EFFECT OF STRUCTURAL DYNAMIC CHARACTERISTICS ON DAMAGE TOLERANCE OF CARBON-FIBRE SANDWICH PANELS

Thesis submitted to the Institute of Space Technology in partial fulfillment of the requirements for the degree of Doctorate of Philosophy in Mechanical Engineering

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August 2018
Effect of Structural Dynamic Characteristics on Damage Tolerance of Carbon-Fibre Sandwich Panels

By

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APPROVAL BY BOARD OF EXAMINERS
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DEDICATION

The work is dedicated to my family and friends. There are number of people without whom this thesis might not have been completed and to whom I am greatly obliged.
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<th>Description</th>
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<tr>
<td>AA</td>
<td>Aluminum Alloy</td>
</tr>
<tr>
<td>AFDL</td>
<td>Air Force Flight Dynamics Laboratory</td>
</tr>
<tr>
<td>ASIP</td>
<td>Aircraft Structural Integrity Program</td>
</tr>
<tr>
<td>ASTM</td>
<td>American Society for Testing and Materials</td>
</tr>
<tr>
<td>AS</td>
<td>Aerospace Standard</td>
</tr>
<tr>
<td>BGC</td>
<td>Breaking Glued Contact</td>
</tr>
<tr>
<td>BVID</td>
<td>Barely Visible Impact Damage</td>
</tr>
<tr>
<td>CC</td>
<td>Compliance Calibration</td>
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<tr>
<td>CFC</td>
<td>Carbon Fibre Composites</td>
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<tr>
<td>CG</td>
<td>Centre of Gravity</td>
</tr>
<tr>
<td>CLS</td>
<td>Cracked Lap Shear</td>
</tr>
<tr>
<td>CZM</td>
<td>Cohesive Zone Modeling</td>
</tr>
<tr>
<td>CTOL</td>
<td>Conventional Take-off and Land</td>
</tr>
<tr>
<td>DTA</td>
<td>Damage Tolerance Analysis</td>
</tr>
<tr>
<td>DCB</td>
<td>Double Cantilever Beam</td>
</tr>
<tr>
<td>DLM</td>
<td>Doublet Lattice Method</td>
</tr>
<tr>
<td>FALSTAFF</td>
<td>Fighter Aircraft Loading Standard for Fatigue</td>
</tr>
<tr>
<td>F&amp;DT</td>
<td>Fatigue and Damage Tolerance</td>
</tr>
<tr>
<td>FEM</td>
<td>Finite Element Method</td>
</tr>
<tr>
<td>ILSS</td>
<td>Inter laminar Shear Strength</td>
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<tr>
<td>LCO</td>
<td>Limit Cycle Oscillation</td>
</tr>
<tr>
<td>MAC</td>
<td>Modal Assurance Criteria</td>
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<tr>
<td>MBT</td>
<td>Modified Beam Theory</td>
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<tr>
<td>Abbreviation</td>
<td>Full Form</td>
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</tr>
<tr>
<td>MCC</td>
<td>Modified Compliance Calibration</td>
</tr>
<tr>
<td>MCT</td>
<td>Multi Continuum Technology</td>
</tr>
<tr>
<td>MOI</td>
<td>Moment of Inertia</td>
</tr>
<tr>
<td>NZ</td>
<td>Vertical Load Factor</td>
</tr>
<tr>
<td>PFA</td>
<td>Progressive Failure Analysis</td>
</tr>
<tr>
<td>SG</td>
<td>Strain Gage</td>
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<tr>
<td>STP</td>
<td>Standard Test Practices</td>
</tr>
<tr>
<td>SIF</td>
<td>Stress Intensity Factor</td>
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<tr>
<td>SERR</td>
<td>Strain Energy Release Rate</td>
</tr>
<tr>
<td>LVDT</td>
<td>Linear Variable Displacement Transducer</td>
</tr>
<tr>
<td>NACA</td>
<td>National Advisory Committee for Aeronautics</td>
</tr>
<tr>
<td>QF</td>
<td>Quality Factor</td>
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<tr>
<td>RVE</td>
<td>Representative Volume Element</td>
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<tr>
<td>UAV</td>
<td>Unmanned Aerial Vehicle</td>
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<tr>
<td>UTS</td>
<td>Ultimate Tensile Strength</td>
</tr>
<tr>
<td>UD</td>
<td>Uni-Directional</td>
</tr>
<tr>
<td>VAT</td>
<td>Variable Angle Toe</td>
</tr>
<tr>
<td>VTOL</td>
<td>Vertical Take-off and Land</td>
</tr>
<tr>
<td>VCCT</td>
<td>Virtual Crack Closure Technique</td>
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### NOMENCLATURE

<table>
<thead>
<tr>
<th>Symbol</th>
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<tbody>
<tr>
<td>$E_1$</td>
<td>Young’s modulus in the first direction</td>
</tr>
<tr>
<td>$E_2$</td>
<td>Young’s modulus in the second direction</td>
</tr>
<tr>
<td>$E_3$</td>
<td>Young’s modulus in the third direction</td>
</tr>
<tr>
<td>$\nu_{12}$</td>
<td>Poisson’s ratio in the first direction</td>
</tr>
<tr>
<td>$\nu_{13}$</td>
<td>Poisson’s ratio in the second direction</td>
</tr>
<tr>
<td>$\nu_{23}$</td>
<td>Poisson’s ratio in the third direction</td>
</tr>
<tr>
<td>$G_{12}$</td>
<td>Shear modulus in the first direction</td>
</tr>
<tr>
<td>$G_{13}$</td>
<td>Shear modulus in the second direction</td>
</tr>
<tr>
<td>$G_{23}$</td>
<td>Shear modulus in the third direction</td>
</tr>
<tr>
<td>$V$</td>
<td>Velocity</td>
</tr>
<tr>
<td>$F$</td>
<td>Frequency</td>
</tr>
<tr>
<td>$G$</td>
<td>Damping ratio</td>
</tr>
<tr>
<td>$\delta$</td>
<td>Displacement</td>
</tr>
<tr>
<td>$G_I$</td>
<td>Strain energy release rate mode-I</td>
</tr>
<tr>
<td>$G_{IC}$</td>
<td>Critical strain energy release rate mode-I</td>
</tr>
<tr>
<td>$P$</td>
<td>Load</td>
</tr>
<tr>
<td>$P_C$</td>
<td>Critical load</td>
</tr>
<tr>
<td>$E$</td>
<td>Fexural modulus of elasticity</td>
</tr>
<tr>
<td>$I$</td>
<td>Moment of inertia</td>
</tr>
<tr>
<td>$C$</td>
<td>Compliance</td>
</tr>
<tr>
<td>$a$</td>
<td>Crack length</td>
</tr>
<tr>
<td>$\delta$</td>
<td>Displacement</td>
</tr>
<tr>
<td>$\delta_C$</td>
<td>Critical displacement</td>
</tr>
</tbody>
</table>
$\Delta$ Correction factor

$K_I$ Stress Intensity in mode-I

$K_{IC}$ Plane strain fracture toughness in mode-I

$X_{1t}$ Tensile strength in fibre direction

$X_{1c}$ Compressive strength in fibre direction

$X_{2t}$ Tensile strength in transverse direction

$X_{2c}$ Compressive strength in transverse

$X_{3t}$ Tensile strength in third direction

$X_{3c}$ Compressive strength in third direction

$S_{12}$ Shear strength in XY plane

$S_{13}$ Shear strength in XZ plane

$S_{23}$ Shear strength in YZ plane

$M_{-1}$ First bending

$M_{-2}$ Second bending

$M_{-4}$ First torsion

$M_{-5}$ Third bending

$\sigma_{\text{normal}}$ Normal stress

$S_n$ Normal strength

$\sigma_{\text{tangent}}$ Tangential stress

$S_t$ Tangential strength

$\Delta^0$ critical relative displacement

$\Theta$ Fibre angle

$S$ Span length

$M$ Bending moment
ACKNOWLEDGEMENTS

All praise to the Almighty who made me able to make it possible. I am thankful to my friends and learning partners, Muhammad Tahir, Aiman Iqbal, Awais Anjum and Nazeer Ahmad Anjum, who played very positive roles along the journey to compel the research work, as we mutually discussed in making the sense of the different challenges which I faced during the completion of different tasks. They provided me encouragement at those particular tough times when it seemed difficult for me to handle the technical challenges. I also offer my gratefulness and gratitude to my supervisors, Dr. M. Zubair Khan and Dr. Asif Israr, for the different ways in which they advised and supported me throughout the work, actually it is very important for the mentor to know when to push and when to let it up. I am especially grateful to one of my senior colleague Dr. Azher Munir whose bold steps rescue people when they are in trouble. I offer special thanks to my other colleagues who supported me in the mechanics of producing this thesis, Dr. M Tayyab, for reading and improving the drafts involved in various corresponding, Mr. Shahid Mehmood for editing, proof reading and helping me to get unstuck with this thesis work on many occasions; and particularly Dr. Masoodur Rehman Shah and his brother Dr. Owaisur Rehman Shah for supporting me at certain times when I was running short of time. Special thanks to my colleague Dilawar Ali who is very expert software engineer, he helped me a lot in the development of fatigue spectrum which is the core subject of this research work. My colleague Hassan Moin helped me when I was badly stuck during ABACUS simulations. One of my best colleagues, Tariq Mehboob provided me a comfort zone to concentrate on my thesis writing. I may not forget the moral support of my boss Engr. Muhammad Ajmal who provides comfortable working environment to his subordinates.
I want to offer thanks to my parents who continued to grow and develop me as a human being with the prime objective of serving others. Actually my parents have always been a source of encouragement and inspiration for me throughout my life. There is again a special thank you my parents for providing me the affection. And also for the countless of ways in which, throughout the life, you have always been actively supporting me in my determination to realize my potential of undertaking the work load. A special thanks to my Father, who characterized to me a ‘living proof’ of ability to continuous hard work and physical struggle. He advised a continuous struggle despite the opposition, the tremendously constraining, repressive and suppressive situations in which we generally exist and lost. Thanks to my dear wife who has always been willing to support me and engaged with the struggle. She always ensure my comfort and is actively promised in standing with me in a special role. A very special thank you my dear wife for your effective and emotional support. It is very important for a man to be easy in his family life for part time study and personal development. Thanks to dear Daud Mehdi my friend, for being so helpful even when being lonely was hard to me, and for your help with the discussions in all the matters of life. This work is for, and because of my friends, family and colleagues. It is dedicated to all our journeys in learning to thrive. At the end I would also like to pay my gratitude to Elsevier for giving me the honor to become the reviewer of International Journal of Fatigue. The reviewer’s certificate is attached as ‘Appendix - A’

Waqas Anwar
Effect of structural dynamic characteristics on fatigue and damage tolerance of aerospace grade composite materials

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SUMMARY

The use of composite materials is rapidly growing in the construction of aerospace structures. Most of the aircraft parts made from composite materials, like wing skins, spoilers, fairings, and flight controls, are being used due to their reduced weight as compared to aluminum parts. New generation large commercial aircraft are being designed with all composite fuselage and wing structure. The main advantages of composites over conventional metallic materials are their high strength to weight ratio, corrosion resistance and tailored stiffness. Uni-directional composites have predominant mechanical properties in one direction and vary with the direction relative to the axis of structure. The aero-elastic properties, such as stiffness and dynamic stability, also depend on the stacking sequence of the plies.

Fatigue and Damage Tolerance (F&DT) assessment of these advanced composite materials is an emerging field of research. In aircraft structural integrity analysis, the damage tolerance and fatigue life is investigated against a cyclic loading spectrum. The particular spectrum includes the stress/loading levels counted during a flight of certain duration. The occurrences of z-axis load factors ‘Nz’ may include higher gravitational acceleration ‘g’ levels. While maintaining a certain g level occurrence at higher angle of attack, wing structure vibrates with the amplitudes of its natural frequencies. The cyclic stress amplitudes of vibration depend upon the natural frequencies of vibrating structure, i.e. lower frequency gives higher amplitudes and vice versa. These fluctuating load amplitudes are superimposed on the higher ‘g’ level mean loads during fatigue analysis. These additional cycles are very critical in DTA studies.
In the present work, the structural dynamic and fatigue life of composite panels have been explored. To improve the dynamic stability, modal parameters of simple carbon fibre sandwich panels have been adjusted by tailoring the fibre orientation angles and stacking sequence. In this way, the effect of change in structural dynamic characteristics on fatigue life of this simplified structure has been demonstrated. The research methodology followed in this work consists of two phases. In the first phase, aero-elastically tailored design was finalized using FEM based modal analysis and Flutter analysis simulations followed by experimental modal analysis. In the second phase, fatigue and damage tolerance behaviour of the samples was investigated using different fracture mechanics based techniques. ASTM’s standard practices were adopted to determine material allowable and fracture properties. Simulation work was performed after proper calibration and correlation of finite element model with experimentally determined static and dynamic behaviour of panels.

It has been observed that the applicable cyclic loading spectra, as major input parameter of fatigue analysis, largely depend upon the natural frequencies, damping and the stiffness of the structure. While maintaining a certain pull-up maneuver, the basic load cycle mainly experienced by the aircraft wings further experience the secondary vibrational loads. It has been observed that the severity level of these additional loads depend upon the first few lower natural frequencies of the wings or stores. Most critical of these vibrational modes to the fatigue life are the bending and torsional modes. Results and conclusion of this exercise may be beneficial while carrying out aero-elastic tailoring of composite aircraft wing. This research work has also a positive contribution towards multidisciplinary structural design optimization of aerospace vehicles.

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CHAPTER 1

INTRODUCTION

In this chapter the significance of structural integrity related issues of aerospace grade composite structures are highlighted. The increasing use of composites in aerospace industry are described in three major applications i.e. the commercial aircraft, military aircraft and UAV’s. It is evident from the literature survey that the trend of using carbon fibre composites in place of metals is exponential. According to aerospace applications, the main advantages and disadvantages of carbon fibre composites over conventional metallic materials are also described briefly. After the brief introduction of composites usage in aerospace applications and the common challenges associated with these materials, the discussion turns toward the scope of thesis work. As mentioned earlier that the objective of this work is to quantify the difference (either increase or decrease) in the service life of a sample carbon fibre composite material as an integral part of aircraft wing skin, due to variation in its natural frequencies of vibration. The justification for the variation in the natural frequencies of vibration of composite sample is established as ‘aero-elastic tailoring’. The aero-elastic tailoring of composite wing is explained in detail along with a couple of examples. The detail includes the structural dynamic model explanation along with mathematical formulations. Structural dynamics is the first subject of the thesis. The second subject is structural integrity of composites primarily focusing on fatigue and damage tolerance. To introduce the basics of this area, fracture mechanic approach is explained along with the conventional theories to handle the fracture in composite parts.
Current development to handle fracture is explained for both metals and composites. At the end of this chapter, the deficient area in the field of study which requires more development has been highlighted, which is the investigation of fatigue life under variable parameters of fatigue load spectra. These variable parameters of fatigue load spectra are strongly associated with structural dynamic characteristics. This chapter concludes with the remarks which justifies the synopsis as a fruitful effort to fill the gaps in this area and provides the link to onward chapters.

1.1 Use of Composites in Aerospace Industry

Aviation is one of the most critical industry for composite materials application. Every year millions of the people travel by the air. During past 5 years alone (2012-2017) air passengers has been increased 50%, which is the extensive forecast of continual growth of this industry.[1] The increasing demand of air travel is one of the reasons for airlines to look for bigger aircraft with higher capacity, such as A-380, which is the largest aircraft in the world. It can carry more than 800 passengers with a maximum takeoff weight 560 Tons. Bigger aircraft rises majestically in the sky, particularly during takeoff and landing extensive forces act on the wing. Under extreme conditions during the flight, the tip of wing may deflect upwards by 7 meters. All the wing tip deflections may only tolerated by aircraft due to the use of strong materials in the spars and other load bearing elements.[2] For past decades, lighter materials have developed with the improved strength and toughness. These are composite materials made of fibres and matrix. The textile machines generally produce the fibres, which is the result of decade’s research. Generally the carbon fibre composites are 50% lighter than the Aluminum. The key of saving weight is the identification of direction of forces action on the structure. Unlike steel or aluminum, the
composite materials manufactured according to the direction of the axes of external forces. Stability of the material can be affected by placing the fibres appropriate to the direction of external forces acting on the structural part. For example, a part stresses by torsion, the fibres direction will be at 45° to the length of the component. The basic composite material generally used in aerospace industry is carbon fiber. Each of the fiber strips consists of 600,000 filaments or 6 Million threads per square meter. [3] They are sewn with special yarn. Five hundred needles sew the fiber bundle with incredible precision and the results is a cloth of extreme resilience. Carbon fiber reinforces plastics are made of these textile have the advantage of getting any complex shape more than the steel or aluminum. The carbon fiber is also tougher than these conventional materials. These factors contributes extensively for weight saving. The carbon fiber plies saturated with the resin are baked to elevated temperature in the special ovens. Depending on the individual plies, the end shape may be different.

1.1.1 Composites in Commercial Aircraft

For A380 aircraft, 20% of the parts of body are made of carbon fiber composite material because the goals were to achieve lower weight, save fuel and reduce emission. [4] Over half of the new airbus A-350 will be made of the same carbon fiber material and it aims to be of 10 tons less weight than A-330. It will allow the airbus engineers to reduce emission up to 30% of the standard value. Finally, it will allow the passenger’s per kilometer cost reasonably low as 5 liters. As shown in the Fig.1-1 and Fig.1-2, Boeing's 787 Dreamliner is made of 50% composite material by weight and 80% of the composite materials by volume. The trend of using composites in commercial aircraft has been increased rapidly. [5] [6] [7]
Fig.1-1. Boeing’s 787 Dreamliner composite material contents by weight (Source: 2013 Boeing fire department release)

Fig.1-2. Commercial aircraft structure trends contents by weight (Source: 2013 Boeing fire department release)
1.1.2 Composites in Military Aircraft

Similar to commercial aircraft, the use of composites in military aircraft has also been increased rapidly. In pursuit of increased performance most aircraft manufacturers now employ carbon fibre composites (CFCs) as an alternative to conventional aluminium alloys. In recent years military requirements have put much greater emphasis on reduced life cycle costs, which in turn is posing new challenges to aircraft designers and manufacturers. Fig.1-3 shows the use of composite material in F-18 aircraft.

![Composite Materials in F/A-18 Structure](image)

**Fig.1-3. Use of composite materials in F/A–18 structure as a percentage of structural weight. (Source Composites in Aerospace Applications, White Paper)**

Fig.1-5 and Fig.1-6 shows the use of composites by weight in civil and military aircraft. In these figures we see that the use of aerospace grade composite materials are almost present in all the major structural members including the skin, doors, spars and interior. In current generation aircraft the primary structural members are particularly being made of carbon fibre composites due to its high strength.
1.1.3 Composites in UAV’s

For the last two decades, Unmanned Aerial vehicles (UAV’s) have become more significant in the battlefield and many other military applications. UAV’s civilian applications have also been increased tremendously for last few years. Farmers use to facilitate their farms and surveillance purposes. Electric supply companies use the UAV’s for verification of their distribution lines. City management officials use it for car traffic observation and environmental monitoring. Other potential applications are increasing day by day as the new manufacturers are entering in the international market. [8] Fig.1-4 shows the used of different composite materials in the construction of Vertical Take-off and Landing (VTOL) UAV.

Fig.1-4. Composites in a VTOL UAV.
Fig. 1-5. Use of composite by weight in military aircraft (Source Avalon Consultancy Services Ltd.)

Fig. 1-6. Use of composite by weight in civil aircraft (Source Avalon Consultancy Services Ltd.)
1.2 Advantages of Carbon Fibre Composite

There are several advantages of carbon fibre composites as compared to conventional metallic materials of aircraft i.e. aluminium and titanium alloys. A few important points are discussed here;

1) Because of increased strength and stiffness of carbon composites, up-to 30% of weight may be reduced without compromising the design strength.

2) CFC plies may be oriented in the load path direction to achieve the higher stiffness in the required orientation without adding additional mass.

3) Formability of complex shapes is quite easy as compared to metallic materials.

4) Placing the uni-directional plies at specific orientation/angles, the directional strength properties may be altered as per requirement of the application.

5) During manufacturing process, the mass is added where needed. It reduces the scrap rate and saves cost. In the bigger assemblies, the parts count is less as compared to metals because the curved surfaces may be easily bonded together without welding.

1.3 Disadvantages of Carbon Fibre Composites

The disadvantages of CFC restrict their use in some of the critical airborne elements. A few important points are discussed from aerospace point of view:

1) Environmental degradation of composites is comparatively higher. Generally the effects of and moisture and temperature (the hygro-thermal effect) is quite severe in case of CFC as compared to metals. The operational environment for a ‘conventional take-off and land’ (CTOL) aircraft becomes very critical in ultra violet (UV) exposed areas. Similarly, very high temperature changes are experienced by ‘vertical take-off and land’ (VTOL) vehicles such as the Harrier aircraft. When the matrix material is
coupled with a heated and humid environment, it absorbs moisture and degrades, giving less favorable condition to the composite load bearing structure. This phenomenon generally occurs at off-shore places.

2) Carbon Fibre Composite is notch-sensitive material. Local yielding around notches such as cut-outs, bolt holes and sharp features etc. cannot take place; Carbon fibre composites are prone to de-lamination especially when exposed to impact damage and furthermore, this delamination cannot usually be detected from naked eye. This delamination considerably reduces the compressive and tensile strength and fracture toughness related mechanical properties of critical parts.

3) The fundamental source of strength in CFC is the ‘carbon fibre’. The values of matrix properties are usually one tenth of the fibre properties. In the majority of engineering applications, fibre reinforcement in the third direction of space coordinates is not used. So, the composites have little resistance to out-of-plane loadings. In this way, the load carrying capability of the third direction becomes dependent only on the matrix, which is quite weaker part of the material.

4) CFCs experience higher variation in theoretical and experimental results as compared to metallic materials. Coefficients of variation generally approaches up to 7.5% for the in-plane loading applications and it becomes as high as 15% for the out-of-plane loading applications, whereas 5% coefficient of variation is generally considered reasonable for metallic applications. [9]

5) The electrical resistance offered by CFC may be 1000times higher than aluminium which making CFC aerospace structures more vulnerable to lightning strike. However, proper static electrical dischargers may be installed to avoid damage. [10]
6) Another disadvantage to CFC is the availability and cost of material. Conventional aerospace material such as Aluminum is readily available to carry out experimentation and analysis at both the industry and academia level.

1.4 Aero-elastic Tailoring of Composite Aircraft Wing

Aero-elastic performance of the wing may be improved by controlling the fiber orientation. The composite wing may give better performance during steady state condition, gust and maneuvering. Each flight condition of the aircraft generates stresses on the wings; these stresses at any condition are load cases for the design. Tailoring may reduce the stresses on the wings. Aero-elastic performance is measured as the flutter speed and divergence speed. There are four novel ways to control aero-elastic performance:

1. To use curvilinear structural members, i.e. curved spars, curved ribs and curved stringers etc. This is relatively difficult scheme to adopt.

2. To use special structural arrangements, i.e. the control of angles between different structural members with respect to one another. It may be the angles between ribs and spars, the angles between spar and stringers and so on.

3. Crenellated skin of varying thickness may also improve the aero-elastic stability. The thickness varies in chord wise and span wise direction as described by J Cooper. [11]

4. The benchmarked concept is the stacking sequence concept which has also been used in this research work.

High fidelity FEM modeling is always carried out for best possible design of either option. The stacking sequence is generally designed to avoid bending and twisting mode coupling of the wing or any other low damping modes of vibration.
Aerodynamic, elastic and inertial forces are the main concern for better stability of aircraft. Traditional collar triangle describes the relation between these forces. Static and dynamic aero-elasticity lies at the bottom and center of the triangle. The triangle focuses on the stability of aircraft. All the parameters mentioned in Collar triangle are the key influencing factors for aircraft design.\[12\]Thermal effects were also taken into account after some high speed flight incidents occurred. As described in Fig.1-7, the Collar’s triangle changed to a tetrahedron adding four new kinds of interactions. The correct terminology for the interaction of all these four forces together would be aero-thermo-dynamo-elasticity.\[13\]

**Fig.1-7. Aero-thermo-dynamo-elasticity diagram**
As shown in Fig.1-8, X-29 aircraft is one of the example of aero-elastic tailoring. It’s the experimental aircraft of early 1980’s. Unbalanced composite lay-ups were used on wing covers to generate washout effects. If the wings bend upward it gives negative angle of attack towards the tip. It becomes beneficial particularly for the forward swept wings because such wings have very low tendency towards divergence speed.

**Fig.1-8.X-29 Aero-elasitically tailored experimental aircraft**

1.5 **Dynamic Stability of Aircraft Structures as a Design Requirement**

Laminated composite material development and reinforced fiber materials are closely related keeping in view their use in aerospace structures construction. These type of materials allow the structural parts construction with high strength to mass ratio. The development of composite material gave the designer a wide number of options for the construction of lightweight structures. FS 24 Phoenix was the first sailplane made of
fiberglass, it was made in West Germany in 1957. The use of composite materials in structural parts of aircraft started increasing by the first use.

Initially, the designers were reluctant to use similar composite materials in aircraft design; however, the physical and material properties of composites were explored to know more about their further uses. Standard procedures were started establishing for standard testing of these materials. [14] The first composite made horizontal stabilizer was developed in 1960 for F-111 aircraft. Flight trials of this aircraft were so successful that the production of later aircraft F-14, 15 and F-16 followed the same concept and a relatively greater amount of composite materials were incorporated in their production with greater success.[15] Currently, there is a need to expand the operational envelopes of these aircraft. To meet this requirement, the weight of the aircraft should be as low as possible, which is only possible by using lighter materials.

Considering the dynamic stability and aero-elastic performance of Unmanned Aerial Vehicles, lighter weight UAV’s is the matter of prime importance. Unmanned Aerial Vehicles are more prone to aero-elastic instability due to their high span wings and being lighter in weight as compared to other conventional, rigid structure and short span aircraft. Possibility to Limit Cycles Oscillations (LCO) is also a greater concern during routine operation of UAV’s. Limit cycle oscillation is the case of more severe nature because the structure may undergo direct fatigue loads due to this kind of aero-elastic instability. In case of conventional aircraft the pilot, and in case of UAV the payload becomes under continuous vibrations due to LCO. The automatic control system of vehicles may also be affected due to LCO frequencies. Flutter is generally closely related to LCO but there is no coupling involved in LCO, which is there in case of flutter between non-stationary
aerodynamic forces and structural response. In this way, the conventional flutter analysis may also be used to predict the LCO in addition to flutter speed. [16]

Now a days, new design and mission requirements of aircraft have been emerged. Structural optimization involves a mandatory procedure to achieve an efficient aero-elastic design. If the structure undergoing design optimization process, and is made of composite materials, then it must undergo aero-elastic tailoring. Aero-elastic tailoring is a design process in which the orientation of directional fibre properties are used in best optimum way. [17]

Aero-elastic tailoring became a hot topic in early 1970’s when X-29 fighter aircraft was under production with the discovery of forward swept wing’s capability to encounter divergence. [18], [19]. That was a time, when aero-elastic tailoring was aiming at only one objective to avoid divergence. Different analytical design tools were used at that time and all the focus was the divergence phenomenon. One deficiency of that time analysis was to overlook the transonic speed region due to complexity of phenomenon. Similarly, the dynamic aero-elasticity and the control surfaces effects were also not studied due to limited analytical models. In 1990’s the active aero-elastic tailoring became the field of study for multi-disciplinary design optimization engineers. [20] With the help of computer programming, United States Air Force Flight Dynamics Laboratory (AFDL) carried out experimentations to demonstrate the effect of different angles of backward swept wings on dynamic stability of aircraft, however the composite material usage was not the objective of that study. [21] To perform unsteady aero-elastic analysis, Doublet Lattice Method (DLM) was introduced in 1971. [22] The said reference work has only presented the results of DLM and focused very little on the mathematical modeling of the technique. Later on
different modified methods were also presented to calculate the flutter speed and divergence in which the instability was shown as a function of aerodynamic influence coefficients. The lamination parameters are ultimately declared as a design variable for dynamic instability. [23], [24]

1.6 Structural Dynamics Model

The coupling of aerodynamic, elastic and inertial forces yields a complex state of stress for the aircraft structure. There are many analysis types, which may be carried out to make an engineering estimate for dynamic stability under these forces. Collar’s triangle not only gives a clear understanding of the interaction between each type of the forces acting during flight of aircraft, it also illustrates the type of analysis required to address the stability. It is very important to define what are the special considerations required by these approaches. The aero-elastic equation of motion is presented with a separation of structural and aerodynamic coefficients.[25] It can be written as following relationship:

\[ M\ddot{x} + C\dot{x} + Kx = F_{ext} \]  

**Equation 1-1**

Where M is the mass matrix of the vibrating system, C is the damping matrix of the system and K is the stiffness matrix. Here, the x is displacement vector. It may be represented in all degree of freedom systems. In case of angular systems, the x may be replaced with \( \theta \). F represents all the external forces. In case of the problem presented in present work, F is the Flutter forces vector. The Flutter forces may be represented as mentioned below.

\[ F_{ext} = G_s^T F_a (t, x, \dot{x}) \]  

**Equation 1-2**

This means that the ‘F’ forces depend upon time and the time derivative. \( G_s \) is the matrix, which gives the displacement from structural grid points to the aerodynamic grid points. It
is generated by the spline transformation. This is due to coupling of structural and aerodynamic forces. [26]

1.6.1 The Structural Matrix

In case of dealing with a linear approach, the assumption of small displacements may be considered in mathematical modeling and the superposition principle becomes valid. This assumption allows us the use of generalized matrices formulation. In this formulation the structural modal in formation is coupled to an aerodynamic operator. Using this approach, the structural part is generally represented by eigen-frequencies and their associated eigen-modes. The structural displacement may be presented as below relation:

\[ x = \phi_e \bar{x}(t) \]  \hspace{1cm} \text{Equation 1-3}

Where, \( \bar{x}(t) \) is the generalized displacement vector and \( \phi_e \) is the eigen-vector matrix, which is obtained from modal analysis of structural model. It may be obtained by FEM solution or experimental modal analysis. The aero-elastic un-damped equation of motion is now becomes as following:

\[ \phi_e^T M \phi_e \ddot{x}(t) + \phi_e^T K \phi_e \bar{x}(t) = \phi_e^T F_{ext}(t) \]  \hspace{1cm} \text{Equation 1-4}

Where,

\[ F_{ext} = G_s^T F_a (t, x, \dot{x}) \]

Hence, the equation of motion becomes:

\[ \bar{M}_e \ddot{x} + \bar{K} \bar{x} = \phi_e^T F_{ext} (t, q) \]  \hspace{1cm} \text{Equation 1-5}

Here, \( \bar{M}_e \) is generalized mass matrix and \( \bar{K} \) is called generalized stiffness matrices. The computation of aerodynamic forces have the basis on boundary conditions which are defined at aerodynamic control points. It also depends upon the displacement vector for
boundary conditions which is given in the FEM model nodal coordinates. The transformation represents the physical coordinates and it is applied to each modal matrix. [27]

### 1.6.2 The Aerodynamic Model

The aerodynamic model gives the external forcing terms of aero-elastic equation of motion through an aerodynamic operator. The first work on the aero-elasticity was flutter oriented problem which used the potential flow theory with an operator which was a function of reduced frequency and it represented the lift force and bending moments at two-dimensional airfoil. [28] An equivalent formulation was also developed by the same team, in which the lift was indicated as a function of reduced time variable. [29] [30] As it has been already mentioned that these theories worked with two-dimensional airfoil models which was the so-called typical section. In such cases, the structural behavior is constrained to only a few displacement modes, namely pitch, plunge and the aileron deflections. These modes consist in one longitudinal displacement, the pitch and three rotations only. After that, the two-dimensional aerodynamic model was presented as a finite wing, it was made by using several typical sections side by side, which consists of the strip method. Now-a-days there are many commercial software tools available for computational fluid dynamics analysis on the basis of finite element modeling in which 3 dimensional modeling is supported. Visualization of the effects are is the most important option during design and analysis phase of any engineering application.
1.6.3 Lamination Parameters

The classical lamination theory is an addition of the standard plate bending and plane stress theories for layered plates with changing stiffness and strength of each ply. As the detailed illustration of this theory is not the scope of this work, however it may be easily found in many of the standard books covering the plates and shells design. According to this theory, the constitutive equation of laminated plate can be described as below:

\[
\begin{align*}
\begin{bmatrix} N \\ M \end{bmatrix} &= \begin{bmatrix} A & B \\ B & D \end{bmatrix} \begin{bmatrix} \varepsilon \\ \kappa \end{bmatrix} \\
N \text{ and } M &\text{ are the distributed tractions and moments, respectively, applied to the plate: Where, } \varepsilon \text{ are mid-plane (membrane) strains, and } \kappa \text{ is the vector of curvatures, second derivatives of the transverse displacements.}
\end{align*}
\]

\[
\text{Equation 1-6}
\]

\[
\begin{align*}
N &= \begin{bmatrix} N_x \\ N_y \\ N_{xy} \end{bmatrix} \\
M &= \begin{bmatrix} M_x \\ M_y \\ M_{xy} \end{bmatrix}
\end{align*}
\]

\[
\text{Equation 1-7}
\]

The submatrix A is called extensional stiffness matrix and is related to the influence of an extensional mid-plane strain \( \varepsilon \) on the in-plane traction \( N \). It can be described as:

\[
A = \sum_{k=1}^{np} \bar{Q}(z_{k+1} - z_k),
\]

\[
\text{Equation 1-8}
\]

Where, \( z_k \) is the distance from laminate mid-plane to the bottom of the \( k^{th} \) ply, \( \bar{Q} \) is the transformed stiffness matrix, and the \( np \) is number of plies. The submatrix ‘B’ is called coupling stiffness matrix, it calculated the contribution of the curvature ‘k’ to the traction and it is there when the plate is not symmetric. This term may be described as:

\[
B = \frac{1}{2} \sum_{k=1}^{np} \bar{Q}(z_{k+1}^2 - z_k^2),
\]

\[
\text{Equation 1-9}
\]
‘D’ is a bending stiffness matrix which may be defined as following relation:

\[ D = \frac{1}{3} \sum_{k=1}^{np} \bar{Q}(z_{k+1}^3 - z_k^3). \]  

**Equation 1-10**

It is worth mentioning that in case of symmetric plates, the term which relates the coupling matrix ‘B’ is zero, once applying an in-plane traction to the plate do not generate a curvature, or bending moment will not generate an extensional normal strain. A detailed review about the formulations of all the matrices presented above may be found in many relevant books about composite materials. [31]

### 1.7 Structural Integrity of Aircraft Composites

Composites are susceptible to damage mainly because of the lack of reinforcement in the through-thickness direction. Detrimental Barely Visible Impact Damage (BVID) usually caused by low-energy impact is a particular case, and it could significantly degrade structural performance due to induced interior delamination.[32] [33]

#### 1.7.1 Fracture Mechanics Approach for Damage Tolerance Analysis

The behavior of a material containing a flaw under stress was first characterized by Griffith. [34] He is considered to be pioneer of fracture mechanics He proposed that the kinetics for crack growth are favored when the surface energy increase supersedes the release of elastic strain energy. At equilibrium the two processes will be equal. He developed a formulation to estimate the amount of energy required to grow an already existing crack in a material. When an object is subjected to the mode I opening of a crack, Griffiths predicted that the critical stress (\(\sigma\)) needed to propagate the crack is calculated as

\[ \sigma = \frac{4E}{(1-\nu)K^2} \]

**Equation 1-11**

Where:
- \(E\) is Young’s modulus
- \(\nu\) is Poisson’s ratio
- \(K\) is the stress intensity factor

\(\sigma\) is the critical stress required for crack propagation.
\[
\sigma = \sqrt{\frac{2E\gamma}{\pi a}} \quad \text{Equation 1-11}
\]

Where \( E \) is the material’s modulus of elasticity, \( \gamma \) is surface energy of material per unit length of the crack, and \( 'a' \) is crack length of edge crack or \( '2a' \) is the crack length for plane crack.

For brittle materials the criterion is valid but the surface energy term accounts only for elastic deformation. Irwin [35] extended the surface energy term to include plastic work, \( \gamma_p \)

\[
\sigma = \sqrt{\frac{2E(\gamma + \gamma_p)}{\pi a}} \quad \text{Equation 1-12}
\]

The work required to create new surfaces was called the crack extension force, \( G \), and the above equation is rewritten as:

\[
G = \sigma^2 \left\{ \frac{(\pi a)}{E} \right\} \quad \text{Equation 1-13}
\]

The stress intensity factor \( K \) is related to the bulk applied stress, \( \sigma_c \), to the flaw size, \( 'a' \) and the loaded specimen geometry through \( 'Y(a/w)' \) term. The mode I fracture toughness for plane strain is defined as;

\[
K_{IC} = Y \sigma_c \sqrt{(\pi a)} \quad \text{Equation 1-14}
\]

Where \( \sigma_c \) is critical far field stress and \( Y \) is a dimensionless parameter which depends on the geometry of crack and loading conditions. \( K_1 \) is the stress intensity factor which is determined experimentally. The stress intensity factor is a material parameter which serves as a measure of crack resistance primarily for brittle materials. For laboratory testing the specific guidelines have been established to measure \( K_{IC} \) in ASTM E399 which permits its use as a design criterion. [36]

Similarly \( K_{II} \) and \( K_{III} \) can also be determined for shearing and tearing loading. The stress state around cracks in different shapes can be expressed in the form of stress intensity
factors. Linear elastic fracture mechanics laws predict that a crack can extend if the stress intensity factor at the crack tip becomes greater than the fracture toughness \( (K_C) \) of that particular material. Hence, the critical applied stress can then be determined once the stress intensity factor at a crack tip is known.[37]

1.7.2 Strain Energy Release Rate

In case of anisotropic materials such as; composites, it becomes difficult to apply linear elastic fracture mechanics laws. Similarly for the cases where the loading or geometry of structure is complex LEFM laws also become difficult to apply. In this scenario, the strain energy release rate approach becomes useful to apply. The strain energy release rate for a mode I crack going through the thickness of a plate can be defined as follows;

\[
G_I = \frac{P}{2t} \left( \frac{du}{da} \right)
\]

Equation 1-15

Where ‘\( P \)’ is applied load, ‘\( t \)’ is the thickness of the plate, ‘\( u \)’ is the displacement at the point of application of the load due to crack opening and ‘\( a \)’ is the crack length for edge cracks or ‘\( 2a \)’ is the crack length for plane cracks. The crack is expected to propagate when the value of strain energy release rate ‘\( G_I \)’ exceeds the critical value of the same i.e. ‘\( G_{IC} \)’. Critical strain energy release rate and Fracture Toughness can be related in plane stress conditions as following;

\[
G_{IC} = \frac{K_{IC}^2}{E}
\]

Equation 1-16

Where, ‘\( E \)’ is the modulus of elasticity of the material. With known initial crack size, a critical stress value can be calculated using strain energy release rate.

The strain energy release rate is the energy dissipated during fracture per unit of the newly created fracture surface area. The energy that must be given to a crack tip to grow
further must be balanced by the energy which is dissipated due to the creation of new surfaces and some other dissipative processes i.e. plasticity.

Similarly the energetic contour path integral (J-integral) is used to calculate the strain energy release rate, or energy per unit fracture surface area. J-integral is independent of the path around a crack. Experimental methods have been developed for measurement of critical fracture properties using laboratory level material specimens. It is assumed that Linear Elastic Fracture Mechanics (LEFM) laws do not hold due to small sizes of specimen and a critical value of fracture energy $J_{IC}$ is determined. The property $J_{IC}$ defines the point at which large scale plastic yielding takes place during crack propagation under 1st mode of loading.[38]

2D composites initially started with the development of cross-ply laminates. Initially they were composed of multiple uni-directional aligned fibres having alternating angles between the plies. Low inter-laminar strength caused the failure of 2D composites in shear mode, it became the first drawback of this material. In contrast with metals, the plane-stress & plane-strain transitions of $K_{IC}$ with respect to specimen thickness which is observed in monolithic materials, was not determined in composites. So, the plastic zone near crack tip which increases the failure resistance was also not observed in composite materials. The fracture toughness of linear-elastic monolithic materials was also independent of fracture test configuration. [39]

1.7.3 Composites Failure Analysis Approach

In micromechanics, the failure of continuous fibre reinforced composite material can be explained by micro level analysis of stresses and strains within each constituent element of composite material i.e. fibre, matrix and the interface between these
constituents. The applicable macro stresses are calculated by ply level analysis. Micromechanics based theory gives more accurate results as compared to phenomenological approach i.e. Tsai Hill and Tsai Wu failure criteria. The phenomenological approach identifies the critical ply whereas the micro mechanics based model identifies not only the critical ply but also the critical constituent within the critical ply. The engineering failure analysis involved in composite material starts from estimating the mechanical behavior of constituents (matrix, fibre and interface) and covers the mechanical behavior of ply, laminate and ultimately the whole structure. It is also called the building block approach as shown in Fig.1-9.

![Fig.1-9. Building block approach for engineering analysis of composite structures](image)

Three elements are mandatory to fully characterize the element level constituents.
1. The constitutive relation is the primary requirement which defines the transient or time-independent response of each constituent to external mechanical and hygro-thermal loadings. Primarily, it is the stiffness and compliance matrices etc.

2. The master curve is second requirement of characterization which defines the time dependent response of the constituent elements under fatigue loads.

3. The applicable failure criterion, which defines the conditions that cause the failure of constituent element.

A micromechanical models links the ply behavior with micro level constituents, so that the properties of lamina can be derived from the properties of constituents. On the other hand, stresses and strains at constituent level can be calculated from macro level stresses and strains of lamina.

![Square and Hexagonal Array](image)

**Fig.1-10. Two different ways to define a unit cell from idealized fibre arrays**

We may start the analysis from micro level, the basic unit cell of unidirectional lamina should be well defined. In ideal situation, all the fibres should be aligned in one longitudinal direction, however, in cross section view, random distribution of fibres is
generally observed, so there is no proper regular pattern in which the fibres are arrayed. In such cases, an ideal scheme of the fibre arrangement in a ply is adopted; resultantly a regular fibre packing pattern is defined. Two common regular patterns of fibre packing are considered in above Fig.1-10. First case is square array and the second case is hexagonal array. Either type can be used as a repetition of single element, which is called a unit cell or Representative Volume Element (RVE). A unit cell ideally responds to the external forces in the same way as the whole array can do. So a unit cell is considered sufficient to model a representation of UD ply.

1.8 Deficient Area That Requires More Development

In aerospace sector, fatigue life estimation on the basis of fracture mechanics has always been a field of concern for design engineers. There is always a challenge to design aircraft parts against fatigue to avoid the undesirable failures prior to completion of proposed designed life of a structure. Initially, composite materials were used only in secondary structure, but with the passage of time, their use in primary structure such as wings and fuselages has increased. Keeping in view the growing use of composite materials in aerospace applications, the fatigue life estimation becomes very critical under different conditions. There is a growing demand for fuel efficient, light weight, and high stiffness structures that should also have fatigue durability and corrosion resistance for the construction of modern aircraft.

A lot of research has been carried out to quantify the effect of different parameters on fatigue life such as the effect of material characteristics, the effect of humidity, the effect of temperature variations and the effect of heat treatments etc. The effect of embedded sensor tube on fatigue life of composite structure under compression fatigue has been
determined by Rajendra Kumar Nahar et al. [40] The effect of voids on quasi-isotropic carbon-fibre reinforced plastic laminates under quasi-static loading has been compared with under cyclic tension loading by Sanjay Sisodia et al. [41] When the composites are used in turbine blades, an advanced fatigue life prediction method was identified by Yun Jung Jang et al, to identify the effect of mean wind speed distribution on the fatigue life of a small-scale wind turbine composite blade. [42] The fatigue cracking behavior of unidirectional reinforced carbon composites with different fiber orientations has been investigated in detail by Piyas Chowdhury et al especially for aerospace applications. [43] A comprehensive Comparative study of the mode-I and mode-II delamination fatigue properties has been conducted by F. Pegorin for aircraft composites. [44] Carbon fiber composite laminate containing symmetric internal ply-drop simulating thickness variation was fabricated and tested by Manjusha M to determine fatigue life under a standard FALSTAFF spectrum load sequence. He concluded that the fatigue life of ply-drop composite was significantly lower than that of plain composite due mainly to initiation and growth of delamination near the ply-drop location. [45] Similarly the carbon epoxy composite T-joints were fabricated and tested by M. M. Thawre et al to determine their fatigue life again under a standard fighter aircraft spectrum load sequence, FALSTAFF with various reference loads. [46] There is always a lot of potential in continuous monitoring of damages in composites to increase operational safety. A lot of research has also been reported on live condition based fatigue life estimation of aerospace vehicles. [47]

One of the most important parameters on which the fatigue life mainly depends is the applicable fatigue load spectrum. In aerospace applications, the critical structural
members of aircraft undergo the fatigue analysis simulations on the basis of standard fatigue loads spectra. These standard fatigue load spectra are different for different types of aircraft such as, fighter aircraft, commercial aircraft and transport aircraft etc. [48] These fatigue load spectra only includes the static load factor occurrences which are generally measured through a low frequency accelerometer installed at the center of gravity of aircraft. A standard fatigue spectrum is generally made through hundreds of flights data.

1.9 Concluding Remarks

Although there are many more factors which should be thoroughly investigated before giving the absolute figure of fatigue life, however one of the aspect which is generally not discussed in detail in the ongoing research is the effect of dynamic characteristics of structures on damage tolerance and fatigue of composite materials. According to known standard LIM-A-8866(C), “The airframe structure shall have unlimited life due to low frequency vibratory loadings. When these low frequency vibratory loadings are combined with the other various airplane loading conditions, (i.e. Pull-ups, banks, high angle of attack, gust etc.) The vibratory loadings shall not cause the structural fatigue life to be degraded for that which results when separately applying the other various loading conditions to the airframe structure.”

According to military standard, the fatigue life should be primarily investigated against various airplane loading conditions due to pull-ups, banks, high angle of attack and gust etc. These primary loads are the same load factor occurrences as discussed earlier. These are always included in the standard fatigue load spectra against which the critical load bearing members are designed. The low frequency vibratory loads are the additional fluctuations which must not be overlooked according to military standard. These
vibrational loads are not available as a standard because the vibrational frequencies and load amplitudes always depend upon the specific structure’s dynamic response. It has never been quantified in existing literature that how the fatigue life may vary according to the application of these low frequency vibratory loads superimposed to the static load factor occurrences. This is the deficient area of research which need more development not only in composites but metallic materials as well.
CHAPTER 2

WORK METHODOLOGY

To demonstrate the hypothesis, a set of activities were planned in a scientific manner. In this chapter the methodology of work has been presented in detail. Firstly, the general approach to handle structural integrity related issues of an aircraft has been discussed. This discussion converges with the same technical question that gives a rise to the significance of thesis work. The work was planned and carried out in 02 phases. The sequence of activities has been presented in the form of 02 flow charts. Each flow chart presents one phase of the research work. The specialized samples were used to undergo experimentation by considering them an integral part of aircraft skin. Their natural frequencies were altered to benefit the dynamic stability and consequently the effect of this change on fatigue and durability was investigated.
2.1 Aircraft Structural Integrity Approach

General Aircraft Structural Integrity Program (ASIP) activities include: static & fatigue aspects, environmental degradation and structural life assessments. Additionally, damage tolerance analysis is often required to prevent the failure of structure as a result of fatigue cracks.[49] The objective of the damage tolerant design requirement is to demonstrate adequate residual strength in the presence of flaws for specified periods of service usage. Fatigue and Damage Tolerance (F&DT) analysis gives the service life of critical load bearing structural members by taking into account the material properties, initial crack/damage level and the applicable cyclic loading spectrum. Commercially available Damage Tolerance Analysis (DTA) software codes are based upon different material models from which NASGRO is considered the latest model in metals. This material model incorporates crack growth rate data of material at different stress ratios.[50], [51] Whereas, in case of composite structures, there are many analytical models under consideration to analyze damage, however the validation of newly developed models is still a challenge. Currently, some phenomenological and micro level theories are being used to predict onset of damage. Besides material failure criteria, there are many other aspects of consideration when dealing with composites, such as the inclusion of thermal stresses in structure, moisture effects, in plane shear strength and non-linear behavior of the material.[52] One of the most important parameters to be considered in aerospace structures is the aerodynamic vibrational loads that depend upon structural dynamic characteristics.[53] In certain cases of aircraft’s critical structural location, the applicable loading spectrum can’t be measured directly by installing accelerometers. One of the limitations may be the inaccessibility of high stress concentration area.
Secondly, the flight testing for fatigue loads calculation is also an expensive activity. In this way, the fatigue spectrum for different critical structural locations is generally generated by using the easily available N-z acceleration data of the sensor installed at center of gravity (CG) of the aircraft. If it is required to carry out F&DT analysis on a critical member at the root of an aircraft wing, the cyclic loading spectrum for this particular member will be governed by the N-z acceleration occurrences count of aircraft and the vibrations of the wing. In this way, any change in natural frequencies of the wing should be considered while declaring the fatigue life. In Fig.2-1, time domain data of 02 accelerometers mounted on wing tip and center of gravity have been presented. As compared to fuselage, the wing vibrates with higher amplitudes while maintaining an occurrence of 5g level.

Keeping in view the close relationship among structural dynamics, static strength and fatigue life of both the metals and composite parts of the aircraft, the effort has been carried out to establish a relationship among these parameters for multidisciplinary design optimization. Failure index of composite part against ply material failure and delamination propagation has been investigated using latest fracture mechanics based techniques such as progressive failure analysis, cohesive zone modeling and virtual crack closure technique.
2.2 Sequence of Activities

In this work, the procedures of all the engineering activities are in line with the standard practices of aerospace industry. However, the demonstration of concept on full scale aircraft wing becomes difficult. That’s why a simple design of carbon fibre sandwich panels along with an aluminium attachment has been used as testing model, as shown in Fig.2-2. This approach is adopted to reduce the overall design space of the work. Such problems become complicated when the output of each parameter does not only depend on the independent design variables, but often on each other as well. Multidisciplinary design optimization concepts are generally proved on simpler models to achieve a better agreement among the parameters under consideration. [54], [55]
The overall methodology adopted in this work comprises of two phases. In the first phase aero-elastically improved design of test model was finalized using finite element based numerical simulations followed by the experimental validation and correlation of results. Dynamic stability was improved by setting up the fibre stacking sequence at ± 45° angles.

For this purpose two sets of tailored and un-tailored composite panels were manufactured and went under extensive experimentation. Structural dynamic characteristics such as natural frequency, damping, generalized stiffness and generalized mass against each vibrational mode of panels were measured for Flutter analysis. This analysis was performed to prove a positive change in aerodynamic stability of panels. Flutter speed was calculated for both the tailored and un-tailored panels. The schematic of various steps adopted in first phase is shown in Fig.2-3. Output of first phase is an experimentally verified aero-elastic design with altered structural dynamic characteristics.
In second phase, where the tailored and un-tailored test samples are available, the static and fatigue strength of both the designs was investigated. The objective of this phase was to investigate the effect of altered dynamic characteristics on structural integrity and fatigue life. Software based simulations for progressive damage analyses were carried out under applicable cyclic loading conditions for both the designs. For updating and fine tuning the FEM model, static response of both designs was also correlated with experimental results. Basic strength parameters and fracture properties of composites were experimentally determined in laboratory using genetic material coupons. Fig.2-4 shows the schematic flow chart adopted in phase-II.
2.3 Concluding remarks

At the end of this chapter it may be safely concluded that being at university level, the research was planned to ensure the working of engineering activities according to the existing aerospace practices. Because, this is a student level research work, so it may possible to get a compromise on some expensive tests, which may otherwise become mandatory while applying the scheme directly to airworthy aerospace products due to strict regulatory requirements for the qualification of flightworthy aircraft parts.
CHAPTER 3
MODELING AND SIMULATIONS

In this chapter, most of the modelling and simulation work involved in the thesis has been explained. First of all, 3D modelling of samples is described along with the technical issues being faced while performing the activity. Finite element model was basically developed to predict the fatigue response of samples. The FE model was correlated and calibrated with 02 types of experimental behaviour. Firstly, the FE model was calibrated and correlated with experimentally measured static response. Secondly, the same FE model got correlated with experimentally measured dynamic response. From aero-elastic point of view, the samples under investigation consist of 02 design types. The first type is un-tailored design and the second is tailored design. The scheme of aero-elastic tailoring used to get better dynamic stability has also been explained. Numerical simulations were also performed to predict the flutter speed of tailored and un-tailored design. The details about finite element model used for direct cyclic fatigue & damage tolerance analysis have been explained at the end of this chapter.

3.1 3D Modeling

An appropriate size of test samples was selected to demonstrate the whole exercise for the proof of concept. The overall dimension of carbon fibre sandwich panel was 490mm x 90mm with three plies of 1mm on both sides. 3mm hexagonal honeycomb aramid paper
(NOMEX) was used as a core material. Fig.3-1 shows the exploded view of the baseline composite sandwich model.

**Fig.3-1. 3D model of the sample**

Modeling the honeycomb core in a right way is very important from FE analysis point of view. Fig.3-2 shows three ways of hexagon array to cut the extruded solid material. With respect to a reference dimension of in-circle, horizontal and angular array is the correct way of modeling the honeycomb core. Whereas, the vertical array produces irrational dimensions among hexagons which further causes meshing error during finite element analysis. Modeling the composite samples for finite element analysis is quite different as compared to metals. In metallic parts, the modeling may work better in finite element model due to homogeneous material. If the material properties of any metallic part get matched with any FE model, the numerical simulation may predict accurate results at a large spectrum of inputs. Whereas in case of composites, the manufacturing process outcome is the actual product which has higher probability to differ in physical response under applied loads when compared to finite element model response.
Test samples were manufactured using carbon fibre layups followed by vacuum bagging process. Fig.3-3 shows the prepared test samples and generic coupons for fracture toughness testing. Keeping in view the hygro-thermal sensitivity of composite material, proper packing was ensured while transporting samples to testing facilities.

Two different FEM based software codes have been used to get the results of numerical simulations. In the first software modeling & simulation was supported in a same package, whereas .IGES format was used for the second software code.
3.2 The Finite Element Model (FEM) Development

An accurate, validated, correlated and updated finite element model is the foremost objective of any engineering analysis. The finite element (FE) based model was developed by using the same constituent material properties obtained through supplier’s data sheets and lamina level laboratory testing.

During manufacturing process, vacuum bagging was used to pull out excessive resin from the honeycomb core. However, 100% resin couldn’t pull out of core; this penetration makes the core material more strong. Fig.3-4 shows the penetrated resin at upper and lower side of core.
In this case, it became necessary to correlate the stiffness of FE model with the same structure as manufactured under real environmental conditions. In simulation software material properties table, the density of honeycomb material was increased in such a way to achieve same macro level response of the panels as the experimental result. This type of FE model correlation becomes essential due to deviation in manufacturing parameters. An ideal FE model cannot be used for realistic engineering analysis without its static or modal response correlation with the actual physical object. The difference between the designed models and manufactured items arises due to manufacturing process limitations. The actual product contains discontinuities which makes the difference in designed stiffness, damping, mass distribution and moments of inertia. As previously described in Fig.3-1, the sample consists of 03 layers of carbon fibre at the upper side and 03 layers at the lower side. The FE model required primarily the elastic moduli of each lamina and Poisson’s ratio. To fulfil this requirement, lamina level testing was performed.

Fig.3-4. Resin penetration in honeycomb core during manufacturing
3.2.1 The Finite Element Model Details

In the FEM modelling, 3D deformable shell planer elements were used to define the model. In material properties phase, elastic engineering constants were defined as mentioned in Table 3-5 (engineering constants of lamina), Table 3-6 (material properties of lamina) and Table 3-7 (mechanical properties of core material). Other analysis parameters were adjusted as following:

- In property module of the software, conventional shell composite layup was defined by giving individual ply details. Thickness of 1mm was defined for each ply along with relevant engineering constants. Ply rotation angles for untailed panels were defied as zero degree for upper plies and 90 degree for lower plies.
- In the assembly module, the plies were joined having instance type fixed at ‘Dependent Mesh’. In the step definition module, static general analysis was selected for static testing correlation. Non-linear effects of large displacements were kept as ‘Off’.
- In mesh module, the general purpose 3-dimensional deformable shell elements (S4R) were used because they provide accurate and robust solution in all loading conditions for thick and thin shell problems as their formulation incorporates thickness change effects as a function of in-plane deformation. [56] [57] Seeds were defined with approximate global size of 0.2mm. Curvature control having maximum deviation factor of 0.1 was introduced with approximate number of elements per circle: 8. Minimum size control was 0.1 by fraction of global size.
- In the load module, the fixed boundary condition was introduced at one end of the sample, i.e. ENCASTRE (U1 =U2 = U3 = UR1 = UR2 = UR3 = 0). Whereas, the load was applied as shell edge load at the same location described in Fig.3-6.
In model calibration activity the physical properties of finite element model were adjusted in such a way to match its behaviour with the manufactured test piece. To get high fidelity model, the models were calibrated using following two techniques:

### 3.2.2 FEM Calibration through Static Load

The panels were loaded as a cantilever beam using a test wall setup. Both the tailored and un-tailored panels were equipped with a strain gauge at root side and a linear variable displacement transducer (LVDT) at the tip side. A hydraulic actuator applied the force at tip side and the response was recorded using a properly configured and synchronised data acquisitioning system. The results were later got matched with the FE model response. Fig.3-6 shows the experimental test setup for finite element model calibration. The strain gages were applied to investigate the stresses developed in test specimen. The strain gage application configuration was half bridge, it compensates the temperature effects and also not sensitive to tensile loads. Fig.3-5 shows the configuration of strain gage used to measure the stresses during force application. The initial unloaded reading of strain gage was neglected due to the presence of some initial bridge unbalancing which sometimes occurs during gage application process.

![Fig.3-5. Strain gage installation configuration](image)

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Fig. 3-6. Structural test setup

Applied Static Load in Downward Direction
The instruments such as strain gauges and LVDTs were properly calibrated before the test. The hydraulic actuator was operated in position control at 20 mm per minute rate. Fig.3-7 shows the extent of linear elastic region calibrated with experimental response. As the samples are the sandwich panels, so there are two steps of load decrease ahead of the linear elastic region. The first down step in the machine load is the failure of lower ply and the second down step in the load is the progressive failure of upper ply of panel. The finite element model was correlated only within the linear elastic response of the panels. Primarily the stiffness characteristics were adjusted in such a way to get a best agreement between numerical and experimental response.

![Finite Element Model Correlation with Experimental Response](image)

**Fig.3-7. Static displacement model correlation within linear elastic limit**

The micro-strains were measured from upper and lower layer of the sample through half bridge configured uni-axial strain gage and its response was compared with the numerical simulations for the calibration of FE model. The micro-strain values were measured manually using P-3 strain indicator. The test data was acquired at constant lab temperature. The experimental data was considered as the reference for finite element mode. In Fig.3-8
static tip deflection measured through LVDT has been compared with the maximum achieved comparable response of finite element model at 15 different load values. Similarly, the correlated micro strain response is shown in shown in Fig.3-9.

![Static Deflection Correlation Curve](image1)

**Fig.3-8. Static deflection correlation curve**

![Micro-Strain Correlation Curve](image2)

**Fig.3-9. Micro-strain correlation curve**
In the following Table 3-1, the correlated values of both the static deflection and micro strain are enlisted along with the percentage errors. An average value of 11.2% absolute was achieved in deflection and 15.9% in micro strain response correlation.

**Table 3-1: Static test results correlation with static simulation results**

<table>
<thead>
<tr>
<th>Load Steps (Newton)</th>
<th>Z-Axis Deflection Experimental (mm x 10)</th>
<th>Z-Axis Deflection Simulation (mm x 10)</th>
<th>Deflection Percentage Error</th>
<th>X-Axis Micro Strain (Experimental)</th>
<th>X-Axis Micro Strain (Simulation)</th>
<th>Strain Percentage Error</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.000</td>
<td>0.000</td>
<td>0.000</td>
<td>0.000</td>
<td>0.000</td>
<td>0.000</td>
<td>0.000</td>
</tr>
<tr>
<td>20.000</td>
<td>31.503</td>
<td>31.749</td>
<td>-0.780</td>
<td>176.923</td>
<td>205.882</td>
<td>-16.368</td>
</tr>
<tr>
<td>40.000</td>
<td>53.906</td>
<td>58.605</td>
<td>-8.716</td>
<td>350.000</td>
<td>441.176</td>
<td>-26.050</td>
</tr>
<tr>
<td>60.000</td>
<td>91.010</td>
<td>101.782</td>
<td>-11.836</td>
<td>530.769</td>
<td>607.843</td>
<td>-14.521</td>
</tr>
<tr>
<td>80.000</td>
<td>185.014</td>
<td>166.067</td>
<td>10.241</td>
<td>553.846</td>
<td>460.784</td>
<td>16.803</td>
</tr>
<tr>
<td>100.000</td>
<td>101.017</td>
<td>113.249</td>
<td>-12.109</td>
<td>915.385</td>
<td>1078.431</td>
<td>-17.812</td>
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<td>120.000</td>
<td>218.623</td>
<td>258.344</td>
<td>-18.169</td>
<td>1007.692</td>
<td>1215.686</td>
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<td>140.000</td>
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<td>284.788</td>
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<td>20.685</td>
<td>1515.385</td>
<td>1541.176</td>
<td>4.594</td>
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<td>200.000</td>
<td>465.041</td>
<td>368.219</td>
<td>20.820</td>
<td>1776.154</td>
<td>2146.078</td>
<td>-20.827</td>
</tr>
<tr>
<td>100.000</td>
<td>450.045</td>
<td>409.132</td>
<td>9.091</td>
<td>606.923</td>
<td>698.431</td>
<td>-15.077</td>
</tr>
<tr>
<td>30.000</td>
<td>614.068</td>
<td>621.880</td>
<td>-1.272</td>
<td>346.154</td>
<td>323.529</td>
<td>6.536</td>
</tr>
<tr>
<td>50.000</td>
<td>970.090</td>
<td>818.264</td>
<td>15.651</td>
<td>438.462</td>
<td>501.234</td>
<td>-14.317</td>
</tr>
<tr>
<td>55.000</td>
<td>1028.899</td>
<td>849.554</td>
<td>17.431</td>
<td>450.009</td>
<td>498.543</td>
<td>-10.785</td>
</tr>
<tr>
<td>Average</td>
<td>Deflection Error</td>
<td>11.2</td>
<td>Micro Strain Error</td>
<td>15.9</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
3.2.3 FEM Calibration through Modal Analysis

To determine the dynamic response of panels, experimental modal analysis was performed with 27 miniature accelerometers. In Fig.3-10, experimental modal analysis setup is shown.

Fig.3-10. Experimental modal analysis setup

The sensor shown in the up corner of Fig.3-10 is a piezoelectric accelerometers along with low noise cable which is connected to the charge amplifier. While performing
the vibration testing, shaker and sensors have been used in such a manner without disturbing the natural characteristics of the structure under test. During calibration phase, before the actual test, it was ensured that the shaker produces stable sinusoidal output frequency characteristics. Further it was ensured that the force output was relatively independent of the vibrational amplitudes of the test samples. The alignment of force application point was re-adjusted for each mode by considering the results of numerical simulations. The shaker used in this test is electro-dynamic device which are based on the Laplace law [58] Table 3-2 and Table 3-3 represent the structural dynamic characteristics of both the un-tailored and tailored design respectively.

Table 3-2.Model characteristics of un-tailored panels

<table>
<thead>
<tr>
<th>Modes No.</th>
<th>Frequency (Hz)</th>
<th>Damping</th>
<th>Normalized Generalized Mass (kg m²)</th>
<th>Normalized Generalized Stiffness (kg m²/s²)</th>
<th>Modes Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>19.1000</td>
<td>0.00913</td>
<td>0.39011</td>
<td>0.1423972E+003</td>
<td>First bending</td>
</tr>
<tr>
<td>2</td>
<td>174.100</td>
<td>0.00526</td>
<td>0.57854</td>
<td>0.1082817E+003</td>
<td>Second bending with two nodal points</td>
</tr>
<tr>
<td>3</td>
<td>474.700</td>
<td>0.02945</td>
<td>0.23665</td>
<td>53.329543E+003</td>
<td>First lateral bending, low quality factor</td>
</tr>
<tr>
<td>4</td>
<td>671.200</td>
<td>0.01089</td>
<td>0.41668</td>
<td>0.1877087E+006</td>
<td>First torsion</td>
</tr>
<tr>
<td>5</td>
<td>1167.40</td>
<td>0.00435</td>
<td>1.00000</td>
<td>1.3623475E+006</td>
<td>Third bending with more nodal points</td>
</tr>
<tr>
<td>6</td>
<td>1280.70</td>
<td>0.00846</td>
<td>0.68585</td>
<td>0.3048500E+006</td>
<td>Second twist</td>
</tr>
</tbody>
</table>
### Table 3-3. Model characteristics of tailored panels

<table>
<thead>
<tr>
<th>Modes No.</th>
<th>Frequency (Hz)</th>
<th>Damping</th>
<th>Normalized Generalized Mass (kg m²)</th>
<th>Normalized Generalized Stiffness (kg m²/s²)</th>
<th>Modes Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>13.4600</td>
<td>0.00906</td>
<td>0.30180</td>
<td>0.0553725E+003</td>
<td>First bending</td>
</tr>
<tr>
<td>2</td>
<td>119.100</td>
<td>0.00426</td>
<td>0.54869</td>
<td>7.7835287E+003</td>
<td>Second bending with two nodal points</td>
</tr>
<tr>
<td>3</td>
<td>329.100</td>
<td>0.02965</td>
<td>0.27310</td>
<td>29.578843E+003</td>
<td>First lateral bending, low quality factor</td>
</tr>
<tr>
<td>4</td>
<td>684.700</td>
<td>0.01887</td>
<td>0.45486</td>
<td>0.2132267E+006</td>
<td>First torsion</td>
</tr>
<tr>
<td>5</td>
<td>997.800</td>
<td>0.00445</td>
<td>1.00000</td>
<td>0.9958675E+006</td>
<td>Third bending with more nodal points</td>
</tr>
<tr>
<td>6</td>
<td>1190.88</td>
<td>0.00746</td>
<td>0.65710</td>
<td>0.9319700E+006</td>
<td>Second twist</td>
</tr>
</tbody>
</table>

The stiffness of honeycomb structure was changed in such a way to match the modal characteristics of FE model with experimental results. Only the four natural modes of vibration were used to correlate with numerical simulation results i.e. the first, second, fourth and fifth mode. Table 3-4 shows the maximum agreement achieved between numerical simulation results and the experimental behaviour of un-tailored panels. The percentage difference between the numerical simulation results and the experimental results is the %age error. The reason for this error is because of the approximation of
numerical solution. However there is always a difference between the stiffness of actual model and a computer model which may be the cause of this error.

Table 3-4. Modal calibration agreement of un-tailored panel

<table>
<thead>
<tr>
<th>Description of Mode</th>
<th>Experimental value</th>
<th>Numerical Result</th>
<th>% Error</th>
</tr>
</thead>
<tbody>
<tr>
<td>First bending (M-1)</td>
<td>19.1000</td>
<td>17.90</td>
<td>-6.28%</td>
</tr>
<tr>
<td>Second bending (M-2)</td>
<td>174.100</td>
<td>167.3</td>
<td>-3.90%</td>
</tr>
<tr>
<td>First torsion (M-4)</td>
<td>671.200</td>
<td>579.0</td>
<td>-13.7%</td>
</tr>
<tr>
<td>Third bending (M-5)</td>
<td>1167.40</td>
<td>1210</td>
<td>+3.65%</td>
</tr>
</tbody>
</table>

Force appropriation method was used to isolate the vibrational modes of the panels. The acquired response consists of resonant and off-resonant components. The applied force at a single frequency is adjusted in such a way that off resonant components of the response are minimized and the respective mode is isolated accurately with almost nil contribution of other modes. After modal isolation using Force Appropriation Method[59], Complex Power Method[60] was used for extraction of modal parameters by giving micro frequency sweep around a particular resonant frequency. According to LIM standard definition “The rationale for the appropriation method is that the application of an ad-hoc generalized force (single signal applied at different actuators) allows the measurement of the response of a single normal mode. As other modes have small contributions, it is then possible to extract the modal characteristics of the considered mode (frequency $\omega_j$, damping ratio $\zeta_j$, modal mass $M_{jIN}$, mode shape $\phi$) with great accuracy.”

It was desired that all the normal modes should be orthogonal to each other. Modal Assurance Criteria (MAC) was utilized to evaluate the accuracy of experimentally
determined modes. MAC gave an indication of the correlation between two shapes. For ideal orthogonal modes, off diagonal terms of MAC matrix should be zero if compared to unit diagonal terms. However, for experimental modes its values up to 15-20% indicate modes are reasonably orthogonal. Fig.3-11 shows the orthogonality of experimentally determined modes.

![Graph showing orthogonality of each mode using MAC](image)

**Fig.3-11. Orthogonality of each mode using MAC**

\[ \Phi_i = \text{is the mode shape vector of the } i^{\text{th}} \text{ mode}, \Phi_j = \text{is the mode shape vector of the } j^{\text{th}} \text{ mode} \]

If, \( i = j \)  \( \text{MAC}=1 \) or 100%,

If, \( i \neq j \)  \( \text{MAC}=0 \)
As per standard LIM-A-8870C (AS) all the calculated off-diagonal elements of the orthogonality matrix should not be more than 10% of unit diagonal element.[61]

Quality Factor (QF) is measure of how well the off resonant components of the complex response are isolated from the resonant components. The relative importance of the measured imaginary and real response at resonance thus provides the information on the quality of force appropriation method.

\[
QF = 1 - (y_i / y) \quad \text{Equation 3-1}
\]

Where \(y_i\) is the off resonant component of response and \(y\) is the total response. The \(QF = 1\) represents the perfect appropriation of the mode. Minimum \(QF = 0.7\) has been taken as a good criteria for this particular testing.

### 3.3 Material Properties

Following generic mechanical properties of composite coupons were determined in laboratory:

1. Compressive & Ultimate Tensile Strength (UTS) of UD lamina as per ASTM standard D3039
2. Bending strength using 3 point & 4 point bend test as per ASTM standard D790
3. Inter laminar Shear Strength (ILSS) of laminate as per ASTM standard D2344

Table 3-5, Table 3-6 and Table 3-7 shows different properties used to develop a finite element model. The determined material properties were also compared with the values presented in various literature for reference purpose. If the determined value was found out of the range from published data, the test was repeated and accuracy of test was improved. Some less critical material properties were used in FEM model as such provided
by the supplier/OEM. However, the critical material properties were determined in laboratory through standard test methods.

Table 3-5. Mechanical properties through testing (Engineering Constants)

<table>
<thead>
<tr>
<th>Carbon Epoxy</th>
<th>Mechanical Property</th>
<th>Direction 1 ±Std. Deviation</th>
<th>Direction 2 ±Std. Deviation</th>
<th>Direction 3 ±Std. Deviation</th>
<th>Source</th>
</tr>
</thead>
<tbody>
<tr>
<td>Carbon Epoxy</td>
<td>Young’s moduli MPa</td>
<td>$E_1 = 153,700 \pm 21,000$</td>
<td>$E_2 = 8,960 \pm 1,500$</td>
<td>$E_2 = 8,960 \pm 1,500$</td>
<td>Testing</td>
</tr>
<tr>
<td>Carbon Epoxy</td>
<td>Poisson’s ratios</td>
<td>$v_{12} = 0.31$</td>
<td>$v_{13} = 0.31$</td>
<td>$V_{23} = 0.52$</td>
<td>Testing</td>
</tr>
<tr>
<td>Carbon Epoxy</td>
<td>Shear moduli MPa</td>
<td>$G_{12} = 4,980 \pm 690$</td>
<td>$G_{13} = 4,980 \pm 690$</td>
<td>$G_{23} = 2,600 \pm 280$</td>
<td>Testing</td>
</tr>
</tbody>
</table>

Table 3-6. Mechanical properties of lamina

<table>
<thead>
<tr>
<th>Mechanical Property</th>
<th>Value</th>
<th>Units</th>
<th>Source</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass Density</td>
<td>1020</td>
<td>kg/m$^3$</td>
<td>Data sheet</td>
</tr>
<tr>
<td>Thermal Expansion Coefficient in 90°</td>
<td>28</td>
<td>Strain/K</td>
<td>Data sheet</td>
</tr>
<tr>
<td>Thermal Expansion Coefficient in 0°</td>
<td>-0.3</td>
<td>Strain/K</td>
<td>Data sheet</td>
</tr>
<tr>
<td>Moisture Expansion Coefficient in 90°</td>
<td>0.3</td>
<td>Strain/K</td>
<td>Data sheet</td>
</tr>
<tr>
<td>Moisture Expansion Coefficient in 0°</td>
<td>-0.01</td>
<td>Strain/K</td>
<td>Data sheet</td>
</tr>
<tr>
<td>Thermal Conductivity</td>
<td>0.2256</td>
<td>W/(m·K)</td>
<td>Data sheet</td>
</tr>
<tr>
<td>Specific Heat</td>
<td>1386</td>
<td>J/(kg·K)</td>
<td>Data sheet</td>
</tr>
</tbody>
</table>
### Table 3-7. Mechanical properties of core material

<table>
<thead>
<tr>
<th>Honeycomb</th>
<th>Mechanical Property</th>
<th>Value</th>
<th>Units</th>
<th>Source</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Young’s moduli Compressive</td>
<td>0.418</td>
<td>GPa</td>
<td>Data Sheet</td>
</tr>
<tr>
<td></td>
<td>Shear Modulus</td>
<td>0.0538</td>
<td>GPa</td>
<td>Data Sheet</td>
</tr>
<tr>
<td></td>
<td>Poisson’s ratios uxy</td>
<td>0.49</td>
<td>Nil</td>
<td>Data Sheet</td>
</tr>
<tr>
<td></td>
<td>Compressive Yield Strength</td>
<td>12.1</td>
<td>MPa</td>
<td>Data Sheet</td>
</tr>
<tr>
<td></td>
<td>Shear Strength</td>
<td>3.24</td>
<td>MPa</td>
<td>Data Sheet</td>
</tr>
<tr>
<td></td>
<td>Thermal Expansion Coefficient</td>
<td>19.4</td>
<td>μ Strain/F&lt;sup&gt;0&lt;/sup&gt;</td>
<td>Data Sheet</td>
</tr>
<tr>
<td></td>
<td>Density</td>
<td>0.128</td>
<td>g/cc</td>
<td>Data Sheet</td>
</tr>
</tbody>
</table>

#### 3.4 Modal Analysis Simulations

First six natural frequencies of the panels were considered significant to analyse the Flutter behaviour. Generally, the lower frequency modes are excited while normal operational environmental conditions of aircraft. However, FEM based modal analysis simulation may extract as many modes as the designer requires. Most of them may not be useful if considered during flutter prediction calculation and may also consume a lot of computational time. First 6 mode shapes determined through numerical simulations of carbon fibre sandwich panel are shown in Fig.3-13.
The FEM Model was analyzed using Abaqus Software. Fig. 3-12 show the GUI of Abaqus Software version 6.12.

Fig. 3-12. FEM analysis in Abaqus software
First six natural frequencies and respective mode shapes of the panels considered significant to analyse the Flutter behaviour are shown in Fig.3-13.
3.5 Aero-elastic Tailoring Scheme

Aero-elasticity is an interaction of aerodynamic, elastic and inertial forces. Structural and aerodynamic behaviour of wing are usually optimised by varying composite fibre angles and layup sequence. Few parameters which are needed to be considered while tailoring are the lift to drag ratio, static strength, flutter and divergence speeds and flight control. Significant work has been done in aero-elastic tailoring since early 1990s and it was first carried out on X-29 aircraft.[62] It was found in the latest research work that these fibre paths increase the dynamic structural performance of VAT panels than that of equivalent uni-directional laminates. [63], [64]

The scheme involved in this work is the change in directional stiffness of panels and the passive aero-elastic coupling between bending and torsion modes. In these composite panels, the fibre angles and layup sequences have been tailored to optimize bending-torsion coupling over entire span. The process was constrained to some extent by the limitation of non-availability of toe steering equipment. In toe steering equipment, the fibres are forced to vary their direction continuously using specialized machines. However, the fibres were arranged to follow a curvilinear paths manually such that the fibre angles and lamina stiffness changes continuously through the plane of each lamina. The fibres were placed in such a way to maintain minimum orthogonality against maximum eigenvectors of banding and torsional modes of the panels. In this way the optimum [+45/-45/0]s stacking sequence was adopted for tailored design and [03/903] adopted for un-tailored design. In first two plies of tailored design, the fibre angle gradually changes from 00 to + and – 45° as the length increases from root to tip. Modal response of both designs was investigated in experimental vibration testing.
Considering the displacement of the aerodynamic control points as $q_a$, the resultant aerodynamic forces at the aerodynamic boxes due to this displacement is defined as following:

$$F_a = q_\infty A(ik)q_a$$  \hspace{1cm} \text{Equation 3-2}

Where, $q_d$ is the dynamic pressure, and $k = \omega b/U$ is the reduced frequency, where $\omega$ is the harmonic frequency, $U$ is the free stream velocity, and $b$ is the reference semi-chord. The aerodynamic influence coefficient matrix $A(ik)$, is a function of the reduced frequency (only incompressible flow is considered here). The generalized external forces are then given by following relation:

$$F_{ext} = \phi_e G_s^T F_a = q_\infty \phi_e A G_s^T A(ik)q_a$$  \hspace{1cm} \text{Equation 3-3}

To complete the formulation of aero-elastic un-damped equation of motion, the displacement $q_a$ is substituted by $\Phi_a \eta = G_s \Phi_e \eta$ and the above equation becomes:

$$\tilde{F}_{ext} = q_\infty \phi_e A G_s^T A(ik)G_s \phi_e \eta$$  \hspace{1cm} \text{Equation 3-4}

In ZONA Tech., a generalized aerodynamic influence coefficient matrix is defined as:

$$Q_A(ik) = \phi_e^T \phi_e A G_s^T A(ik)G_s$$  \hspace{1cm} \text{Equation 3-5}

Both the tailored and un-tailored panels design was analyzed through Flutter analysis which was performed using a commercial software code.

3.6 Flutter Analysis

Flutter Analysis was performed to check the instability of panels subjected to different Mach from 0.5 to 1.2 at different altitudes. Velocity vs. Frequency (V-F) and Velocity vs. Damping (V-G) curves of un-tailored design at 0.7 Mach number are presented in Fig.3-14 and Fig.3-15 respectively. Whereas, the V-F curves of tailored design are
presented in Fig.3-16. The air speed at x-axis is the equivalent air speed mentioned in meter per seconds. It is obvious in Fig.3-14 that 02 natural frequencies are being coupled at 1500 m/s speed. The coupling modes are the bending and torsional. In Fig.3-15, the damping of same modes is decreasing.

![Fig.3-14. V-F curves for un-tailored panels at Mach 0.7](image)

It may be observed in Fig.3-16 that the coupling of same modes of vibration is happening at 1900 m/s speed, which is the indication of better dynamic stability.
Flutter analysis was carried using ZONA6 and ZONA7 for subsonic and supersonic respectively. ZONA7 is the modified form of ZONA51 employs the acceleration potential approach for thin plate type of lifting surfaces. Software’s flutter module contains two flutter calculation techniques, the K-method and the g-method. The g-method is
generalized K-method and P-K method for true damping predictions. P-K is only valid at the condition of zero damping, zero frequency or linearly varying generalized aerodynamic forces (Qij) with respect to reduced frequency. In fact, if (Qij) is highly nonlinear, P-K method may produce unrealistic roots due to its inconsistent formulation. Prior to the computation of flutter predictions, transformation of modal data over aerodynamic grid was also carried out and mode shape animations were checked to verify the spline of sensors over the aerodynamic grid. To be more conservative during Flutter analysis, zero structural damping was considered for the analysis purpose.

![Fig.3-17. Flutter speed on realistic flight envelop at Mach 0.7](image)

As shown in VF & VG curves, two modes are critical, the bending mode and the torsion mode. Un-tailored design shows the flutter air speed at 1500m/s Veas (2900 Knots) whereas tailored design shows it at 1900m/s Veas (3700 Knots). However from Fig.3-17, it can be seen that this speed is far from achievable region even at 0 meters height.
3.6.1 Working Principle of Dublett-Lattice Method

In the present work, a commercially available software module is used for the aeroelastic analysis. Dublett-Lattice Method (DLM) has been used due to its simplicity in computations, especially when we are dealing with the complex configurations. In this method, a pair of continuous pressure doublet sheet are replaced by a set of lattice pairs with finite length. The programming simplicity is obtained when all panels are treated equally, independent of their proximity to panel limits (leading and trailing edges). The lattice is placed at \(1/4\) of the chord length at each panel and the up-wash \(\omega(x, y, z)\) is calculated at \(3/4\) of the chord length, in the middle span of each panel. DLM is used to perform the unsteady calculations, since it is fast and corrections can be included with comparatively little effort. The sample is modeled as a rectangular half with a symmetry plane. DLM relates the downwash and the pressure distribution \(\Delta C_p\) through an aerodynamic influence coefficients matrix \(\text{AIC}\)

\[
w_\alpha = \text{AIC}^{DLM} \cdot \Delta C_{p\alpha}^{DLM}. \tag{3-6}
\]

The experimental equivalent is

\[
w_\alpha = \text{AIC}^{Exp} \cdot \Delta C_{p\alpha}^{Exp} \text{ with } \text{AIC}^{Exp} := C \cdot \text{AIC}^{DLM}. \tag{3-7}
\]

Therefore, the elements of the diagonal correction matrix \(c_{ii}\) can be calculated as

\[
c_{ii} = \frac{[w_\alpha]_i}{[\text{AIC}^{DLM} \cdot \Delta C_{p\alpha}^{Exp}]_i}. \tag{3-8}
\]

Directly transferred to the heave results, the corrected pressure differences read:

\[
\Delta C_{p_h}^{DLM, direct} = (C \cdot \text{AIC}^{DLM})^{-1} \cdot w_h. \tag{3-9}
\]
One advantage of DLM is the high computational efficiency. However, DLM is only reliable for subsonic, attached flow. Since DLM cannot predict viscous effects and transonic phenomena such as recompression shocks, corrections have to be introduced.

### 3.6.2 Validation of Software Code

The commercially available software code used to predict flutter speed of panels was first validated against reliable known data. The reference data publication used to validate the software provides both the input and output parameters of flutter analysis. The standard wing planform employed had an aspect ratio of 4.0, a taper ratio of 0.6, and 45° of quarter-chord sweepback angle. Following is the description of reference report:

“North Atlantic Treaty Organization Advisory Group for Aerospace Research and Development AGARD

Report No. 765 AGARD Standard Aero-elastic Configurations for Dynamic Response

I - Wing 445.6

By: E. Carson Yates, Jr

Interdisciplinary Research Office NASA Langley Research Center Hampton, VA 23665-5225, USA”

Experimentally proved ‘Frequency vs. Speed” and “Damping vs. Speed” curves given in the reference report are measured against following 05 input frequencies of the wing:

- Mode-1 Frequency at 14.1201 Hz
- Mode-2 Frequency at 50.9125 Hz
- Mode-3 Frequency at 68.9416 Hz
- Mode-4 Frequency at 122.2556 Hz
- Mode-5 Frequency at 160.5492 Hz
3.7 FEM for Direct Cyclic Fatigue & Damage Tolerance Analysis

The FE analysis was performed in Abaqus (Version 6.13) to compare the fatigue life of samples against mixed mode loading spectra keeping other parameters constant in all cases under investigation. The load cases are discussed in Table 5-2 & Table 5-3.

- **Assumption:** Enhanced VCCT criteria has been used to investigate the damage tolerance which is based on the assumption that the strain energy released when a crack is extended by a certain amount is the same as the energy required to close the crack by the same amount as shown in

The enhanced VCCT criterion is very similar to the original VCCT. As in the original VCCT criterion, the fracture criterion is the general case involving Mode I, II, and III.

\[ f = \frac{G_{\text{equiv}}}{G_{\text{equivC}}} \geq 1.0. \]

Equation 3-10

Where \( G_{\text{equiv}} \) is the equivalent strain energy release rate calculated at a node, and \( G_{\text{equivC}} \) is the critical equivalent strain energy release rate calculated based on the user-specified mode-mix criterion and the bond strength of the interface. The crack-tip node will de-bond when the fracture criterion reaches the value of 1.0.

Actual sample consists of 6 layers above and below the honeycomb layer, the crack growth possibility was considered as in-plane between any of the two layers. For numerical simulation purpose first two layers have been considered for crack growth analysis. The static testing has been performed at lamina level to for material properties of composite layers. However the honeycomb properties get changed after excessive penetration of resin. In this way the actual honeycomb properties to be used in FEM remains questionable until the FEM calibration activity is performed. In this activity, mainly the honeycomb layer’s
properties were altered to get the similar response of panels as demonstrated during experimental testing as shown in Fig.3-18.

![Fig.3-18. Source of material properties for fracture analysis](image)

3.7.1 Part Definition Module

The sample part having 90mm x 490 mm dimensions was modeled using 3D deformable Shell elements as shown in Fig.3-19. Its configuration is similar to CLS specimen undergoing mainly the in-plane loading and the crack growth i.e. XY plane. However 15% load has also been applied in Z direction.

![Fig.3-19. Part definition showing the damage, loading and boundary conditions](image)

Master and Slave surfaces were defined between 02 layers of the sample. Potential crack surfaces were modeled as slave and master contact surfaces. The predetermined crack surfaces were assumed to be initially partially bonded so that the crack tips can be
identified. A set of nodes was identified which makes the initially bonded part of the slave surface only.

3.7.2 Property Module

In this Module, CFRP material elastic lamina properties were defined with the same properties determined through static testing at lamina level as shown in Table 3-5. Shell Homogeneous sections were assigned to both parts with 1mm shell thickness. In ‘Assign Section’ tab, bottom surface of first lamina and top surface of second lamina was selected because the contact has to be defined between these two surfaces.

3.7.3 Assembly Module

In this module both the lamina were glued together. Datum points were created to define the crack tip position and load application points at free edge of lower surface.

3.7.4 Field Output Definition

In the field output request window, following options were selected as shown in Fig.3-20:

- DBT, time at bond failure
- OPENBC, opening behind crack tip at bond failure
- ENRRT, strain energy release rate
- EFENRRTR, effective energy release rate ratio
- BDSTAT, bond state etc.
3.7.5 Interaction Module

It is one of the most important module in composite material analysis regarding crack growth. In this module, the contact definition was carried-out as shown in Fig.3-21.

- Standard surface to surface contact was defined.
- Sliding formulation was selected as ‘Small sliding’.
- Default values selected for ‘Slave adjustment’, ‘Surface smoothing’, and ‘Bonding’
- In ‘Clearance’ tab, Uniform clearance value of slave surface was selected as 1e-8.

In contact interaction properties, following parameters were defined:

- Fracture criteria: VCCT
- Direction of crack growth relative to local direction: Maximum tangential stress
- Mixed mode behaviour: BK
- Tolerance: 0.2
- Viscosity: 0.0
If it would have been unstable de-bonding scenario, the specific tolerances for unstable crack propagation may be enabled which helps the model convergence in a better way. However, this option has not been used here. In this analysis shell elements are used which generally do not create mesh convergence issues. [65] The critical strain energy release rates were selected on the basis of testing (for Mode-I) and literature (for Mode-II & III ratio to Mode-I) [66] [67] [68]

- Critical Strain Energy Release Rate(Mode-I): 0.6 KJ/m²(Nmm⁻¹)
- Critical Strain Energy Release Rate(Mode-II): 1.7KJ/m²(Nmm⁻¹)
- Critical Strain Energy Release Rate(Mode-III): 1.7KJ/m²(Nmm⁻¹)
- Exponent ‘n’ for BK model calculations: 1.6

3.7.6 Mesh Elements Definition

As the first-order elements generally work best for crack propagation analysis, so the three dimensional deformable shell element (S4R) has been used in this analysis.
Equally distributed mesh spacing was used. Both surfaces were meshed with same element and bonded together over entire span except the edges used for boundary conditions. Mesh convergence was achieved at 0.2 mm mesh spacing when a further refinement of 0.1 mm required significantly more simulation time with less than 1% difference in results. As depicted in Fig.3-22. Shell elements have an advantage to reduce overall simulation time due to less number of nodes. Once the convergence of mesh is achieved with reasonable confidence, the rest of iterations were performed at the same value of mesh size.

![Mesh convergence graph](image)

**Fig.3-22. Mesh convergence graph**

### 3.7.7 Boundary Conditions and Interactions

In the boundary conditions of the specimen, one edge of the specimen kept fixed, whereas, the opposite edge was free to move. The fixed edge was fully constrained except the translational degree of freedom in Y direction. A continuum distributing coupling was defined in constraints as shown in Fig.3-23. No rotational degree of freedom was selected.
Bonded nodes were also defined in Interaction Module. Total 10044 bonded nodes were created, it excluded the boundary nodes where the load and fixed degree of freedom is required to be applied. A concentrated load was defined in Load Manager against the one of the static mean loads of spectrum.

3.7.8 Crack Definition

The crack was defined in interaction module. In ‘Special’ tab, crack was created using VCCT deboning option. Initiation step and contact pair interaction was also selected as it has already been defined. De-bonding force may be set at ‘Step’ or ‘Ramp’ because we may not see the de-bonding happening in this case. However, in this scenario the ‘Step’ force was selected.
3.7.9 **Amplitudes Definition**

Periodic loading amplitudes were defined in ‘Amplitude Definition Module’ which acts as superimposed loading on the static mean loads. Against a certain step time the following parameters were defined in Amplitude module:

- Circular frequency: $2 \times \pi \times F$, where $F$ is the frequency in Hz
- Starting time: 2 sec at each occurrence
- Initial Amplitude: ‘Nz’ occurrence mean load
- Parameter A: Cosine Factor
- Parameter B: Sine Factor

3.7.10 **Step Definition**

The crack propagation capability was activated within the step definition to specify that crack propagation may occur between the two surfaces that are initially partially bonded. 02 steps were defined in ‘Step Manager’ of Abaqus. The first is static general step and the second one is direct cyclic step. Direct cyclic load has been implemented after static general step. Basic options are defined as following:

- Static general time period was defined as 0.2 sec
- Direct cyclic time period was defined as 2 sec.
- The nonlinear geometry option was kept off because Abaqus is not able to deliver nonlinear geometric behaviour with direct cycles.

Incrimination option was selected as following:

- Fixed increment was selected with maximum number of increments: 100 for static general and 10000 for direct cyclic.
• Increment size: 0.02

• Maximum number of iterations: 3 (Higher number of iterations were not selected because the nonlinear behaviour of structure has not been supported in direct cyclic fatigue analysis)

Number of Fourier Terms were increase to the following parameters:

• Initial: 50

• Maximum: 50

• Increment: 5 (Default Value)

Fatigue Option was defined as following:

• Low cycle fatigue analysis option has been included.

• In the forward damage estimation, the maximum and minimum cycle increment size was set as 3 and 6 respectively.

• Maximum number of cycles were selected as 10,000 cycles.

• Damage extrapolation tolerance was set as: 1

Calculation Options are following:

• In the equation solver, default matrix storage was selected.

• The ‘Step Manager’ of Abaqus shows total 02 steps.

• The first is static general step and the second one is direct cyclic step.

3.7.11 Paris Law Implementation; the Illustrative of Phenomenon

Progressive delamination growth at the interfaces in laminated composites subjected to sub-critical cyclic loadings can be simulated by using low cycle fatigue
criterion. This criterion can be used only in a low-cycle fatigue analysis using the direct cyclic approach. [69] [70] [71] The onset and delamination growth are characterized by using the Paris law, which relates the relative fracture energy release rate to crack growth rates as illustrated in Fig.3-24.

![Fatigue crack growth explained by Paris law](image)

**Fig.3-24. Fatigue crack growth explained by Paris law**

The fracture energy release rates at the crack tips in the interface elements are calculated based on the above mentioned VCCT technique. [72] The Paris regime is bounded by the energy release rate threshold, $G_{\text{thresh}}$, below which there is no consideration of fatigue crack initiation or growth, and the energy release rate upper limit, $G_{\text{pt}}$, above which the fatigue crack will grow at an accelerated rate. $G_c$ is the critical equivalent strain energy release rate calculated based on the user-specified mode-mix criterion and the bond strength of the interface. [73] In a low-cycle fatigue analysis the onset of the fatigue crack growth criterion is characterized by $\Delta G$, which is the relative fracture energy release rate when the structure
is loaded between its maximum and minimum values. [74] The fatigue crack growth initiation criteria is defined as shown in following Equation 4.7:

$$ f = \frac{N}{C_1 \Delta G_{c_2}} \geq 1.0 $$

Equation 3-11

Where, $C_1$ and $C_2$ are material constants and $N$ is the cycle number. The interface elements at the crack tips will not be released unless the above equation is satisfied and the maximum fracture energy release rate, $G_{max}$, which corresponds to the cyclic energy release rate when the structure is loaded up to its maximum value, is greater than $G_{thresh}$.

Once the onset of delamination growth criterion is satisfied at the interface, the delamination growth rate, $da/dN$, can be calculated based on the relative fracture energy release rate, $\Delta G$. The rate of the delamination growth per cycle is given by the Paris law if the following condition is met:

$$ G_{thresh} < G_{max} < G_{pl}, $$

Equation 3-12

The rate of crack growth is defined as following:

$$ \frac{d\alpha}{dN} = C_3 \Delta G^{C_4}, $$

Equation 3-13

Where, $C_3$ and $C_4$ are material constants.

If $G_{max} > G_{pl}$, the interface elements at the crack tips will be released by increasing the cycle number count, $dN$, by one only. In Paris law the values of $C_1$ & $C_2$ are used to calculate crack onset. $C_3$ & $C_4$ are used for crack propagation. Reliable source for fatigue crack growth parameters data was used for this analysis. [67] The parameters of fatigue include $C1 = 2.8e-9$, $C2 = -12.415$, $C3 = 2.44e+6$, $C4 = 10.61$, $r1 = 0.353$ and $r2 = 0.9$. The applied fatigue load spectra details are presented in Table 5-2, Table 5-3 and Fig.7-2 to Fig.7-5.
3.7.12 Job Definition Module

A job was defined with all the above mentioned parameters and submitted for solution run.

3.7.13 Postprocessor Visualization Module

Total energy release rate was calculated by adding the parameters denoting energy release rate 1, energy release rate 2 and energy release rate 3 i.e. ENRRT11+ ENRRT12+ ENRRT13. In this way, a session step is created in Abaqus. The GUI is shown in Fig.3-24.

![Fig.3-24. Total calculated strain energy release rate](image)

As already discussed under heading 3.7.8, the crack was defined using VCCT deboning option. Fig.3-26 shows the crack tip generated by un-bonded nodes of two layers. The images showing nature of damage during experimental crack growth in DCB specimen are presented in Fig.4-11 and Fig.4-12, indicating pure delamination without fibre breakage.
3.8 Concluding Remarks

The numerical simulations work involved in the plan has been carried out using commercially available software codes. Before getting useful results through finite element based numerical simulation, the FE model has been correlated and calibrated with the experimentally determined static and dynamic response of samples. The dynamic stability of samples has also been improved by altering the ply angles. The tailored design of samples has been verified through Flutter analysis simulations. FE model to predict direct cyclic fatigue & damage tolerance behaviour of samples has also been developed. The results of analysis are presented in chapter 6.
CHAPTER 4

EXPERIMENTAL WORK

In this chapter, most of the experimental work involved in the thesis has been explained in detail. Keeping in view the criticality of mode-I crack growth, the details of experimental procedure followed to measure critical strain energy release rate in crack opening direction \( (G_{IC}) \), has been explained first. The details of test specimen, testing parameters and test data analysis schemes have been presented. Because the sample is a sandwich panel, it contains unidirectional plies of carbon fibre at both sides of honeycomb core. The static testing was also performed to measure the elastic moduli and allowable yield limit and ultimate strength of plies. Vibration testing details of the samples have also been discussed in this chapter. The significant vibration testing techniques are also explained such as, modal isolation technique, modal parameter extraction technique and orthogonality check for the extracted modes.

4.1 Fracture Toughness Testing

For high fidelity engineering analysis, standard and reliable material characterization data of laminated composites is also needed as input to analytical models of structures to predict onset of failure. Fracture toughness is one of the most important parameter to analyse fatigue strength. Double cantilevered beam specimen was used with laminated composites to measure fracture toughness and strain energy release rate data under cyclic
loading. Generic specimen test data of composite material is not only applicable to a single component but in multiple field applications. [75], [76]Fig.4-1 shows one of the double cantilever beam specimens used for interlinear fracture toughness testing of unidirectional lamina.

During mathematical modelling section, it is illustrated that the ply angle of laminated composite, is directly related to the stiffness of structure. Similarly the stiffness of structure is also directly related to the strain energy release rate of the laminated composites. Keeping in view these direct relationships, the stiffness is omitted from both sides of the relationship and the fibre angles have been directly correlated with the strain energy release rates of the composite panels. The fracture toughness can be predicted for composite lay-up in terms of the elastic constants and the fiber failing strain. Keeping in view the significance of fracture toughness of material for damage tolerance analysis, this property has been carefully determined in laboratory environment under controlled conditions.

**Fig.4-1.Double Cantilever Beam (DCB) test specimen**

Inter-laminar fracture toughness is one of the most important material properties for damage tolerance analysis. Composite structures are usually prone to inter-laminar fracture that is the failure between the lamina or plies. Common cause of these failure is the impact load or the stresses on pre-existing cracks due to shear load. In-plane shear load usually
leads to Mode-II fracture and the compression loads lead towards layers buckling.[77] Similar to Mode-I, the Mode-II inter laminar fracture toughness is also very important material characterization in fracture mechanics. There are two basic configurations for the DCB specimens, the constant width and the width tapered DCB (WTDCB). The width of the tapered specimen is designed so that a/b is constant, where ‘a’ is crack length and ‘b’ is specimen width. The strain energy release rate derived from the analysis of a WTDCB specimen is independent of crack length, and the crack grows under constant load. Fig. 4-2 shows the testing of mode-I inter-laminar fracture toughness testing under ambient environment. Due to non-availability of thermal chamber, the hygro-thermal effects have not been measured experimentally. However, reliable material database have been used for reference where applicable.

Fig. 4-2. Double Cantilever Beam (DCB) test specimen undergoing testing
4.1.1 Test Specimen for $G_{IC}$

As it has already been discussed that all the testing involved in this research work was performed as per ASTM standard. Double Cantilever Beam (DCB) test was used for determination of Mode-I inter-laminar fracture toughness. It is important to mention that the DCB test has been standardized by ASTM only for unidirectional composites.[78] However, these guidelines are also used for testing of woven fibre composites.[79], [80] The geometry and dimensional parameters of the test coupon model is shown in Fig.4-3. Specimen dimensions are $B=25\text{mm}$, $h=3\text{mm}$, $L=135\text{mm}$, $a=30\text{mm}$, All the dimensions were set according to ASTM Standard D5528.

![Fig.4-3. Standard dimensional parameters of test coupon](image-url)
4.1.2 Testing Parameters

A rectangular specimen was used with constant thickness $h$, width $B$, and length $L$. The specimen was produced with pre-implanted and non-adhesive Teflon tape which serves as an initial crack. Keeping in view the Mode-I pattern, an opening load was applied using special loading strips at the delaminated end of the coupon. The real time applied load and displacement data was recorded.

- 0.0004in (10.16 Micron) thin Teflon paper was used as a delamination insert to give fairly a sharp crack tip. The tape was inserted in the sample while manufacturing.
- Load was applied in displacement control mode of machine. Loading rate = 1mm per minute. In lower loading rate crack tip monitoring becomes easy even with naked eye. A microscopic observation may further reduce the reading errors.
- When fracture started, the coupon was unloaded at 90% of the critical load and the growth of delamination was measured.
- 10 mm equally spaced vertical tick marks on both sides of coupon were used as a reference of crack growth measurements.
- The measurement of delamination was recorded with the help of equally spaced vertical marks on both sides of the specimen, starting from the tip of initial crack.
- While initial loading, 3.3mm (0.13in) measurement difference was observed from both sides of coupon. So the test was stopped and the fixtures were realigned.
- DCB test data was quite stable when the specimen was fully unloaded for re-alignment.
- At the split-up, the coupon was examined under optical microscope for any noticeable imprint against successive load steps to correct the recorded crack extension values.
4.1.3 Special Considerations of Testing

- The loading tabs were glued to the coupon with a polyurethane adhesive which did not de-bond under the applicable testing load.

- The loading tabs were carefully aligned with the specimen to ensure uniform crack growth. Generally, the resultant measured crack growth becomes un-equal at sides.

4.1.4 Test Data Analysis

There are three common methods for DCB test data reduction; the first is Modified Beam Theory (MBT), the second one is Compliance Calibration (CC) and the third method is modified compliance calibration (MCC). Strain energy release rate has been calculated using all the three formulations. In Fig.4-6, the results of strain energy release rate calculated using all the three methods are plotted against crack growth on a single graph.

In MBT approach, it is assumed that classical beam theory is applicable in stresses and deformations calculation. So, both the arms of the double cantilever beam specimen are considered as cantilever beams. Using this method, the relationship for deflection of beam tip can be calculated as following:

\[ \delta = \frac{(Pa^3)}{(3EI)} \]  \hspace{1cm} \text{Equation 4-1}

Where, E is the flexural modulus of elasticity, and I is the area moment of inertia of a single arm. Using classical theory, the compliance of the whole specimen is following:

\[ C = \frac{2\delta}{P} = \frac{(2a^3)}{3EI} \]  \hspace{1cm} \text{Equation 4-2}
The displacement per unit applied force is called the compliance, $C$. It is invers of the slope of the $Q$ (Load) vs. $q$ (Displacement) curve. Fig.4-4 shows the graph between normalized load and the vertical displacement. Crack extension relation with displacement is shown in Fig.4-5. The equation for $G$ doesn’t depend on whether the loads or displacements are fixed. However, for fixed displacement, $G$ is a decreasing function of crack length ‘$a$’ while for fixed load, $G$ increases with crack length.

$$G = \frac{1}{2} Q^2 \left(\frac{\partial C}{\partial s}\right)$$  \hspace{1cm} \text{Equation 4-3}

Differentiating compliance equation w.r.t. crack length and substituting into above equation gives the following relationship;

$$G = \frac{P^2 a^2}{BEI}$$  \hspace{1cm} \text{Equation 4-4}

This equation can be made independent of $E$ and $I$ values by substitution of Eq.5.1 into Eq.5.3. The final result gives an expression for mode-I inter-laminar fracture toughness;

$$G_{IC} = \frac{3P c \delta c}{2Ba}$$  \hspace{1cm} \text{Equation 4-5}

This relation of fracture toughness depends only on the specimen geometry, critical load $P_C$ and displacement $\delta_C$. As per ASTM Standard D5528, above mentioned relationship will overestimate $G$ value, because in actual the arms of double cantilever beam specimen undergoes a finite rotation at the delamination front. To address this issue, the crack length used in above relation is usually increased by a correction factor “$\Delta$” giving a new length as $(a + \Delta)$. The new crack length when substituted into above Equation, the new relationship becomes the following.

$$G_{IC} = \frac{3P c \delta c}{2B (a + \Delta)}$$  \hspace{1cm} \text{Equation 4-6}
\( G_{IC} \) may be related with \( K_{IC} \), by using the following relationship.

\[
G_{IC} = (K_{IC})^2 / E
\]

Equation 4-7

Fig. 4-4. The experimental results (Load vs. Displacement)
Fig. 4-5. The experimental results (Crack extension vs. Displacement)

Fig. 4-6. Material Property (Strain energy release rate)
4.2 Static Testing

Static strength properties were also determined by experimental testing. Table 4-1 shows the static strength related properties of composite material under investigation. Fig.4-7 shows the physical samples used to determine basic mechanical properties in addition to fracture toughness testing. Following strength properties of the composite samples were determined in different planes.

- Tensile strength
- Compressive strength
- Shear strength

4.2.1 Test Specimen Dimensions

250 x 25.4 mm (standard dimension as per ASTM D-3039) rectangular coupons were used for tensile testing. For 0° fibre orientation angle, specimen width was 12.7mm and gage length 138 mm, whereas for 90° fibre orientation angle, specimen width was 24.4 mm and gage length 125mm.

For flexural testing, ASTM standard D790 was followed. Length of specimen is dependent upon thickness. Span to thickness ratio of 16 to 1 was selected. 3 point flexure fixture with variable span setting options was used to carry out the test. Due to non-availability of reflectometer, only the load versus machine cross head travel was recorded and used for 3 point bend calculations. For unidirectional carbon fibre with 0° fibre orientation angle length to thickness ratio (L/t) was 38/1. For unidirectional carbon fibre with 90° fibre orientation angle length to thickness ratio (L/t) was 24/1. Three samples for each type of testing were used. Tests carried out at room temperature in lab environment.
Fig. 4-7 (b). Standard composite samples for tensile and flexural testing

The tensile, compressive and shear strength of composite coupons in third direction were not determined in laboratory. However the same average percentage difference of these strength values was incorporated in the standard published database which was calculated after testing of other parameters. Fig. 4-8 shows the installed strain gauge on uni-directional fibre. Fig. 4-9 shows the breaking pattern of uni-directional fibre.
Table 4-1. Strength properties of carbon fibre composite material (CFC) material

<table>
<thead>
<tr>
<th>Property</th>
<th>Mean ± Std. Deviation</th>
<th>Units</th>
<th>Source</th>
</tr>
</thead>
<tbody>
<tr>
<td>Tensile Strength in Fibre Direction X1t</td>
<td>2123e6 ±180e6 with tensile strain 1.9%</td>
<td>N/m^2</td>
<td>Testing</td>
</tr>
<tr>
<td>Compressive Strength in Fibre Direction X1c</td>
<td>103e6</td>
<td>N/m^2</td>
<td>Sourced out Testing</td>
</tr>
<tr>
<td>Tensile Strength in Transverse Direction X2t</td>
<td>48e6 ±3e6</td>
<td>N/m^2</td>
<td>Testing</td>
</tr>
<tr>
<td>Compressive Strength in Transverse Direction X2c</td>
<td>1588e6</td>
<td>N/m^2</td>
<td>Sourced out Testing</td>
</tr>
<tr>
<td>Tensile Strength in Third Direction X3t</td>
<td>243e6</td>
<td>N/m^2</td>
<td>Same % less from data sheet</td>
</tr>
<tr>
<td>Compressive Strength in Third Direction X3c</td>
<td>243e6</td>
<td>N/m^2</td>
<td></td>
</tr>
<tr>
<td>Shear Strength in XY plane S12</td>
<td>129e6 ±15e6</td>
<td>N/m^2</td>
<td>Testing</td>
</tr>
<tr>
<td>Shear Strength in XZ plane S13</td>
<td>130e6 ±17e6</td>
<td>N/m^2</td>
<td>Testing</td>
</tr>
<tr>
<td>Shear Strength in YZ plane S23</td>
<td>88e6</td>
<td>N/m^2</td>
<td>Same % less from data sheet</td>
</tr>
</tbody>
</table>

Table 4-2. UD carbon fibre layer properties used during analysis

<table>
<thead>
<tr>
<th>Mechanical Property</th>
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<th>ET</th>
<th>ET’</th>
<th>GLT</th>
<th>GLT’</th>
<th>GTT’</th>
<th>vLT</th>
<th>vLT’</th>
<th>vTT’</th>
</tr>
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<tr>
<td>Mean Value</td>
<td>153700</td>
<td>8960</td>
<td>8960</td>
<td>4980</td>
<td>4980</td>
<td>2600</td>
<td>0.31</td>
<td>0.31</td>
<td>0.52</td>
</tr>
</tbody>
</table>
4.3 Vibration Testing (Experimental Modal Analysis)

To validate the tailoring, experimental modal analysis was performed with 27 miniature accelerometers. Fig. 4-10 shows the application of accelerometers on a composite panel along with their standard minimum resistance cables which connect them with charge amplifiers and data acquisitioning system. In Fig. 3-10 (Modelling and Simulation
Chapter), experimental setup is shown along with sensor size comparison to a coin. The sensor shown in Fig.4-10 there are piezoelectric accelerometers along with 5 meters long low noise standard cable which is connected to the charge amplifier. The sensitivity of accelerometer is 6pC/g with 2-3000 Hz frequency range. Its own resonance frequency is 8500Hz. The same scheme of instrumentation may also be used to assess the morphology of in-built damage in structure by monitoring the change in structural dynamic characteristics.

Fig.4-10.Installation of accelerometers on panels

It is another important technical observation that the shaker and sensors should be used in such as manner without disturbing the natural characteristics of the structure under test. During calibration phase, before the actual test, it was ensured that the shaker produce a stable sinusoidal output frequency characteristics. Further it was ensured that the force
output was relatively independent of the vibrational amplitudes of the test samples. The shaker used in this test is electro-dynamic shakers which are based on the Laplace law. A coil is inserted inside the thin air gap of a magnetic field. It provides a sine wave, random or shock force as required. The force range is 3-5000N with +/- 5mm stroke. Vibration shaker applied the same force independent of its own motion throughout the frequency sweep. During structural vibration analysis, forces were applied and the resulting response function gave almost the similar behaviour which was modelled in finite element based software code using theoretical basis.

First, the frequency response function was measured against an impulse excitation to get an estimate of all the natural frequency locations. Afterwards a detailed experimental analysis was performed from 50-1500Hz frequency range sine sweep. After acquiring amplitude-frequency response curve, the sweep frequency increments were selected in such a way that no important resonant peaks were overlooked. Alternate vibrator locations were also employed to check whether the node lines would passed through the first selected vibrator location. Table 3-2 and Table 3-3 represents the structural dynamic characteristics of both the un-tailored and tailored design respectively.

4.3.1 Modes Isolation Technique

Force appropriation method was used to isolate the vibrational modes of the panels. It is also called the sine dwell method or phase resonance method. It is considered as a traditional method for vibrational testing which uses a single sinusoidal signal to apply force at different points of the object. This method allows the measurement of the response at single normal mode. The complex response of different accelerometers is acquired. This acquired response consists of resonant and off-resonant components. The applied force at
a single frequency is adjusted in such a way that off resonant components of the response are minimized and the respective mode is isolated accurately with almost nil contribution of other modes. Force appropriation method provided unique opportunity for testing the samples, to closely interact with the structure on mode by mode basis.

### 4.3.2 Modal Parameter Estimation Technique

After modal isolation using Force Appropriation Method, Complex Power Method was used for extraction of modal parameters by giving micro frequency sweep around a particular resonant frequency. The modal parameters consist of natural frequency, normalized generalized mass, normalized generalized stiffness and damping of each mode. Complex power of the structure is evaluated as a product of velocities of the measurement points and applied forces.

### 4.3.3 Quality and Orthogonality Check

It was desired that all the normal modes should be orthogonal to each other. Modal Assurance Criteria (MAC) was utilized to evaluate the accuracy of experimentally determined modes. MAC gave an indication of the correlation between two shapes. For ideal orthogonal modes, off diagonal terms of MAC matrix should be zero if compared to unit diagonal terms. However, for experimental modes its values up to 15-20%, indicates modes are reasonably orthogonal. Fig.3-11 (in Modeling and Simulation Chapter) shows the orthogonality of experimentally determined modes.

### 4.4 Scanning Electron Microscopy

After fracture toughness testing of double cantilever beam specimen, the breaking pattern was observed to estimate the failure pattern of specimen. Scanning electron
microscopy results at 80X zoom showed a clear mode I breaking pattern. To be substituted in modified beam theory, as such no finite rotation ‘Δ’ was incorporated at delamination front. Fig.4-11 shows the microscopic view of fibres after separation at 80X zoom. Fig.4-12 shows the naked eye image of uni-directional DCB crack plane indicating pure delamination with no fibre breakage pattern.

**Fig.4-11 SEM image of fractured DCB specimen at 80X zoom**

**Fig.4-12 Crack growth plane of DCB specimen showing the nature of damage**
4.5 Concluding remarks

Experiments are the essence of any research work. However, the experimentation requires more resources than the simulation work. It may be concluded safely that the test results acquired from all the experimentation are quite logical. Primarily there are two types of results reported in this chapter, the first is the mechanical properties of generic test coupons and the second is static and dynamic response of specialized samples. The numerical results of FEM based simulations follow the same trend as measured through experimental testing. One to one comparison of simulations and experimental based static response and modal frequencies has already been presented in Chapter 3.
CHAPTER 5

FATIGUE LOADING SPECTRUM

Development of fatigue load spectrum from flight test data is the most important activity in fatigue analysis. This particular area is also the direct focus of this research work. In this chapter, different phases of data analysis involved to finalize an applicable fatigue load spectrum have been discusses. The data used to develop a loading spectrum was acquired through a few hundred sorties of a particular aircraft which flew different routine missions. Each mission contributes to the spectrum according to severity of flight loads occurred during the flight. Rain flow cycle counting algorithm was developed to measure the normal acceleration counts and race track algorithm was developed for data reduction purpose. All the flight load factor data was acquired through an accelerometer mounted at the center of gravity of aircraft.

5.1 Applicable Load Spectrum

One of the most important input parameters to analyse fatigue and damage tolerance of structures and the main focus of this work, is the applicable loading spectrum. In case of aircraft F&DT design, the baseline loading data is acquired from a sensor mounted at the centre of gravity of any aircraft which is called ‘Nz Spectrum’. Nz describes the vertical axis loading or gravitational pull of aircraft, here the positive Nz is taken as upward gravitational pull. The sign convention of Nz is positive upward in American notation and negative upward in British notation as shown in Fig.5-1.
The sensor type used to acquire the Nz acceleration at centre of gravity may be an accelerometer. The selected load factors are 2g, 3g and 4g levels. At each load factor, there are particular vibrational loadings which are applied to represent the flutter. The race track method has been adopted for data reduction purpose. Appropriate omission and truncation levels have been selected as per ASTM STP1006 and current state of the art [81], [82]. In general engineering applications where the loads are applied in simply linearly cyclic way, the number of cycles may be easily calculated using counting algorithms such as rain flow counting or level cross counting. However, in case of aircraft where the loading becomes irregular the cycle’s start point and ending point location becomes difficult to identify. Using finite element based stress analysis; the maximum stress value on critical structural location was calculated at worst load case i.e. 245MPa. Therefore, a particular stress multiplication factor (245) was included for application of normalized fatigue load spectrum. This loading spectrum has been further used in progressive damage analysis. Fig.5-2 shows the general ‘fighter aircraft loading standard for fatigue’ (FALSTAFF) random loadings spectrum having maximum normalized value 1.
5.2 Development of Loading Spectrum

A simple methodology has been used to extract fatigue loads spectrum from a typical flight tests data. The steps included in this phase are following:

- First of all the statistically analyzing the raw flight data
- Filtering the data through low pass filter using MATLAB tools
- Data reduction to eliminate low amplitude cycles which do not affect the fatigue

The spectrum generated through this approach primarily contains the fluctuating load amplitude information and secondarily the time for which a particular load level is maintained by the aircraft. The scheme has also been validated by applying the same on raw data of aircraft and resultant was compared with the already developed standardized fatigue load spectrum. Sample sizing to generate a representative spectrum is very important because a same spectrum becomes applicable to overall aircraft fleet because it contains all the types of missions that an aircraft may fly. [83] Typical fatigue spectrums are developed to get sinusoidal loadings with uniform levels from the randomly varying
flight loads. In actual phenomenon it is very difficult to apply and control the random variable loads on a structure using hydraulic actuators during laboratory testing. So there is a need to determine an average load spectrum from randomly varying flight loads with known frequency contents. Normally the uniform loads that are applied by using actuators are in the form of a sine wave. Fig. 5-3 shows a single occurrence fatigue spectrum that is formed for estimating the fatigue life of a material / structure having constant cyclic load so that it will be applied to a material / structure conveniently in laboratory. This fatigue spectrum is developed for a specific mean load corresponding to a specific load factor. The raw data of accelerometer has been arranged for number of flights, so that it may be representative of the entire usage of aircraft. Generally all types of missions are included in the raw data.

![Acceleration vs Time graph of a specified load factor](image)

**Fig.5-3.Single load factor occurrence of a fatigue loading spectrum**

A fatigue spectrum computation usually consists of following attributes.

- The mean value of load factor Nz.
- High energy low frequency vibrational contents
• The amplitude of particular low frequency vibrational loads
• Exceedance diagram which tells the number of time a particular g level exceeded
• Time duration for a particular load occurrence

5.2.1 Flight Data Acquisitions

The accelerometers are installed on center of gravity sometimes, additional accelerometers are used to verify the trend of CG data. Data drift and data clipping may be adjusted through post analysis by considering the response of additional accelerometers. Temperature compensations may be critical at high altitude flights, to counter this issue, strain gages are configured in full bridge configuration. Fig.5-4 shows the schematic.

![Data acquisition schematic diagram](image)

5.2.2 Development of Applicable Load Spectra

After finding all the required parameters for development of a fatigue spectrum, a known data set was generated in the first step. Data of three frequencies (F1, F2, and F3) and three load factors (L1, L2 and L3) are generated at a rate of 1000 Hz, shown in Fig.5-5(a). Power spectral density PSD of the input signal is calculated using Welch's averaged modified period gram method of spectral estimation. Fig.5-5 (b) shows the PSD
plot of the signal. Three frequencies F1, F2, F3 in the original signal can be seen easily in PSD plot. It is important to find out load factor exceedances in the random data and to time tag the data for which it exceeds any particular load factor. Fig.5-6 shows the filtered response for one frequency extracted from raw data. Fig.5-7 shows the output of a certain load factor occurrence which contains all the basic three frequency contents. Table 5-1 gives load RMS values against a certain load factor occurrence. Similar is the procedure for rest data.

Fig.5-5. Loads data analysis; (Fig ‘a’ is Raw Data, ‘b’ is Power Spectral Density Plot, ‘c’ is Averaging Filtering and ‘d’ is Specific Load Level
Fig. 5-6. Filtered response for one frequency extracted from raw data

Fig. 5-7. Output of a certain load factor occurrence which contains all the basic three frequency contents. (a) Before filtering, (b) After filtering

Table 5-1 Root mean square values of amplitudes for a particular load factor

<table>
<thead>
<tr>
<th>RMS Calculation</th>
<th>L1</th>
<th>L2</th>
<th>L3</th>
<th>L4</th>
<th>L5</th>
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</thead>
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<tr>
<td>Frequencies F1</td>
<td>8.00</td>
<td>8.20</td>
<td>8.40</td>
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<td>F2</td>
<td>4.05</td>
<td>4.20</td>
<td>4.35</td>
<td>...</td>
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</tr>
<tr>
<td>F3</td>
<td>1.65</td>
<td>1.68</td>
<td>1.71</td>
<td>...</td>
<td>...</td>
</tr>
</tbody>
</table>
5.3 **Effect of Structural Stiffness on Loading Amplitudes**

In un-tailored design, the focus of designer remains primarily on the static strength of the panels. To enhance the static strength of panels, the stiffness of structure is kept higher, resulting in the higher natural frequencies of the system. When there are higher natural frequencies of structure, the amplitudes of vibration is always decreased. Consequently, the applicable fatigue loading amplitudes are also decreased. This design gives a better fatigue life. Secondly, when the designer focuses on aero-elasticity tailored design, the stiffness of first bending mode is always compromised by enhancing the torsional mode stiffness. With the decrease of first bending mode stiffness the fatigue load cycles amplitude increases. This results into the lower fatigue strength.

It was observed that, during bending of panels there is a shear load between plies which causes the delamination to grow. So the load amplitudes against first bending mode are directly related to $G_{IIIC}$ of the material, which was taken as 2.5 times the $G_{IC}$ value. The reason for keeping it higher is to be more conservative, because no experimental testing has been performed for $G_{IIIC}$. However, in different composite materials $G_{IIIC}$ to $G_{IC}$ ratio varies from 2 to 5 depending upon the composition of materials. [84], [85], [86] The results of fatigue and damage tolerance were quite different in tailored and un-tailored design because the applicable spectrum was different in both the design cases due to change in structural dynamic characteristics.

5.4 **Change in Loading Spectrum Characteristics**

The fatigue life analysis has been carried out by varying the characteristics of a standard baseline fatigue spectrum for both the tailored and un-tailored panels. As it has already been discussed that the characteristics of a loading spectrum are dependent upon
vibration frequencies, stress ratio and loading amplitudes. Table 5-2 and Table 7-1 show the different cases for the iterations of fatigue load spectra application with their changed characteristics for un-tailored and tailored panels respectively. It is very important to mention here, that the vibrational frequencies considered for application in the loading spectra are the three most critical frequencies. The criticality of the modes of vibration were selected on the basis of the quality factor of the mode and the flutter critical scenario. The modes of vibration selected for fatigue life investigations are the ‘First bending’, ‘second bending with two nodal points’ and the ‘first torsion’. All the three modes with their respective natural frequencies were superimposed in all the three Nz load factors in the baseline spectrum for both the un-tailored and tailored panels. The fluctuating load amplitudes, are mentioned as “percentage of mean load” are calculated against each frequency through regression analysis of accelerometers data acquired during experimental modal analysis of panels. As it has already been illustrated in the work methodology (Chapter 3) the test samples were prepared to demonstrate the change of their dynamic characteristics. Respective critical frequencies of both the panels were incorporated in the applicable fatigue loading spectrum.
Table 5-2. Test cases of fatigue spectrum application for Un-tailored design

<table>
<thead>
<tr>
<th>Test Case No.</th>
<th>Title</th>
<th>Mean Loads (Nz Pull)</th>
<th>Amplitudes of Fluctuating Load</th>
<th>Frequencies of Vibration</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.</td>
<td>Base line spectrum</td>
<td>4, 3 and 2g levels</td>
<td>14.1, 2.3 and 0.5% of the mean load respectively</td>
<td>19.1, 171.1 and 674.2 Hz</td>
</tr>
<tr>
<td>2.</td>
<td>Only mean loads with no vibration</td>
<td>Same</td>
<td>Nil</td>
<td>Nil</td>
</tr>
<tr>
<td>3.</td>
<td>Double load amplitudes about the same mean loads and the same frequencies</td>
<td>Same</td>
<td>Double</td>
<td>Same</td>
</tr>
<tr>
<td>4.</td>
<td>Half load amplitudes about the same mean loads and the same frequencies</td>
<td>Same</td>
<td>Half</td>
<td>Same</td>
</tr>
<tr>
<td>5.</td>
<td>Double frequencies with same loads</td>
<td>Same</td>
<td>Same</td>
<td>Double</td>
</tr>
<tr>
<td>6.</td>
<td>Half frequencies with same loads</td>
<td>Same</td>
<td>Same</td>
<td>Half</td>
</tr>
</tbody>
</table>
Table 5-3. Test cases of fatigue spectrum application for Tailored design

<table>
<thead>
<tr>
<th>Test Case No.</th>
<th>Title</th>
<th>Mean Loads (Nz Pull)</th>
<th>Amplitudes of Fluctuating Load</th>
<th>Frequencies of Vibration</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.</td>
<td>Base line spectrum</td>
<td>4, 3 and 2g levels</td>
<td>15.2, 4.1 and 0.4 % of the mean load respectively</td>
<td>13.46, 119.1 and 684.7 Hz</td>
</tr>
<tr>
<td>2.</td>
<td>Only mean loads with no vibration</td>
<td>Same</td>
<td>Nil</td>
<td>Nil</td>
</tr>
<tr>
<td>3.</td>
<td>Double load amplitudes about the same mean loads and the same frequencies</td>
<td>Same</td>
<td>Double</td>
<td>Same</td>
</tr>
<tr>
<td>4.</td>
<td>Half load amplitudes about the same mean loads and the same frequencies</td>
<td>Same</td>
<td>Half</td>
<td>Same</td>
</tr>
<tr>
<td>5.</td>
<td>Double frequencies with same loads</td>
<td>Same</td>
<td>Same</td>
<td>Double</td>
</tr>
<tr>
<td>6.</td>
<td>Half frequencies with same loads</td>
<td>Same</td>
<td>Same</td>
<td>Half</td>
</tr>
</tbody>
</table>

5.5 Procedure to Establish Loading at the Scale of Structure

As described in chapter 2, the samples under investigation are considered as the integral part of aircraft wing skin. As shown in Fig.5-8 the samples may treated as critical portion of wing skin probably at the lower side and its location may be near the wing root. It is very clear to understand that the fatigue loads against which the sample is to be investigated will be governed by primarily the Nz loads and secondarily by the structural vibrational response of wing and stores. It is very extensive and expensive activity to measure the flight loads of aircraft wing and aircraft store to perform fatigue analysis.
In aerospace related engineering research activities, a particular field of research may not be isolated from other associated activities which are somehow remain connected as a mandatory input requirement or parallel procedure. Similarly, the aircraft flight loads measurement activity is also associated in a same manner. Following is the brief description of the activity through which the aircraft wing and store loads were measured and the flight loads fatigue spectrum was developed to be applied for fatigue analysis.

### 5.6 Aircraft Wing and Store Loads Measurement

In this activity the scheme followed to measure in flight loads is based upon NACA report for strain gages calibration to measure in flight loads of aircraft NACA-TR-1178. A special method was developed for calibration of strain gages installed at main spar, rear spar and front spar of an aircraft. Extensive calibration loads were applied at different wing stations and calibration factores were developed by solving the stiffness matrix. During flight, the shear loads an bending moments were measured by multiplying the same calibration factors with the strain gages response. The loads model development highlights are enlisted as following:
• The load calculation model was based upon 06 full bridge configuration strain gages applied to measure bending strain at the root of 03 spars.

• 2V Excitation current was provided through calibrated data acquisitioning system.

• The data was acquired at 500 Hz measurement frequency to notice all the vibrational fluctuations in addition to stable load shifts. However appropriate high frequency filters (the moving average filters) were used to refine stable load factor data.

• Strain gauge data corrected by subtracting initial offsets of the full bridges which was assumed to be due to initial unbalance of bridge during application of gages.

• The load calculation model were developed on the basis of average 1 second data at stable point during application of calibration loads.

• Shear Force units were considered as 'Kg' and Bending Moment units as 'Kg-m'

• Push load applied in upward direction having positive sign convention & vice versa

The flight load calculation models were finalized using the calibration data with computational error of less than 5%. The relation between strain gauge output µ and structural loads i.e. shear force ‘V’, bending moment ‘M’ and torque ‘T’ at the surface on which strain gage can be applied has been established with sufficient precision by a linear equation with coefficients ‘αV’, ‘αM’ and ‘αT’. In a typical calibration process involving ‘j’ bridges and ‘n’ different calibration load cases, the output ‘µki’ of the ‘i-th’ sensor to the ‘k-th’ load case follows the General form of the load equations Equation 5-1:

\[
\forall (k,i) \in [1,n] \times [1,j], \ \mu_{ki} = \alpha_{V,i} V_k + \alpha_{M,i} M_k + \alpha_{T,i} T_k \quad \text{Equation 5-1}
\]

The principle of superposition is applicable due to linear system which says that the strain at a particular location as a result of applied load at several structural locations may be the algebraic sum of the strains due to the same loads which may be applied individually.
If a shear force ‘V’ is applied at a point having ‘x, y’ coordinates, will give a bending moment ‘M = V * y’ and a torque ‘T = V * x’. By substituting the values into above mentioned equation and including additional terms with coordinates and using the inverse relation, a linear function of the outputs of ‘j’ bridges becomes as

Equation 5-2.

\[
\{Vx^r y^s\} = [\alpha]^{-1} [\mu] \Rightarrow \forall k \in [1, n], \begin{bmatrix} V_k \\ M_k \\ T_k \end{bmatrix} = \begin{bmatrix} \beta_{V,1} & \beta_{V,2} & \ldots & \beta_{V,j} \\ \beta_{M,1} & \beta_{M,2} & \ldots & \beta_{M,j} \\ \beta_{T,1} & \beta_{T,2} & \ldots & \beta_{T,j} \end{bmatrix} \begin{bmatrix} \mu_{k,1} \\ \mu_{k,2} \\ \vdots \\ \mu_{k,j} \end{bmatrix}
\]

Equation 5-2

By considering the number of ‘j’ bridges and ‘n’ calibration load cases at the aircraft wing, the least squares linear equation is developed, which may give the loading coefficients as a primary requirement for fatigue load measurement during flight. ‘\beta \{V, M, T\}, i’ may be calculated by solving the following equation.

\[
\Xi^T \begin{bmatrix} V'_1 \\ V'_2 \\ \vdots \\ V'_n \end{bmatrix} = \Xi^T \Xi \begin{bmatrix} \beta_{V,1} \\ \beta_{V,2} \\ \vdots \\ \beta_{V,j} \end{bmatrix} \left(\text{when } \Xi^T \Xi \text{ invertible} \right) \Rightarrow \begin{bmatrix} \beta_{V,1} \\ \beta_{V,2} \\ \vdots \\ \beta_{V,j} \end{bmatrix} = [\Xi^T \Xi]^{-1} \Xi^T \begin{bmatrix} V'_1 \\ V'_2 \\ \vdots \\ V'_n \end{bmatrix}
\]

Equation 5-3

where \(\Xi = \begin{bmatrix} \mu_{11} & \mu_{12} & \ldots & \mu_{1j} \\ \mu_{21} & \mu_{22} & \ldots & \mu_{2j} \\ \vdots & \vdots & \ddots & \vdots \\ \mu_{n1} & \mu_{n2} & \ldots & \mu_{nj} \end{bmatrix}\)

If we apply ‘n’ different load cases having known shear force values ‘Vn’ at ‘n’ different loading points of the aircraft wing, ‘n’ values of the bending moments and torque may be calculated by above mentioned equations ‘M = V * y’ and ‘T = V * x’, which may further give the values of \{\beta M, i\} and \{\beta T, i\} The least square solution in Equation 5-3 exists only if the determinant of \([\Xi^T \Xi]^{-1}\) is greater than zero, which essentially need the strain gages having the similar response characteristics as in calibration procedure.
Shear force and bending moment calibration factors calculated using above mentioned equations for 06 strain gages applied at wing root are presented in Table 5-4

<table>
<thead>
<tr>
<th></th>
<th>SG_2</th>
<th>SG_3</th>
<th>SG_4</th>
<th>SG_10</th>
<th>SG_11</th>
<th>SG_12</th>
</tr>
</thead>
<tbody>
<tr>
<td>Bending Moment</td>
<td>-1.492</td>
<td>-3.115</td>
<td>-0.973</td>
<td>-6.956</td>
<td>-1.502</td>
<td>-0.765</td>
</tr>
</tbody>
</table>

Fig.5-9 Shows single flight measured loads of aircraft wing through calibration factors, these are the loads used for fatigue spectrum development as illustrated in Chapter 7.

![Normalized Load](image)

**Fig.5-9. The aircraft wing loads measurement data for one flight**

Similarly, the store loads were measured by applying the strain gages on the pylon surface. The technical scheme followed for store loads and vibrations measurement is the same as mentioned by Robert Richard of Rafael Ltd. [88] Six full bridge configuration strain gages were applied on the pylon for the measurement of vertical ‘Z-axis’, lateral ‘Y-axis’ and drag ‘x-axis’ loads of aircraft store during flight. These are the secondary loads in addition to wing loads which directly affect the sample under investigation.
The pylon instrumentation for store loads is shown in Fig.5-10. The applied calibration loading scheme is presented in Fig.5-11 where 46 individual load cases are required.

A1 & A2 are the Z-axis sensitive strain gages in Full Bridge- III configuration

B1 & B2 are the Y-axis sensitive strain gages in Full Bridge- I configuration

C1 & C2 are the X-axis sensitive strain gages in full bridge- III configuration

*Fig.5-10. The instrumented pylon for store loads*

*Fig.5-11. The instrumented pylon for store loads*
The calibration factors were calculated on the basis of Equation 5-3 are presented in Table 5-5. The calibration loads data is presented Fig.5-12 for only pure lateral and angular loads. Similar data of vertical loads was used for calculation of calibration factors.

<table>
<thead>
<tr>
<th>Vertical ‘Z-axis’ Load Factors</th>
<th>SG_A1</th>
<th>SG_A2</th>
<th>SG_B1</th>
<th>SG_B2</th>
<th>SG_C1</th>
<th>SG_C2</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>-1.396</td>
<td>-26.820</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>Lateral ‘Y-axis Load Factors</td>
<td>0</td>
<td>0</td>
<td>3.179</td>
<td>-1.959</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>Drag ‘X-axis Load Factors</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0.0827</td>
<td>-30.361</td>
</tr>
</tbody>
</table>

To verify the accuracy of calibration factors, 03 known check loads were applied at multi points of pylon calibration beam near center of gravity. The micro-strain values were acquired at stable time and multiplied with the calibration factors to recover the applied loads. These check loads were applied in Positive & Negative angles and Z direction. The average of 03 check loads as shown in Fig.5-13 gave the recovered load error 9.7%.

Fig.5-12. The store loads calibration data

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Fig. 5-13. Calibration check loads for model verification

Fig. 5-14. Store loads during one flight
As shown in Fig. 5-14, the store loads are measured for one flight using the same calibration factors. A fraction of sum of the total measured wing loads and store loads becomes the input to the sample under investigation for estimating its fatigue and damage tolerance behavior. The same flight loads are also measured to develop a fatigue spectrum for the laboratory testing and fatigue qualification of other aerospace parts.

5.7 Concluding Remarks

The fatigue load spectrum has been developed by using data analysis techniques as illustrated in this chapter. Outcome of this activity is the number of counts for each load factor that occurred during overall flights data. This information describes that how many cycles of a particular load should be applied to simulate a certain time duration in fatigue analysis. There are two parameters of an applicable fatigue load spectrum, the number of occurrences and the fluctuating loading amplitudes to be superimposed to each occurrence. The methodology presented in this chapter for development of a fatigue load spectrum only describes the first parameter of spectrum which is the number of occurrences. The second parameter which defines the fluctuating loads is generally determined through vibration testing of particular wing or stores.
CHAPTER 6

DAMAGE TOLERANCE ANALYSIS

This chapter is dedicated to the characteristics of damage in composite structures and the techniques to model and analyse those damages using finite element base numerical simulations. In the first part of this chapter, the general types of manufacturing-induced damages and service-induced damages in composites are enlisted. In the second part, ply material failure and de-lamination methodologies to analyze damage of composites have been explained. In ply material failure, progressive failure analysis details are given. In case of de-lamination analysis, virtual crack closure technique, cohesive zone modelling, breaking glue and element separation methodologies have been discussed. The merits and de-merits of different elements for damage growth analysis application are explained. Mainly the Shell, Solid and Solid Shell elements are presented. The failure criteria used to analyse damage onset are also explained in this chapter. The mathematical model developed on the basis of fibre angle orientation and resultant decrease of fatigue life is presented at the last.

6.1 Damage and Defects in Composites

In fatigue and damage tolerance studies, the analysis is carried out against the specific initial damages. The use of composite materials in aerospace applications is associated with two types of damages; manufacturing induced and service induced damages. The necessary detail of both the types is discussed as following.
6.1.1 Manufacturing induced defects

Manufacturing defects generally include the following:

1. Resin rich areas
2. Voids or pits
3. Delaminations
4. Resin starved regions
5. Blisters and air bubbles
6. Thermal decomposition
7. Wrinkles etc.

Manufacturing induced damage also includes other anomalies, like microcracking, porosity, and certain complex delaminations resulting from processing discrepancies. During mass production of composites some unwanted features on the critical parts may also be observed such as, inadvertent edge cuts, surface gouges and damaged fastener holes etc. Contaminated bond-line surfaces, such as a prepreg backing paper or separation film, that is inadvertently left between fibre plies during lay-up may also be observed. Inadvertent (non-process) damage can occur in some exposed parts or critical components during assembling or transportation. Manufacturing damages are generally considered less critical than the service damages.

A component becomes resin-rich if too much resin is used while manufacturing. For non-structural applications this phenomenon is not necessarily damaging, however it adds additional weight. A component is called resin-starved if too much amount of resin is bled off in the curing process or if the enough amount of resin is not applied during the wet-layup process. Resin-starved areas becomes evident by the fibres that are visible to the
surface. General sources of manufacturing induced defects generally include the following processes:

- Mishandling
- Improper cure or processing
- Tool drops
- Improper machining
- Improper drilling
- Contamination
- Substandard material
- Improper sanding
- Mislocation of holes or details
- Inadequate tooling

The extent of damage controls the residual strength. The extent and orientation of damage is critical to damage tolerance of composite structures. A few types of damages critical from fatigue point of view are discussed in detail.

6.1.2 Service Induced damages

In-service defects of composite are generally more critical in F&DT studies. Structure in aerospace applications may include the following in-service damages:

1. Fatigue
2. Environmental degradation
3. De-bonding
4. Impact damage
5. Fiber fracturing
6. Cracks from local overloads
7. Delaminations
8. Erosions

Nondurable design details (e.g., improper core edge close-outs) also lead to liquid ingression. Improper vacuum bagging scheme is also a major cause of core material defects. During improper vacuum bagging the additional resin is stucked in the core material and it makes the overall structure with uneven stiffness.

6.2 Damage Assumption & Incorporation

There is number of options to assume an initial damage type in composite material, as shown in Fig.6-1. A limited survey of operation data of in-service composite aircraft wings was conducted to explore the type of reported field damages on UAV structures. Generally reporter composite damages are fibre breakage, ply delamination and fibre pullouts. The primary cause of composite panels damage is the heat and moisture of environment in which aerospace structure is being used. The extent of damage in composite structures also varies on the basis of missions being flown by the aircraft. Generally the UAV’s are prone to lower gravitational pulls, so the probability of fatigue damages becomes lower as compared to fighter aircraft.

On the basis of available data, ply delamination ‘Type 03’ (as shown in green colour of the following Fig.) was selected as the initial damage/ crack which may be due to interface of metallic attachment with composite material (as shown in Fig.2-2 of Methodology Chapter). The damage was characterized as an initial delamination of length 15mm at one direction of the panels. Damage incorporation depends upon the selected F&DT technique as discussed comprehensively in the next section.
6.3 Damage Tolerance Analysis Techniques

Composite panels were analysed against progressive damage analysis keeping in view the applicable loadings scenario against their application. Ply failure, crack propagation and delamination prediction gave the failure index and strength ratio using different failure criteria. First ply linear failure analysis has been considered as the ground level for these composite panels. The critical result of such failure analysis is the stress redistribution after first ply failure and its continuous calculations as the plies fail progressively. Residual strength continuously decreased as the delamination grew. Four different methods have been used to analyse the failure. Fig.6-2 shows that the five technologies fall under two categories: Ply material failure and delamination.

Both the shell and solid elements can be used to analyse the failure scenarios. If the layered composite laminate is defined as a shell element, mainly the in-plane behaviour is investigated such as bending. However, progressive failure analysis can also be performed
using shell elements. When loading becomes three dimensional and the load path involves the third dimension, it becomes necessary to calculate stresses and strains in third direction. In such applications we use solid elements to determine stresses at each ply. Solid elements sometimes don’t bend very well due to limitations of their shape function, in these cases of solid elements under bending; a special string function is added to make them solid-shell elements. The same technique was used to model proper bending in tailored and un-tailored panels. Fig.6-3 illustrates three types of elements which can be used to define a complete structure in finite element based techniques. Although the finite element based numerical solution may be run for any of the element option, however, proper selection of elements gives improved results. In finite element analysis, new elements are being added rapidly in different software packages according to the application. The elements with more nodes are generally very good from accuracy point of view but their computational time increases drastically when applied to honey comb structure. The element mesh in honey comb core material becomes time taking due to extensive modelling non linearity involved.
Fig.6-2. Five technologies to analyse composite structure failure

The detailed results of delamination growth under applicable loading, using all the five methodologies have been discussed in the last chapter along with other results.
6.3.1  Progressive Failure Analysis (PFA)

In progressive failure analysis, the failure criteria offered in different FEM based software packages are Puck, Hashin, Tsai Wu, Max Stress and Max Strain.[89] Multiple criteria were used in panels analysis, first were used to determine failure while the others were used to calculate failure indices only. As the plies start to fail layer by layer, and when a particular ply fails, it became necessary to deactivate it by scaling down its strength moduli; Young’s modulus ‘E’ and Shear modulus ‘G’ to make the ply soft. Hence the ply was taken out of the load path and the load was distributed to the other plies and surrounding structure. Hence, the crack or delamination propagated by deactivation the elements. This is the similar technique to the classical analytical ply failure methodology which uses to discount the one by one ply as the failure of each ply grows. At each step, the remaining plies become less in number.

Fig. 6-3. Different elements to model a composite structure
6.3.2 Virtual Crack Closure Technique (VCCT)

Virtual crack closure technique is a fracture mechanics based model. In linear fracture mechanics a crack begins to grow when $G$ (strain energy release rate) becomes greater than $G_c$ (inter laminar fracture toughness as determined in previous section). VCCT is one of the methods to calculate energy release rate which has been frequently implemented in commercially available software packages. In Fig. 6-4, a mesh is shown which has an existing crack in the model. When the structure is loaded in such a way that the crack tries to open up with displacement ‘$u$’, there is a force ‘$F$’, acting at crack tip. With existing crack ‘a’ the energy release rate can be calculated as following relation as it has already been discussed in literature survey;

$$G = \frac{F \times u}{2}$$  \hspace{1cm} \text{Equation 6-1}

All the three modes of crack propagation can be incorporated in this technique, i.e. opening, shear and tear mode. However, only the first mode has been analyzed due to availability of critical strain energy release rate only in mode-1 direction. VCCT propagates the crack in different ways, the first way is with glued interface, where a crack is modelled between two contact bodies glued together. The cracked area is not joined and the nodes of rest portion are glued together, in this way the crack tip nodes are properly identified. The path of crack can be seen in graphics of latest FE based commercial software codes. Ultimately the two joined bodies are unglued when VCCT criteria is met. [90]Simulation results also include the step by step video graphics to clearly analyse the separation of elements and
their path. The area of interest may be focused in the video graphics to see the detailed behaviour of separation.

Fig. 6-4. A general VCCT model

On the basis of calculated stresses, the program decides where the crack has to go. If Quad mesh is selected, the crack travels through 00 or 900. However, if triangular mesh is adopted, the crack growth will be along different path. The most important feature of VCCT adopted in the analysis of panels is its growth with re-meshing. As the crack is growing, the computer program looks ahead of crack tip at maximum stresses and determines in which way they are going. Hence, it continues to re-mesh the body in front of crack tip and the crack grows toward its natural direction. [91] VCCT actually calculates energy-release rate, with the assumption that the energy needed to separate a surface is the
same as the energy needed to close the same surface. This is the basic philosophy of this method. It is equally applicable to metals and composites.

6.3.3 Cohesive Zone Model (CZM)

Cohesive zone modelling uses a special class of inner face elements to simulate delamination and crack growth. [92] The libraries of these special inner face elements to characterize the interface behaviour in FEM codes make CZM different from other analysis technologies. The inner face elements are sandwich between structural solid elements. The elements are defined by looking into the traction stresses on top and bottom face of the elements; the shear and normal traction stresses.[93], [94] It is required to map the traction stresses to the relative displacement between top and bottom faces. The relative displacement is plotted on the horizontal axis and stresses on vertical axis as shown in Fig.6-5.

![Cohesive Zone Model](image)

**Fig.6-5. A general Cohesive Zone Model (Traction stress vs. Displacement)**
The Fig. describes a bilinear model, when the top and bottom face of structure deforms, traction stresses goes linearly. The critical relative displacement is defined by the user i.e. $\Delta^0$. When the solution reaches the final maximum relative displacement, the modelled element is totally damaged. The area underneath the curve is cohesive energy or the critical energy release rate, $G_{IC}$ (already calculated in previous section). Material behaviour becomes irreversible when $\Delta$ becomes greater than $\Delta^0$. Exponential or linear-exponential models may also be used for this behaviour. Ply and adhesive layers are meshed independently using higher order elements and then ‘contact’ option is used to glue the adhesive layer and plies together. Both the bodies can have different mesh densities. In comparison with VCCT, the CZM involved no special requirement to model an initial crack. The progressive failure index of cohesive zone has been shown in coming section as a comparison of results with other techniques.

### 6.3.4 Breaking Glued Contact (BGC)

This includes most comfortable stress criteria in which calculated normal and tangential stresses are compared for a glued interface.

$$
(\sigma_{\text{normal}} / S_n)^m + (\sigma_{\text{tangent}} / S_t)^n > 1
$$

**Equation 6-2**

Where m, n, $S_n$ and $S_t$ are the user defined parameters. This method requires the information about the area where delamination will occur to look at normal and tangential / shear interface stresses. $S_n$ and $S_t$ are allowable normal and shear stresses. FEM software calculates the developed stress to allowable stress ratios. The exponent m & n defines the interaction between these stresses. If it is taken as 1, there will be linear relation between
two terms. If the values are 2 and 2, it will end up with circular interaction. When the result is equal to 1, FEM software starts to release the contact as a result of failure.

6.3.5 Delamination Method (Elements Separation)

This is also stress based method with same criteria as discussed in braking glue contact. In BGC we have to know where the delamination will occur and a glued joint is defined there. However, in delamination method, we don’t have to know where the crack may occur. The stress criteria may occur anywhere in the meshed body and the elements will split. This split may occur either in the same material or between the two plies. In FEM software, we have inserted interface elements where the mesh is split, in this way the power of analysis was increased by joining CZM and stress based delamination technology. The comparative outcome of all the methods has also been presented in results and conclusion chapter.

6.4 Failure Criteria

The type of onset failure analysis criteria adopted in this work is non-interactive model which does not take into account the interaction between stresses and strains acting on a lamina of panels. This model resulted into some errors in the strength predictions, when multi-axial states of stresses were applied. These are the typical maximum strain and maximum stress criteria. The second approach adopted to predict onset of failure was the interactive model. It took into account the interactions between stresses and strains acting on the lamina. Although there are different models available in literature such as Hashin-Rotem, Hashin (Only) and Puck. [95] In contrast with phenomenological approach, these micro level criteria consider both the fibre and matrix failure and sensitive towards tension and compression loadings.
The Hashin criterion is well incorporated in FEM software codes. This was applied under three dimensional state of stress as shown in the following equations. For the matrix failure mode, a quadratic attitude was selected because the linear criterion cannot accurately predict the material behaviour, and a polynomial of higher degree became very complex while modelling. Additionally, the effect of the shear stress was also taken into account in the fibre tension mode.

Fibre tension mode (When $\delta_1 > 0$)

$$\left( \frac{\delta_1}{\delta_{1(ult)}} \right)^2 + \left\{ \frac{(\delta_{12})^2 + (\delta_{13})^2}{(\delta_{12(ult)})^2} \right\} = 1$$ \hspace{1cm} \text{Equation 6-3}

Fibre failure in compression (When $\delta_1 < 0$)

$$-\delta_1 = \delta_{1(ult)}$$ \hspace{1cm} \text{Equation 6-4}

Matrix failure in tension (When $(\delta_2 + \delta_3) > 0$)

$$\left\{ \frac{(\delta_2 + \delta_3)}{\delta_{2(ult)}} \right\}^2 + \left\{ \frac{(\delta_{23})^2 - (\delta_2\delta_3)}{\delta_{23(ult)}} \right\}^2 + \left\{ \frac{(\delta_{12})^2 + (\delta_{13})^2}{(\delta_{12(ult)})^2} \right\} = 1$$ \hspace{1cm} \text{Equation 6-5}

Matrix failure in compression (When $(\delta_2 + \delta_3) < 0$)

$$\left\{ \frac{(\delta_2(ult)}{2\delta_{23(ult)}} \right\}^2 - 1 \right\} \left\{ \frac{(\delta_2 + \delta_3)}{\delta_{2(ult)}} \right\} + \left\{ \frac{(\delta_2 + \delta_3)}{2\delta_{23(ult)}} \right\}^2 + \left\{ \frac{(\delta_{23})^2 - (\delta_2\delta_3)}{\delta_{23(ult)}} \right\} \left\{ \frac{(\delta_{12})^2 + (\delta_{13})^2}{(\delta_{12(ult)})^2} \right\} = 1$$ \hspace{1cm} \text{Equation 6-6}

Hashin uses three modes of matrix cracking having different angle between fracture plane and lamina in addition to the different type of loading causing fracture. Other criteria which are reported in various literature have been studied and may also be used in future work such as: Cuntze theory,[96] Yamada Sun equation,[97] Koop Michaeli criterion,[98] Kroll Hufenbach theory[99] Sun Tao equation, [100]Zinoviev criterion,[101] Gosse theory, [102]and Hart Smith relationship. [103] Another criterion is the modified Kinetic theory,
where, the calculated constituent stresses and strains are used in combination with kinetic theory to predict fatigue life in large-scale composite structures under variety of complex loadings that actually occur during the flight of aircraft. This method in software codes like Abaqus and Helius MCT, joins the fatigue analysis with different finite element based tools and requires less composite fatigue characterization data in comparison to that mentioned in previous sections. In addition to its fast simulation, it also provides the needed estimates between physics based analysis and mechanical structural analysis. [104], [105]

6.5 Mathematical Modeling

Based upon the results of extensive experimentation and high fidelity numerical simulations, an empirical relationship has been developed to model the fatigue behaviour of composite panels against gradual change of fibre orientations. The newly developed model takes into account the following manufacturing parameters;

a. Rate of change of fibre orientation per unit span length
b. Strain energy release rate as a primary measure of damage tolerance
c. The effect of core thickness on results

During the development phase of mathematical model given in Eq-8.7, first, fibre angles were related with the ply stiffness and the ply stiffness was related to strain energy release rate. Secondly, the common parameter of ply stiffness was cancelled out and the ply angles were directly related to the strain energy release rate of the panels.

\[ \frac{d\theta}{dS} = C (\Delta G)^n - \left( \frac{1}{2} \frac{My}{I} \right) \]  \hspace{1cm} \text{Equation 6-7}

Where, \( \theta \) is the fibre angle and \( S \) is the span length. ‘C’ and ‘n’ are material dependant empirical parameters. The developed mathematical model is based upon the assumption that most of the dynamic instabilities are the result of coupling between bending and torsion
modes of vibration. In case of any other predicted phenomenon of dynamic instability, a
detailed vibrational testing must be performed for qualification of aerospace grade structure
as per applicable Standard LIM-8870 (C).

6.6 Concluding Remarks

Finite element based damage growth analysis in composites and the selection of
appropriate failure criterion is an emerging field. There are different approaches to
determine composite damage onset. Phenomenological approach based finite element
analysis has been adopted to compare the fatigue life of samples. The damage tolerance
analysis requires an initial damage in the structure which has been determined on the basis
of limited field data survey. A de-lamination between two plies of carbon fibre sandwich
panel was considered as initial damage.
CHAPTER 7
RESULTS AND CONCLUSION

7.1 Synthesis of Experimental and Simulation Work

On the basis of experimental and simulation work, investigations were carried out to explore the fatigue and damage tolerance behaviour of composite samples with different parameters of fatigue load spectrum which mainly depends upon structural dynamics. In the experimental work presented in chapter 4, the aero-elastically tailored design of carbon fibre sandwich panels was finalized by altering the fibre orientations of different plies. The altered design resulted with an increased flutter speed as compared to the un-tailored design. The dynamic instability was investigated through Flutter analysis simulations and the comparison of both the designs has been reported in chapter 3. This activity was performed to rationalize the basis of change in structural dynamic characteristics of a sample structure.

After getting a validated tailored and un-tailored design, the samples were manufactured and further experimental testing was performed. The experimental modal analysis of both the samples resulted with different natural frequencies and amplitudes of vibration. The basic three natural frequencies were integrated with the fatigue load spectrum which was developed for an input to fatigue analysis. The development of fatigue load spectrum has been discussed in chapter 5. To investigate the effect of these altered fatigue load spectrum characteristics, fatigue and damage tolerance analysis was performed.
using finite element based simulations. The details of finite element model has been presented in chapter 3. Numerical simulation to predict static or fatigue behaviour of any structural part requires some material properties as an input parameter. The basic and most critical material properties were determined through experimental testing as reported in chapter 3 and chapter 4. Critical strain energy release rate is one of the most important material property required to determine material failure in finite element simulations. This is a directional property, its value varies in all the three modes of crack propagation. However being most critical among all the three modes, only the opening mode critical strain energy release rate has been determined experimentally. In addition to that, ply level static moduli of elasticity and the strength properties have also been determined experimentally to be incorporated as an input to the finite element model. The resultant fatigue life of tailored and un-tailored design has been presented followed by the fatigue life comparison of a same sample with altered parameters of applicable fatigue load spectrum. The altered parameters include the vibrational amplitudes and frequencies.

7.2 Results and Discussions

Keeping the same material properties (as described in Chapter 3), under the application of respective loading spectrum (as described in Chapter 5), with the most likely initial damage orientation, the fatigue and damage tolerance has been analysed using the applicable failure criteria (as described in Chapter 6). The tailored and un-tailored designs gave different Fatigue and Damage Tolerance (F&DT) results. The fatigue life of un-tailored panels under the baseline spectra as mentioned in Table 5-2 and Table 5-3 is 9.3% higher than the tailored panels. Fig.7-1 shows the graphical comparison of lives for both the un-tailored and tailored design.
In addition to the baseline fatigue life comparison, simulations were carried out with different characteristics of the applicable fatigue-loading spectrum as mentioned under the heading 5.4 and the results are presented in comparison with the baseline spectrum. Fig.7-2 shows a base line spectrum with 03 superimposed loading frequencies.

Fig.7-2. Baseline Fatigue load spectrum with 03 superimposed fluctuating cycles
Fig. 7-3. Fatigue load spectrum without superimposed fluctuating cycles

A fatigue load spectrum without having any superimposed fluctuating vibrational loading frequencies is shown in Fig. 7-3. In the figure, only 03 occurrences of normal load factor are visible i.e. 4g, 3g and 2g load factor. These loads are repeated for several times.

Fig. 7-4. Fatigue load spectrum with double and half amplitudes of superimposed fluctuating cycles
In Fig.7-4, the same baseline fatigue load spectrum has been presented with double and half of the fluctuating load amplitudes. In actual the spectrum contains a large number of cycles, however the minimum numbers are shown for better visibility. Similarly in Fig.7-5, the same load spectrum has been presented with double and half of the fluctuating load frequencies. The net effect of higher or lower vibrational frequencies is just the increase or decrease of applied number of cycles. Table 7-1 shows the detail of each test case along with the fatigue life results. Fig.7-6 shows the graphical representation of the same results.

Fig.7-5. Fatigue load spectrum with double and half frequency of superimposed fluctuating cycles
Table 7-1. Fatigue life results with different spectrum characteristics

<table>
<thead>
<tr>
<th>Test Case No.</th>
<th>Title of Applicable Load Spectrum</th>
<th>Life in terms of spectrum pass</th>
<th>Difference of Results</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.</td>
<td>Base line spectrum</td>
<td>2892</td>
<td>Reference Value</td>
</tr>
<tr>
<td>2.</td>
<td>Only mean loads with no vibration</td>
<td>3331</td>
<td>15 % higher fatigue life</td>
</tr>
<tr>
<td>3.</td>
<td>Double load amplitudes about the same mean loads and the same frequencies</td>
<td>2674</td>
<td>7.5% less fatigue life</td>
</tr>
<tr>
<td>4.</td>
<td>Half load amplitudes about the same mean loads and the same frequencies</td>
<td>3109</td>
<td>7.5% higher fatigue life</td>
</tr>
<tr>
<td>5.</td>
<td>Double frequencies with same loads</td>
<td>2840</td>
<td>1.8 % less fatigue life</td>
</tr>
<tr>
<td>6.</td>
<td>Half frequencies with same loads</td>
<td>2944</td>
<td>1.8 % higher fatigue life</td>
</tr>
</tbody>
</table>

Fig.7-7 shows the effect of change in fibre angle upon percentage increase of dynamic stability and resultant decrease of fatigue life. The presented envelop covers the relationship among fibre angles, dynamic stability and fatigue life in terms of percentage. It may be inferred that the composite panels exhibit maximum dynamic stability (axis-B) at optimum fibre angle (axis-A) of 45 degree. Similarly, the fatigue life (axis-C) gradually decreases with the increase of fibre angle from 0° to 90°. This envelop may help to design a suitable configuration of aerospace structure keeping in view the multidisciplinary constraints.
Fig. 7-6. Fatigue life comparison with changed spectrum characteristics.

On the basis of numerical iterations using progressive failure and delamination analysis techniques under FALSTAFF spectrum, results are presented in Fig. 7-8. It was observed that virtual crack closure technique gave the most conservative results of failure index after fifteen return periods of standard FALLSTAFF spectrum. One return period contains 16000 cycles. Complete failure corresponds to failure index = 100. In addition to fatigue failure, it is also very important to mention that whenever carbon fibre composite is required to be joined with aluminium alloys (as in case of test specimens) there is always a probability of corrosion. To avoid it, a thin glass fibre layer may be introduced between joining materials.
Fig. 7-7. Effect of fibre angle on dynamic and static stability

Fig. 7-8. Failure index comparison among DTA techniques
7.3 Conclusions

Based upon the experimentation and simulation work performed on a carbon fibre sandwich panels, following conclusions may be drawn safely for the application to similar structural members of an aircraft while performing F&DT analysis.

- Wing roots are fatigue critical locations which undergo low frequency fatigue loads due to aircraft manoeuvres and high frequency superimposed vibrational fatigue loads due to aerodynamic flow separation at high angle of attacks. The similar superimposed fatigue loads may also be due to gust vibrations.

- Structural dynamic characteristics are altered in aero- elastically tailored design to achieve higher speed of aircraft without flutter. This approach mostly compromises the stiffness of wing’s bending mode. Decreased stiffness gives lower natural frequency and consequently higher fatigue loading amplitudes during flight. Hence, the fatigue life and damage tolerance of optimum tailored design becomes lower than the un-tailored design.

- In most of the test cases for aircraft wings or tails, dynamic instability is governed by the coupling of first torsional and the first bending modes of vibration. Keeping in view the vulnerability of these modes, the fibre direction in the plies may be equally aligned towards maximum eigenvectors of both the modes. In this way, the stiffness of these modes may be kept higher to prevent their flutter.

- Fatigue analysis of different aircraft parts is generally carried out using cycles count from standard normal acceleration (Nz) data of CG sensor. However, the vibration of associated structures (heavy weight) must not be overlooked and the additional stress reversals due to vibration of associated structures should also be applied at higher CG
load factors. The fatigue life results without superimposed fluctuating loads are much higher than the results by considering the vibrations effect.

- Modal analysis simulations for additional fatigue load contributing structures may be carried out to estimate additional loading amplitudes, for their incorporation to give combined cyclic loading spectrum. Fatigue analysis is generally performed to critical parts such as joints and spars near aircraft wing root.

- If the material under investigation is required to be used in complicated parts, the devised methodology of incorporating structural dynamics in fatigue load spectrum may overcome the extensive additional LIM-Standard requirements of separate experimental buffet and vibration testing for qualification of aerospace structure.

- While performing the failure analysis simulations, Virtual Crack Closure Technique may be adopted as the most conservative approach among other techniques.

- While incorporating final factor of safety, FE model calibration and correlation factors may also be included for greater confidence in the design.

7.4 Future Work

Micro mechanics based failure criteria to analyse composite structures, as discussed in literature survey, may also be explored for comparison with phenomenological approach. Additionally the loading spectrum with Environmental effects, i.e. EN-STAFF may also be applied to incorporate the hydrothermal effects of composite structure for better accuracy and more realistic results.
7.5 Limitation of the Work and Proposal to Overcome:

Limitations of the work primarily include the availability of flight loads data exactly at the critical structural location being investigated against fatigue and damage tolerance. This limitation may be overcome by proper instrumentation at the location of interest during flight testing. This instrumentation may include the installation of 3-axis strain gage rosettes to measure stress state at certain points and the application of full bridges strain gages at critical location to measure the bending strains. These strains may be further converted to in flight loads after proper ground based calibrations. This setup also requires a dedicated on board data acquisitioning system. This data may be used for direct measurement of applicable fatigue load spectrum. In this case there may be the only requirement of applying race track algorithm for data reduction and rain flow cycle counting algorithm to calculate number of cycles.
REFERENCES


Appendix- A
International Journal Of Fatigue

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"Waqas Anwar"
has acted as a reviewer for the journal during 2017

The Editors of International Journal Of Fatigue
Elsevier, Amsterdam, The Netherlands
Appendix - B
Effect of structural dynamic characteristics on fatigue and damage tolerance of aerospace grade composite materials

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A R T I C L E   I N F O

Article history:
Received 2 October 2016
Received in revised form 24 December 2016
Accepted 19 January 2017
Available online 23 January 2017

Keywords:
Structural dynamics
Composite materials
Fatigue analysis
Multidisciplinary design optimization

A B S T R A C T

In aircraft structural integrity analysis, the damage tolerance and fatigue life is investigated against a cyclic loading spectrum. The particular spectrum includes the stress/loading levels counted during a flight of certain duration. The occurrences of load factors may include higher gravitational acceleration ‘g’ levels. While maintaining a certain g level occurrence at higher angle of attack, wing structure vibrates with the amplitudes of its natural frequencies. The cyclic stress amplitudes of vibration depend upon the natural frequencies of vibrating structure, i.e. lower frequency gives higher amplitudes and vice versa. To improve the dynamic stability, modal parameters of simple carbon fibre sandwich panels have been adjusted by tailoring the fibre orientation angles and stacking sequence. In this way, the effect of change in structural dynamic characteristics on fatigue life of this simplified structure has been demonstrated. The research methodology followed in this work consists of two phases. In the first phase, aero-elastically tailored design was finalized using FEM based modal analysis and unsteady aerodynamic analysis simulations followed by experimental modal analysis. In the second phase, fatigue and damage tolerance behaviour of material was investigated using different fracture mechanics based techniques. ASTM’s standard practices were adopted to determine material allowable and fracture properties. Simulation work was performed after proper calibration and correlation of finite element model with experimentally determined static and dynamic behaviour of panels.

It has been observed that the applicable cyclic loading spectra, as major input parameter of fatigue analysis, largely depend upon the natural frequencies, damping and the stiffness of the structure. The results and discussions of the whole exercise may be beneficial while carrying out aero-elastic tailoring of composite aircraft wing. This research work has also a positive contribution towards multidisciplinary structural design optimization of aerospace vehicles. © 2017 Elsevier Masson SAS. All rights reserved.

1. Introduction

General Aircraft Structural Integrity Program (ASIP) activities include: static & fatigue aspects, environmental degradation and structural life assessments. Additionally, damage tolerance analysis is often required to prevent the failure of structure as a result of fatigue cracks [1]. The objective of the damage tolerant design requirement is to demonstrate adequate residual strength in the presence of flaws for specified periods of service usage. Fatigue and Damage Tolerance (F&DT) analysis gives the service life of critical load bearing structural members by taking into account the material properties, initial crack/damage level and the applicable cyclic loading spectrum. Commercially available Damage Tolerance Analysis (DTA) software codes are based upon different material models from which NASGRO is considered the latest model in metals. This material model incorporates crack growth rate data of material at different stress ratios [2,3]. Whereas, in case of composite structures, there are many analytical models under consideration to analyse damage, however the validation of newly developed models is still a challenge. Currently, some phenomenological and micro level theories are being used to predict onset of damage. Besides material failure criteria, there are many other things of consideration when dealing with composites, such as the inclusion of thermal stresses in structure, moisture effects, in plane shear strength and non-linear behaviour of the material [4]. One of the most important parameters to be considered in aerospace structures is the aerodynamic vibrational loads that depend upon structural dynamic characteristics [5]. In certain cases of aircraft’s critical structural location, the applicable loading spectrum can’t be measured directly by installing accelerometers. One of the limitations may be the inaccessibility of high stress concentration area.

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http://dx.doi.org/10.1016/j.ast.2017.01.012
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Secondly, the flight testing for fatigue loads calculation is also an expensive activity. In this way, the fatigue spectrum for different critical structural locations is generally generated by using the easily available N-2 acceleration data of the sensor installed at centre of gravity (CG). If it is required to carry out F&D analysis on a critical member at the root of an aircraft wing, the cyclic loading spectrum for this particular member will be governed by the N-2 acceleration occurrences count of aircraft and the vibrations of the wing. In this way, any change in natural frequencies of the wing should be considered while declaring the fatigue life. In Fig. 1, time domain data of 02 accelerometers mounted on wing tip and centre of gravity have been presented. As compared to fuselage, the wing vibrates with higher amplitudes while maintaining an occurrence of 5 g level.

Keeping in view the close relationship among structural dynamics, static strength and fatigue life of both the metals and composite parts of the aircraft, the effort has been carried out to establish a relationship among these parameters for multidisciplinary design optimization. Failure index of composite part against ply material failure and delamination propagation has been investigated using latest fracture mechanics based techniques such as: progressive failure analysis, cohesive zone modelling and virtual crack closure technique.

2. Research methodology

In this research, the pattern of engineering work is in line with the standard practices of aerospace industry. However, the investigations have not been performed on proper aircraft wing. A simplest design of carbon fibre sandwich panels along with an aluminium attachment has been used for the proof of concept, as shown in Fig. 2 this approach is adopted to reduce the overall design space of work. Such problems become complicated when the output of each parameter does not only depend on the independent design variables, but often on each other as well. Multi-disciplinary design optimization studies are usually conducted to achieve a better agreement among the parameters under consideration [6,7].

The overall methodology adopted in this work comprises of two phases. In first phase aero-elasticity improved design of panels was finalized using finite element based numerical simulations followed by the experimental validation and correlation of results. Dynamic stability was improved by setting up a certain orientation of carbon fibres.

For this purpose two sets of tailored and un-tailored composite panels were manufactured and went under extensive experimentation. Structural dynamic characteristics such as natural frequency, damping, generalized stiffness and generalized mass against each vibrational mode of panels were the input parameters to unsteady aerodynamic analysis. This analysis was performed to prove a positive change in aerodynamic stability of panels. Flutter speed was calculated for both the tailored and un-tailored panels. Flow chart of engineering activities adopted in first phase is shown in Fig. 3. Output of first phase is an experimentally verified aero-elastic design with altered structural dynamic characteristics.

In second phase, where the tailored and un-tailored test articles are available, the static and fatigue strength of both the designs was investigated. The objective of this phase was to investigate the effect of altered dynamic characteristics on structural integrity and fatigue life. Software based simulations for progressive damage analyses were carried out under applicable cyclic loading conditions for both the designs. For updating and fine tuning the FEM model, static response of both designs was also correlated with experimental results. Basic strength parameters and fracture properties of composites were experimentally determined in laboratory using genetic material coupons. Fig. 4 shows the activities flow chart adopted in phase-II.

3. Modelling & simulations

An appropriate size of test articles was selected to demonstrate the whole exercise for the proof of concept. The overall dimension
of carbon fibre sandwich panel was 490 mm × 90 mm with three plies of 1 mm on both sides. 3 mm hexagonal honeycomb was used as a core material. Fig. 2 shows the carbon fibre composite panel with AA 7075-T6 attachment.

Modelling the honeycomb core in a right way is very important from FE analysis point of view. Fig. 5 shows three ways of hexagon array to cut the extruded solid material. With respect to a reference dimension of in-circle, horizontal and angular array is the correct way of modelling the honeycomb core. Whereas, the vertical array produces irrational dimensions among hexagons which further causes meshing error during finite element analysis.

Test articles were manufactured using carbon fibre layups followed by vacuum bagging process. Fig. 6 shows the prepared test articles and generic coupons for fracture toughness testing. Keeping in view the hygro-thermal sensitivity of composite material, proper packing was ensured while transporting samples to testing facilities.

Two different FEM based software codes have been used to get the results of numerical simulations. In the first software modelling & simulation was supported in a same package, whereas IGES format was used for the second.

### 3.1. Finite element model development

An accurate, validated, correlated and updated finite element model is the foremost objective of any engineering analysis. The finite element (FE) based model was developed by using the same constituent material properties obtained through supplier’s data sheets and lamina level laboratory testing.

During manufacturing process, vacuum bagging was used to pull out excessive resin from the honeycomb core. However, 100% resin couldn’t pull out of core; this penetration makes the core material more strong. Fig. 7 shows the penetrated resin at upper and lower side of core.

In this case, it became necessary to correlate the stiffness of FE model with the same structure as manufactured under real environmental conditions. The density of honeycomb material was increased in such a way to achieve same macro level response of the panels. This type of FE model correlation becomes essential due to deviation in manufacturing parameters. An ideal FE model cannot be used for realistic engineering analysis without its static or modal response correlation with the actual physical object. The difference between the designed models and manufactured items arises due to manufacturing process limitations. The actual product contains discontinuities which makes the difference in designed stiffness, damping, mass distribution and moments of inertia.

In model calibration activity the physical properties of finite element model were adjusted in such a way to match its behaviour with the physical object. To get high fidelity model, the panels were calibrated using following two techniques:
3.1.1. Calibration through static load

The panels were loaded as a cantilever beam using a test wall setup. Both the tailored and un-tailored panels were equipped with a strain gauge at root side and a linear variable displacement transducer (LVDT) at the tip side. A hydraulic actuator applied the force at tip side and the response was recorded using a properly configured and synchronised data acquisitioning system. The results were later got matched with the FE model response. Fig. 8 shows the experimental test setup for finite element model calibration.

The instruments such as strain gauges and LVDTs were properly calibrated before the test. The hydraulic actuator was operated in position control at 20 mm per minute rate. Fig. 9 shows the extent of linear elastic region calibrated with experimental response. As the samples are the sandwich panels, so there are two steps of load decrease ahead of the linear elastic region. The first down step in the machine load is the failure of lower ply and the second down step in the load is the progressive failure of upper ply of panel. The finite element model was correlated only within the linear elastic response of the panels. Primarily the stiffness characteristics were adjusted in such a way to get a best agreement between numerical and experimental response.

3.1.2. Calibration through modal analysis

To determine the dynamic response of panels, experimental modal analysis was performed with 27 miniature accelerometers. In Fig. 10, experimental modal analysis setup is shown.

The sensor shown in the figure is 1.5 gram weight piezoelectric accelerometers along with low noise cable which is connected to the charge amplifier. It is an important technical observation that modal shaker and sensors should be used in such a manner without disturbing the natural characteristics of the structure under test. During calibration phase, before the actual test, it was ensured that the shaker produces a stable sinusoidal output frequency characteristics. Further it was ensured that the force output was relatively independent of the vibrational amplitudes of the test articles. The shaker used in this test is electro-dynamic shakers which are based on the Laplace law [8]. Table 1 and 2 represent the structural dynamic characteristics of both the un-tailored and tailored design respectively.

The stiffness of honeycomb structure was changed in such a way to match the modal characteristics of FE model with experimental results. Only the four natural modes of vibration were used to correlate with numerical simulation results i.e. the first, second, fourth and fifth mode. Table 3 shows the maximum agreement achieved between numerical simulation results and the experimental behaviour of un-tailored panels.

Force appropriation method was used to isolate the vibrational modes of the panels. The acquired response consists of resonant and off-resonant components. The applied force at a single frequency is adjusted in such a way that off resonant components of the response are minimized and the respective mode is isolated accurately with almost nil contribution of other modes. After modal isolation using Force Appropriation Method, Complex Power Method was used for extraction of modal parameters by giving micro frequency sweep around a particular resonant frequency.

It was desired that all the normal modes should be orthogonal to each other. Modal Assurance Criteria (MAC) was utilized to evaluate the accuracy of experimentally determined modes. MAC gave an indication of the correlation between two shapes. For ideal
orthogonal modes, off diagonal terms of MAC matrix should be zero if compared to unit diagonal terms. However, for experimental modes its values up to 15–20%, indicate modes are reasonably orthogonal. Fig. 11 shows the orthogonality of experimentally determined modes.

Φᵢ = the mode shape vector of the ith mode
Φⱼ = the mode shape vector of the jth mode
If, i = j MAC = 1 or 100%,
If, i ≠ j MAC = 0 or 0%

As per applicable standard A-8870C (AS) all the calculated off-diagonal elements of the orthogonality matrix should not be more than 10% of unit diagonal element [9].

Quality Factor (QF) is measure of how well the off resonant components of the complex response are isolated from the resonant components. The relative importance of the measured imaginary and real response at resonance thus provides the information on the quality of Force Appropriation Method.

\[
QF = 1 - \frac{y_j}{y} \quad (1)
\]

where \(y_j\) is the off resonant component of response and \(y\) is the total response. The \(QF = 1\) represents the perfect appropriation of the mode. However, minimum \(QF = 0.7\) has been taken as a good criteria for this particular testing.

3.2. Material properties

Following generic mechanical properties of composite coupons were determined in laboratory:

![Orthogonality of each mode using MAC graph.](image)

**Table 1**
Model characteristics of un-tailored panels.

<table>
<thead>
<tr>
<th>Modes No.</th>
<th>Basic parameters</th>
<th>Normalized generalized mass (kg.m²)</th>
<th>Optimization parameter</th>
<th>Modes description</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Frequency (Hz)</td>
<td>0.00913</td>
<td>0.39011</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Damping ratio</td>
<td>0.1423972E+003</td>
<td>1.3623475E-007</td>
<td>First bending</td>
</tr>
<tr>
<td>1</td>
<td>19.100</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2</td>
<td>1741.00</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>3</td>
<td>474.700</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>4</td>
<td>671.200</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>5</td>
<td>1167.40</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>6</td>
<td>1280.70</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

**Table 2**
Model characteristics of tailored panels.

<table>
<thead>
<tr>
<th>Modes No.</th>
<th>Basic parameters</th>
<th>Normalized generalized mass (kg.m²)</th>
<th>Optimization parameter</th>
<th>Modes description</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Frequency (Hz)</td>
<td>0.00906</td>
<td>0.30180</td>
<td></td>
</tr>
<tr>
<td></td>
<td>Damping ratio</td>
<td>0.0553725E+003</td>
<td>0.9958675E-007</td>
<td></td>
</tr>
<tr>
<td>1</td>
<td>13.460</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>2</td>
<td>119.100</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>3</td>
<td>329.100</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>4</td>
<td>684.700</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>5</td>
<td>997.800</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>6</td>
<td>1190.88</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

**Table 3**
Modal calibration agreement of un-tailored panel.

<table>
<thead>
<tr>
<th>Description of vibrational mode</th>
<th>Experimental frequency value (Hz)</th>
<th>Numerical result value (Hz)</th>
<th>% Error between FEM &amp; Exp. value</th>
</tr>
</thead>
<tbody>
<tr>
<td>First bending (M-1)</td>
<td>19.1000</td>
<td>17.90</td>
<td>–6.28%</td>
</tr>
<tr>
<td>Second bending (M-2)</td>
<td>174.100</td>
<td>167.3</td>
<td>–3.90%</td>
</tr>
<tr>
<td>First torsion (M-4)</td>
<td>671.200</td>
<td>579.0</td>
<td>–13.7%</td>
</tr>
<tr>
<td>Third bending (M-5)</td>
<td>1167.40</td>
<td>1210</td>
<td>+3.65%</td>
</tr>
</tbody>
</table>

1. Compressive & Ultimate Tensile Strength (UTS) of UD lamina as per ASTM standard D3039
2. Bending strength using 3 point & 4 point bend test as per ASTM standard D790
3. Inter laminar Shear Strength (ILSS) of laminate as per ASTM standard D2344

**Table 4** shows different properties used to develop a finite element model.

3.3. Modal analysis simulation results

First six natural frequencies of the panels were considered significant to analyse the unsteady aerodynamic behaviour. Generally, the lower frequency modes are excited while normal operational environmental conditions of aircraft. However, FEM based modal analysis simulation may extract as many modes as the designer...
Table 4

Mechanical properties (engineering constants).

| Carbon/epoxy | 
|-----------------|-----------------|-----------------|-----------------|
| Mechanical property | Direction 1 | Direction 2 | Direction 3 | Source |
| Young's moduli MPa | $E_1 = 153,700$ | $E_2 = 8,960$ | $E_3 = 8,960$ | Testing |
| Poisson's ratios | $\nu_{12} = 0.31$ | $\nu_{13} = 0.31$ | $\nu_{23} = 0.52$ | Testing |
| Shear moduli MPa | $G_{12} = 4980$ | $G_{13} = 4980$ | $G_{23} = 2600$ | Testing |

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<td>Mass density</td>
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</tr>
<tr>
<td>Thermal expansion coefficient in $0^\circ$</td>
<td>$-0.3$</td>
<td>Strain/K</td>
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<td>Moisture expansion coefficient in $90^\circ$</td>
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<tr>
<td>Thermal conductivity</td>
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<td>W/(m$\cdot$K)</td>
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<tr>
<td>Specific heat</td>
<td>1386</td>
<td>J/(kg$\cdot$K)</td>
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| Honeycomb | 
|-----------------|-----------------|-----------------|-----------------|
| Mechanical property | Value | Units | Source |
| Young's moduli compressive | 0.418 | GPa | Data sheet |
| Shear modulus | 0.0538 | GPa | Data sheet |
| Poisson's ratios $\nu_{xy}$ | 0.49 | Nan | Data sheet |
| Compressive yield strength | 12.1 | MPa | Data sheet |
| Shear strength | 3.24 | MPa | Data sheet |
| Thermal expansion coefficient | 19.4 | $\mu$ Strain/K | Data sheet |
| Density | 0.128 | g/cc | Data sheet |

**Fig. 12.** First six mode shapes determined through FE analysis.

4. Aero-elastic tailoring

Aero-elasticity is an interaction of aerodynamic, elastic and inertial forces. Structural and aerodynamic behaviours of wing are usually optimised by varying composite fibre angles and layup sequence. Few parameters which are needed to be considered while tailoring are the lift to drag ratio, static strength, flutter and divergence speeds and flight control. Significant work has been done in aero-elastic tailoring since early 1990s and it was first carried out on X-29 aircraft [10].

The scheme involved in this work is the change in stiffness of panels and the passive aero-elastic coupling between bending and torsion modes. In these composite panels, the fibre angles and layup sequences have been tailored to optimize bending–torsion coupling over entire span. The process was constrained to some extent by the limitation of non-availability of toe steering equipment. The fibres have been arranged to follow a curvilinear paths requires. Most of them may not be useful if considered during flutter prediction calculation and may also consume a lot of computational time. Significant mode shapes determined through numerical simulations of carbon fibre sandwich panel are shown in Fig. 12.
such that the fibre angles and lamina stiffness changes continuously through the plane of each lamina. It was found in the latest research work that these fibre paths increase the dynamic structural performance of VAT panels than that of equivalent uni-directional laminates [11,12].

The fibres were placed in such a way to maintain minimum orthogonality against maximum eigenvectors of vulnerable modes i.e. first bending and first twist. In this way the optimum [±45, −45/0], stacking sequence was adopted for tailored design and [0/90], adopted for un-tailored design. In first two plies of tailored design, the fibre angle gradually changes from 0° to + and −45° as the length increases from root to tip. Modal response of both the tailored and un-tailored designs were investigated in experimental vibration testing.

5. Unsteady Aerodynamic Analysis

Unsteady Aerodynamic Analysis was performed to check the instability of panels subjected to different Mach from 0.5 to 1.2 at different altitudes. Velocity vs. Frequency (V-F) and Velocity vs. Damping (V-G) curves of un-tailored design at 0.7 Mach number are presented in Figs. 13 and 14 respectively. Whereas, the V-F curves of tailored design are presented in Fig. 15. The air speed at x-axis is the equivalent air speed mentioned in meter per seconds.

Prior to the computation of flutter predictions, transformation of modal data over aerodynamic grid was also carried out and mode shape animations were checked to verify the spline of sensors over the aerodynamic grid.

Flutter analysis was carried using ZONAS and ZONAT for subsonic and supersonic respectively. ZONAT is the modified form of ZONAS1 employs the acceleration potential approach for thin plate type of lifting surfaces. ZAERO flutter module contains two flutter calculation techniques, the K-method and the g-method. The g-method is generalized K-method and P-K method for true damping predictions. P-K is only valid at the condition of zero damping, zero frequency or linearly varying generalized aerodynamic forces (Q_{ij}) with respect to reduced frequency. In fact, if (Q_{ij}) is highly nonlinear, P-K method may produce unrealistic roots due to its inconsistent formulation. To be more conservative during unsteady aerodynamic analysis, zero structural damping was considered for the analysis purpose.

As shown in VF & VG curves, two modes are critical, the bending mode and the torsion mode. Un-tailored design shows the flutter air speed at 1500 m/s Veas (2900 Knots) whereas tailored design shows it at 1900 m/s Veas (3700 Knots). However from Fig. 16, it can be seen that this speed is far from achievable region even at 0 m height.

6. Material characterization

For engineering analysis, standard and reliable material characterization data of laminated composites is also needed as input to analytical models of structures to predict onset of failure. Fracture toughness is one of the most important parameters to analyse fatigue strength. Double cantilever beam specimen was used with laminated composites to measure fracture toughness and strain energy release rate data under cyclic loading. Generic specimen test data of composite material is not only applicable to a single analysis, but in multiple field applications [13,14]. Fig. 17 shows one of the double cantilever beam specimens used for inter-laminar fracture toughness testing of unidirectional lamina installed on testing machine.

Composite structures are usually prone to inter-laminar fracture that is the failure between the lamina or plies. Common cause of theses failure is the impact load or the stresses on pre-existing cracks due to shear load. In-plane shear load usually leads to Mode-II fracture and the compression loads lead towards layers buckling [15]. So, Mode-I and Mode-II inter laminar fracture
toughness are the most important material characterization in fatigue and fracture mechanics analysis.

Fig. 17 shows the testing of mode-I inter-laminar fracture toughness testing under ambient environment. Due to non-availability of thermal chamber, the hygro-thermal effects have not been measured experimentally. However, reliable material database has been used for reference where applicable.

There are three common methods for DCB test data reduction: the first is Modified Beam Theory (MBT), the second one is Compliance Calibration (CC) and the third method is Modified Compliance Calibration (MCC). Strain energy release rate has been calculated using all the three formulations. In Fig. 18, the results of strain energy release rate calculated using all the three methods are plotted against crack growth on a single graph.

In this research work all the tests were performed as per ASTM Standard. It is important to mention that the DCB test has been standardized by ASTM only for unidirectional composites [16]. However, these guidelines are also used for testing of woven fiber composites [17,18].

Static strength properties were also determined by experimental testing. Table 5 shows the static strength related properties of composite material under investigation. Fig. 19 shows the physical samples used to determine basic mechanical properties in addition to fracture toughness testing.

The tensile, compressive and shear strength of composite coupons in third direction were not determined in laboratory. However the same average percentage difference of these strength values was incorporated in the standard published database which was calculated after testing of other parameters.

7. Applicable load spectrum

One of the most important input parameters to analyse fatigue and damage tolerance of structures, is the applicable loading spectrum. The baseline loading data is acquired from a sensor mounted at centre of gravity of any aircraft which is called ‘NZ Spectrum’. There are number of cycles counting techniques to analyse fatigue spectrum from raw accelerometer data. These techniques include the level cross counting method, peak counting and simple range counting method etc. In this work, Rain flow counting algorithm was developed in MATLAB software and the effective fatigue cycles were calculated at 03 load factors. The selected load factors are 2g, 3g and 4g levels. At each load factor, there are particular vibrational loadings which are applied to represent the flutter. The race track method has been adopted for data reduction purpose. Appropriate omission and truncation levels have been selected as per ASTM STP1006 and current state of the art [19,20].

Using finite element based stress analysis; the maximum stress value on critical structural location was calculated at worst load case. Therefore, a particular stress multiplication factor was included for application of normalized fatigue load spectrum. This loading spectrum has been further used in progressive damage analysis.

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<td>Testing</td>
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<td>Compressive strength in fibre direction X1c</td>
<td>103e6</td>
<td>N/m²</td>
<td>Testing</td>
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<td>48e6</td>
<td>N/m²</td>
<td>Testing</td>
</tr>
<tr>
<td>Compressive strength in transverse direction X2c</td>
<td>1588e6</td>
<td>N/m²</td>
<td>Testing</td>
</tr>
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<td>Tensile strength in third direction X3t</td>
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<td>N/m²</td>
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<td>Shear strength in YZ plane S23</td>
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Fig. 17. Double Cantilever Beam (DCB) test specimen undergoing testing.

Fig. 18. Material property (Strain energy release rate).

Fig. 19. Standard composite samples for tensile and 3 point bend test.
Fig. 20. Standard normalized fatigue loading spectra.

Fig. 21. Initial damage characterization for composite laminate. (For interpretation of the references to colour in this figure, the reader is referred to the web version of this article.)

Fig. 22. Five technologies to analyse composite structure failure.

On the basis of available data, ply delamination type 03 (as shown in green colour of above figure) was selected as the initial damage/ crack which may be due to interface of metallic attachment with composite material (as shown in Fig. 2). The damage was characterized as an initial delamination of length 15 mm at one direction of the panels. Damage incorporation depends upon the selected F&D technique as discussed comprehensively in the next section.

9. Damage tolerance analysis techniques

Composite panels were analysed against progressive damage analysis keeping in view the applicable loadings scenario against their application. Ply failure, crack propagation and delamination prediction gave the failure index and strength ratio using different failure criteria. First ply linear failure analysis has been considered as the ground level for these composite panels. The critical result of such failure analysis is the stress redistribution after first ply failure and its continuous calculations as the plies fail progressively. Residual strength continuously decreased as the delamination grew. Four different methods have been used to analyse the failure. Fig. 22 shows that the five technologies fall under two categories: ply material failure and delamination.

Both the shell and solid elements can be used to analyse the failure scenarios. If the layered composite laminate is defined as a shell element, mainly the in-plane behaviour is investigated such as bending. However, progressive failure analysis can also be performed using shell elements. When loading becomes three dimensional and the load path involves the third dimension, it becomes necessary to calculate stresses and strains in third direction. In such applications we use solid elements to determine stresses at each ply. Solid elements sometimes don’t bend very well due to limitations of their shape function, in these cases of solid elements under bending; a special string function is added to make them solid-shell elements. The same technique was used to model proper bending in tailored and un-tailored panels.

Fig. 23 illustrates three types of elements which can be used to define a complete structure in finite element based techniques.

9.1. Progressive Failure Analysis (PFA)

In Progressive Failure Analysis, the failure criteria offered in different FEM based software packages are Puck, Hashin, Tsai Wu, Max Stress and Max Strain [24]. Multiple criteria were used in panels analysis, first were used to determine failure while the others were used to calculate failure indices only. As the plies start to fail layer by layer, and when a particular ply fails, it became necessary to fail...
to deactivate it by scaling down its strength moduli; Young's modulus 'E' and shear modulus 'G' to make the ply soft. Hence the ply was taken out of the load path and the load was distributed to the other plies and surrounding structure. Hence, the crack or delamination propagated by deactivation the elements.

9.2. Virtual Crack Closure Technique (VCCT)

Virtual crack closure technique is a fracture mechanics based model. In linear fracture mechanics a crack begins to grow when G (strain energy release rate) becomes greater than Gc (inter laminar fracture toughness as determined in previous section). VCCT is one of the methods to calculate energy release rate which has been frequently implemented in commercially available software packages. In Fig. 24, a mesh is shown which has an existing crack in the model. When the structure is loaded in such a way that the crack tries to open up with displacement 'u', there is a force 'F', acting at crack tip. With existing crack 'a' the energy release rate can be calculated as following:

\[ G = F \times u / 2a \]

(2)

All the three modes of crack propagation can be incorporated in this technique, i.e. opening, shear and tear mode. However, only the first mode has been analysed due to availability of critical strain energy release rate only in mode-1 direction. VCCT propagates the crack in different ways, the first way is with glued interface, where a crack is modelled between two contact bodies glued together. The cracked area is not joined and the nodes of rest portion are glued together, in this way the crack tip nodes are properly identified. The path of crack can be seen in graphics of latest FE based commercial software codes. Two joined bodies are unglued when VCCT criteria is met [25].

On the basis of calculated stresses, the program decides where the crack has to go. If Quad mesh is selected, the crack travels through 0° or 90°. However, if triangular mesh is adopted, the crack growth will be along different path. The most important feature of VCCT adopted in the analysis of panels is its growth with re-meshing. As the crack is growing, the computer program looks ahead of crack tip at maximum stresses and determines in which way they are going. Hence, it continues to re-mesh the body in front of crack tip and the crack grows toward its natural direction [26].

9.3. Cohesive Zone Model (CZM)

Cohesive zone modelling uses a special class of inner face elements to simulate delamination and crack growth [27]. The libraries of these special inner face elements to characterize the interface behaviour in FEM codes make CZM different from other analysis technologies. The inner face elements are sandwiched between structural solid elements. The elements are defined by looking into the traction stresses on top and bottom face of the elements; the shear and normal traction stresses [28,29]. It is required to map the traction stresses to the relative displacement between top and bottom faces. The relative displacement is plotted on the horizontal axis and stresses on vertical axis as shown in Fig. 25.

The figure describes a bilinear model, when the top and bottom face of structure deforms, traction stresses goes linearly. The critical relative displacement is defined by the user i.e. \( \Delta^0 \). When the solution reaches the final maximum relative displacement, the modelled element is totally damaged. The area underneath the curve is cohesive energy or the critical energy release rate. \( G_{IC} \) (already calculated in previous section). Material behaviour becomes irreversible when \( \Delta \) becomes greater than \( \Delta^0 \). Exponential or linear-exponential models may also be used for this behaviour. Ply and adhesive layers are meshed independently using higher order elements and then ‘contact’ option is used to glue the adhesive layer and plies together. Both the bodies can have different mesh densities. In comparison with VCCT, CZM involved no special requirement to model an initial crack. The progressive failure index of cohesive zone has been shown in coming section as a comparison of results with other techniques.
9.4. Breaking Glued Contact (BGC)

This includes most comfortable stress criteria in which calculated normal and tangential stresses are compared for a glued interface.

\[(\sigma_{\text{normal}}/S_n)^m + (\sigma_{\text{tangential}}/S_t)^n > 1\]  \[\text{(3)}\]

where \(m, n, S_n\) and \(S_t\) are the user defined parameters. This method requires the information about the area where delamination will occur to look at normal and tangential/shear interface stresses. \(S_n\) and \(S_t\) are allowable normal and shear stresses. FEM software calculates the developed stress to allowable stress ratios. The exponents \(m\) & \(n\) define the interaction between these stresses. If it is taken as 1, there will be linear relation between two terms. If the values are 2 and 2, it will end up with circular interaction. When the result is equal to 1, FEM software starts to release the contact as a result of failure.

10. Failure criteria

The type of onset failure analysis criteria adopted in this work is non-interactive model which does not take into account the interaction between stresses and strains acting on a lamina of panels. This model resulted into some errors in the strength predictions, when multi-axial states of stresses were applied. These are the typical maximum strain and maximum stress criteria. The second approach adopted to predict onset of failure was the interactive model. It took into account the interactions between stresses and strains acting on the lamina. Although there are different models available in literature such as Hashin–Rotem, Hashin (only) and Puck [30]. In contrast with phenomenological approach, these micro level criteria consider both the fibre and matrix failure and sensitive towards tension and compression loadings.

The Hashin criterion is well incorporated in FEM software codes. This was applied under three dimensional state of stress as shown in the following equations. For the matrix failure mode, a quadratic attitude was selected because the linear criterion cannot accurately predict the material behaviour, and a polynomial of higher degree became very complex while modelling. Additionally, the effect of the shear stress was also taken into account in the fibre tension mode.

Fibre tension mode (when \(\delta_1 > 0\))

\[\delta_1/\delta_{1\text{ult}} = \left(\frac{(\delta_{12})^2 + (\delta_{13})^2}{(\delta_{12})^2}\right)^{1/2} = 1\]  \[\text{(4)}\]

Fibre failure in compression (when \(\delta_1 < 0\))

\[-\delta_1 = \delta_{1\text{ult}}\]

Matrix failure in tension (when \(\delta_2 + \delta_3 > 0\))

\[\left\{\left(\delta_2 + \delta_3\right)/\delta_{2\text{ult}}\right\}^2 + \left\{\left(\delta_{23}\right)^2 - \delta_{23\text{ult}}\right\}/\delta_{23\text{ult}}^2 = 1\]  \[\text{(5)}\]

Matrix failure in compression (when \(\delta_2 + \delta_3 < 0\))

\[\left\{\left(\delta_{2\text{ult}}/\delta_{23\text{ult}}\right)^2 - 1\right\} \left\{\left(\delta_2 + \delta_3\right)/\delta_{2\text{ult}}\right\} + \left\{\left(\delta_2 + \delta_3\right)^2/\delta_{23\text{ult}}^2\right\} + \left\{\left[(\delta_{23})^2 - (\delta_{2}\delta_3)\right]/(\delta_{23\text{ult}})^2\right\} + \left\{\left(\delta_{12}\right)^2 + (\delta_{13})^2\right\}/(\delta_{12\text{ult}})^2 = 1\]  \[\text{(6)}\]

Other criteria which are reported in various literature have been studied and may also be used in future work such as: Cuntze theory [31], Yamada Sun equation [32], Koop Michaeli criterion [33], Kroll Hufenbach theory [34], Sun Tao equation [35], Zinoviev criterion [36], Goss theory [37], and Hart Smith relationship [38].

Another criterion is the modified Kinetic theory, where, the calculated constituent stresses and strains are used in combination with kinetic theory to predict fatigue life in large-scale composite structures under variety of complex loadings that actually occur during the flight of aircraft. This method in software codes like Abacus and Helius MCT, joins the fatigue analysis with different finite element based tools and requires less composite fatigue characterization data in comparison to that mentioned in previous sections. In addition to its fast simulation, it also provides the needed estimates between physics based analysis and the mechanical structural analysis [39,40].

11. Mathematical modelling

Based upon the results of extensive experimentation and high fidelity numerical simulations, an empirical relationship has been developed to model the fatigue behaviour of composite panels against gradual change of fibre orientations. The newly developed model takes into account the following manufacturing parameters;

a. Rate of change of fibre orientation per unit span length
b. Strain energy release rate as a primary measure of damage tolerance
c. The effect of core thickness on results

During the development phase of mathematical model given in Eq. (1), first, fibre angles were related with the ply stiffness and the ply stiffness was related to strain energy release rate. Secondly, the common parameter of ply stiffness was cancelled out and the ply angles were directly related to the strain energy release rate of the panels.

\[d\theta/dS = C(\Delta G)^n - (1/2(My/I))\]  \[\text{(8)}\]

where, \(\theta\) is the fibre angle and \(S\) is the span length, ‘C’ and ‘n’ are material dependant on empirical parameters. The developed mathematical model is based upon the assumption that most of the dynamic instabilities are the result of coupling between bending and torsion modes of vibration. In case of any other predicted phenomenon, a detailed vibrational testing must be performed for qualification of structure as per applicable Standard 8870.

12. Results and discussions

Keeping the same material properties, under the application of respective loading spectrum, fatigue and damage tolerance was analysed using all the above mentioned techniques and failure criteria. The tailored and un-tailored designs gave different Fatigue and Damage Tolerance (F&DT) behaviours. Fig. 26 shows the effect of change in fibre angle upon percentage increase of dynamic stability and resultant decrease of fatigue life. The strength envelope covers the relationship among fibre angles, dynamic stability and fatigue life in terms of percentage. It may be inferred that the composite panels exhibit maximum dynamic stability (axis-B) at optimum fibre angle (axis-A) of 45°. Similarly, the fatigue life (axis-C) gradually decreases with the increase of fibre angle from 0 to 90°. This envelope may help to design a suitable configuration of aerospace structure keeping in view the multidisciplinary constraints.

On the basis of numerical iterations using above mentioned progressive failure and delamination analysis techniques, limited results are presented in Fig. 27. It was observed that virtual crack closure technique gave the most conservative results of failure index after fifteen return periods of standard FALLSTAFF spectrum. One return period contains 16,000 cycles. Complete failure corresponds to failure index = 100.
13. Conclusions

Keeping in view the rapid increase of using composite materials in aerospace applications, this exercise to predict mechanical behaviour of a simple part covers the necessary considerations while performing an engineering analysis. Based upon all the experimentations and simulation work performed on a carbon fibre sandwich structure, following conclusions may be drawn safely for their application to similar components of an aircraft wing or UAV structure:

a. Structural dynamic characteristics are altered in aerodynamically tailored design to achieve higher speed of aircraft without flutter. This approach mostly compromises the stiffness of wing’s bending mode. Decreased stiffness gives lower natural frequencies and consequently higher fatigue loading amplitudes during flight. Hence, the fatigue life and damage tolerance of optimum tailored design becomes lower than the un-tailored design.

b. In most of the test cases for aircraft/UAV wings or tails, dynamic instability is governed by the coupling of first torsional and the first bending mode of vibration. Keeping in view the vulnerability of these modes, the fibre direction in the plies may be equally aligned towards maximum eigenvectors of both the modes. In this way, the stiffness of these modes may be kept higher to prevent their flutter.

c. Fatigue analysis of different aircraft parts is generally carried out using cycles count from standard normal acceleration data of CG sensor. However, the vibration of associated structures (wing or stores) must not be overlooked and the additional stress reversals due to vibration of associated structures should also be applied at higher CG load factors.

d. Modal analysis simulations for fatigue load contributing structures may be carried out to estimate additional loading amplitudes for their incorporation to combined cyclic loading spectrum. Fatigue analysis is generally performed to critical parts such as joints and spars near aircraft wing root.

e. While performing the failure analysis simulations, Virtual Crack Closure Technique may be adopted as the most conservative approach among other fracture mechanics techniques.

f. While incorporating final factor of safety, FE model calibration and correlation factors may also be included for greater confidence in the design.

Conflict of interest statement

It is certificated that no conflict of interest is involved in this work. There is no involvement of any organization or entity with any financial interest. It is also certified that no intellectual interest (such as personal or professional relationships, affiliations, knowledge or beliefs etc.) in the matter or material have been discussed in this manuscript.

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