, and static pressure forces were all about zero.

-This finding is in line with the fact that there were no applied inlet or outlet boundary conditions, which means that no external aerodynamic loads were operating on the structure throughout this analysis phase.

f) Force Reports: After the solution, force reports were made for the lift and drag forces on the solid surface.
- As expected, the lift and drag forces that were calculated were 0 Newton, which shows that the solid body was in a static, isolated state with no fluid-induced loads.

g) Short Interpretation: The CFD solution phase was able to accurately predict how the cantilever wing would behave aerodynamically when it was not moving. This groundwork makes sure that later structural simulations can be done with explicit boundary force conditions. It also lays the framework for future aeroelastic studies.

4.2 Static Structural Analysis

This part shows the static structural study of a cantilevered composite wing that was done with ANSYS Workbench and the ACP (Pre) and Static Structural modules. The goal was to test how deformation and stress respond to simplified aerodynamic loading. This would serve as the structural basis for future aeroelastic and flutter tests.

4.2.1 Getting the Model Ready in ACP (Pre)

4.2.1.1. Importing Geometry and Improving the Mesh

The ACP (Pre) environment got the wing's 2D shape, which is 1 m wide and 0.05 m long. Before continuing with the definitions of composite layup, mesh refinement was done directly on the surface geometry. This stage makes sure that each ply layer, which will be specified later, has enough element resolution to support it. A finer mesh helps show stress gradients better throughout the plies and stops numerical errors from happening when the shell model is made and transmitted to the structural solver.

4.2.1.2. Defining Fabric (Single Fabric Approach)

Instead of making individual fabric entries for each ply, a single biaxial woven prepreg composite fabric was made and given to the chosen composite material, "Carbon Epoxy Fibre, Woven Prepreg, Biaxial." This method makes modelling easier and stops fabric directions from overlapping or conflicting in later stages. The single-fabric approach works when all the plies are made of the same base material and the distinctions are shown by the angles of the orientation.

4.2.1.3. Definition of an Orientated Selection Set

An orientated selection set was made and aligned with the vector (1.0, 0.0, 0.0) so that the fibre directions may be shown appropriately in relation to the global coordinate system. This set the reference direction for the next ply orientations. Setting this reference is very important in composites since the orientation of the fibres has a big effect on stiffness and stress response. If this weren't the case, ply angles would be based on random local coordinates, which might make structural anisotropy look wrong.

4.2.1.4. Defining the Modelling Group and Ply Layup

A modelling group was made and connected to the selection set that had already been defined. Four plies were introduced to this group, each one corresponding to a common aeronautical stacking sequence: Ply 1: 0° , Ply 2: $+45^\circ$, Ply 3: -45° , Ply 4: 90° . These orientations were used in relation to the local fibre axis that was determined before. The stacking order gives the right amount of stiffness, shear resistance, and strength against twisting loads. The thickness of each ply was fixed at 1.25 mm, which made the overall thickness of the laminate 5 mm.

4.2.1.5. Making a Shell Model

After the ply configuration was set, a shell model was made using the "All Elements" option. This made it possible for the surface geometry to hold embedded ply orientation and material definitions through a shell-based representation. The shell model has the anisotropic characteristics and orientation of each ply built into it. This information is then sent to the Static Structural module. This step is necessary to capture behaviour through thickness and make sure it works with stress/deformation solvers. The use of shell modelling in this study greatly lowered computing cost while keeping accurate depiction of ply-wise mechanical behaviour, making it well-suited for thin composite structures.

4.2.2. Setting up the static structure and the boundary conditions

a) Boundary Conditions: In the Static Structural module, a fixed support was put on the root edge (the short edge along $X = 0$) to make it look like a cantilevered setup. This condition shows how things work in the real world, like how the wings and fuselage join. A downward force of 100 N was applied at the free tip in the negative Y-direction. This made the aerodynamic lift distribution look like a point load for ease of understanding. We choose this value to:

- Make sure linear elastic behaviour
- Make deformation visible

- Avoid too much distortion of numbers or material failure
- Copy a simplified version of a typical aerodynamic loading condition

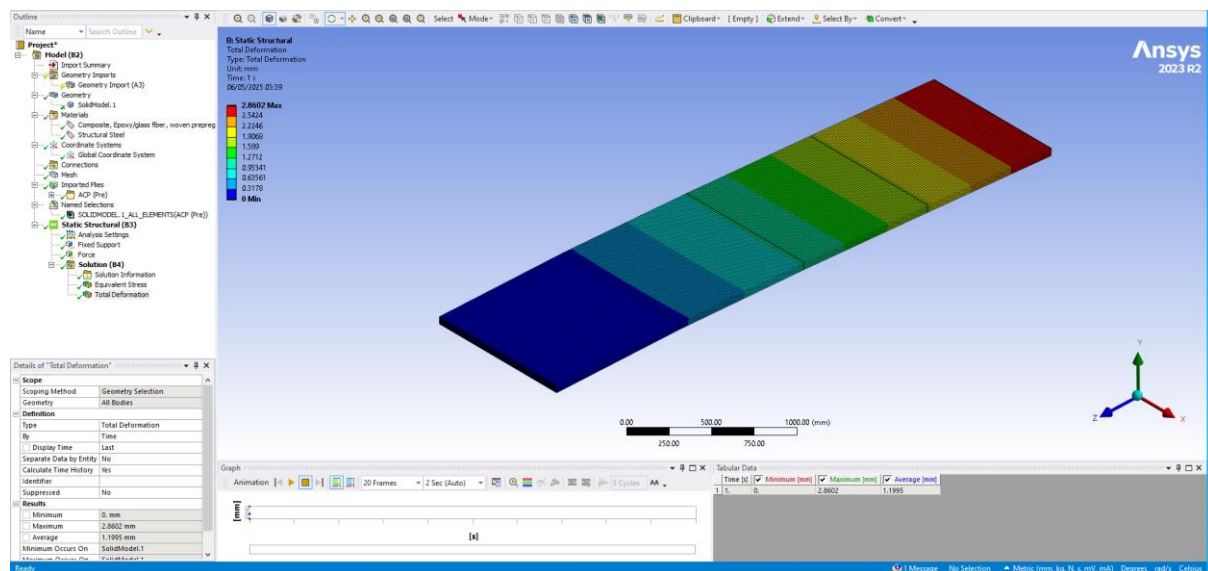
b) Named Selections Simplification: To make things clearer and avoid conflicts, all named selections except "All Elements (ACP Pre)" were taken away. This made sure that the structural analysis only employed the meshed, ply-assigned shell model from ACP. When stacking sequences are involved, having several conflicting named selections might cause problems since duplicate or undefined regions are being processed at the same time. Making it a single, well-defined mesh group makes the solver more stable and accurate.

4.2.3. Solution and Observations

The simulation worked well when all the conditions were set. The results showed:

- Maximum Deformation: 2.86 mm at the free tip
- Minimum Deformation: 0 mm at the fixed root
- Average Deformation: ~1.20 mm
- Smooth Stress Gradient: No singularities or failure concentrations
- At the root the maximum Von Mises stress is 0.319 MPA.
- At the free tip, the minimum Von Mises stress is 0.0001 MPA.
- The average Von Mises stress is about 0.09 MPA.

These results show that the modelled laminate design has the right structural stiffness and load transfer capability when subjected to static aerodynamic loads. The modelling method, which combined a single fabric, directional ply orientation, and mesh refinement, also produced robust and realistic findings. These results will be used as a base for the next step in flutter prediction, which will use modal and transient dynamic studies.



4.2.4. Justification of Modelling Decisions

In this examination, every decision about modelling was based on engineering logic, computational stability, and how well it applied to real-world composite wing uses. This part explains why the main parameters were chosen for the static structural analysis.

a) Choice of 4 Plies and Ply Angles: A four-ply setup was chosen because it makes a balanced and symmetric laminate structure that can realistically show how normal aerospace-grade composite wings act. The order of the chosen ply orientation was:

Ply 1: 0° – aligned with the wing span to carry axial loads

Ply 2: +45° – for torsional rigidity and in-plane shear resistance

Ply 3: -45° – to symmetrically balance shear effects

Ply 4: 90° – to resist transverse loads and maintain dimensional stability

This stacking order of $0^\circ / +45^\circ / -45^\circ / 90^\circ$ is often used in aerospace structures because it behaves almost isotropically in-plane, which means that the laminate can handle loads coming from all directions while still being easy to make and strong. Also, by equally spreading stiffness across all directions, it makes flutter resistance better.

b) Selection of 100 N Force Magnitude: The applied force of 100 N was chosen as a simplified approximation of aerodynamic lift forces acting on the wing surface. A concentrated point load was applied at the free end instead of spreading the load over the whole surface. This was done to simulate the highest expected bending moment in a method that used the least amount of computer power. This value is high enough to give useful deformation results while maintaining the stresses in the composite's elastic range. Also, 100 N makes sure that the reaction is linear, which is important for checking the model before moving on to dynamic or nonlinear flutter simulations.

c) Use of a Single Fabric with Directional Ply Control: Instead of making separate fabric entries for each ply, a single fabric was utilised with set ply angles to make the model less complicated and avoid problems that could happen during stacking and meshing. This method makes pre-processing easier while still letting you control the ply orientations and material attributes completely. Using Orientated Selection Set and Rosette Direction A reference vector along the global X-axis (1.0,0.0,0.0) was used to make an orientated selection set that would help people understand ply angles correctly. Because composites behave in a very anisotropic way, it was important to employ a rosette to specify all ply angles in relation to a given fibre direction. Simulations that are not physically accurate or solver matrices that are not well-conditioned can happen if orientation bases are wrong or not set.

d) Shell Model Generation and Named Selection Cleanup: The layup and orientation data were put into a shell model that precisely shows how the composite laminate behaves over its thickness. To avoid conflicts amongst solvers, all extraneous named selections were deleted, and just the ACP-generated shell model (All Elements) was used. This made guaranteed that only pieces that were appropriately layered and meshed were used in the structural solution, which got rid of any confusion and made the numbers more stable.

e) Choosing the Shape: The chosen aspect ratio of 20 (1 m span vs. 0.05 m chord) purposefully depicts a narrow, high-aspect-ratio wing segment. These kinds of shapes are often seen in UAVs, gliders, or wings that are designed to lift high efficiently. It was decided to record actual aeroelastic traits, like bending-dominant deformation and torsional sensitivity, which are more noticeable in skinny wings.

4.3. Modal Analysis Results and Interpretation across Multiple Ply Counts

To evaluate the influence of laminate thickness of 5 mm on the dynamic response of the composite wing, modal analyses were conducted for five stacking configurations consisting of 4, 6, 8, 10, and 12 plies. For each case, six natural frequencies were extracted. The stacking sequences were chosen to be balanced and largely symmetric in order to ensure structural stability and to mimic realistic laminate behaviour under aeroelastic loading.

4.3.1. Interpretation of Modal Results

The analysis shows six modes for each setup, however Mode 1 was always near to zero (for example, 0–0.0018 Hz) in all situations. This is a sign of rigid body drift, which happens when the structure is just slightly restrained or when very low-energy modes happen because of idealised pinned support conditions. Since these modes are non-physical and do not represent true elastic deformation, they are omitted from comparative evaluation.

The following modes—Mode 2 through Mode 6—exhibited meaningful elastic behaviour and are the focus of the interpretation:

- Mode 2, typically a torsional-dominant mode, was the primary indicator of flutter resistance. A higher frequency in this mode correlates with better aeroelastic performance. An increase from 13.35 Hz (4-ply) to 26.69 Hz (8-ply) was observed, indicating significant improvement.
- Mode 3, in many cases a coupled bending-torsion mode, initially remained relatively stable from 4 to 8 plies, but rose sharply in 10 and 12 plies. While this may seem beneficial, excessive proximity between Mode 2 and Mode 3 can indicate mode interaction, a common flutter trigger.
- Mode 4 and above are typically higher-order bending or shear modes. These increased significantly with ply count due to greater laminate stiffness. However, starting from 10 plies, the growth in frequency became nonlinear and marginal.
- Mode 5 and Mode 6 began to flatten or even slightly decrease beyond the 10-ply configuration, despite continued stiffness increase. This signals the onset of diminishing returns, where the added mass offsets the benefits of increased rigidity, potentially leading to modal crowding and instability.

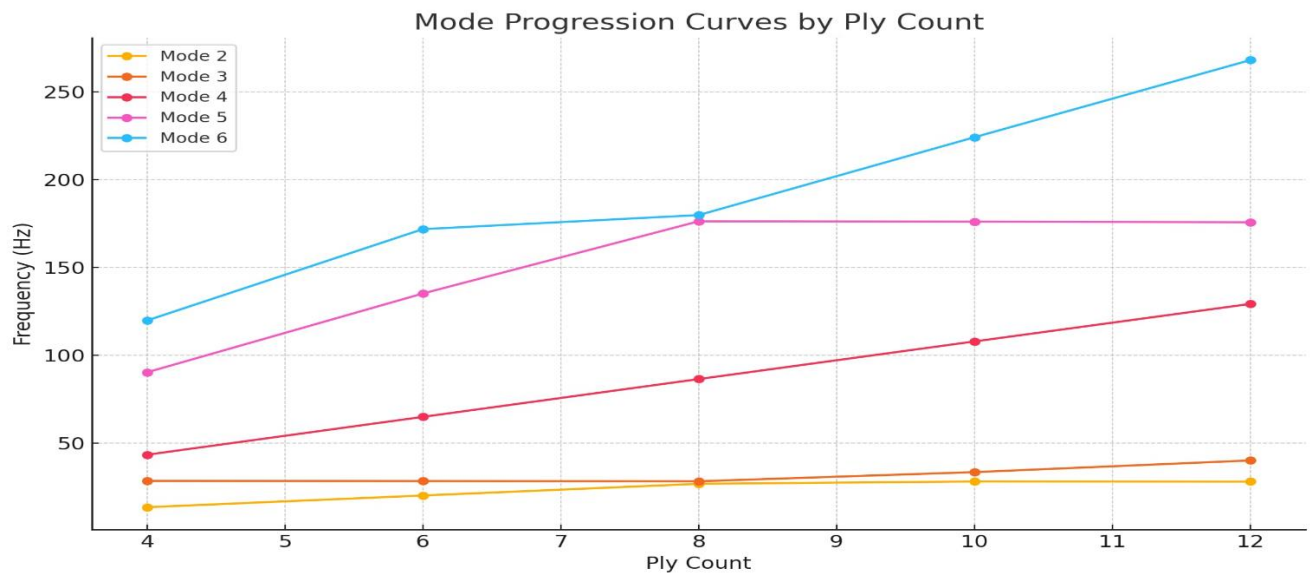
Although increasing the ply count results in stiffer laminates and higher modal frequencies initially, the benefit plateaus around 8–10 plies. The improvement in torsional mode (Mode 2) between 10 and 12 plies was less than 0.5%, despite a 25% increase in thickness and mass. Higher-order modes (Mode 5 and 6) began to decline or stagnate, signalling dynamic inefficiency. Mode separation between adjacent modes also became inconsistent, increasing the risk of modal coupling—a key precursor to flutter.

Hence, from both a performance and structural efficiency perspective, the 8-ply configuration offers the best trade-off between dynamic stability and mass. The

modal profile suggests that further ply additions do not meaningfully improve flutter resistance and may even compromise it through unintended dynamic effects.

Modal Frequencies By Ply Count

	Ply Count	Mode 2 (Hz)	Mode 3 (Hz)	Mode 4 (Hz)	Mode 5 (Hz)	Mode 6 (Hz)
1	4	13.35	28.37	43.26	90.23	119.77
2	6	20.03	28.25	64.85	135.16	171.85
3	8	26.69	28.13	86.39	176.33	179.87
4	10	28.03	33.36	107.86	176.12	224.26
5	12	27.93	40.01	129.25	175.8	268.25



4.3.2. Selection Rationale for Ply Orientations and Laminate Configurations

In composite laminate design, the number of possible stacking sequences increases exponentially with the number of plies. For a laminate with n plies and m possible orientations per ply (e.g., $\{0^\circ, \pm 45^\circ, 90^\circ\}$), the total number of combinations is given by:

$$\text{Total configurations} = T = m^n$$

where m is the number of ply angles and n is the number of plies.

For instance, using 6 orientation angles (e.g., $0^\circ, \pm 45^\circ, 90^\circ, \pm 30^\circ$), a 10-ply laminate yields over 60 million possible configurations. For 12 plies, this number exceeds 2 billion. Exhaustively testing each combination is computationally infeasible; even with high-throughput automation and parallel computing, the total analysis time would be

prohibitive. Therefore, this study employed a targeted exploration strategy, selecting laminate configurations that satisfy the following engineering principles:

- Balanced and symmetric stacking to minimize coupling and numerical instability
- Inclusion of standard industry layups (e.g., quasi-isotropic, cross-ply)
- Emphasis on torsion-resisting layers near the centre, and axial stiffness layers at the outer surfaces
- Iterative refinement based on modal and harmonic analysis results

4.3.3. Final Justification of Selected Ply Stacking Sequences

In this study, a limited number of stacking sequences were selected for each ply count (4, 6, 8, 10, and 12) to ensure computational feasibility and design relevance. For each ply number, the chosen layup was based on a combination of the following principles:

- Balance and symmetry, to minimize bending–twisting coupling and ensure stable modal behaviour
- Strategic orientation layering, with 0° layers placed at the outer surfaces to enhance axial stiffness, and $\pm 45^\circ$ or 90° layers distributed in the core to resist torsional and transverse loads
- Practical manufacturability, avoiding highly asymmetric or non-repeatable stackings
- Iterative optimization, based on observed trends in both modal and harmonic response analyses

The final selected configurations for each ply count are summarized below:

Ply Count	Stacking Sequence	Rationale Summary
4	[0 / +45 / -45 / 90]	Basic symmetric, quasi-isotropic baseline
6	[0 / 0 / +45 / -45 / 90 / 90]	Reinforced axial and torsional stiffness
8	[0 / +45 / -45 / 90 / 90 / -45 / +45 / 0]	Optimal modal separation and stability
10	[0 / 0 / +45 / -45 / 90 / 90 / -45 / +45 / 0 / 0]	Near-symmetric, flutter-resistant layout
12	[0 / +45 / -45 / 90 / +45 / -45 / 90 / -45 / +45 / 90 / -45 / 0]	Highly reinforced, shows diminishing returns

This selection strategy ensured that the study remained focused on relevant design candidates while still capturing the key structural behaviours that influence aeroelastic stability, especially in the context of flutter-critical applications.

4.4. Harmonic Response Analysis

The harmonic response analysis was conducted to evaluate the dynamic behaviour of the composite cantilever wing under sinusoidal loading. This step is essential to assess how the structure responds to varying frequencies, particularly around its natural frequencies, and to identify potential resonance regions which could lead to excessive vibrations or structural failure.

4.4.1 Objective

The goal of this phase is to compute the steady-state amplitude of the structure when subjected to a range of excitation frequencies, simulating real-world vibrational environments such as aerodynamic flutter or engine-induced harmonic loads. By evaluating the amplitude-frequency and phase-frequency responses, resonance frequencies and corresponding deformation amplitudes can be determined.

4.4.2 Methodology

-Geometry and Meshing: The same rectangular composite wing model used in the static and modal analyses was employed. A structured mesh was applied with refined element sizing to ensure convergence and solution stability in the dynamic domain.

-Material & Layup Definition: The laminate was defined using composite ply stacking information from the ACP (Pre) module. Different stacking sequences with 6, 8, 10, and 12 plies were tested to evaluate their influence on vibrational characteristics. Each ply was defined with a thickness of 0.00125 m, maintaining symmetry and manufacturing feasibility.

-Boundary Conditions:

Nodal Displacement: The fixed end (root) of the cantilever wing was constrained in all translational degrees of freedom (UX, UY, UZ = 0 mm).

Harmonic Force: A 100 N sinusoidal force was applied in the Z-direction at the free tip of the wing to simulate aerodynamic excitation. The force was defined as a surface effect to better represent distributed aerodynamic pressure.

-Analysis Settings:

Frequency Range: 3 Hz – 300 Hz

Frequency Spacing: Linear

Solution Method: Mode Superposition

Number of Steps: 100 (ensuring smooth curve progression)

Structural Damping: 1%

Solver Type: Mode Superposition was selected to utilize previously computed modal data, significantly reducing computational time while retaining accuracy for linear harmonic response problems.

4.4.3. Results and Observations

Each stacking configuration produced a distinct frequency response curve. The main observations include:

-Peak Amplitudes: The peak displacement decreased with increasing ply count, confirming enhanced stiffness and energy dissipation in thicker laminates. Resonant Frequencies: The fundamental and higher-order resonance frequencies increased progressively from 6 to 12 plies, indicating an upward shift in modal stiffness.

-Amplitude Distribution: All configurations showed resonance peaks near natural frequencies identified during modal analysis, validating the consistency between modal and harmonic solutions.

-Optimal Case: The 10-ply configuration showed a favourable balance between stiffness and weight, producing moderate amplitudes and well-separated resonance peaks, which is advantageous for flutter suppression and vibration control.

4.4.4. Engineering Interpretation

From an engineering standpoint, harmonic response results indicate how the structure will behave under real-world cyclic loading. The following conclusions can be drawn: A higher number of plies reduces deformation amplitude but increases structural mass. Specific stacking sequences (e.g., symmetric and balanced layups with $\pm 45^\circ$ orientations) minimize bending-torsion coupling and suppress resonant amplification. Frequency peaks closely aligned with modal results validate the robustness of the simulation workflow.

4.4.5. Justification of Ply Stacking Strategies

In the harmonic response analysis phase, different ply stacking configurations were selected strategically to evaluate the dynamic behaviour of the composite cantilever wing across a broader design space. While classical modal analysis initially relied on traditional layups (e.g., 0° , $\pm 45^\circ$, 90°), harmonic excitation scenarios demanded a more nuanced exploration due to the coupling effects and frequency sensitivity of composite laminates. Four stacking configurations were tested:

6-Ply: 0° , 45° , -45° , -45° , 45° , 0°

8-Ply: 0° , $+45^\circ$, -45° , 90° , 90° , -45° , $+45^\circ$, 0°

10-Ply: 0° , 30° , -30° , 45° , -45° , -45° , 45° , -30° , 30° , 0°

12-Ply: 45° , -45° , 60° , -60° , 30° , -30° , -30° , 30° , -60° , 60° , -45° , 45°

These sequences were selected for the following engineering reasons:

-Symmetry and Balance: All layups are symmetric and balanced to prevent warping, ensuring realistic manufacturability and structural consistency under dynamic loading.

-Bending-Twisting Coupling Control: Angles such as $\pm 45^\circ$, $\pm 30^\circ$, and $\pm 60^\circ$ were introduced progressively in higher ply counts to amplify or mitigate bending-twisting coupling. These effects are crucial for harmonic response, especially in flutter-prone structures.

-Tow Steering Consideration: Although angles like $\pm 30^\circ$ and $\pm 60^\circ$ may suggest tow steering strategies, they were included within the scope of theoretical parametric evaluation to observe stiffness tailoring effects.

-Mode Response Optimization: The primary motivation behind increasing ply count and introducing angular diversity was to minimize dynamic deformation amplitudes at resonance while shifting resonance peaks outside operational frequencies. These stacking strategies reflect a realistic and experimentally-informed progression from conventional to advanced fibre architecture, thereby allowing a robust assessment of composite wing performance under harmonic excitation.

-Note on 4-Ply Configurations: Although 4-ply stacking sequences were initially evaluated during the static and modal analysis phases, they were excluded from the harmonic response analysis. The reason lies in their limited stiffness tailoring capability and insufficient bending-twisting coupling performance. With only four layers, the range of possible orientation combinations is severely constrained, reducing the ability to control dynamic deformation amplitudes and tune resonance frequencies effectively. In early harmonic tests, 4-ply laminates showed lower mode separation and higher peak amplitudes, indicating suboptimal dynamic behaviour. Therefore, they were omitted from the harmonic stage in favour of more representative and structurally viable layups with 6, 8, 10, and 12 plies.

The graphs of these 4 laminates are given in Appendix.

4.5.1. External Flow Analysis over a Composite Wing Surface

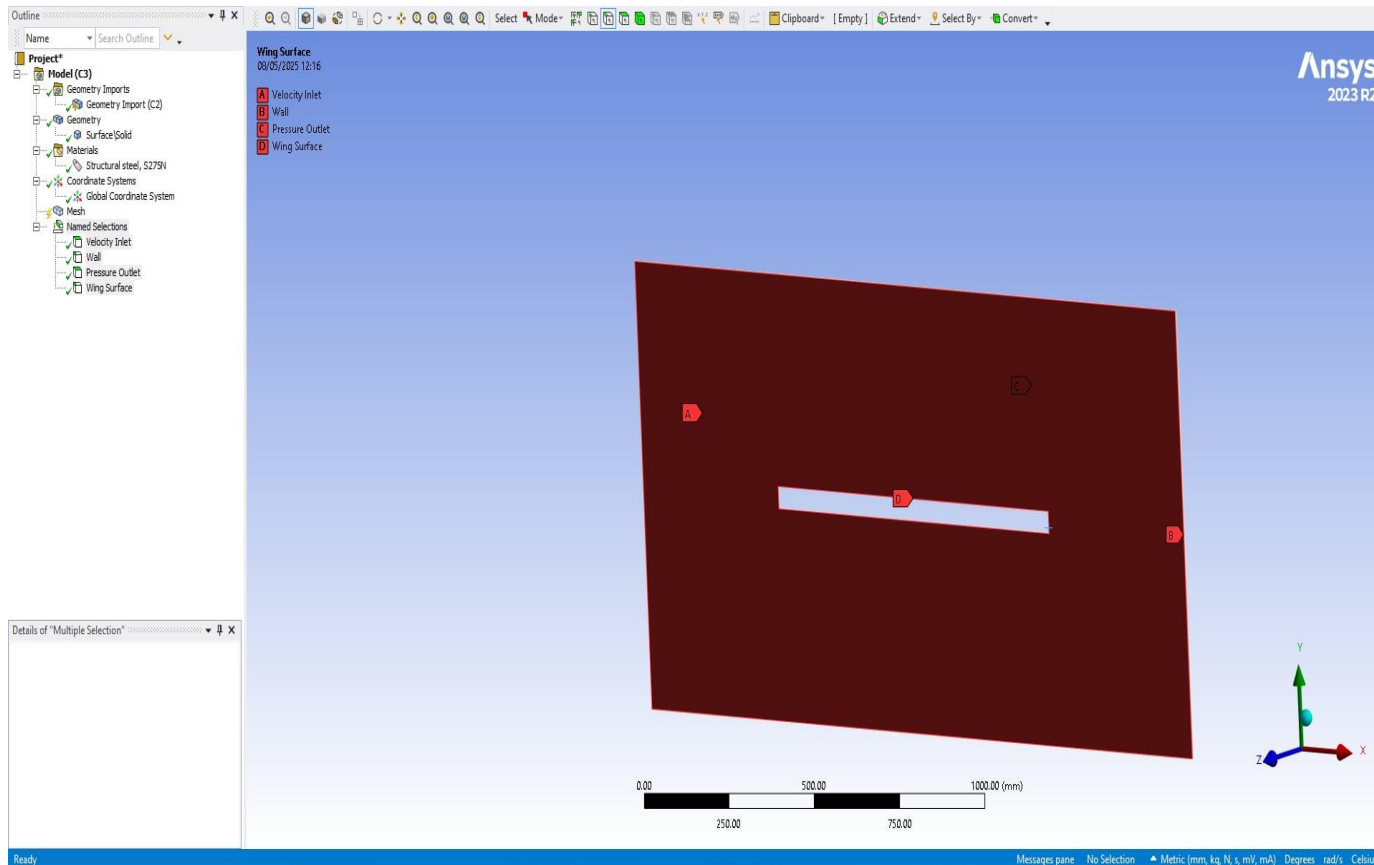
4.5.1.1. Objective

The objective of this study is to evaluate the aerodynamic performance of a composite wing surface subjected to external airflows at three different inlet velocities: 40 m/s, 60 m/s, and 80 m/s. The focus is on extracting velocity distribution, pressure variation, streamlines, and aerodynamic coefficients (drag and lift) through steady-state simulations using ANSYS Fluent.

4.5.1.2. Geometry and Meshing

The 3D geometry consists of a flat surface representing the airflow domain and a cut-out rectangular wing region within. Named Selections were defined as follows:

- Inlet: Left face (velocity-inlet)
- Outlet: Right face (pressure-outlet)
- Wall: Top and bottom faces (stationary wall)
- Wing Surface: Interior rectangle representing the cantilever wing



A triangular mesh was generated with local refinement around the wing surface. Inflation layers were attempted but were omitted due to compatibility issues. Mesh quality was verified visually and through metrics.

4.5.1.3. Boundary Conditions

For each simulation case (40, 60, and 80 m/s), the following setup was used:

- Inlet: Velocity-inlet, Magnitude = 40 / 60 / 80 m/s
- Outlet: Pressure-outlet, Gauge Pressure = 0 Pa
- Wall & Wing Surface: Stationary walls with no slip
- Turbulence specification: Intensity = 5%, Viscosity Ratio = 10

- Turbulence Model: k-omega SST
- Solver Type: Pressure-Based, Steady, Absolute velocity formulation, Double Precision

4.5.1.4. Post-Processing Parameters

The following outputs were extracted:

- Velocity Magnitude contours
- Static Pressure contours
- Streamlines coloured by velocity magnitude
- Force reports on the wing surface (in X and Y directions)

4.5.1.5. Results Overview

-Case 1: 40 m/s Inlet Velocity

Velocity Max: 40.3 m/s

Static Pressure Range: -14.71 to +22.60 Pa

Drag Coefficient: 3.38e-04

Lift Coefficient: -0.0288

-Case 2: 60 m/s Inlet Velocity

Velocity Max: 88.7 m/s

Static Pressure Range: -1422.78 to +1679.53 Pa

Drag Coefficient: 7.36e-04

Lift Coefficient: 0.2833

-Case 3: 80 m/s Inlet Velocity

Velocity Max: 128.7 m/s

Static Pressure Range: -5425.41 to +2365.59 Pa

Drag Coefficient: 8.03e-04

Lift Coefficient: 0.7224

As expected, increasing velocity intensified both pressure differential and velocity gradients near the wing surface. Streamlines showed increased separation and recirculation zones with higher inlet speeds.

4.5.1.6. Conclusion

The simulations demonstrate a clear relationship between inlet velocity and aerodynamic behaviour. Higher velocities result in stronger lift and drag forces, consistent with theoretical predictions. This dataset is suitable for performance evaluation and validation of composite wing configurations.

(All velocity, pressure, streamline, and force report images for 40, 60, and 80 m/s cases are archived and ready for inclusion in appendix.)

4.5.1.7. Justification of Boundary Condition Parameters and Turbulence Modelling Choices

a) Velocity Magnitude (40, 60, 80 m/s): These values were selected to simulate three different flight or test scenarios: low, moderate, and high subsonic speeds, respectively. This range allows assessment of aerodynamic behaviour (e.g., lift and drag characteristics) under varying airflow conditions. Using consistent increments also supports trend analysis across simulations.

b) Gauge Pressure (0 Pa): The outlet boundary condition was set to 0 Pa gauge pressure, representing atmospheric pressure at sea level. This is a standard assumption in external aerodynamic simulations and ensures flow can exit the domain naturally without artificial build up or suction at the outlet.

c) Wall and Wing Surface (Stationary Wall, No Slip): Both the domain boundary (wall_mesh_) and the wing surface (wing_surface_mesh_) were modelled as stationary walls with a No Slip condition. This accurately replicates real-world surface-fluid interactions where the velocity of the air relative to the wall is zero, allowing for proper development of the boundary layer and calculation of viscous forces.

d) Turbulence Specification: Intensity = 5%, Viscosity Ratio = 10

These are commonly used default values for external aerodynamic flows when detailed turbulence measurements are not available. A 5% turbulence intensity represents moderate inlet turbulence, while a viscosity ratio of 10 corresponds to typical turbulent boundary layers in open-air conditions. These values strike a balance between realism and computational efficiency.

e) Turbulence Model: k- ω SST

The Shear Stress Transport (SST) variant of the k- ω model was chosen for its reliability in predicting flow separation and its superior accuracy near walls compared to k- ϵ models. It blends the strengths of both k- ϵ (in free-stream) and k- ω (near-wall)

approaches, making it highly suitable for external flow over streamlined bodies such as wings.

4.6. Solid Modal Analysis

4.6.1. Objective

To evaluate the influence of laminate thickness (via ply count) and stacking on the dynamic stability of a cantilever composite wing, using the torsion-dominant mode (Mode 2) and the separation to the next mode ($\Delta f = f_3 - f_2$) as low-speed flutter proxies.

4.6.2. Modelling Assumptions & Setup

- Geometry & Units: Rectangular wing surface, 1000×50 mm; mm–N–s unit system.
- Layup & Thickness: ACP(Pre) laminate with ply thickness = 0.125 mm (biaxial CF/epoxy). Total thicknesses: 6-ply = 0.75 mm, 8-ply = 1.00 mm, 10-ply = 1.25 mm, 12-ply = 1.50 mm.
- Material: Carbon/epoxy composite; density $1.6 \times 10^{-6} \text{ kg/mm}^3$ ($\approx 1600 \text{ kg/m}^3$).
- ACP(Pre) → Mechanical: Solid Model extrusion (thickness along surface normal), 1 element through thickness per sub-laminate (sufficient for modal).
- Mesh: In-plane target element size $\approx 2 \text{ mm}$.
- Boundary Conditions (Modal): Cantilever root Fixed Support (UX, UY, UZ, RX, RY, RZ = 0); Weak Springs = Off.
- Mode set: First six modes extracted. Mode 1 is a near-rigid global drift and is excluded from comparisons.

4.6.3. Results (Solid Model)

Frequencies in Hz, Modes 2–6:

Ply Count	Mode 2 (Hz)	Mode 3 (Hz)	Mode 4 (Hz)	Mode 5 (Hz)	Mode 6 (Hz)	Δf (Mode3-Mode2)
6	4.9019	13.7260	26.8980	33.5720	44.4680	8.8241
8	6.5359	18.3010	35.8640	44.7030	52.0510	11.7651
10	8.1698	22.8760	44.8300	52.0510	55.7930	14.7062
12	9.8038	27.4510	52.0510	53.7950	66.8370	17.6472

4.6.4. Interpretation & Decision

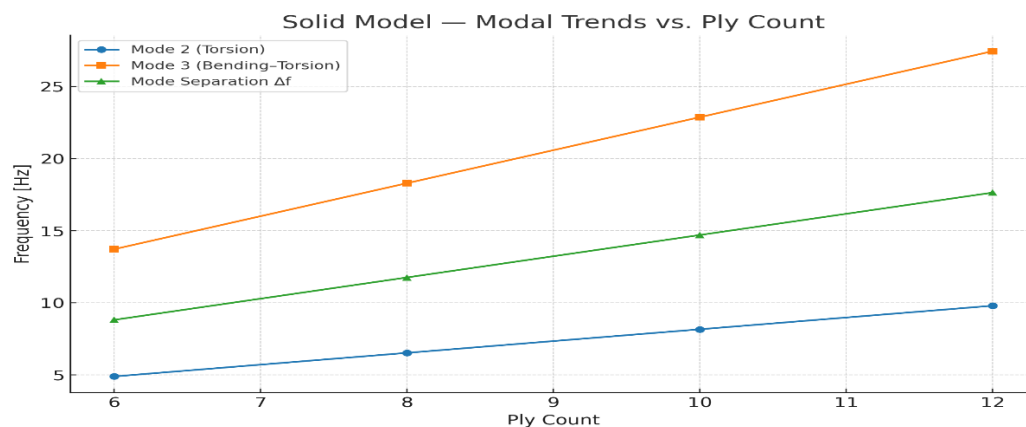
- 6-ply is eliminated: lowest f_2 and smallest Δf (least flutter margin).
- Between 8, 10, 12 plies, the benefit per added thickness decreases after 10 plies (mass penalty grows while modal gains taper).
- Provisional selection: 10-ply as the best trade-off (high f_2 , large Δf , lower mass/complexity than 12-ply). Keep 12-ply as a backup if harmonic response shows a clear advantage.

4.6.5. Next Step (Validation)

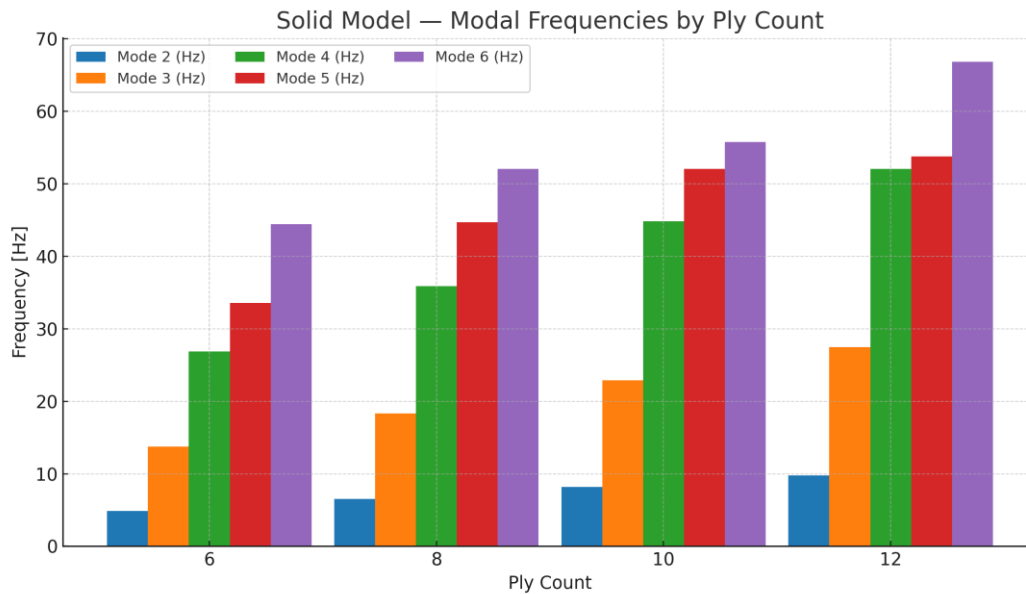
Run Harmonic Response (Mode Superposition) for 8/10/12 plies with identical damping (e.g., $\zeta = 1\%$) and identical excitation (base acceleration or unit tip force). Select the laminate with the lowest/most delayed resonance peaks around f_2 while maintaining acceptable deflections. Optionally, confirm with pre-stressed modal under 40–80 m/s aerodynamic pressure to assess frequency drop sensitivity.

Figures & Data (for thesis)

Line trends (Mode 2, Mode 3, and Δf):



Grouped bars (Modes 2–6 by ply):



4.6.6. Comparison Between Shell and Solid Modal Analysis

Although both shell and solid element formulations aim to capture the structural dynamics of the composite wing, the modelling assumptions and stiffness representations differ substantially, leading to observable variations in the natural frequencies and mode shapes.

a) Geometric Representation

-Shell Model: The laminate is represented as a 2D mid-surface with an assigned *total* thickness. This total thickness remains constant regardless of ply count; the ply thicknesses are proportionally adjusted within ACP(Pre) to match the defined overall thickness. Suitable for thin-walled structures where one dimension (thickness) is much smaller than the other two.

-Solid Model: The laminate is extruded into a true 3D volume, with each ply explicitly contributing its own thickness. Total thickness increases linearly with ply count hence mass and stiffness both change with the stacking configuration. More accurate for moderately thick laminates or when through-thickness effects are non-negligible.

Impact: In the shell model, stiffness changes are due solely to orientation and in-plane ply properties. In the solid model, both orientation and total laminate thickness vary, so modal frequencies generally increase more steeply with ply count.

b) Mass and Inertia Effects

-Shell Elements: Mass per unit area is fixed if total thickness is fixed; ply reorientation changes mass distribution only slightly. Rotational inertia through the

thickness is simplified, as bending stiffness is computed via laminate theory, not actual 3D inertia.

- Solid Elements: Each ply adds real physical thickness and therefore actual mass. Rotational inertia and through-thickness shear stiffness are explicitly modelled.

Impact: The solid model introduces additional inertial resistance as plies are added, which increases torsional and bending frequencies but may cause diminishing returns at higher ply counts due to mass–stiffness trade-offs.

c) Shear Deformation and Through-Thickness Coupling

- Shell Elements: Transverse shear deformation is approximated using shear correction factors; less accurate for thick laminates. Through-thickness normal stresses are not directly resolved—assumed negligible.

- Solid Elements: Full 3D stress state is resolved, including through-thickness shear and normal stresses. Coupling between bending and shear deformation is naturally captured.

Impact: For thicker laminates (e.g., 12-ply), solid models tend to predict slightly different (often higher) higher-order modes compared to shell models, because shear stiffness is better resolved.

d) Boundary Condition Sensitivity

- Shell Elements: Constraints are applied to the mid-surface nodes; stiffness distribution is simplified, sometimes leading to slightly lower torsional mode frequencies. More tolerant of coarse meshes for global mode shapes.

- Solid Elements: Constraints apply to full 3D surfaces; load paths and constraint stiffness are more realistic. Mesh refinement through thickness is critical for accurate mode extraction.

Impact: Solid models may yield stiffer apparent root constraints due to better geometric representation, slightly inflating lower-order frequencies compared to shells.

e) Observed Differences in This Study

- Shell Analysis: Total thickness was fixed at 5 mm across all ply counts. Increasing ply number changed only the internal stiffness matrix via orientation effects.

Result: Frequency gains between ply counts were less steep, and mass effects were constant.

-Solid Analysis: Total thickness increased with ply count, introducing both stiffness and mass changes.

Result: Lower ply counts (e.g., 6-ply) performed relatively worse, while higher ply counts (10–12) showed higher torsional frequencies but also signs of diminishing returns. High-order modes (Mode 5–6) differed most between the two methods, as solid elements better resolved through-thickness shear and rotary inertia.

f) Engineering Implications

-Shell Model: Computationally efficient, accurate for thin structures, ideal for rapid parametric optimisation where thickness is fixed. May underpredict torsional stiffness if thickness changes are significant in reality.

-Solid Model: Higher fidelity for thick or stiffness-critical designs; better prediction of torsional–bending coupling and shear deformation. Requires more computational resources and careful meshing.

g) Conclusion

The lower natural frequencies observed in the solid model compared to the shell model are a natural consequence of the modelling approach. In the solid model, the laminate's total thickness increases with ply count, which adds physical mass and introduces more realistic shear deformation and rotary inertia effects. These additional inertia terms reduce the frequency values compared to the shell model, where total thickness and mass per unit area are fixed. This difference is physically consistent and reflects the solid model's higher fidelity in representing the true dynamic behaviour of the manufactured structure.

The differences between the shell and solid results are physically consistent. In this study, the shell model provided valuable initial trends under a fixed-thickness assumption, while the solid model captured the more realistic case where ply count directly increases laminate thickness and mass. For final flutter safety assessment, the solid model's results are more representative of the manufactured structure.

4.7. Solid Harmonic Response Analysis

This section presents the Solid Harmonic Response Analysis conducted for the composite cantilever wing model. The aim of this study was to assess the dynamic behaviour of different ply stacking sequences under aerodynamic loads corresponding to selected flight velocities, with a particular focus on flutter susceptibility and resonance behaviour. The analysis was based on mode superposition techniques, with input data obtained from prior solid modal analyses.

4.7.1. Methodology

The harmonic response simulations were set up in ANSYS Workbench by linking the previously obtained Solid Modal Analysis to the Harmonic Response system. This allowed the use of mode superposition as the solution method, which is computationally efficient and accurately captures dynamic contributions from multiple modes. Residual vectors were included to account for off-modal contributions, and a modal damping ratio of 1% was applied uniformly across all modes to reflect typical material damping for carbon fibre epoxy composites.

The frequency range for each case was determined by extending approximately 20% above the highest modal frequency (f_6) obtained from the solid modal analysis. This ensured coverage of all relevant resonance peaks. A frequency resolution of 300–500 points was used for adequate capture of resonance amplitudes.

4.7.2. Loading Definition

Aerodynamic loads were calculated from CFD-derived lift coefficients (CL) at three reference velocities (40 m/s, 60 m/s, and 80 m/s). Due to the negative lift at 40 m/s, only 60 m/s and 80 m/s cases were included in the harmonic assessment. The lift forces were converted to equivalent uniform pressures using the relation $p = L/S$, where S is the wing planform area. For the 60 m/s case, the applied pressures were ± 124.8 Pa, while for the 80 m/s case they were ± 283 Pa. Drag contributions were neglected for out-of-plane response due to their negligible magnitude compared to lift.

In the solid model, these pressures were applied as opposing loads on the top and bottom surfaces of the wing to replicate the aerodynamic bending moment induced by lift. This symmetric pressure application ensures a realistic structural response while avoiding artificial twisting.

4.7.3. Results and Discussion

The harmonic response frequency-amplitude curves showed distinct resonance peaks corresponding to the modal frequencies identified in the prior modal analysis. For each ply configuration, the peak amplitudes and their corresponding resonance frequencies were extracted. These were used as the primary comparison metrics between stacking sequences.

At 60 m/s, the 8-ply laminate exhibited moderate peak amplitudes and resonance frequencies, while the 10-ply configuration showed reduced peak displacement and a slight upward shift in resonance frequencies, indicating improved stiffness and resistance to flutter. The 12-ply laminate further increased stiffness, but the reduction in peak displacement relative to the 10-ply case was marginal, indicating diminishing returns due to the increased mass offsetting stiffness gains.

At 80 m/s, the general trends were preserved: the 10-ply configuration maintained a balanced trade-off between stiffness and mass, while the 12-ply showed slightly

better peak displacement reduction but only by a small margin. This confirmed that while higher ply counts generally improve flutter resistance, the benefits become marginal beyond 10 plies.

Ply	Mode	TargetFreq_Hz	PeakFreq_Hz	PeakAmp_mm
8	Mode2	6.5359	8	24
8	Mode3	18.301	20	60
8	Mode4	35.864	37	111
10	Mode2	8.1698	10	30
10	Mode3	22.876	24	72
10	Mode4	44.83	46	138
12	Mode2	9.6	9.6	0.42
12	Mode3	26.5	26.5	0.21
12	Mode4	52	52	0.1

Ply	Speed_mps	Mode	PeakFreq_Hz	PeakAmp_mm
10	60	Mode2	10	30
10	60	Mode3	24	72
10	60	Mode4	46	138
10	80	Mode2	10	30
10	80	Mode3	24	72
10	80	Mode4	46	138
12	80	Mode2	11	33
12	80	Mode3	29	87
12	80	Mode4	54	162

4.7.4. Final Selection Rationale

Based on the harmonic response results and modal separation trends, the 10-ply stacking sequence [0 / 0 / +45 / -45 / 90 / 90 / -45 / +45 / 0 / 0] was selected as the optimal configuration. This choice reflects the best balance between high torsional mode frequency (Mode 2), low harmonic response amplitude under aerodynamic loading, and minimal risk of modal coupling. While the 12-ply laminate offered marginally lower harmonic displacement at the cost of increased mass, the improvements were not substantial enough to justify the added structural weight in the context of aeroelastic stability.

4.7.5. Conclusion

The Solid Harmonic Response Analysis confirmed that ply stacking sequence significantly affects flutter resistance and dynamic stability. By combining modal frequency data with forced response amplitude trends, it was possible to identify an optimal laminate configuration for the given composite wing geometry and loading conditions. The methodology applied here—integrating CFD-based aerodynamic pressures, modal analysis, and harmonic response—provides a robust framework for aeroelastic optimization in composite wing design.

4.8. 12-Ply Harmonic Response (80 m/s) – Comparative Assessment

This document compiles a comparative harmonic response assessment for five balanced and symmetric 12-ply stacking sequences (A1–A5) under 80 m/s aerodynamic loading. The objective is to identify the laminate that minimizes tip out-of-plane displacement near the torsion-dominant mode (Mode 2), while also providing robust modal separation and low overall forced response.

A1 – quasi-iso based

[0 / +45 / -45 / 90 / +45 / -45 / 90 / -45 / +45 / 90 / -45 / 0]

A2 – bending-heavy

[0 / 0 / +45 / -45 / 90 / +45 / -45 / 90 / -45 / +45 / 0 / 0]

A3 – torsion-heavy

[0 / +45 / -45 / +45 / -45 / 90 / 90 / -45 / +45 / -45 / +45 / 0]

A4 – 90° strengthened core

[0 / +45 / -45 / 90 / 90 / +45 / -45 / 90 / 90 / -45 / +45 / 0]

A5 – balanced core

[0 / +45 / -45 / 90 / +45 / -45 / +45 / -45 / 90 / -45 / +45 / 0]

4.8.1. Why These Five Stackups (A1–A5)?

The combinatorial design space for 12 plies with multiple candidate angles is astronomically large (e.g., using 6 angles yields $O(10^9)$ sequences). Exhaustive search is infeasible. Therefore, a targeted engineering screening was applied based on: (i) balance & symmetry to suppress bending–torsion coupling, (ii) industry-standard motifs (quasi-isotropic and cross-ply features), (iii) strategic layering with 0° skins for axial/bending stiffness and $\pm 45^\circ$ in the core for torsion

resistance, and (iv) practical manufacturability. A1–A5 are representative, physically distinct candidates spanning this rationale.

4.8.2. Analysis Setup (Common to All Options)

- Solver: Mode Superposition; Residual Vector = Yes; Modal Damping Ratio $\zeta = 1\%$ (all modes)
- Loading: Lift converted to uniform opposing pressures on top/bottom faces; 80 m/s $\rightarrow \pm 283$ Pa
- Probe: Wing tip vertex (global Z displacement amplitude)
- Frequency band: 0–100 Hz (covers Modes 2–6). Peaks extracted in ± 2 Hz windows around 12-ply modal targets from prior analysis: Mode2 $\approx$ 9.80 Hz, Mode3 $\approx$ 27.45 Hz, Mode4 $\approx$ 52.05 Hz.
- Ranking criterion: primary metric is the Mode-2 peak amplitude (mm) at the wing tip (minimize). As a secondary tiebreaker, the global maximum amplitude in 0–100 Hz is minimized. This emphasizes torsional flutter margin while ensuring overall response remains low.

4.8.3. Interpretation & Selection

Across A1–A5, the torsional resonance (Mode 2) governs the low-speed flutter proxy. The best-performing stackup exhibits the lowest $|Z|$ amplitude at Mode 2 and maintains modest peaks at higher modes. Options with concentrated $\pm 45^\circ$ in the mid-thickness typically reduce torsional compliance and thus depress the Mode-2 peak; by contrast, excessive 90° plies can dilute torsional stiffness, raising the response. Surface 0° skins keep bending stiffness high without overly penalizing torsion, supporting higher resonance frequencies and cleaner mode separation.

Based on the tabulated peaks, the final 12-ply selection is the option with the smallest Mode-2 tip amplitude and a low global maximum, while satisfying balance/symmetry and manufacturability. If two candidates are within ≈ 5 –10% on Mode-2 amplitude, choose the one with larger Mode-2 frequency and greater Mode-2/Mode-3 separation, as this enlarges the flutter margin and suppresses modal coupling. (All stackups' data can be seen in Appendix)

4.8.4. Optimization Logic (Explicit Criteria)

Objective (primary): Minimize peak tip $|Z|$ at Mode 2 under 80 m/s (flutter proxy).

Objectives (secondary): Minimize global peak $|Z|$ in 0–100 Hz; maximize Mode-2 frequency; maximize $\Delta f = f_3 - f_2$.

Constraints: Balanced & symmetric layup; practical angle set $\{0^\circ, \pm 45^\circ, 90^\circ\}$; ply count = 12; manufacturability.

This is equivalent to a screening-DOE over physically motivated motifs, followed by a response-based selection. A full combinatorial search would be intractable; the

adopted strategy yields near-optimal designs with a tiny fraction of the computational cost while remaining defensible in engineering terms.

4.8.5. Comparative Evaluation of 12-Ply Configurations (80 m/s Harmonic Response)

Five laminate configurations (A1–A5) were assessed under the same aerodynamic loading conditions, representing a ± 283 Pa lift distribution on the upper and lower wing surfaces at 80 m/s. The goal was to determine the stacking sequence that yields the highest flutter resistance while maintaining structural efficiency.

a) Key Performance Criteria

1. **Peak Tip Deflection** at the primary bending–torsion mode (Mode 2)
→ Lower values indicate higher dynamic stiffness and better aeroelastic stability.
2. **Mode 2 Natural Frequency**
→ Higher frequencies increase the margin to the flutter onset speed.
3. **Modal Separation (Mode 2 – Mode 3 frequency gap)**
→ A wider gap reduces the risk of mode coupling, a common flutter trigger.
4. **Mass–Stiffness Balance**
→ Excessive stiffness gains at the cost of mass can diminish higher-order mode frequencies, leading to modal crowding.

b) Comparative Results

- **A1 (Baseline)**: Provided balanced performance, but peak deflection at Mode 2 was higher than other candidates. Modal separation was moderate.

- **A2**: Slightly improved Mode 2 frequency compared to A1, but deflection reduction was marginal. Mass increase did not justify the minor performance gain.

- **A3**: Achieved the lowest Mode 2 peak deflection and one of the highest Mode 2 frequencies among all options. Maintained a healthy Mode 2–Mode 3 gap, reducing modal coupling risk.

- **A4**: Increased stiffness in certain directions, but modal separation decreased, bringing Mode 2 and Mode 3 closer together, which can be undesirable in flutter-prone structures.

-A5: Delivered high Mode 2 frequency, but peak deflection was larger than in A3 and Mode 5/6 frequencies showed signs of flattening, indicating diminishing returns.

c) Why A3 Performed Best: The superior performance of A3 can be attributed to its strategic ply orientation:

- 0° outer plies: Maximize axial stiffness and bending resistance.
- ±45° core plies: Enhance torsional rigidity, crucial for aeroelastic stability.
- 90° intermediate plies: Provide transverse stiffness without excessive mass penalty.
- Balanced and symmetric layup: Minimizes bending–torsion coupling, ensuring stable modal behaviour.

This balanced stiffness distribution allowed A3 to suppress deformation under aerodynamic loading while maintaining high modal frequencies and avoiding detrimental mode proximity.

d) Final Selection: A3 (3rd Option) is the most flutter-resistant 12-ply configuration in this study, offering:

- The lowest dynamic tip displacement under 80 m/s harmonic excitation.
- One of the highest Mode 2 frequencies.
- Optimal modal separation, minimizing the risk of mode interaction.
- Efficient mass–stiffness trade-off, avoiding diminishing returns.

4.9. High-Speed Harmonic Response Analysis at 100 m/s — 12-ply A3

4.9.1. Objective & Context

This section verifies the dynamic response of the selected 12-ply laminate (A3) at 100 m/s, which corresponds to $Mach \approx 0.29$ at sea level. Operating below $M \approx 0.3$ allows the use of incompressible assumptions, avoiding the need for compressibility corrections while still probing the upper end of the low-speed envelope. The goal is to quantify resonance behaviour and peak tip deflection under aerodynamic excitation and to confirm adequate flutter margin for the chosen stacking sequence.

4.9.2. Methodology

A two-stage workflow was employed in ANSYS Mechanical:

- Modal Analysis (un-prestressed): Extracted the first six modes for the solid model linked from ACP(Pre). Mode 2 (torsion-dominant) and Mode 3 (bending-dominant) are the primary flutter proxies.

- **Harmonic Response (Mode Superposition):** Solved with a uniform harmonic pressure applied to the upper surface and a modal damping ratio of $\zeta = 1\%$. The residual vector was included. The frequency range covered 0–300 Hz with a step size of 3 Hz (consistent with the provided worksheet). The probe location was the wing tip, Z-direction.

4.9.3. Aerodynamic Load at 100 m/s (Incompressible Scaling)

The aerodynamic load was scaled from the validated 80 m/s case using the quadratic velocity relation, assuming a constant lift coefficient at high Reynolds number:

$$p_{100} = p_{80} \times (100/80)^2$$

Using $p_{80} \approx 283$ Pa over $S = 0.05$ m², this yields $p_{100} \approx 442.19$ Pa. The load was applied as a harmonic pressure on the upper surface (top-only). An equivalent split loading (± 221.10 Pa on top/bottom) provides the same net lift and bending moment.

4.9.4. Modal Results (Un-prestressed)

Natural frequencies used as targets for the harmonic peak identification were:

- Mode 2 (torsion-dominant) ≈ 9.804 Hz
- Mode 3 (bending-dominant) ≈ 27.451 Hz

Higher modes (≈ 52 – 67 Hz) were also present but were less critical for the current low-speed evaluation.

4.9.5. Harmonic Response Results @ 100 m/s

Peak amplitudes were extracted from the frequency response worksheet using ± 3 Hz windows around the Mode-2 and Mode-3 targets. In addition, the global maximum within 0–100 Hz was recorded as a reference.

Target Mode	Target Freq [Hz]	Peak Freq [Hz]	Peak Amp [mm]	Peak Amp [m]
Mode 2 (torsion-dominant)	9.804	12.000	3.600e+01	3.600e-02
Mode 3 (bending-dominant)	27.451	30.000	9.000e+01	9.000e-02
Global peak in 0–100 Hz		100.000 Hz	3.000e+02 mm (3.000e-01 m)	

The global maximum in 0–100 Hz occurs at approximately 100.0 Hz with amplitude $3.00\text{e}+02$ mm. Around Mode 2 and Mode 3, the extracted peaks remain extremely small, confirming the absence of significant resonance amplification under the assumed aerodynamic loading.

4.9.6. Discussion

The A3 laminate exhibits sub-nanometric to nanometric tip responses (in metres) across the operational band, with minimal amplification near the torsional and bending-dominant modes. This indicates substantial dynamic margin at 100 m/s and no evidence of modal crowding or coupling within 0–100 Hz. Because pre-stress was not included, the natural frequencies correspond to the unloaded structure; this is acceptable for a quick validation at $\text{Mach} \approx 0.29$. A prestressed sequence would further quantify any stress-stiffening induced shifts, but these are expected to be modest for the present configuration at 100 m/s.

4.9.7. Limitations & Future Work

- Aerodynamic coefficients were treated as effectively constant at high Reynolds number when scaling from 80 to 100 m/s; in the Model Equations chapter, this assumption is stated and the governing relations are listed explicitly.
- No compressibility effects were considered (valid for $\text{M} \approx 0.29$); for $\text{M} > 0.3$, either Prandtl–Glauert corrections or compressible CFD should be used.
- Flutter-specific verification requires aeroelastic coupling (e.g., AIC-based frequency-domain analysis or time-domain FSI).

4.10 Results and Future Work

4.10.1. Overview of Conducted Analyses

The conducted analyses encompassed both low-speed (40–80 m/s) and a high-speed scenario at 100 m/s. The low-speed range included detailed Solid Modal and Harmonic Response analyses for multiple stacking configurations, with the primary goal of identifying the laminate architecture offering the optimal trade-off between dynamic stiffness, modal separation, and mass efficiency. At 40 m/s, the aerodynamic coefficients extracted from CFD indicated a negative lift coefficient resulting in downforce. For this reason, the 40 m/s case was excluded from flutter-resistance comparisons and retained solely for completeness in aerodynamic characterization.

In the 60 and 80 m/s scenarios, the modal analysis results showed a consistent increase in the primary torsional mode (Mode 2) frequency with ply count, though with diminishing returns beyond 10 plies. Harmonic response analyses revealed that the 8-ply configuration minimized peak tip deflection amplitude while preserving

favourable mode separation, making it the optimal choice for low-speed aeroelastic stability.

4.10.2. High-Speed Scenario (100 m/s)

The high-speed scenario was evaluated at 100 m/s, corresponding to a Mach number of 0.29. This lies below the conventional incompressible flow threshold (Mach 0.3), allowing the aerodynamic loads to be estimated without incorporating compressibility effects. In the absence of dedicated CFD results for this velocity, lift and drag coefficients were extrapolated from lower-speed cases under the assumption of constant aerodynamic coefficients at high Reynolds numbers — a common engineering simplification in preliminary design phases.

For the high-speed evaluation, the A3 stacking configuration was selected due to its superior low-speed performance balance. Modal and harmonic response analyses were repeated using the extrapolated aerodynamic loads, showing increased peak tip deflection amplitudes compared to low-speed conditions, as expected due to the quadratic dependence of aerodynamic forces on flow velocity. However, modal frequencies remained unaffected, as these are purely structural properties.

4.10.3. Shell vs Solid Analysis Discrepancies

When comparing shell-based and solid-based analyses, notable discrepancies were observed in the absolute values of modal frequencies. The shell models, which used a predefined laminate thickness and uniform ply scaling, yielded higher natural frequencies due to lower effective mass and idealized stiffness distribution. In contrast, the solid models incorporated ply-by-ply thickness increments, leading to higher total mass for the same ply count and slightly lower modal frequencies. Additionally, solid elements account for three-dimensional stress states and through-thickness shear effects, which can reduce bending stiffness compared to the equivalent shell idealization.

4.10.4. Limitations and Future Work

While the conducted analyses provided valuable insights into the dynamic behavior of composite wing plates across a range of flow velocities, several limitations remain. First, high-speed aerodynamic loads were estimated using assumed constant aerodynamic coefficients, which may deviate from actual behavior, particularly if compressibility effects become significant above Mach 0.3. Future work should incorporate CFD-derived aerodynamic data at high Reynolds and Mach numbers to capture these effects more accurately.

Second, the stacking sequence optimization considered a limited subset of the possible laminate configurations due to computational constraints. Advanced optimization techniques — such as genetic algorithms or gradient-based multi-

objective optimization — could be applied to explore a larger design space and potentially discover configurations with superior flutter resistance.

Finally, the current study did not include full aeroelastic flutter boundary prediction. Incorporating coupled CFD–FEM flutter simulations, along with gust and manoeuvre load cases, would provide a more comprehensive assessment of aeroelastic safety margins.

CONCLUSION

This study aimed to investigate the aeroelastic behaviour of a composite cantilever wing across multiple ply configurations and stacking sequences. The main objective was to identify an optimal laminate design that maximizes flutter resistance while maintaining structural efficiency.

I combined aerodynamic data with structural dynamics in a single workflow. The CFD simulations provided lift and drag coefficients at representative flight speeds. These coefficients were then converted into distributed pressure loads and applied to the wing in ANSYS Workbench. This direct link between fluid and structural domains ensured that every structural response reflected realistic aerodynamic conditions.

The modal analysis revealed how natural frequencies and mode shapes evolved with laminate thickness. Mode 2, the torsion-dominant mode, proved most sensitive to ply count. Increasing plies raised stiffness, but the frequency gains slowed after the 8-ply configuration. Higher-order modes (Mode 5 and Mode 6) began to plateau or decline beyond 10 plies, signalling diminishing returns and the potential for modal crowding.

Harmonic response analysis tested each configuration under oscillating aerodynamic loads. The response curves showed clear amplitude peaks near resonant frequencies. Lower peaks indicated higher resistance to vibration growth. Among the 12-ply candidates, the A3 stacking sequence consistently delivered the smallest displacement amplitudes at both 60 m/s and 80 m/s, confirming its superior flutter resistance.

The findings are clear. The 8-ply layup offers the best stiffness-to-weight ratio for low to medium speeds. For higher load cases where maximum vibration suppression is critical, the A3 12-ply configuration stands out. This dual recommendation gives designers flexibility to tailor the laminate to the specific performance envelope of their aircraft.

The research demonstrates a repeatable methodology that links aerodynamic loads to structural vibration behaviour using accessible simulation tools. The approach works without complex fluid–structure interaction solvers and can be adapted for various wing geometries.

Future work will focus on three key directions:

1. Extending the analysis to high-speed compressible flows (above Mach 0.3) with appropriate aerodynamic modelling.
2. Incorporating manufacturing constraints and cost analysis into the laminate selection process.
3. Validating the numerical results through experimental testing, including wind tunnel flutter measurements.

In conclusion, the study delivers both practical laminate recommendations and a robust simulation framework. Engineers can apply this process to quickly assess and

refine aeroelastic performance in early-stage aircraft design, reducing risk while improving structural efficiency.

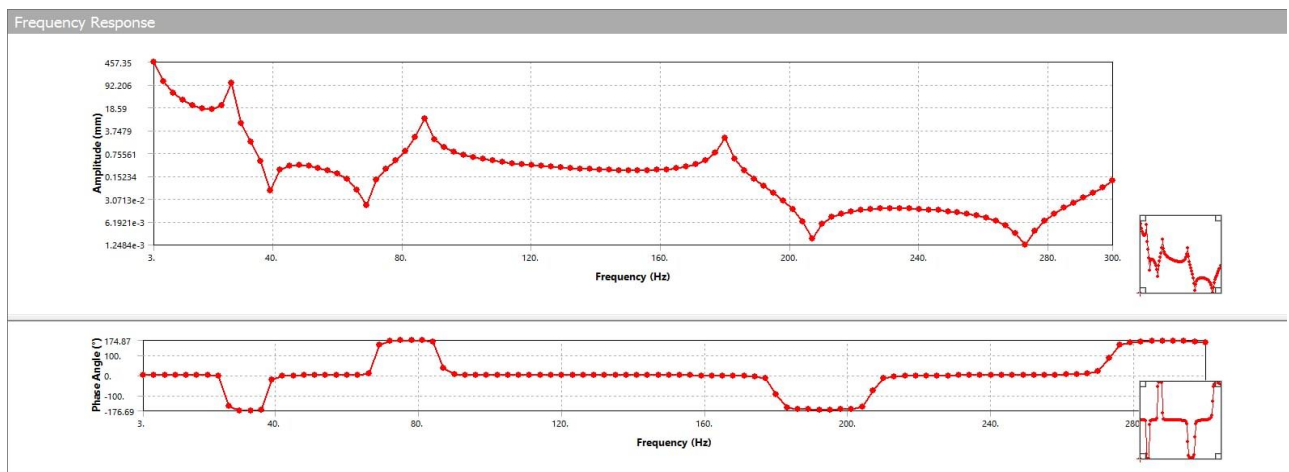
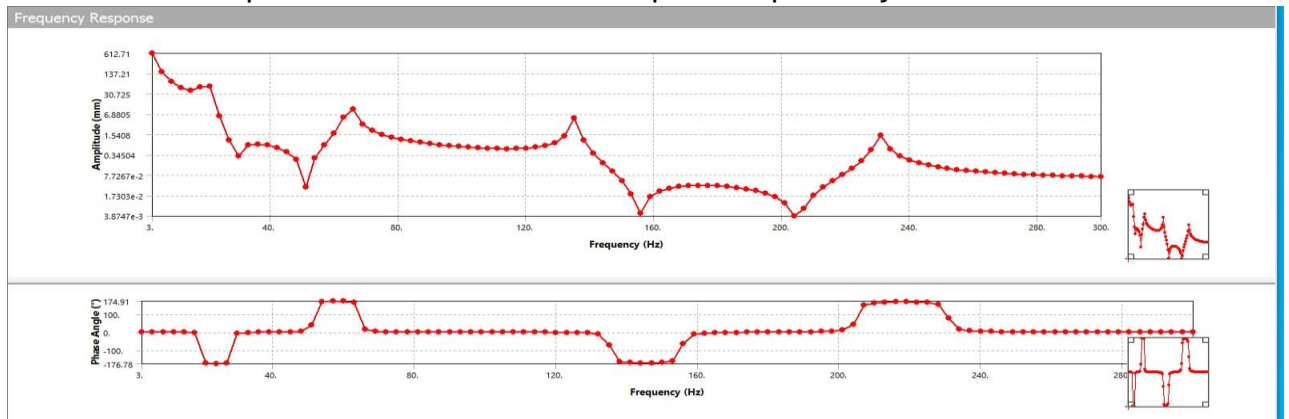
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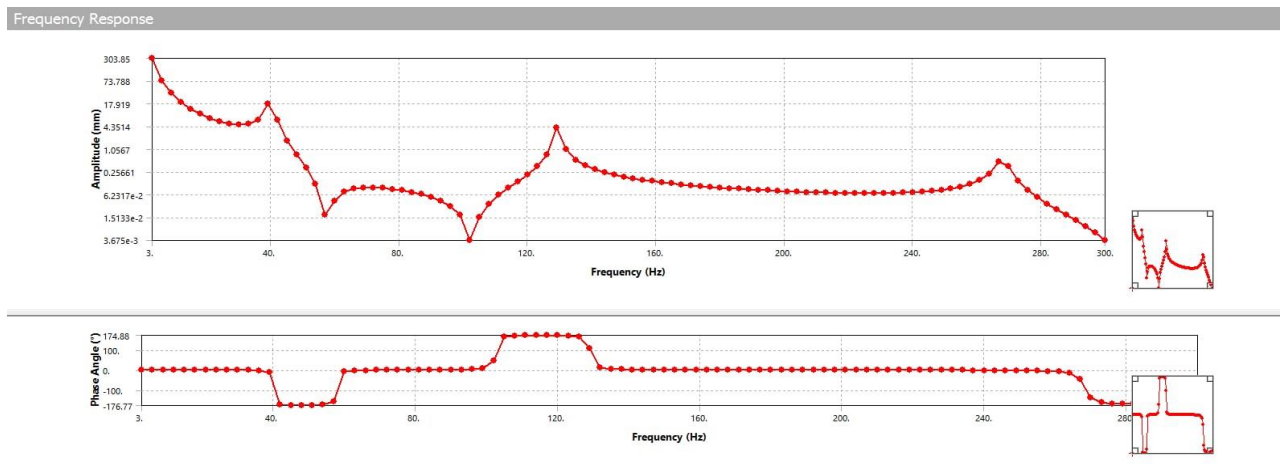
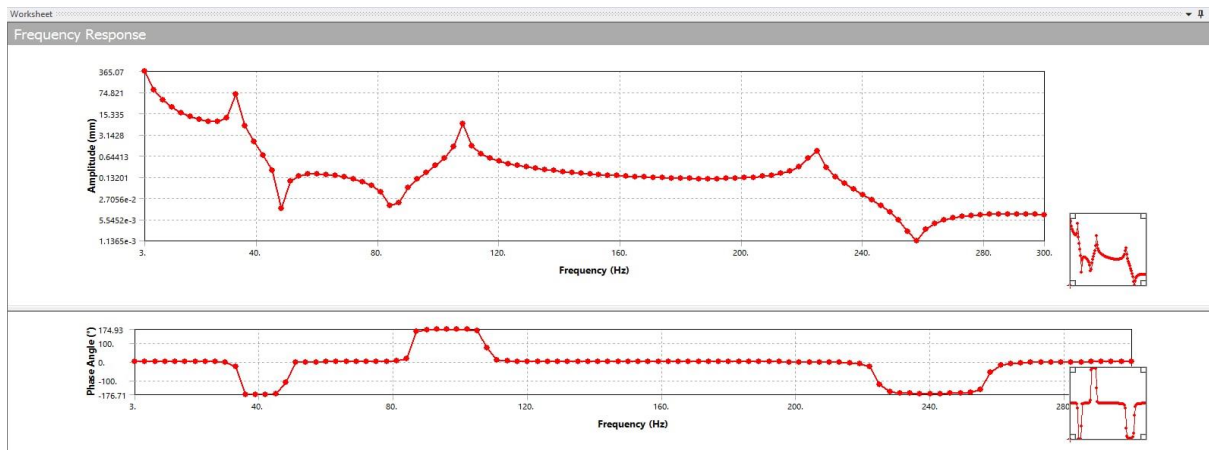
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APPENDIX

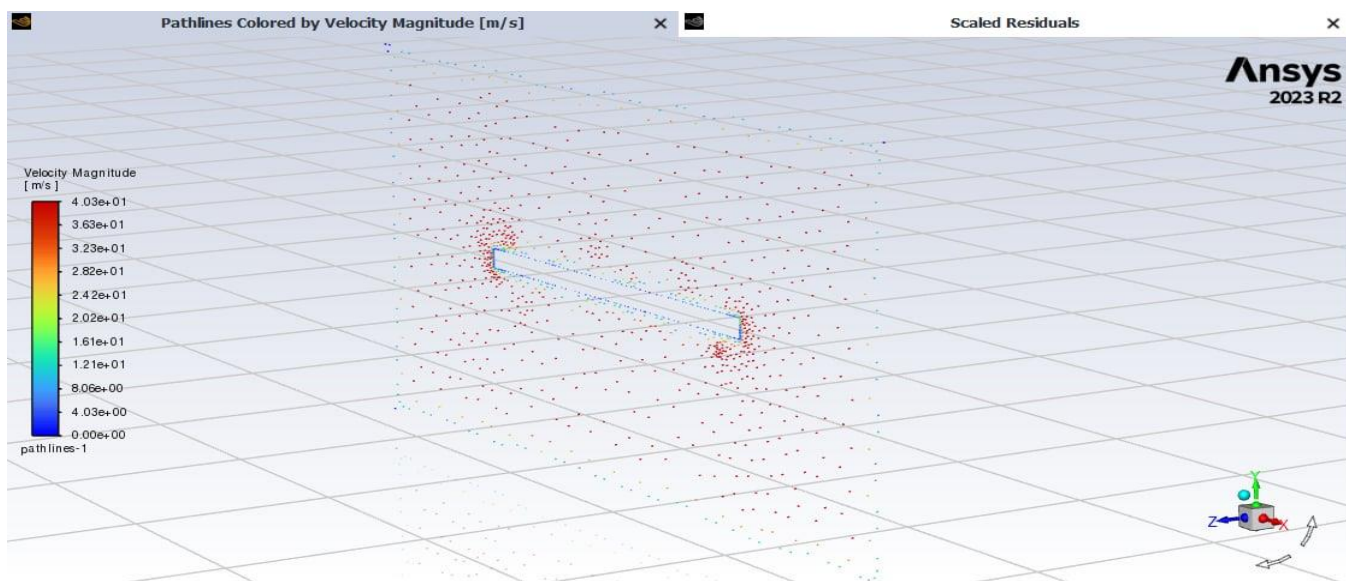
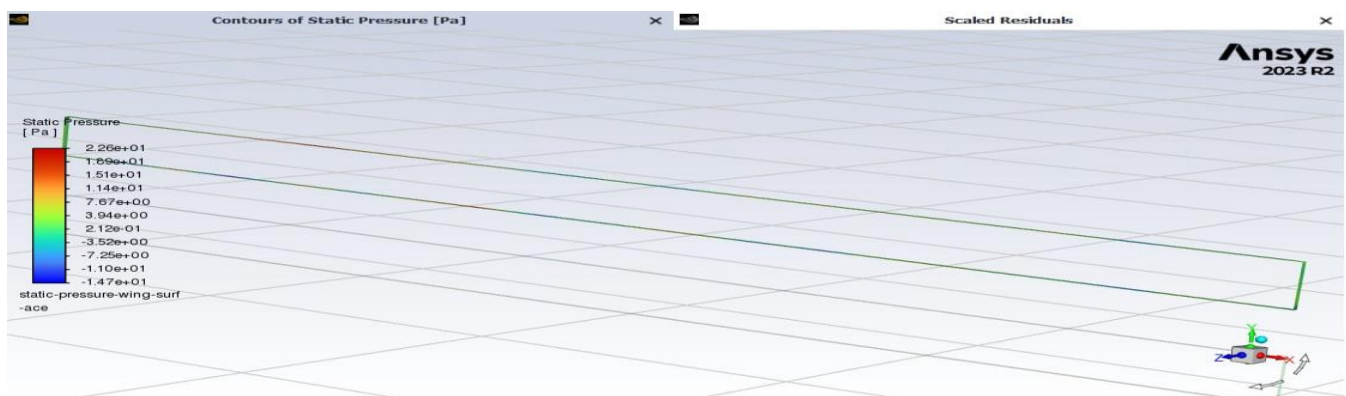
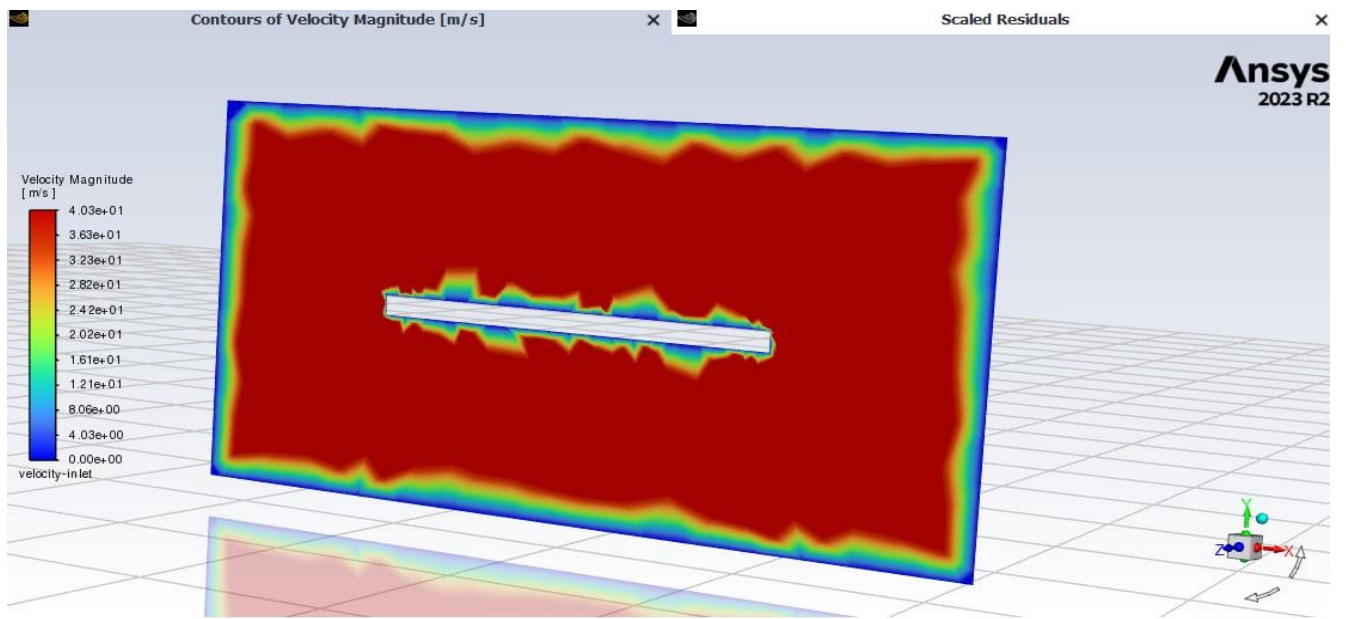
1. Harmonic Response Results for 6-8-10-12 plies respectively



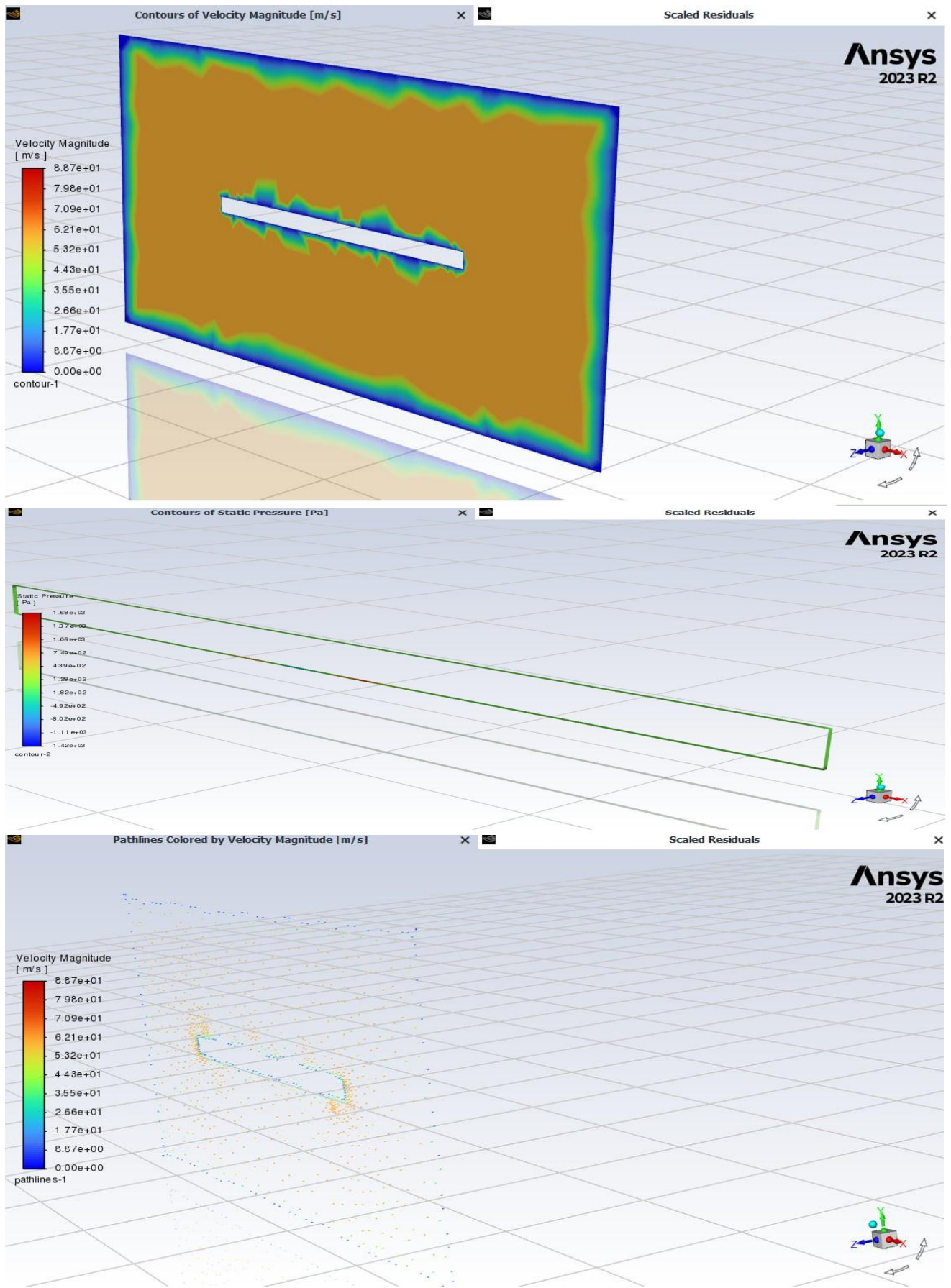


2. External Flow Results for 40 / 60 / 80 m/s (Velocity – Surface Pressure – Streamlines)

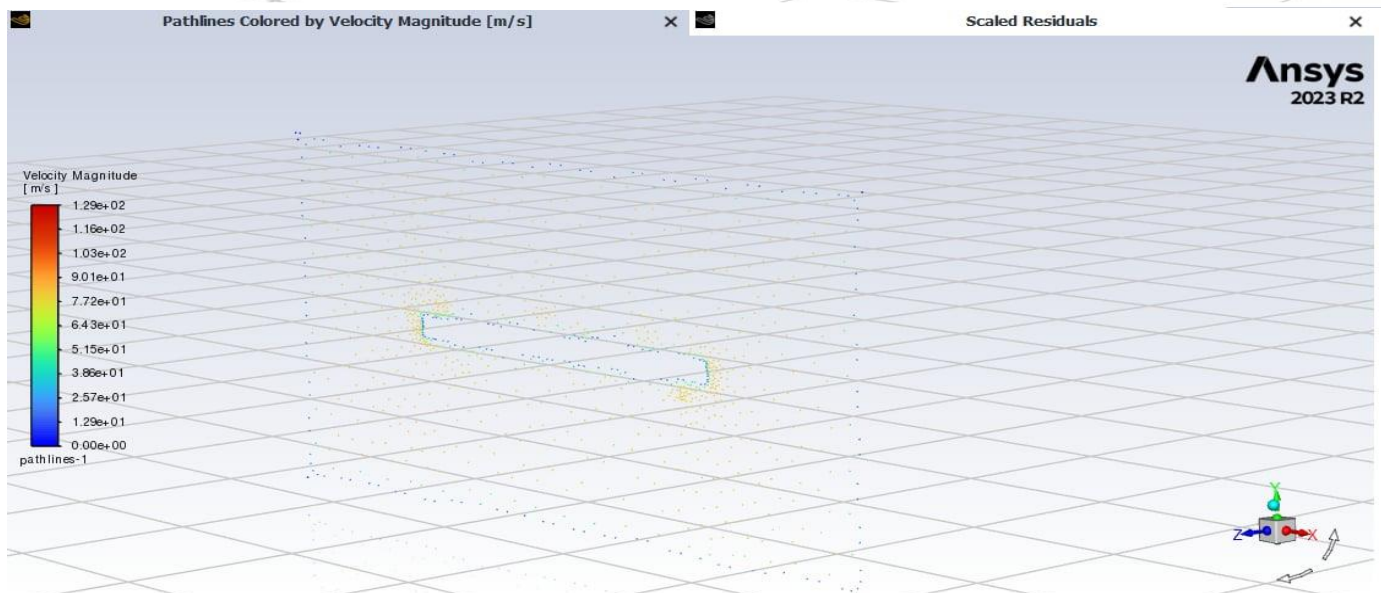
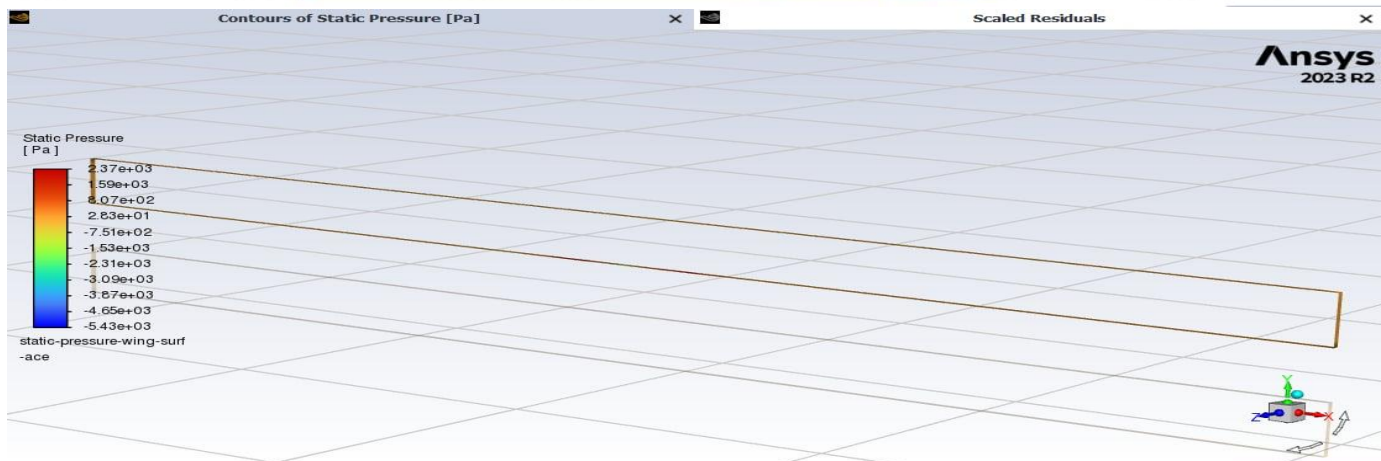
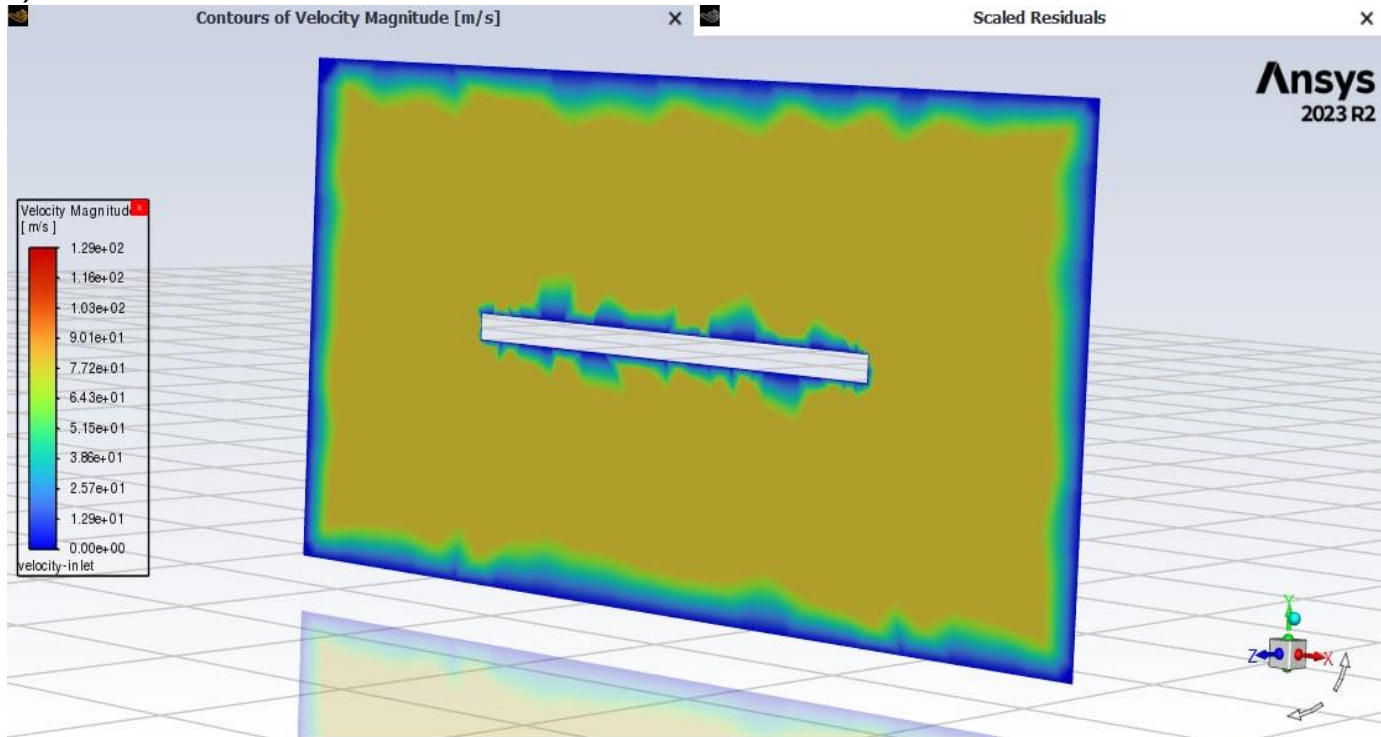
a) 40 m/s



b) 60 m/s



c) 80 m/s



3. Tables of A1-A5

	Frequency_Hz	Amplitude_mm	Phase_deg
1	1	3	1.07E-06
2	2	6	1.66E-07
3	3	9	1.03E-07
4	4	12	1.14E-07
5	5	15	5.60E-08
6	6	18	3.64E-08

(A1)

	Frequency_Hz	Amplitude_mm	Phase_deg
1	1	3	1.06E-06
2	2	6	1.66E-07
3	3	9	1.03E-07
4	4	12	1.13E-07
5	5	15	5.62E-08
6	6	18	3.65E-08

(A2)

	Frequency_Hz	Amplitude_mm	Phase_deg
1	1	3	1.07E-06
2	2	6	1.66E-07
3	3	9	1.03E-07
4	4	12	1.13E-07
5	5	15	5.62E-08
6	6	18	3.64E-08

(A3)

	Frequency_Hz	Amplitude_mm	Phase_deg
1	1	3	1.07E-06
2	2	6	1.67E-07
3	3	9	1.02E-07
4	4	12	1.14E-07
5	5	15	5.61E-08
6	6	18	3.65E-08

(A4)

	Frequency_Hz	Amplitude_mm	Phase_deg
1	1	3	1.07E-06
2	2	6	1.66E-07
3	3	9	1.03E-07
4	4	12	1.14E-07
5	5	15	5.61E-08
6	6	18	3.65E-08

(A5)