Kaunas University of Technology
Faculty of Mechanical Engineering and Design

Impact Damage Analysis of Aircraft Composite Structures at Low Velocity

Master's Final Degree Project

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Kaunas, 2018

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Vehicle Engineering (621E20001)

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Declaration of Academic Integrity

I confirm that the final project of mine, Cyril Varghese Thankachen, on the topic "Impact Damage Analysis of Aircraft Composite Structures at Low Velocity " is written completely by myself; all the provided data and research results are correct and have been obtained honestly. None of the parts of this thesis have been plagiarised from any printed, Internet-based or otherwise recorded sources. All direct and indirect quotations from external resources are indicated in the list of references. No monetary funds (unless required by law) have been paid to anyone for any contribution to this project.

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KAUNAS UNIVERSITY OF TECHNOLOGY
FACULTY OF MECHANICAL ENGINEERING AND DESIGN
Study programme VEHICLE ENGINEERING (621E20001)

TASK ASSIGNMENT FOR FINAL DEGREE PROJECT OF MASTER STUDIES

Given to the student: Cyril Varghese Thankachen

1. Title of the Project

- Impact Damage Analysis of Aircraft Composite Structures at Low Velocity
- Orlaivių kompozitinių struktūrų smūginio suirimo tyrimas esant mažiems greičiams

2. Aim and Tasks of the Project

To study the effects of the impact damages on the composite aircraft fuselage.

1. Review and analyses various scientific literature on composite damage mechanisms.
2. Formulate tests to study the effects of impact damages.
3. Build Finite Element models to carry out simulations to obtain results.

3. Initial Data:

- Literature review of Impact Damage analysis on Composite Fibre cards.
- Review of European Aviation Safety Agency Legislations for heavy aircrafts.
- Methodology for Finite Element investigation of Impact Damage on Composite Structures.

4. Main Requirements and Conditions

- Dassault Systemes SOLIDWORKS TM 2016 Student Edition.
- ANSYS Inc.18.1 CAE Software.
- The current model has been drafted for conditions that do not use flexible supports for holding the test sections.

5. Structure of the Text Part

- Introduction.
- Literature Review and theory – overview.
- Research Methodology and creation of Finite Element Models.
- Results of experimentation, discussions, and recommendations
- Conclusions
- List of References
- Appendices

6. Structure of the Graphical Part

7. Consultants of the Project:

Student: ....*Cyril Varghese Thankachen*..

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Supervisor..*Assoc.Prof.Dr. Kilikevicius Sigitas*...

(Name, Surname, Signature, data)

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(Name, Surname, Signature, data)

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Research field and area: Transport Engineering (E12), Engineering Science.
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Summary

In this thesis, a study of the effects of low velocity impact damages on Aircraft Composite Structures, will be carried out using Finite Element Modelling to understand the behaviour of the material. Two test models were tested upon to replicate the effects of load criteria, while an aircraft is subjected to high energy/low velocity impact. The test sections are 1 metre by 1 metre for the for the fuselage panel with stringer reinforcements and 1metre by 2 metres for the flat plate with cut-out sections. This was done to replicate the results in real – life working conditions. The use of the composite application module in ANSYS, Inc. 18.1 software has been heavily used in the thesis. The skin thickness was kept the same to match the real model of Airbus A350 XWB.

Cyril Varghese Thankachen. Orlaivių kompozitinių struktūrų smūginio suirimo tyrimas esant mažiems greičiams. Magistro baigiamasis projektas / vadovas, doc. Prof. Dr. Kilikevičius Sigitas, Mechanikos inžinerijos ir dizaino fakultetas, Kauno technologijos universitetas.
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Raktiniai žodžiai: kompozicinis, smūginė žala, ANSYS Inc., EASA CS-25.
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Santrauka

Šiame baigiamajame darbe analizuojamas nedidelio greičio smūgių žalos poveikis orlaivio kompozitinėms konstrukcijoms naudojant Galutinio Elemento Modeliavimą, siekiant suprasti medžiagos elgseną. Du bandymo modeliai buvo išbandyti pakartotinai apkrovos kriterijų poveikiui, o orlaiviui buvo taikomas didelis energijos / mažo greičio poveikis. Tiriamieji skyriai yra 1 metras ir 1 metras fiuzeliažo skydui su stringerio sutvirtinimais ir 1 metras - 2 metrai plokščiajai plokštei su išpjovomis. Tai buvo padaryta, siekiant atkartoti rezultatus realiame gyvenime. Baigiamajame darbe buvo labai naudojamas sudėtinis taikomųjų programų modulis "ANSYS, Inc. 18.1" programinėje įrangoje. Odos storis buvo toks pat, kaip ir realus "Airbus A350 XWB" modelis.

Introduction

The Aerospace industry is one industry that pushes the boundary of new technological advancements. From the early days of flight, the aerospace industry has seen growths in terms of speed, weight carrying capacity, and structural integrity. The usage of cloth, paper, etc., in the earlier days of aircraft design to the usage of metal, and the eventual usage of metallic alloys in the construction of aircraft structure was brought about due to industry's need for improved high performance structural materials. And true to its nature, the Aerospace Industry has continued-on with its research for such materials and thereby, have settled on the fact that composite materials tend to exhibit exceptional strength-to-weight(density) ratios and noteworthy physical properties. [16]

In essence, composite materials are designed from two or more constituent materials which exhibit different physical and chemical properties. The combined effect of the materials together differs quite variedly when compared to the properties of the individual component. The increased demand for the usage of composite materials is driven by the fact that it leads to a considerable decrease in aircraft structural weight. A decrease in structural weight decreases the fuel consumption of the aircraft; thereby, reducing the direct operating cost by increasing efficiency of the aircraft.

The most stand out product is the carbon-fibre reinforced plastic, or CFRP. They are essentially made up of carbon (reinforcing)fibers, locked into place with a plastic(epoxy) resin. This results in a product that offers higher strength-to-weight ratio when compared to metal and metal alloys and has shown to be less sensitive to corrosion and fatigue. In short, it is lighter than aluminum, stronger than iron, and corrosion-resistant when compared to both steel and iron. [15]

The Aerospace industry is widely known for its extensive testing of components before it can be used as a part of an aircraft, and therefore, the composite materials are subjected to nothing different. These composite materials are tested using different methodologies – real world tests as well as computer modelled testing.

The type of test that is carried out depends on many factors. Real scale testing would be the best possible method to determine the performance of the material. This can have a negative effect as that would require incurring heavy Research and Development (R&D) costs. Computer modelling was previously considered to be difficult to work with due to its lack of accuracy. The complex nature of the composite material made the accurate modelling of the structure problematic. But the current variety of software suites have been able to model the structure, as well as, mathematically calculate the performance, according to industry standards. This approach is highly favored as the cost of testing new materials can now be reduced to a greater extent. The current methodology follows using NDT testing to upgrade the inbuilt library of different Computer-Aided Engineering (CAE) software; to help understand the composite material behavior for a wider range of applications. [17]

The extensive use of composite material has also resulted in new challenges. The impact damage is a new area in composite material that needed researching and careful accounting for the impact-related damages. There are two types of impact damages: High Velocity Impact Damage and Low Velocity Impact Damages. High Velocity Impact Damages include mid-air bird strike, bullets or any other forms of high velocity projectiles. The previously used, metal-fashioned aircraft have a comprehensive and detailed service history and the workforce: manufacturers, regulators, and operators, around the globe are quite capable enough to handle situations involving damages to such aircraft. The current generation aircraft are making a move towards composites in their construction and that brings about new challenges as the effective directory for solving problems is being updated with each new day the composites are in service. [18]

Aim

To study the effects of the impact damages on the composite aircraft fuselage.

Tasks

To achieve the aforementioned aim, following steps will need to be followed to have a successful research in the same:

1. Review and analyses various scientific literature on composite damage mechanisms.
2. Formulate tests to study the effects of impact damages.
3. Build Finite Element models to carry out simulations to obtain results.

1. Literature Review of existing approaches in Finite Element Modelling of Composites

Repair of Damage in Aircraft Composite Sound-absorbing panels (by Aleksandr N. Anoshkin, Valeriy Y. Zuiko, Mikhail A. Tashkinov, Vadim V. Silberschmidt) focused on study of mechanical behaviour of the aircraft engine's sound-absorbing panels (SAPs) exposed to in-service damage. The study aimed at providing techniques for local repairs of the damaged SAPs and estimation of their structural residual strength, post-repair. The analysis involved studying fiberglass laminate panels with tubular core and perforated elements which were then subjected to a finite-element (FE) analysis. It also considered a through rupture as an in-service damage. The study suggested a vacuum-less method for local repair of defects using specialized equipment. The study involved modelling of sample SAPs models such as – pre-damage, post-damage, and post-damage plus repaired model. [1]

Damage Identification in Aircraft Composite Structures: A case study using various Non-Destructive Testing Techniques (by Andrzej Katunin, Krzysztof Dragan, Michał Dziendzikowskib). This involved the study of various methods to observe the damage caused to the polymeric composites that are used in the construction of modern day aircrafts. The study of the effectiveness in detection was done and documented under different categories such as: identification and localization of damage during early stages, complexity of damage and cost of inspection. The researches carried out a study on three composite models: glass-fibre reinforced plastic, hybrid composite with a core made of the same material and aluminum alloy sheets used as face layer, carbon fibre-reinforced plastic vertical stabilizer of a military aircraft that was extracted – model had barely visible impact and delamination Employed usage of PZT (Lead Zirconate Titanate) sensing, thermography, ultrasonic and vibration-based inspection to study the credibility of the tests and their usage in different environmental condition for inspection of aircraft elements. [2]

Fibre Reinforced Composites in Aircraft Construction (by C. Soutis). The paper reviewed the recent advances in composite material usage increase over conventional metallic aircraft structures and components. Study concluded with the end note that the advances will continue to grow up to 50 or more percent of the total aircraft construction. The increase will only be affected by the cost of manufacturing, on a larger scale. [3]

Structural Health Monitoring Techniques for Aircraft Composite Structures (by K. Diamanti, C. Soutis). Discussion of Health Monitoring Techniques for Aircraft Composites Structures currently available, based on the usage: Non-Destructive Evaluation and Structural Health Monitoring method. Emphasis on the study of emerging technologies like Embedded Ultrasonic NDE and suitability for laminated composite structures. The embedded ultrasonic is based on the PZT principles and provide active structural health monitoring, i.e., ability to evaluate the health of the laminates, on-demand and prediction of the remaining life. Finally, the use of laser doppler vibrometry (LDV), comparative vacuum monitoring (CVM) are some of the available SHM methods available in the aircraft industry. [4]

Stiffness and Fracture Analysis of Laminated Composites with off-axis matrix cracking (by Maria Kashtalyan). The effects of static or fatigue in-plane tensile loading which cause a phenomenon called Intralaminar damage mode. The matrix tends to crack parallel to the fibers in the off-axis plies. The reduction of the laminate stiffness and strength and the eventual development of other damage modes such as delamination. The study involved analysis of the dependence of the stiffness properties and the release rate of the strain energy, on the ply orientation and crack density for glass/epoxy and carbon/epoxy laminates. A discussion on the stability of a fracture criterion for a mixed mode, to predict the strain that could onset cracking. The study showed that the fracture process of composite laminates that are subjected to static or fatigue tensile loading is based on the accumulation, sequentially, of intralaminar damage and interlaminar damage. The study revealed that effects of reduction of the axial and transversal moduli, due to matrix cracking is more significant in angle-ply than in cross-ply laminates. The opposite is true for the shear characteristics. Overall, it can be understood that the matrix cracking can result in an increase in the Poisson's ratio. [5]

Analysis of Fatigue Damage in Composite Laminates (by K.L. Reifsnider, A Talug). The study of damage modes and damage mechanisms due to degradation of laminated composite materials due to service loading conditions. The development of a state which controls the stress state and strength state of the degraded laminate. The paper attempts to make a general case for the damage development in laminated plates under cyclic (fatigue) loading. The effects of the laminae orientation, stacking sequence, and properties of the laminae to discover a 'characteristic damage state' such plates. It discusses, in detail, the nature of this characteristic damage state, its formation, and further influence on the strength, stiffness, and overall life. [6]

Damage of Composite Materials (by Thomas Jollivet, Catherine Peyrac, Fabien Lefebvre). The rapid increase in composite materials being the material of choice for construction of aircraft components has proved to present never before seen challenges, in terms of understanding the behaviour of the material. The study involved the understanding of damage mechanisms which are complex and come in variance due to the mechanical properties of the material. These are caused by the anisotropy, processing defects, heterogeneity. The final understanding that to design composite parts, a clear understanding of the micro-mechanisms of damage modes is necessary. The article focused on the damage mechanisms of different thermoset and thermoplastic composites. The principal features of composites, due to fatigue was investigated. [7]

Finite Element Simulation of Low Velocity Impact Damage in Composite Laminates (by K. R. Jagtap, S. Y. Ghorpade, A. Lal, B. N. Singh). Studying the behaviour of carbon/epoxy laminated plates when subjected to low velocity impact loading. Using the maximum stress failure criteria, the impact damage can be predicted – time of initiation. The impact simulation was performed using the finite element software LS-DYNA with 3-D solid elements. The study of failure modes such as delamination and matrix cracking were carried out. The effect of mesh size on force applied and deformation was observed. The effect of various parameters such as boundary conditions (clamped or simply supported boundary conditions), impactor velocity (impact energy) were examined for the study. The FE software was able effectively simulate the scenario for a low velocity impact. [8]

Low Velocity Impact Modelling in Composite Laminates capturing Permanent Indentation (byC. Bouvet, S. Rivallant, J. J. Barrau). The paper deals with the modelling of permanent indentation due to impact damage. The study involves an elaborated model to simulate the different impact damage types that develop during low velocity/low energy impact. The three identified damage types are: fibre failure, matrix cracking, and delamination which were then simulated. The simulation of interlaminar (interface delamination) failure is based on the interface elements fracture mechanics. Intra-laminar damage, i.e., matrix cracking is simulated using interface elements based on failure criterion. Degradation in volume elements is used to study fibre failures. The paper intended on using an originality-based model in order to simulate the permanent indentation after impact due to the introduction of a "plastic-like" model which is introduced in the matrix cracking elements. This model type was based on experimental observations, showing matrix cracking debris which can cause blockage of crack closure. The experimental part is evidently necessary in order to validate the results of the simulation and therefore certain experimentations were carried out for the same. The modelling of such a system requires meshing in finer quality and an abundance in terms of time available for calculation. This could be a problem if the model is a complex-geometrical shape Therefore, the best approach would be to model the affected area and solving the rest of the area using classical finite element model for the remaining structure. [9]

A Model for Predicting Damage in Graphite/Epoxy Laminated Composites Resulting from Low-Velocity Point Impact (by Hyung Yun Choi, fu-kuo Chang). Study of impact damage of graphite/epoxy laminated composites caused by foreign objects at low-velocity. The primary concern was the observation of matrix cracking and delamination resulting from a point-nose impactor. The modelling was done to predict the initiation of the damage based on the laminate configuration, laminate material properties, and the mass of the impactor. The model consisted of a stress analysis and a failure analysis. An Explicit Dynamics FEA was done to calculate the stresses and strains inside the composites due to impact resulting from a point nose impactor. Proposal of failure criterion for predicting the initial interplay matrix cracking and size of the interface delamination in the composites. A side by side experimentation was carried out to verify the FE analysis. [10]

Damage Tolerance of Laminated Composites containing an Open Hole and Subjected to Compressive Loadings: Part 1 – Analysis (by fu-kuo Chang, Larry Lessard). An analytical investigation to study the damage in laminae of composites when the structure contains open holes and are subjected to compressive forces. To simulate the in-plane response of the laminates, from the initial loading to the final collapse, a progressive damage model was developed for the investigation of the same. The model was subjected to stress analysis and failure analysis. The stresses and strains inside the laminae were calculated using a non-linear FE analysis method. This theory is based on Finite Deformation Theory with consideration for material and nonlinearities due to geometric construction. Failure mode is predicted using different failure criteria in the failure analysis, along with material degradation. Satisfactory results were found when compared between test results and predicted results. [11]

The Influence of Stacking Sequence on Laminate Strength (by N. J. Pagano and R. Byron Pipes). An approach to study the sequence of specific layer orientations which could lead to optimum protection against delamination due to uniaxial loadings such as static and fatigue. [12]

Failure Criteria for Unidirectional Fibre Composites (by Zvi Hashin). The expression of quadratic stress polynomials, in terms of transversely isotropic invariants of the applied average stress rate. This allowed for the establishment of the Three-dimensional failure criteria for unidirectional failure criteria. The identified Four modes of failure – compressive, tensile, fibre and matrix modes- were modelled separately. [13]

Composite Damage Metrics and Inspection (by Zoltan Mikulik and Peter Haase). The paper focused on understanding the effects of high energy low velocity impact damage for aircraft structure. Simulation of a scenario which occur typical during the day-to-day working of the aircraft. Studying of damage caused due to ground vehicles and equipment assisted in establishing impact conditions. The outcomes of the research showed a potentially high safety threat in terms of internal structural damage caused by high energy low velocity impact, which could go undetected while performing a normal visual inspection. [14]

1.1. Theoretical overview

1.1.1. Impact Damage

Low Velocity Impact Damages are caused by bodies/objects that move at a considerably slower velocity when compared to High Velocity objects. It includes ground service equipment (commonly known as GSE – typically 3,000 to 10,000 kg) such as ground vehicles, passenger stair truck, jet/aero bridge, belt/cargo loaders amongst other vehicles. The incurred damages fall under a category called Potential Operational Threats & Damages. This has been identified by the NASA Aviation Safety Reporting System (ASRS) and has been properly documented. The database has been heavily contributed-to by aviation personnel such as pilots, as well as controllers, ramp-workers, technicians, flight attendants, and others. The industry study put the risk factor value at 56 per cent of the damages that an aircraft might incur during its operational life. Therefore, an aircraft stands at a much higher risk percentage - damage, due to uncontrolled and improperly trained handling of airside equipment.

1.2. Impact Damage Classification by Severity – Composite Aircraft Structure

Composite structures present major challenges, in terms of accidental damage. The presence of flaws should not hinder safe operations. Designs are made with damage tolerance in mind – by selecting appropriate damage resistant materials (resin system), identification and understanding of damage sources and types, knowledge and understanding of damage propagation mechanisms, and damage criticality. The lay-up of composites, frame and stringer pitch, attachment details, features for crack arrest, and redundancy in structural geometry, affect the damage tolerance. Understanding of damage, and the ability to predict failure modes, determines the usage of Composite materials, for manufacturing of aircraft components and they have certain requirements such as the ability to be detected during normal inspection of the structures. [19]

Based on the below illustrated Figure 1, the Acceptable Means of Compliance (AMC) classifies types of damages, for composite aircraft parts, into five different categories. They are:

Category 1 – The allowable damage that may/can go undetected during schedule inspections. This includes the low energy Barely Visible Impact Damage (BVID), manufacturing defects that are within the allowable limits or damage caused in-service, that does not affect the eventual load carrying capacity by degrading it over the aircraft’s reliable service life.

Category 2 – This type of damage can be detected reliably during scheduled or instructed inspections. Deep scratches, visible damage-due-to-impact, debonding or delamination that are detectable, are typical examples of this category.

Category 3 – Ramp personnel detection of damage, within a few flight cycles. Pre-flight walk arounds reveal these large visual impact damage or other kinds of damages. The aircraft requires features that provide sufficient damage tolerance capabilities during its design stage, to allow for a Category 3 damage, that helps it to retain load levels that are limited, for a short-time detection interval.

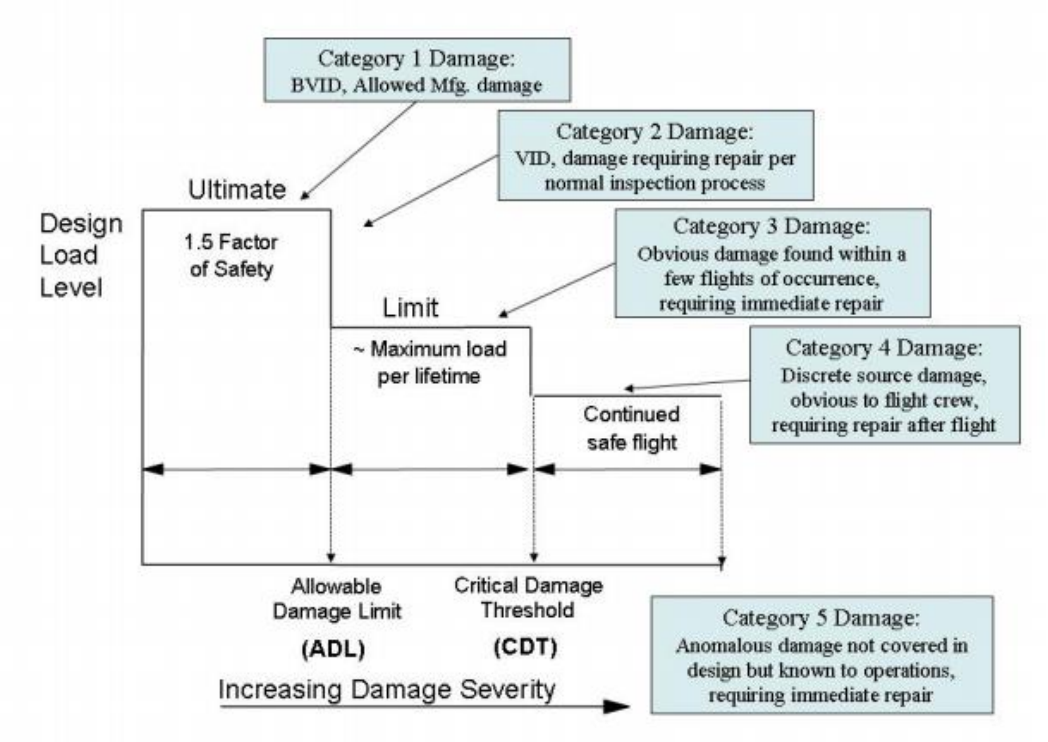


Figure 1. Load Levels Different Designs versus Severity of Damage [25]

Category 4 – Damage due to bird strike, bursting of tyre or weather conditions that affect a safe flight (hail storm) are classified under Category 4 damage. These are discrete damage that can limit flight manoeuvres.

Category 5 – Damage caused to the aircraft by unpredictable conditions on ground or flight, which are not covered in the design criteria, fall under Category 5. A high velocity impact caused by a ground vehicle that severely damages the fuselage of the aircraft, overload conditions, loss of parts mid-flight, hard landings, etc are examples of this category. [19]

1.3. Ground Handling Operations

1.3.1. Ground Handling Operations Overview

As illustrated in Figure 2, it can be seen that ground operations around an aircraft can be filled with complex manoeuvres due to various types of vehicles. The number of equipment and the range of the operations for a commercial airline are:

a. Onboard services – seat modifications, cleaning, catering, in-flight entertainment servicing.
b. Ramp services – supervision, towing of aircraft, engine start-up, marshalling.
c. Aircraft services On-ramp – maintenance, re-fueling, ground power supply, wheel check, de-icing, cleaning.
d. External Equipment and services – Passenger steps/aero bridge, loading of catering, cargo loaders, postal mail and baggage loading. [19]

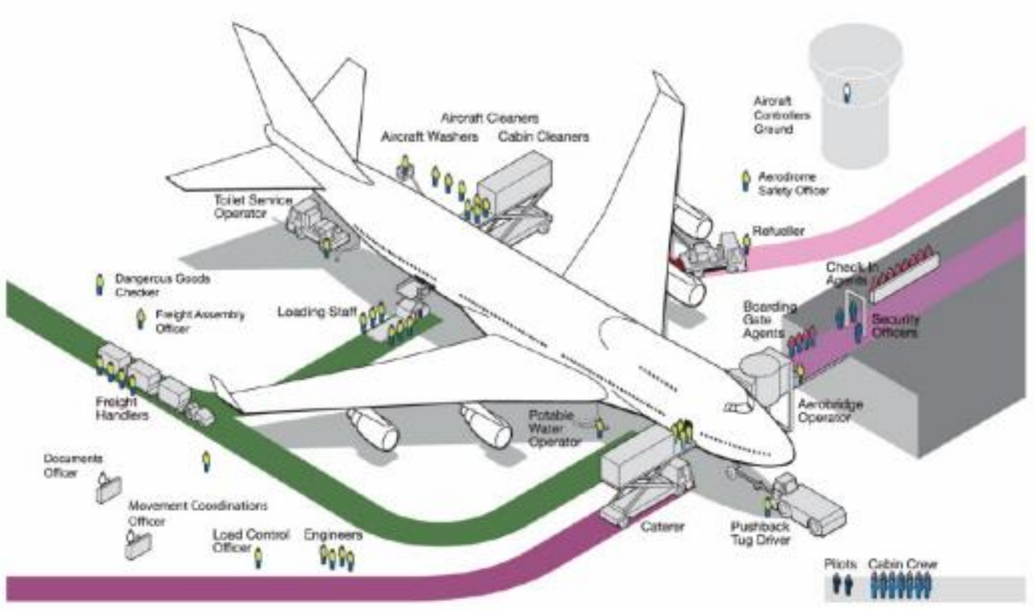


Figure 2. Ground Handling Operations Complexity [26]

Complex operations, coupled with a callous lack of safety, could result in damages due to incidents that involve cargo movements hitting the aircraft in various places.

1.4. Review – Ground Handling Incidents

The Australian Transport Safety Bureau (2010) submitted a comprehensive report which presented a review of the ground operations incidents. The report showed accidents that occurred between January 1998 and December 2008. The number stood at 398 ground incidents which involved large civil aircraft. Out of these, the ground operations related occurrences were around 75 per cent and 25 per cent of these incidents resulted in aircraft damage. The Figure 3 and Figure 4 shows a breakdown of the location-of-the-ground occurrences, as identified by the Australian Transport Safety Board (ATSB) and the NASA ASRS. A staggering 28 per cent was found to be occurring near the airport-aircraft gate in the Australian report when compared to the 46 per cent in the ASRS report. The damage caused are due to the high energy/low velocity blunt impacts when an aircraft is at the gate while it is being prepared for its take-off roll or after landing – before the disembarking of passengers or unloading of cargo/baggage. [19]

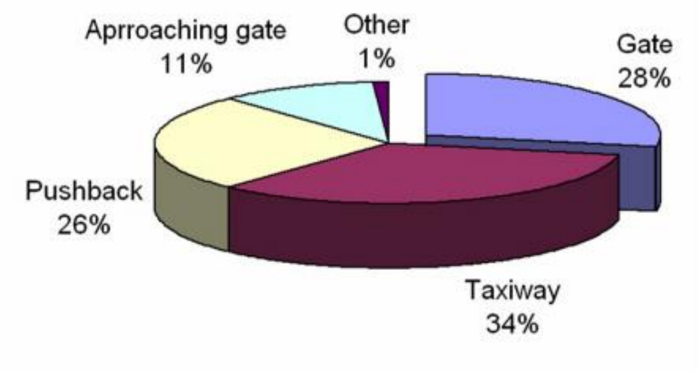


Figure 3. Location of the Ground Operation Incidents [27]

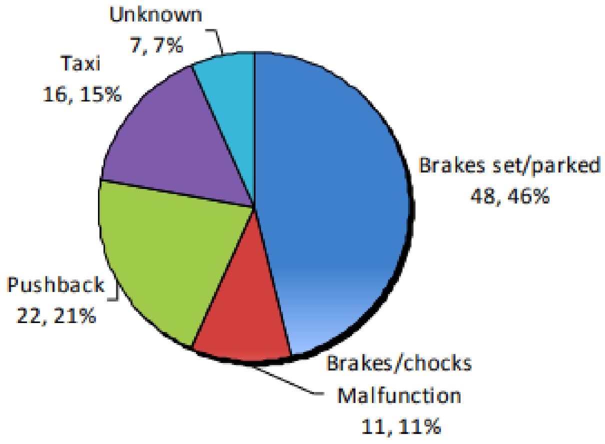


Figure 4. State of the Aircraft – Breakdown for 104 ASRS Report [28]

While the aircraft were stationary, the aircraft was damaged in 45 per cent of these incidents. The ground vehicles were found to be major cause of damages. The below Table 1 shows a summary of the types of vehicles that collide into the aircraft at the gates.

Table 1. Incident summary for Ground Vehicles by Vehicle Type

Vehicle causing damage	**Number**	**Per cent**
Cargo/container loader	8	24.2
Stairs (mobile)	8	24.2
Catering truck	4	12.1
Aerobridge	3	91
Passenger lifter	3	9.1
Belt loader	3	9.1
Tug	2	61
Baggage trolley	1	1.3
Fuel truck	1	1.3
Total	33	100

Most impact damages are repaired via the usage of patch repairs, that are carried out externally, as shown in Figure 5. If the patch repair tends to hinder the aerodynamics of the aircraft, it is then done internally. An internal repair runs the risk of costing higher than an external repair. Parts of fuselage that have cut outs such as doors, cargo bay doors, are areas of high stress concentrations and therefore, need to be suitably reinforced with the use of additional materials such as titanium. [19]

Figure 5. Example of Patch Repair [29]

1.5. Blunt Impact Energy Level

The continuous movements of the ground handling equipment and vehicles could result in impact or damage of the aircraft due to an accident. As mentioned before, impact damages are caused by equipment like mobile stairs, belt loaders, cargo loaders, and aerobridges. The typical velocity of a service vehicle was investigated by Kim (2010). The analysis showed that during normal operations, a velocity of 1 metre per second is common for a vehicle which is 1 m away from the fuselage. The closest distance, such as 100 millimetres, had the vehicles moving around with a speed equal to 0.5 metres per second.

The kinetic energy, in an accidental impact, is directly influenced by the mass of the vehicle and its velocity at the time of impact. A review of the market-available mobile stair vehicles and belt loaders for narrow and wide body aircraft showed that depending on the size, servicing type, vehicle type, and engine type, the mass of the vehicle could be in the range of 2500 kilograms to 5500 kilograms.

Based on the data obtained from the 2010 Sydney airport incident (a 3.5-ton loader unexpectedly accelerated towards the fuselage from a distance of 2.7 metres at a velocity of 1 metre per second) – The indentation left was a 0.8 millimetres score on the fuselage. The event resulted in another impact that occurred between the empennage section and the push-up stairs. That resulted in a 0.5 millimetres dent in the skin of the fuselage. If the kinetic energy were to be calculated, based on the assumptions that the impact occurred at a velocity of 1 metre per second, the value so obtained will be,

$$E_k = \frac{1}{2} * mass * V^2 \quad (1)$$

$$= \frac{1}{2} * 3500 * 1^2$$

= 1750 Joules

Where,

V is the velocity of the vehicle.

The ground vehicle velocity data is decided by reviewing airport operations safety catalogue.

To consider a range of energy levels for the impact, based on the findings shown in Figure 6, researchers tend to bracket the impact between a 2000 kilograms vehicle, impacting the fuselage at velocity of 0.3 metres per second (lower limit) and a 5000 kilograms vehicle, impacting at a velocity of 1.2 metres per second (upper limit). This puts the range between 90 Joules and 3600 Joules.

Further guidelines can be obtained from the results of Kim et al. (2010). The work involves a composite panel, reinforced by stringers that was subjected to Quasi- Static test, using an OEM (original equipment manufacturer) rubber bumper. The results showed large deformations in the panel, delamination between the skin and the stringers, shear tie facture and continuous cracking. However, no visible external damage or permanent deformation was observed. The load-displacement curve showed that 700 Joules was the minimum requirement to generate such damage. [19]

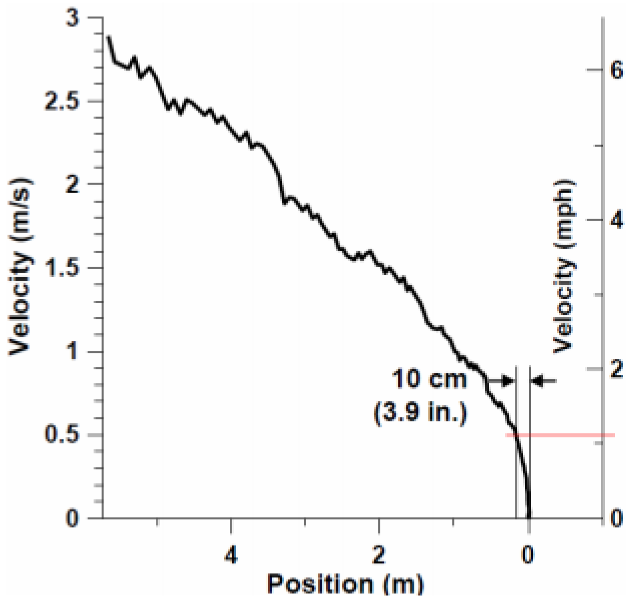


Figure 6. Approach velocity of Ground Vehicles [30]

The research was furthered in 2011 that involved testing on a larger three-frames composite panel and was subjected to a loading range between 955 Joules to 2500 Joules and was expected to produce wide spread internal damage, with permanent deformation on the external surface. [15]

Based on the available incident data and test results, an energy level of 900 Joules to 3000 Joules could create a characteristic failure mode in the composite structure, in accidents involving ground service vehicle incidents. [19]

2. Research Methodology for the Finite Element Approach

An establishment of a credible design for the test panel required an extensive review of the of the diverse designs and materials used in CS – 25 aircraft fuselages. The CS – 25 is a legislation, passed by the European Aviation Safety Agency (EASA) and is known as the Certification Specifications for Large Airplanes CS – 25. This applicability of the Airworthiness code is for the turbine powered large aircraft. The Subpart C – Structure, subsection coded CS 25.301 Loads describes the strength requirements in terms of limit loads, i.e., the maximum loads to be expected while in service and the ultimate loads, a value that is calculated by multiplying the value of the limit load by the factors of safety. [19]

The legislation prescribes that the structure must be able to support loads without permanent deformation that could be detrimental to the service life of the aircraft. The legislation requires the structure to be able to withstand ultimate loads without failure for three seconds. However, a dynamic study to show the proof of strength, does not require the three seconds rule to be implied. The requirement for static tests includes the ultimate deflections and ultimate deformation, due to ultimate loads. When analytical methods are employed to show compliance with the ultimate load conditions, the tests must reflect –

1. The deformation which does not create significant disturbance in performance.
2. The full accountability of the deformations while analyzing.
3. The methodology and the assumptions used need to satisfy the effects of the deformations.
4. The flexibility of the structure requires consideration for the rate of application of loads such that the load cases, in a transient case, which produce transient stresses that are higher than the stresses caused by static loads and therefore, need to be carefully considered while in the analysis. [19]

The proof of structure must show compliance with the strength and deformation requirements for each critical loading conditions. When a static or dynamic test is used to show compliance with the CS – 25 requirements, for flight structures, appropriate material correction factors must be applied to the results of the test. This also requires consideration for structures that are made up of multiple elements, such that, failure of one element results in a redistribution of the loads via alternative load paths.

The main objectives of this thesis study include:

1. To ensure that the aircraft test section is designed according to the prescription of the EASA CS – 25 legislations.
2. To ensure the material properties are as close to the originally used ones to simulate results as close as possible to the real-world scenario.
3. To study the propagation patterns of the stresses incurred due to loading conditions.

Research Methods

This study is an investigation study based on the various testing procedures followed by researchers around the globe to test the use of composite materials as viable resource for creation of aircraft components. A well-defined research method as proposed by the legislations is used in this work. There are five sub-sections in this thesis as shown by the Figure 7 below.

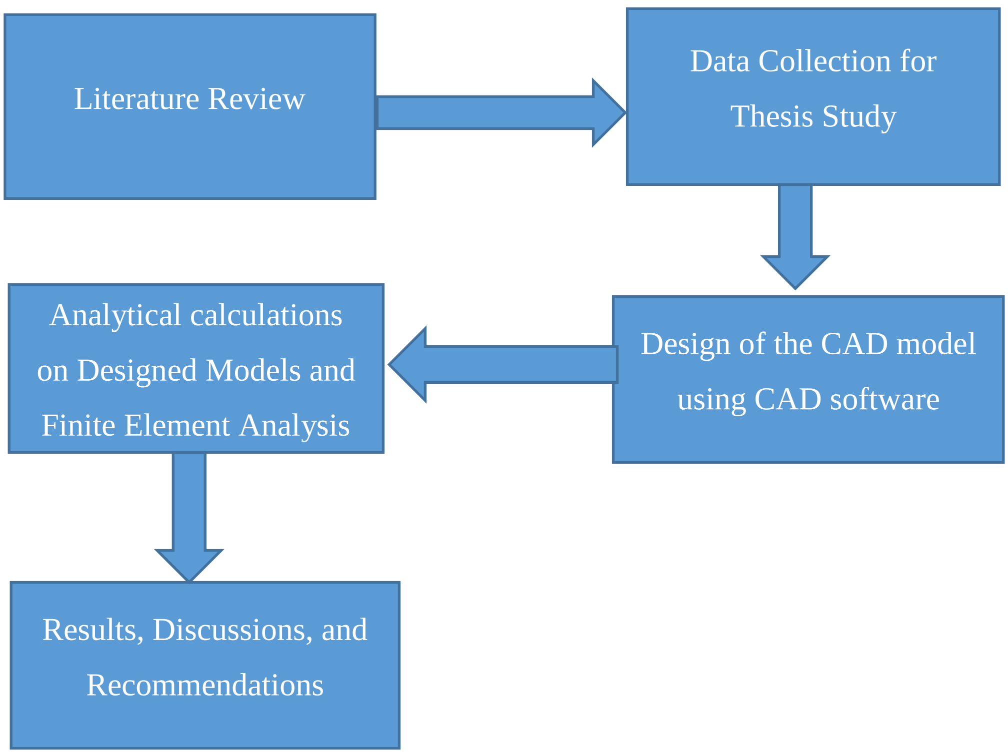


Figure 7. Research Methodology

In the above-mentioned methodology, it can be clearly seen that the analysis process begins with a review of the various literatures that have been published on the selected topic or the topics closely related to the field of study. The section gives an insight into the various conditions in which the tests were carried out. Looking at the discussions and the recommendations can really help in designing a test that best suit the analysis needs.

The follow-up procedure would be to design the components that will be tested. Based on the current data available for the airline industry, the commercial aircraft model Airbus A350, manufactured by Airbus SE is chosen. The aircraft is one of the latest advancement in terms of the usage of Composite materials, thereby, proving an ideal candidate to simulate under tests. A closer look into the design criteria shows that the model for testing can be designed using the Dassault Systèmes SOLIDWORKS Corp student version 2016 software. The calculation of forces to be applied on the test section were calculated using the methods employed by the European Aviation Safety Agency report for damage analysis due to different velocities.

2.1. Airbus A350/Airbus A350 XWB (stylised form of Extra Wide Body)

The Airbus A350 XWB is the newest product from the Airbus SE company in the wide body jetliner market. Its maximum external fuselage diameter stands at 5.97m (Airbus A350 2011). The aircraft has gone through several design changes and major revisions in terms of parameters. Initially, as shown in Figure 8, the aircraft was predominantly designed to have a more classical metallic body frame. The regions of frames, ribs, gear bays, and floor beams were designated to be built using Aluminium or Aluminium-lithium alloys, which would account for nearly 20 per cent, by weight, of the aircraft. [19]

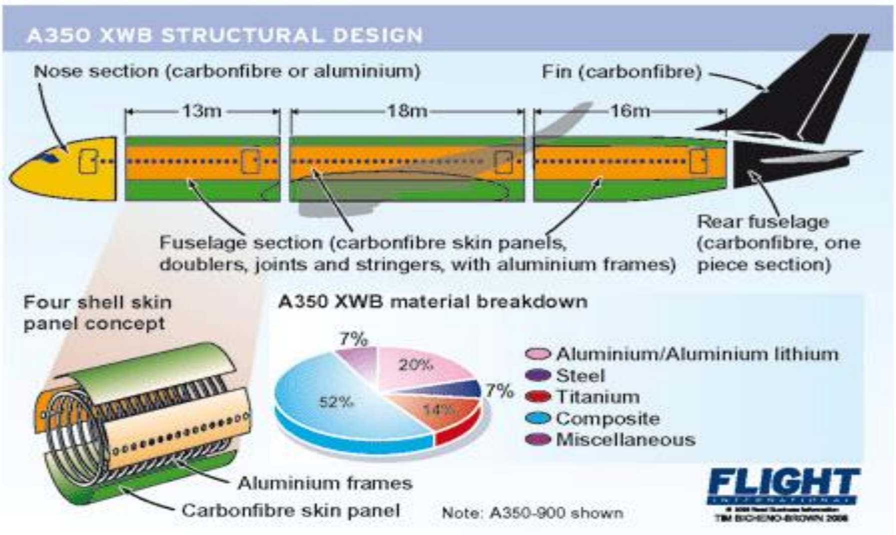


Figure 8. Initial Design Lay-Up of Airbus A350 XWB [31]

Owing to wide-spread criticism from airline operators and general public, Airbus SE decided to switch to a composite frame design. Reports showed that there were key changes such switching from metallic to carbon fibre fuselage frames. The fuselage cross beams were kept metallic. A revision parameter can be seen in the below illustrated Figure 9. [19]

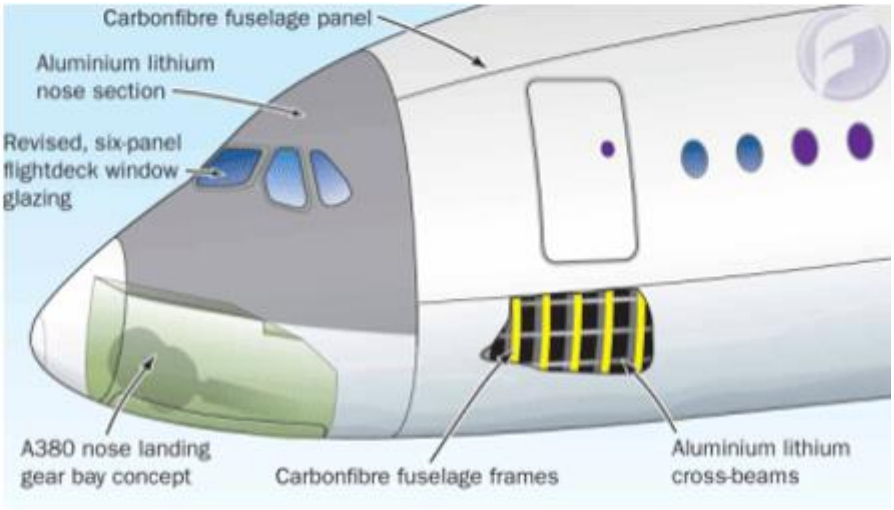


Figure 9. Design Revision of Airbus A350 XWB [32]

The new revised design had Carbon Fibre Reinforced Polymer (CFRP) stringers, bonded to the fuselage skin panels (Airbus A350 XWB update 2010). This resulted in a reduction in the overall number of fabricated parts and fasteners. The fuselage section consists of large carbon fibre composite panels. To meet the design criteria for crashworthiness, certain frames and stringers that are located in high stress zones have fabricated from titanium. [19]

According to an update from the from Airbus A350 design, titanium accounts for nearly 15 per cent of the weight of the aircraft. They have been used near high loaded frames, door surrounds, construction of landing gear, engine pylons, and in regions that maybe susceptible to high-velocity bird strike impacts. The information with regards to the usage of composite materials and titanium is available through the Airbus website. The titanium section of the aircraft can be seen in the below illustrated Figure 10. The website shows that the Airbus A350 XWB fuselage panels, stringers, frames, window frames, clips, and doors are made from Composite Fibre Reinforced Polymer. [19]

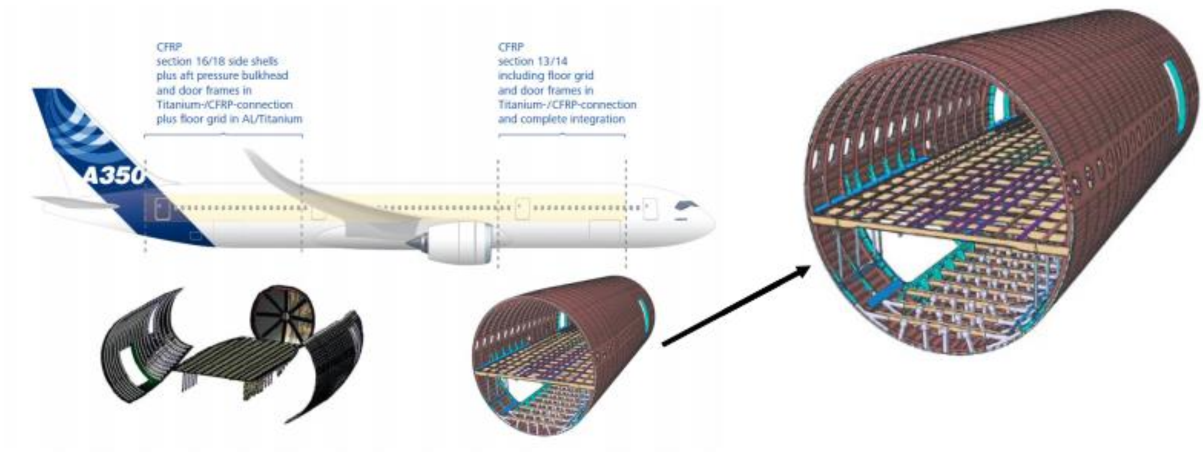


Figure 10. Airbus A350 Titanium usage [33]

The fuselage of the Airbus A350 is manufactured using a construction method that employs the usage of multi-panel. This involves fuselage sections being built from four composite panels – Shorter panels on the upper and lower surfaces connecting two long side-panels. The stringers are shaped based on their location in the aircraft and can vary between T – or Omega shaped. They are either straight or curved depending on the fuselage cross-sectional difference. The stringers are bonded to the skin of the fuselage panels instead of being moulded as a part of each panel. Examples of such stringers is shown in the Figure 11. The extended flange width can be seen. It provides a flush surface for the attachment of composite clips. [19]

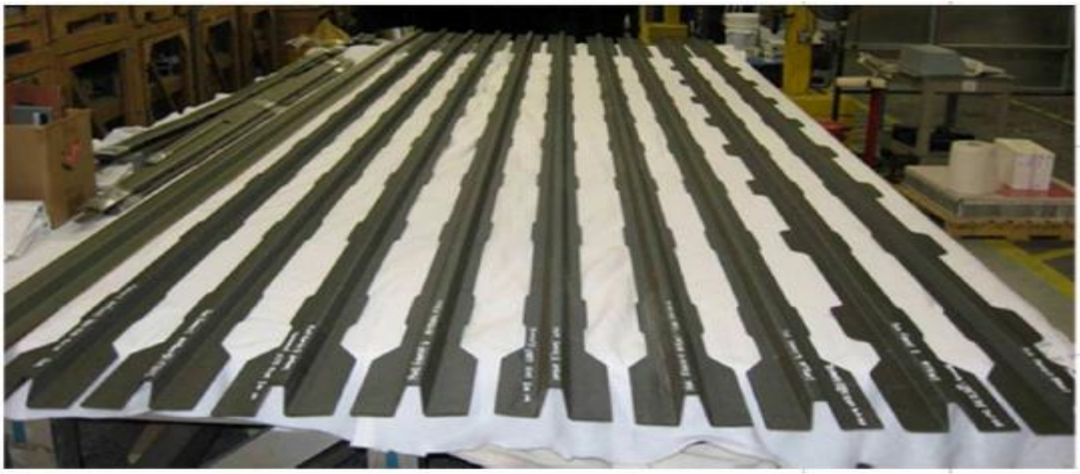


Figure 11. Omega – profile stringers used in Airbus A350 XWB [34]

The skin panels and fuselage frames are not bonded together, instead are joined using special mix of thermoplastic clips and fasteners specially made for such interface. Based on the variant of the Airbus A350, the number of frames could vary. The Airbus A350-1000 has 11 more frames than the Airbus A350-900 version. The frames have a pitch of 635mm between them. [19]

2.2. Summary of Fuselage Design under CS – 25 Legislations

To summarize the findings of the research papers and the inputs by Manufacturers.

Materials

About 50 per cent, by weight, modern aircrafts have imbibed the use of CFRP in the designing of panels, frames, and stringers. The doors tend to have titanium reinforcing. This has come as latest advancement when considering the fact that almost all aircrafts produced by Airbus and Boeing, in the mid-to-late 90s were made out of metals. [15]

Geometry

The frame cross-section generally consists of C – or Z – profiles. The dimensions of the frames are height = 85 – 100 millimetres, thickness = 2 – 3 millimetres, and the flange width = 25 millimetres.

The Z- profile stringers have a height of 30 millimetres, thickness of 2 millimetres, and a flange width of 15 millimetres.

Omega-profile stringers have a height of 25 – 35 millimetres, thickness of 1.5 – 2.0 millimetres, head width of 25 millimetres and a total floor width between 100 – 130 millimetres.

Depending on location, the skin thickness may vary between 1.0 to 2.6 millimetres. [19]

Frame pitch

The older convention has the frames pitched between 457.2 millimetres – 533.4 millimetres. The current models have an extended frame pitch between 610 millimetres – 635 millimetres. [19]

Stringer pitch

The stringer pitch is between 150 – 250 millimetres. [19]

2.3. Composite Fibre Failure Theory

Material failures occur when components are subjected to higher stresses and strain values than they can handle. Different theories are applicable to the study of the various stresses and strains generated in the material. Based on these theories, materials are designed to withstand loads without failing, thereby ensuring safety and longevity of the components. The theories also help to understand the material behaviour and properties. Failure is defined as a "state in which a material cannot perform its intended job". It includes fracture, where the material break into two or several parts, buckling, and matrix cracking.

Carbon Fibre can be studied under the Maximum Stress-Theory, Maximum Strain – theory, Tsai – Hill theory, and Tsai – Wu theory due to their validity for anisotropic materials. ("Anisotropic materials are those materials whose properties, such as Young's Modulus, change with direction along the object").

The stresses and strains need to be calculated in the principal material direction. This is done by calculating stresses in individual plies. The ply stresses are calculated using:

$$\{\sigma\}_{x-y} = [T_\sigma]\{\sigma\}_{1-2} \tag{2}$$

The ply strain is calculated using,

$$\{\varepsilon\}_{1-2} = [Q]_{1-2}\{\varepsilon\}_{1-2} \tag{3}$$

Where $[T_\sigma]$ and $[Q]_{1-2}$ are the stiffness matrices for the respective ply. [24]

2.3.1. Maximum Stress

The Maximum Stress Criterion has set limits for the five failure modes for a material. These are: longitudinal tensile, compressive, transverse tensile, transverse compressive, and shear stress. So mathematically speaking, a failure could occur if,

$$\sigma_1 \geq \hat{\sigma}_{1T} \text{ or } \sigma_1 \leq \hat{\sigma}_{1C} \text{ or } \sigma_2 \geq \hat{\sigma}_{2C} \text{ or } \sigma_2 \leq \hat{\sigma}_{2C} \text{ or } \tau_{12} \geq \hat{\tau}_{12}$$

The above relations show that if any of the corresponding stress values very to exceed the prescribed limits, it could result in a failure of the material. The fact that CFRP is laid at an angle θ to provide directional stiffness, the angle that it is oriented at need to be accounted for. For example, a single stress that σ_x that is acting at an angle θ require some modification in the equation 2 stat ed before, to obtain the value. [24]

The simplification of the equations 2 and 3 would lead to a relationship between the stresses and the angle of orientation of the fibre.

$$\sigma_1 = \sigma_x \, cos^2 \theta \tag{4}$$

$$\sigma_2 = \sigma_y \, sin^2 \theta \tag{5}$$

$$\tau_{12} = -\sigma_x \sin\theta \cos\theta \tag{6}$$

The above equations show that there are three possible results that can help determine failure.

$$\sigma_x = \frac{\sigma_{1T}}{cos^2 \theta} \text{ - Fibre Failure} \tag{7}$$

$$\sigma_x = \frac{\sigma_{2T}}{sin^2 \theta} \text{ - Transverse Failure} \tag{8}$$

$$\sigma_x = \frac{-\tau_{12}}{\sin\theta \cos\theta} \text{ - Shear Failure} \tag{9}$$

The equations cane be then plotted in a graph format for behavioral estimation of composites. The graphical method did show that the shear stress is more likely when the ply angle is 45 degrees to the normal orientation. Transverse failure occurs when the angle approaches 90 degrees, and fibre breakage could occur at smaller angles of force application. [24]

2.3.2. Maximum Strain

The calculation of the maximum strain, the components of stresses are replaced by strain components. Failure can occur if,

$$\varepsilon_1 \geq \hat{\varepsilon}_{1T} \text{ or } \varepsilon_1 \leq \hat{\varepsilon}_{1C} \text{ or } \varepsilon_2 \geq \hat{\varepsilon}_{2C} \text{ or } \varepsilon_2 \leq \hat{\varepsilon}_{2C} \text{ or } \gamma_{12} \geq \hat{\gamma}_{12}$$

To determine maximum strain in the principal direction, when stresses act at an angle, both σ_1 and $\sigma_2$must be considered to give the strain values. Therefore, the below shown equations can help calculate the corresponding strain values. [24]

$$\varepsilon_1 = \frac{\sigma_1}{E_{11}} - \upsilon_{21}\frac{\sigma_2}{E_{22}} \tag{10}$$

$$\varepsilon_2 = \frac{\sigma_2}{E_{22}} - \upsilon_{12}\frac{\sigma_1}{E_{11}} \tag{11}$$

$$\gamma_{12} = \frac{\tau_{12}}{G_{12}} \tag{12}$$

Solving the equations for the stresses, it is so possible to obtain the strains in the principal directions.

$$\varepsilon_1 = \frac{\sigma_1}{E_{11}} - \upsilon_{21}\frac{\sigma_2}{E_{22}} = \left(\frac{\cos^2\theta}{E_{11}} - \frac{\upsilon_{21}}{E_{22}}\sin^2\theta\right)\sigma_x \tag{13}$$

and

$$\varepsilon_2 = \frac{\sigma_2}{E_{22}} - \upsilon_{12}\frac{\sigma_1}{E_{11}} = \left(\frac{sin^2\,\theta}{E_{22}} - \frac{\upsilon_{12}}{E_{11}}cos^2\,\theta\right)\sigma_x \tag{14}$$

2.3.3. Tsai – Hill Criterion

The Tsai – Hill Criterion provides with a method for calculating fibre values when considering the multi-axial stress forces acting on an isotropic material. The Tsai – Hill criterion is shown below equation.

$$\left(\left(\frac{\sigma_1}{\hat{\sigma}_1}\right)^2 - \frac{\sigma_1\sigma_2}{\hat{\sigma}^2{}_1} + \left(\frac{\sigma_2}{\hat{\sigma}_2}\right)^2 + \left(\frac{\tau_{12}}{\hat{\tau}_{12}}\right)^2\right) \geq 1 \tag{15}$$

By reviewing the equation 15, we see that just one value needs to be satisfied for the criterion to hold when compared to the five detailed version that were discussed before this. This allows for one result to be produced for failure stress. [24]

But Tsai – Hill Criterion comes with a drawback that it individually cannot predict the type of failure. To determine the failure stress, when the forces are applied at different angles, non-correspondent to the direction of the fibre, the Tsai – Hill criterion is coupled with stress equations, taking into account the angle θ into consideration. The resulting equation is: [24]

$$\left(\frac{\sigma_x\cos^2\theta}{\hat{\sigma}_1}\right)^2 - \frac{\sigma_x^2\cos^2\theta\sin^2\theta}{\hat{\sigma}_1^2} + \left(\frac{\sigma_x\,sin^2\,\theta}{\hat{\sigma}_2}\right)^2 + \left(\frac{\sigma_x\sin\theta\cos\theta}{\hat{\tau}_1^2}\right)^2 \tag{16}$$

2.3.4. Tsai – Wu Criterion

The conventional assumption involved things like homogeneity and linear elasticity up to the failure of the material. The Tsai -Wu Criterion was proposed in context of anisotropic materials, by the usage of quadratic polynomial expressions for the different stresses and with a tensorial coefficient for each corresponding stress. It simplified the equation from a comprehensive yet less practical form. [24]

Generally, Anisotropic materials are rarely used in real-life situations. The materials which more closely resembles the above-mentioned case are the orthotropic materials. As a result, a familiar form of Tsai – Wu criterion can be employed to find the failure function in the material's principal axes. [24]

$$F = F_{11}\sigma_1^2 + F_{22}\sigma_2^2 + F_{33}\sigma_3^2 + 2F_{23}\sigma_2\sigma_3 + 2F_{13}\sigma_1\sigma_3 + 2F_{12}\sigma_1\sigma_2 + F_1\sigma_1 + F_2\sigma_2 + F_3\sigma_3 + F_{44}\tau_{23}^2 + F_{55}\tau_{13}^2 + F_{66}\tau_{12}^2 \quad (17)$$

The above equation 17 reflects the material symmetries in a material that is orthotropic. The forces are contracted forms of 2nd and 4th raked tensors. There are other tensors that have not been included as they tend to vanish due to material symmetry. The material is said to be safe if,

$$F < 1$$

While the failure due to critical condition occurs when,

$$F = 1$$

The equation 17 can be further simplified by using the principle of Transverse isotropy and the behaviour can be used as satisfactorily sufficient description of the Uni-directional composites.

$$F_{33} = F_{22}, F_{13} = F_{12}, F_3 = F_2, F_{55} = F_{66}, and\ F_{23} = F_{22} - \frac{1}{2}F_{44} \quad (18)$$

This can be used to further reduce the Tsai – Wu Criterion to make the equation into a simpler form. [21]

2.3.5. Unidirectional Fibre Composites – Hashin's Failure Criteria

When more than one stress component is used to determine the interacting failure criteria and estimate the different failure modes, is known as Hashin's criteria. These criteria were originally developed for Uni-directional composite fibers and therefore, when applied to other types, involves some form of approximation. It is usually applied for two-dimensional classical lamination approach for calculation of point stresses with ply discounting as the material degradation model. Hashin's criteria involves four failure modes and related to fibre and matrix failure. The criteria can be used for three-dimensional problems by using the maximum stress criteria for the transverse stress component. [22]

1. Tensile Fibre Failure for $\sigma_{11} \geq 0$

$$\left(\frac{\sigma_{11}}{X_T}\right)^2 + \frac{\sigma_{12}^2 + \sigma_{13}^2}{S_{12}^2} = \begin{cases} \geq 1 \; Failure \\ < 1 \; No\, failure \end{cases} \tag{19}$$

2. Compressive Fibre Failure for $\sigma_{11} < 0$

$$\left(\frac{\sigma_{11}}{X_c}\right)^2 = \begin{cases} \geq 1 \; Failure \\ < 1 \; No\, failure \end{cases} \tag{20}$$

3. Tensile matrix failures for $\sigma_{22} + \sigma_{33} > 0$

$$\frac{\left(\sigma_{22}+\sigma_{33}\right)^2}{Y_T^2} + \frac{\sigma_{23}^2 - \sigma_{22}\sigma_{33}}{S_{23}^2} + \frac{\sigma_{12}^2 + \sigma_{13}^2}{S_{12}^2} = \begin{cases} \geq 1 & \text{failure} \\ < 1 & \text{no failure} \end{cases} \tag{21}$$

4. Compressive matrix failures for $\sigma_{22} + \sigma_{33} < 0$

$$\left[\left(\frac{Y_C}{2S_{23}}\right)^2 - 1\right]\left(\frac{\sigma_{22}+\sigma_{33}}{Y_C}\right) + \frac{\left(\sigma_{22}+\sigma_{33}\right)^2}{4S_{23}^2} + \frac{\sigma_{23}^2 - \sigma_{22}\sigma_{33}}{S_{23}^2} + \frac{\sigma_{12}^2 + \sigma_{13}^2}{S_{12}^2} = \begin{cases} \geq 1 & \text{failure} \\ < 1 & \text{no failure} \end{cases} \tag{22}$$

5. Interlaminar tensile failure for $\sigma_{33} > 0$

$$\left(\frac{\sigma_{33}}{Z_T}\right)^2 = \begin{cases} \geq 1 \; Failure \\ < 1 \; No\, failure \end{cases} \tag{23}$$

6. Interlaminar compression failure for $\sigma_{33} < 0$

$$\left(\frac{\sigma_{33}}{Z_C}\right)^2 = \begin{cases} \geq 1 \; Failure \\ < 1 \; No\, failure \end{cases} \tag{24}$$

Where, σ_{ij}denotes the stress components and the tensile and compressive allowable strengths for lamina are denoted by subscripts T and C, respectively. Similarly, X_C , Y_C, and $Z_{C.}$ denotes the allowable tensile strengths in three respective material directions. X_T, Y_T, Z_T denotes the allowable tensile strengths in three respective material directions. Further, S_{12}, S_{13}, and S_{23} denote the shear strengths in the material's principal directions. [22]

2.4. Computer Aided Design of the Aircraft Model

The Dassault Systèmes SOLIDWORKS Corp student version 2016 software was chosen to design the model of the aircraft. The relatively simple construction allowed the usage of SOLIDWORKS rather than other high-end software such as CATIA (Computer – Aided Three-Dimensional Interface Application) or Siemens NX. The design model has been kept as close as possible to the original dimensions. Many different variations of test panels were designed to simulate results based on the load cases. The models are,

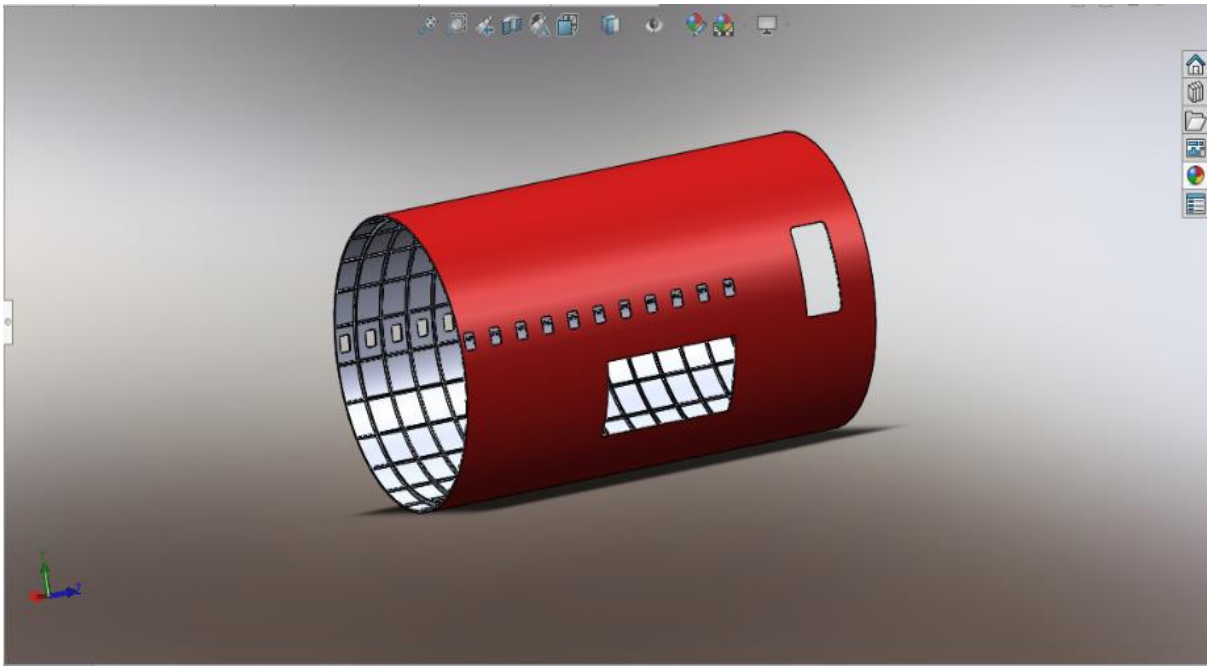


Figure 12. Fuselage section of the Airbus A350 XWB with no Cargo-bay separation

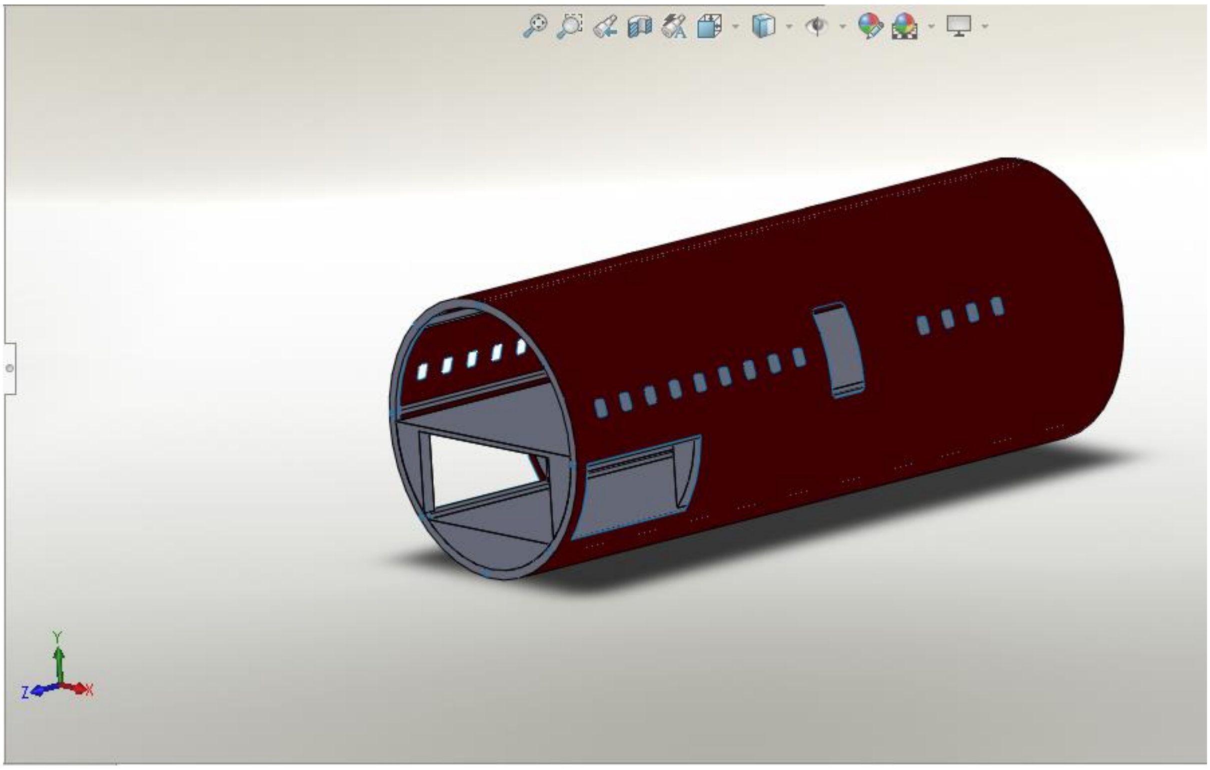


Figure 13. Fuselage section of the Airbus A350 XWB with cargo-bay separation

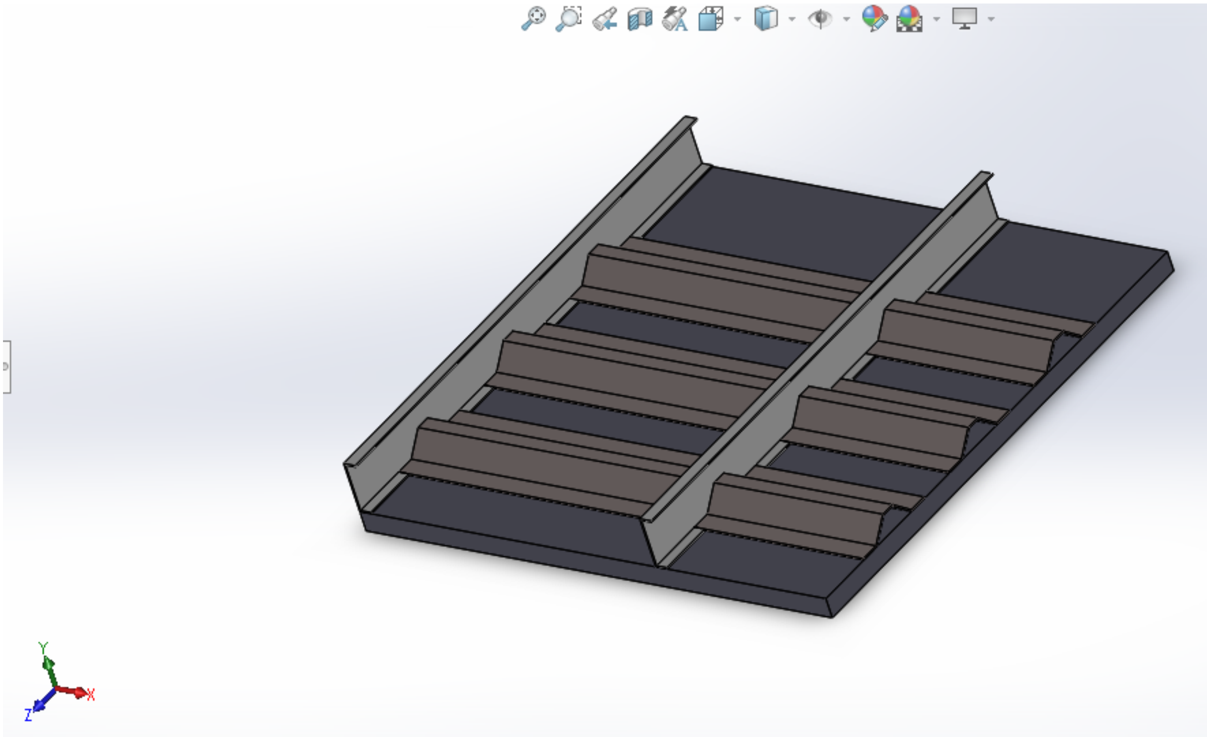


Figure 14. Test Section Cut – out

The parts used to create the above assembly are,

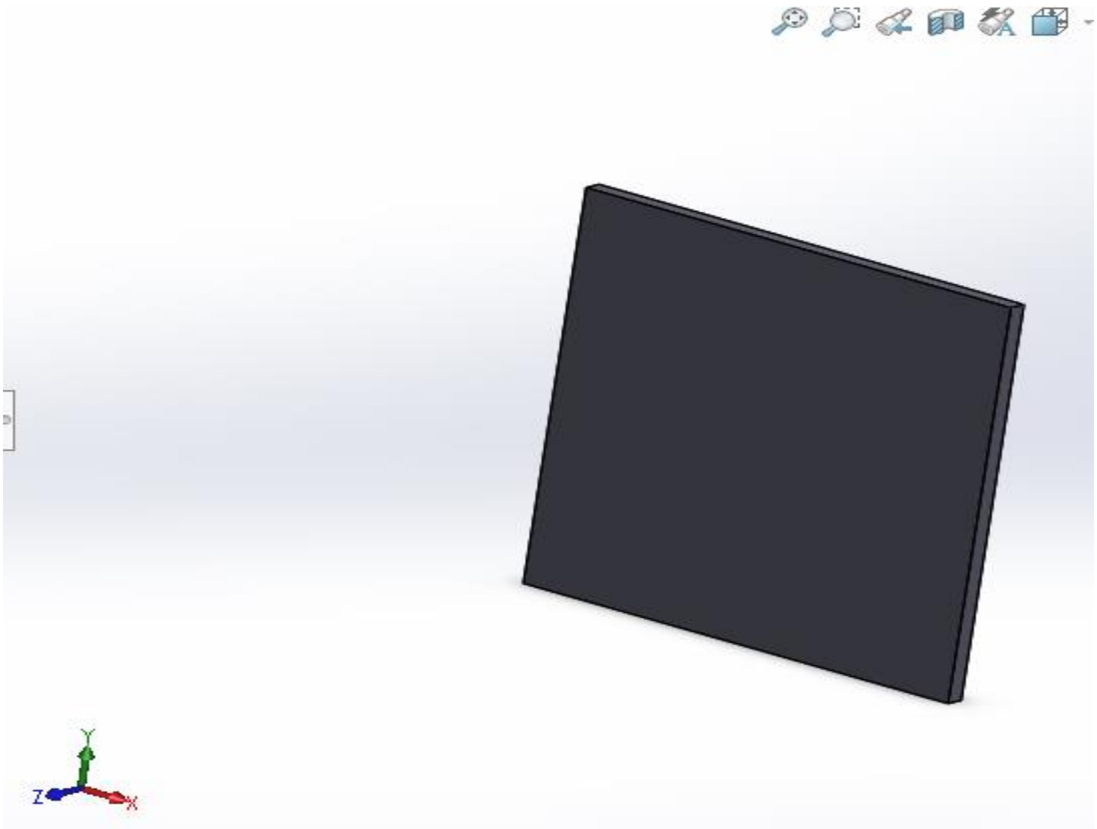


Figure 15. Fuselage Skin Panel

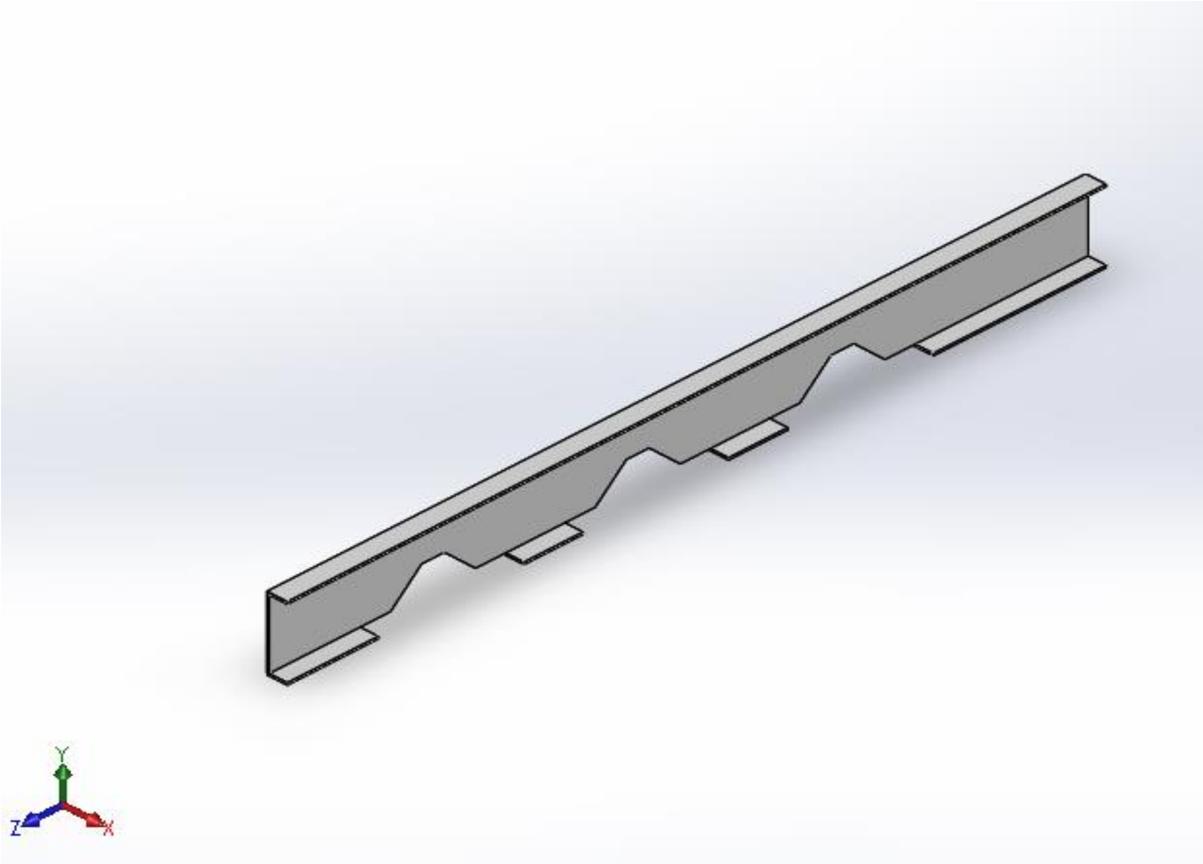


Figure 16. Fuselage Frame

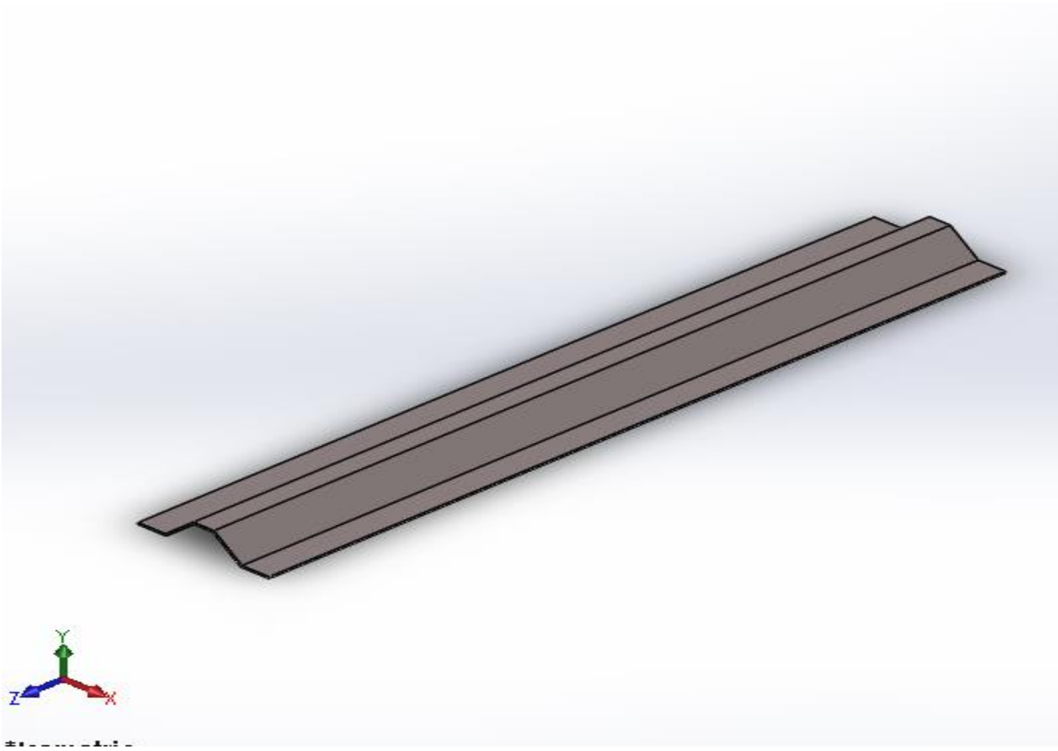


Figure 17. Fuselage Stringer

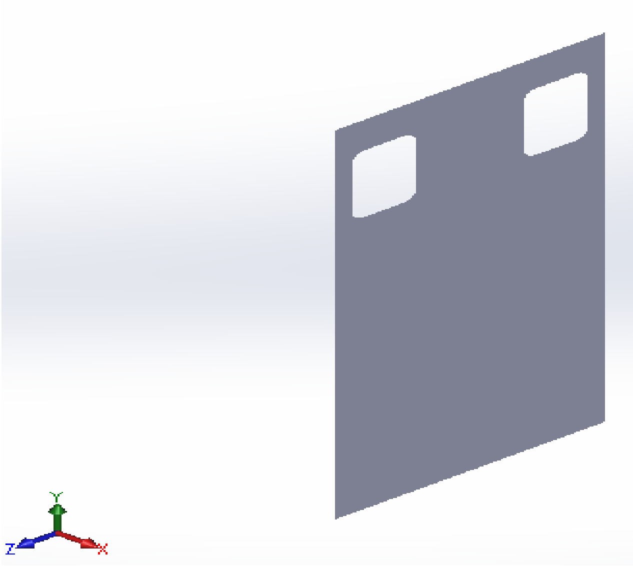


Figure 18. Planar test section with window cut-outs for ANSYS ACP

2.5. Description of the FE models

The simulation begins with a preliminary testing where the FE model was not subjected to any material failure modes, property degradations, plasticity or part connection failures. The objective of the test was to obtain the generic elastic behaviour of the test panel.

Table 2. Test Panel Stiffeners

	Test Panel Model
Number of Frames	2
Number of Stringers	3

Materials

The material that will be used for the analysis is the Composite Material Epoxy Carbon Woven (230Gpa) Prepreg. From the ANSYS, Inc. 18.1 software Engineering Data library. Thickness is 0.00025 metres.

FE Mesh

The shell sections were generated on the surface using the program-controlled option of ANSYS 18.1 software. With the minimum edge limit kept at 3.0 millimetres. and the size function kept at Program-controlled option. The relevance centre was kept coarse and was applied to the entire assembly.

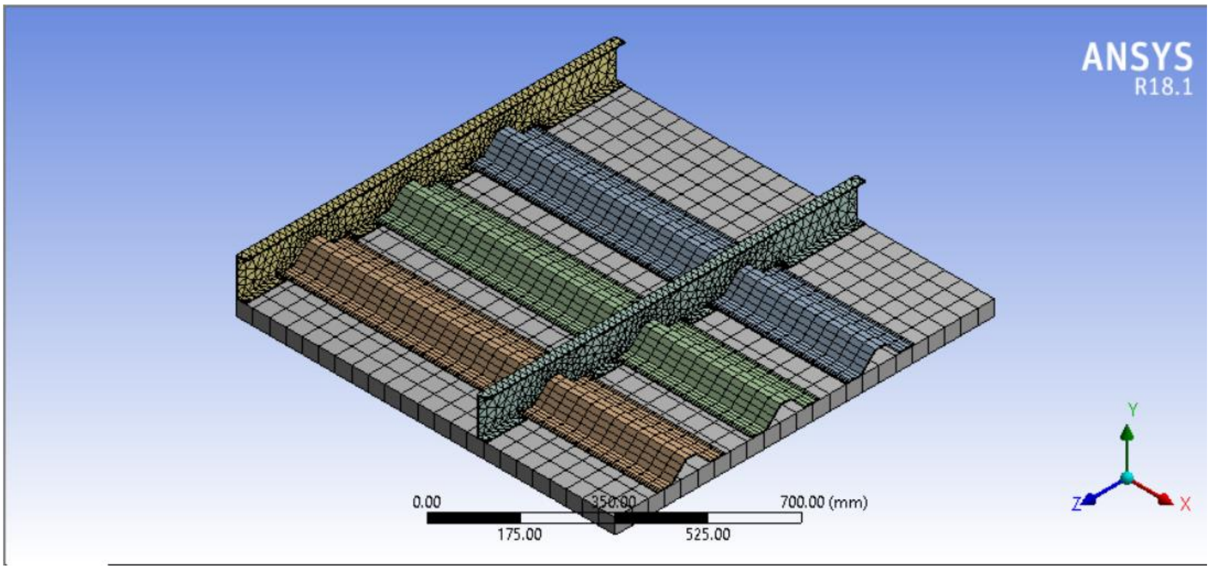


Figure 19. Meshed View of the Test Assembly Panel

2.5.1. Boundary Conditions

The test panel was fixed in all degrees of freedom. This is slightly different from the real-world scenario were the adjoining fuselage section adds into the elastic behaviour of the panel and provides flexibility. Fixing all degrees of freedom means the panel would act a lot stiffer than the actual case. This can help understand the behaviour of the material as larger deformation could mean the structure would fail in the actual situation.

Using the data derived from various sources, the mass of the impacting vehicle is used to obtain values of the impacting forces. The ground vehicle velocity data is decided by reviewing airport operations safety catalogue.

2.6. Preparation of the Ply Lay-up using ANSYS, Inc. 18.1 ANSYS Composite Pre/Post (ACP)

The defining of the layers was done by using the ANSYS ACP module. ANSYS provides a simulation workflow that is complete for the design of composite structures. This process has been kept very similar to the original manufacturing process of a composite material. This is how it follows:

- Base material definition. Along with fabrics, and predefined stack-ups.
- Material orientation based on geometric attributes.
- Local and global ply definition as when fabrics are laid onto the mold.

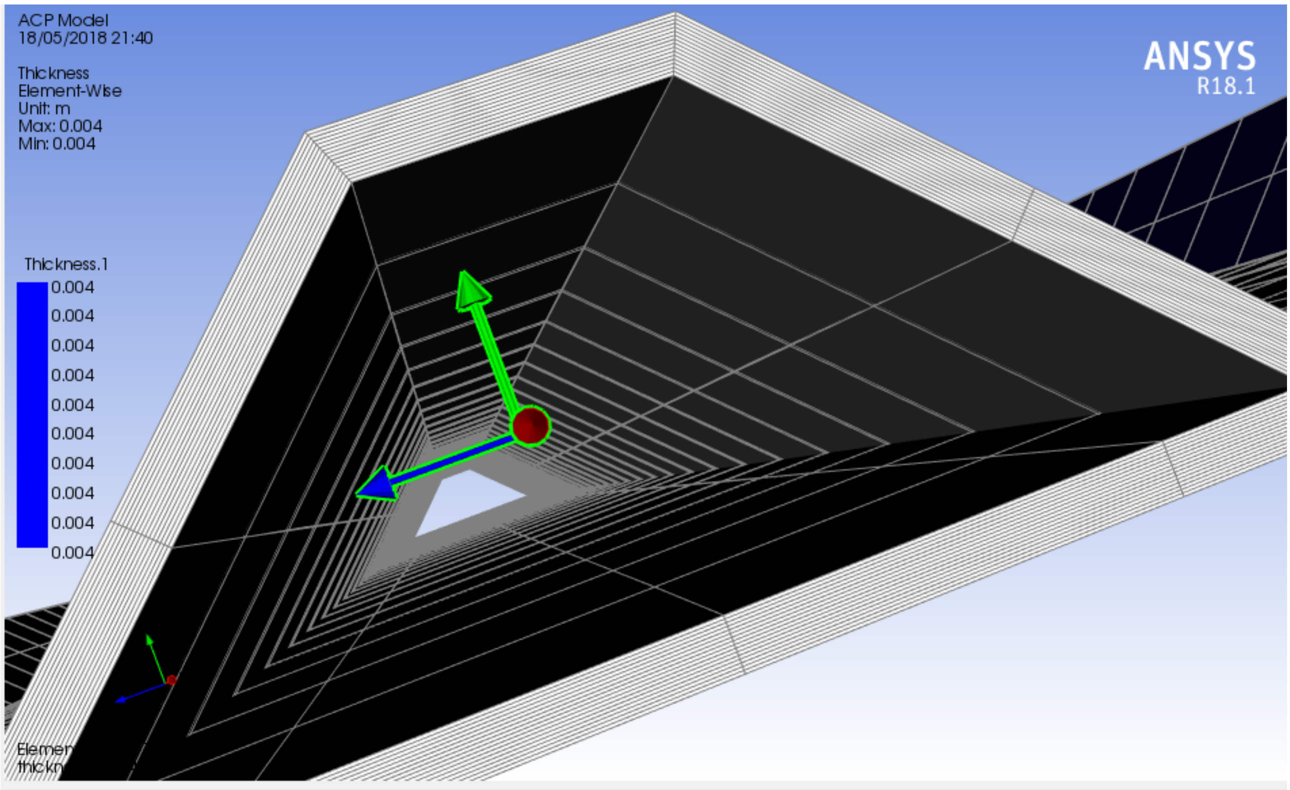


Figure 20 Cross – sectional view of the Material Ply lay- up

Steps involved in designing the Material Lay-up using ANSYS, Inc. ACP 18.1:

1. The initial step involves importing the geometry into the ANSYS Workbench User-Interface. The model so obtained needs to be meshed before it is imported into the ACP module.
2. Engineering Data such as the selection of material is then carried out. The composite menu gives different types of composite materials that can be made into various fibre – types.
3. It is better to create named-selections as that would allow to navigate between different part while applying mesh characteristics as well as fabric orientation scale.
4. The ACP module has various options in the Pre-processor section. The sections to be mindful are the: ACP model, Element Sets, Rosettes, Oriented Selection Sets.

Rosettes – These are coordinate systems that are used to set the direction of reference of Oriented Selection Sets. The main 0° lay-up is defined by the Rosettes. It is compatible with the Global coordinate system as the location of the origin and directions of Rosettes match up with the Global coordinates. They are independent of the meshing tool even though a meshed element cell is required to define its properties.

One or more Rosettes can be defined for the same geometry to account for the different planes of the geometry. This is further helpful as the corresponding Rosettes is used for setting the Reference

Direction in an Oriented Element Set (OSS). Rosettes and Oriented Selection Sets are perpendicular to each other. Shown in Figure 22 & 23. [23]

Oriented Selection Sets – Oriented Selection Sets are element sets, with additional information about the orientations of the element. The orientation direction of an Element Set is responsible for the eventual stacking up direction of the layer. The 0°-degree direction is dependent on the Reference Direction set by the Oriented Selection Sets. For the associated lay-up. The dialogue box gives an option to select the offset direction from the selected face, thereby allowing full control over the layer stack-up. Shown in Figure 24.

5. The final steps involve defining the of Modelling Groups. This module involves choosing the corresponding oriented sets, the ply material, which in this case would be the stack up that we have used. Using the stack up function allows for creation of multiple layers with ease as they act like single lamina.

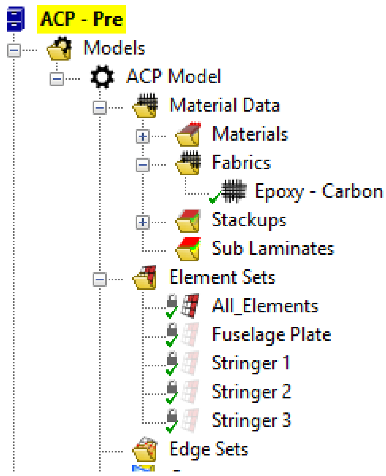


Figure 21. Element tree for ACP Model and Element Sets

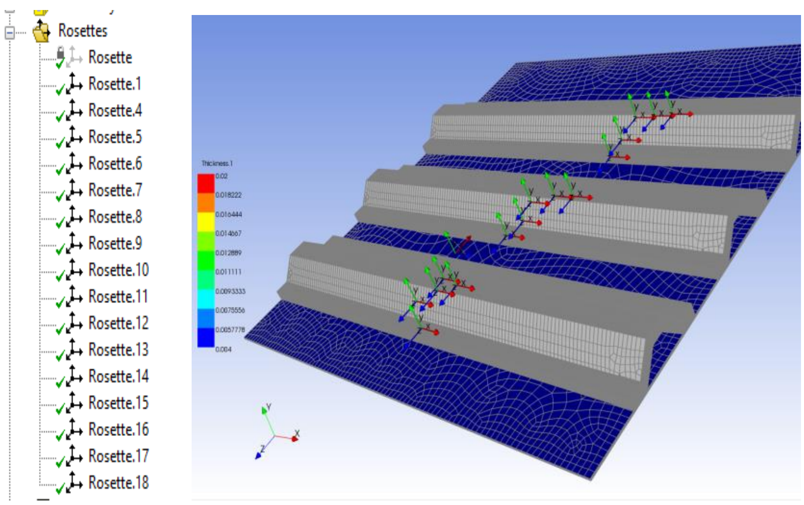


Figure 22. Rosettes creation and Subsequent Location on the plate

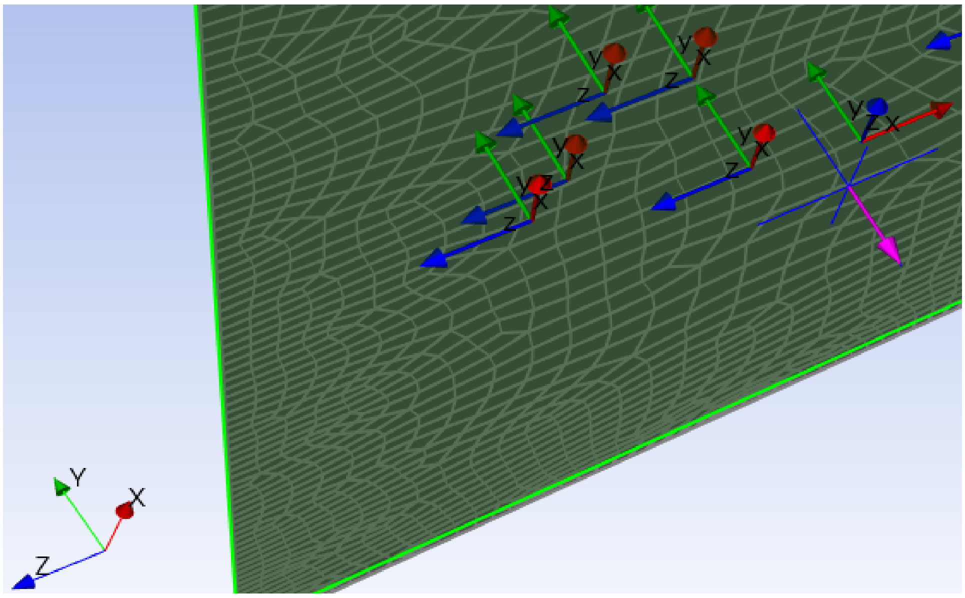


Figure 23. The Pink arrow shows the direction of the layer stack-up. (Top to down)

2.7. Calculations

2.7.1. Calculations of Impact Loads

The major calculations involved in this thesis is the amount of impact load that acts on the body (test section). The test section needs to display strength to with-stand the impact loads and show safe performances. Using Equation 1, we can calculate the loads that act upon on the Test Panel.

First Load Bracket – Calculation of Impact Load for a Vehicle weighing 2000 kilograms. Velocity at a set constant of 1 metre per second.

$$E_{k1} = 0.5*2000*1$$
$$= 1000 \text{ Joules}$$

Assuming that the vehicle decelerated over a distance of 1 metre. Thereby using the Force acting and work done relation, we can find the impact load.

$$Work\ Done\ (W) = Force * Displacement$$
$$1000 = F * 1 \tag{25}$$
$$\text{Force} = 1000 \text{ Newtons}$$

Second Load Bracket – Calculation of Impact Load for a Vehicle weighing 3000 kilograms. Velocity is again set at a constant of 1 metre per second.

$$E_{k2} = 0.5*3000*1$$
$$= 1500 \text{ Joules}$$

Assuming that the vehicle decelerated over a distance of 1 metre. Thereby using the Force acting and work done relation, we can find the impact load.

$$1500 = F * 1$$
$$\text{Force} = 1500 \text{ Newtons}$$

Third Load Bracket – Calculation of Impact Load for a Vehicle weighing 5000 kilograms. Velocity is again set at a constant of 1 metre per second.

$$E_{k3} = 0.5*5000*1$$
$$= 2500 \text{ Joules}$$

Assuming that the vehicle decelerated over a distance of 1 metre. Thereby using the Force acting and work done relation, we can find the impact load.

$$2500 = F * 1$$
$$\text{Force} = 2500 \text{ Newtons}$$

Note: *All calculations have been carried out in accordance to the formulae and techniques used in the study conducted at the University of Chicago in San Diego and the European Aviation Safety Agency test methodology.*

2.8. Finite Element Calculations using ANSYS

The final-step in the thesis is the calculation of the deformation and stresses generated due to the previously mentioned load values. A computer simulation is used to determine the strengths of the structure. The Computer Simulation method has been catching up for ANSYS and other Computer Aided Engineering Software as the material library and the deformation characteristics are being fed into the database of the software. The current methodology of fabricating a test section for the sole purpose of testing has been proving costly to be carried out regularly.

For the Finite Element Analysis (FEA) part, ANSYS Inc. – ANSYS Workbench 18.1 is used. ANSYS Workbench allows a user to run simulations involving structural problems, Computational Fluid Dynamics problems, Implicit and Explicit methods for solving Finite Element problems. Finite Element study is required to study the propagation of the stresses and deformations. These values can then be checked with a standard material, such aluminium to determine the effectivity of the materials being used. The finite element method creates small elements of the test section and then it is subjected to analysis in FEA software. The Mesh accuracy determines the accuracy of the results. ANSYS 18.1 was chosen as it had the required ACP module to design the Composite lay-up.

In this report, the test section will be subjected to Static Structural Analysis and Transient Structural Analysis. Static Structural Analysis is chosen to study the behaviour when a continuous force is applied. Transient Structural Analysis is done to study the effect of the load when it is applied for a short-period of time. The response of the structure to these conditions is crucial.

2.8.1. Test 1: Static Analysis of the Composite Panel

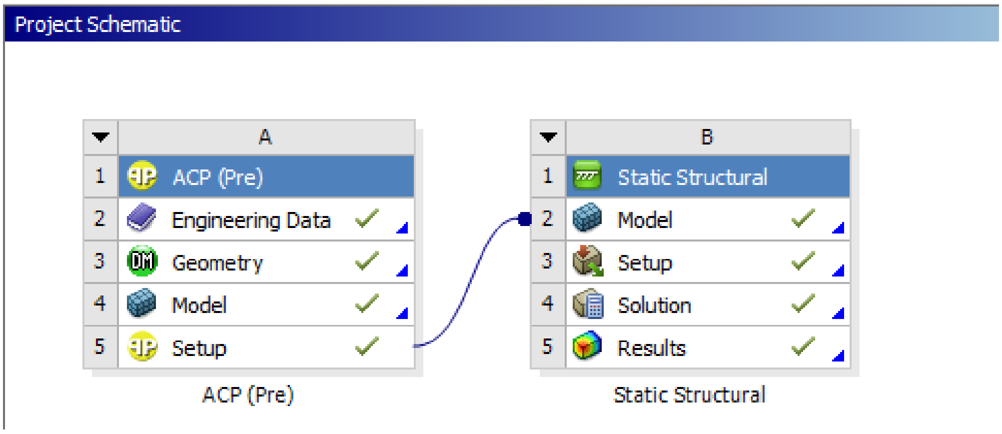


Figure 24. ANSYS Workbench Project Schematic

Imported Geometry

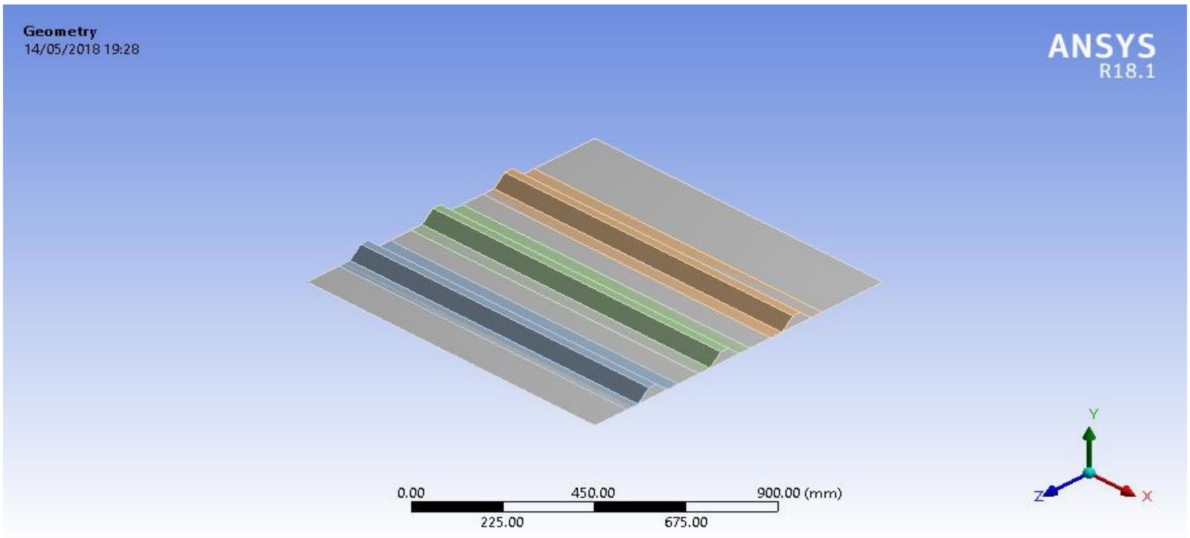


Figure 25. ANSYS Imported Geometry in Mechanical Workbench

As shown in the Figure 25. The model has been imported into ANSYS Workbench for analysis. The ACP pre/post means that the model will have the material properties already defined.

Boundary Conditions

Fixtures

As shown in the Figure 26., the four sides of the test section have been chosen for the addition of the Fixed Support criteria. Usually an aircraft fuselage section panel will tend to show more elasticity as it tends to be supported by other section panels. By adding a fixed support, we are limiting the degree of freedom of the edges and are concerned more about the inner-middle portion of the section.

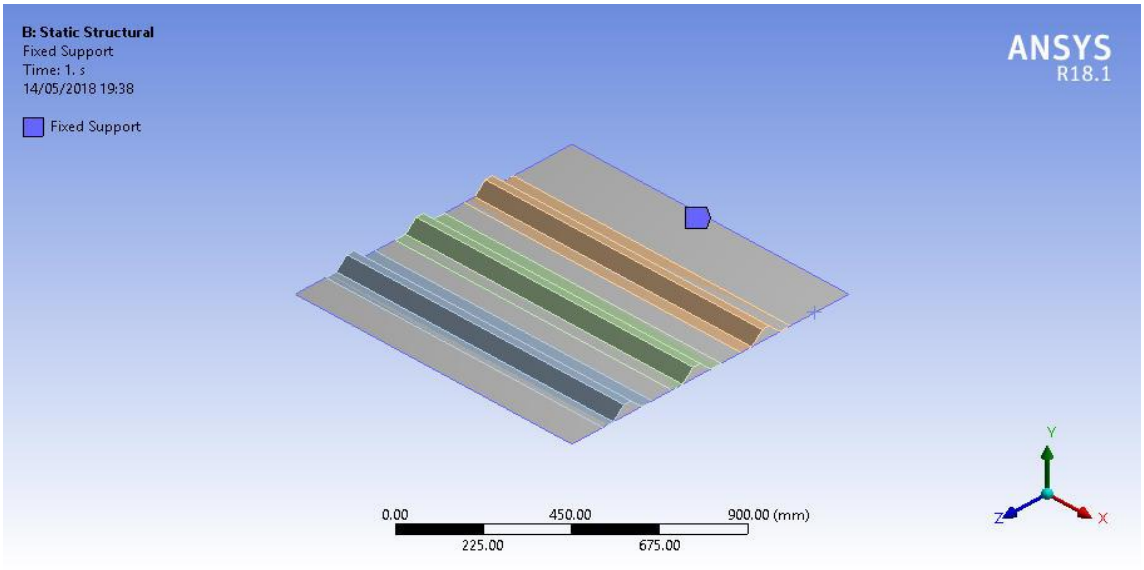


Figure 26. Fixed Support

Loads

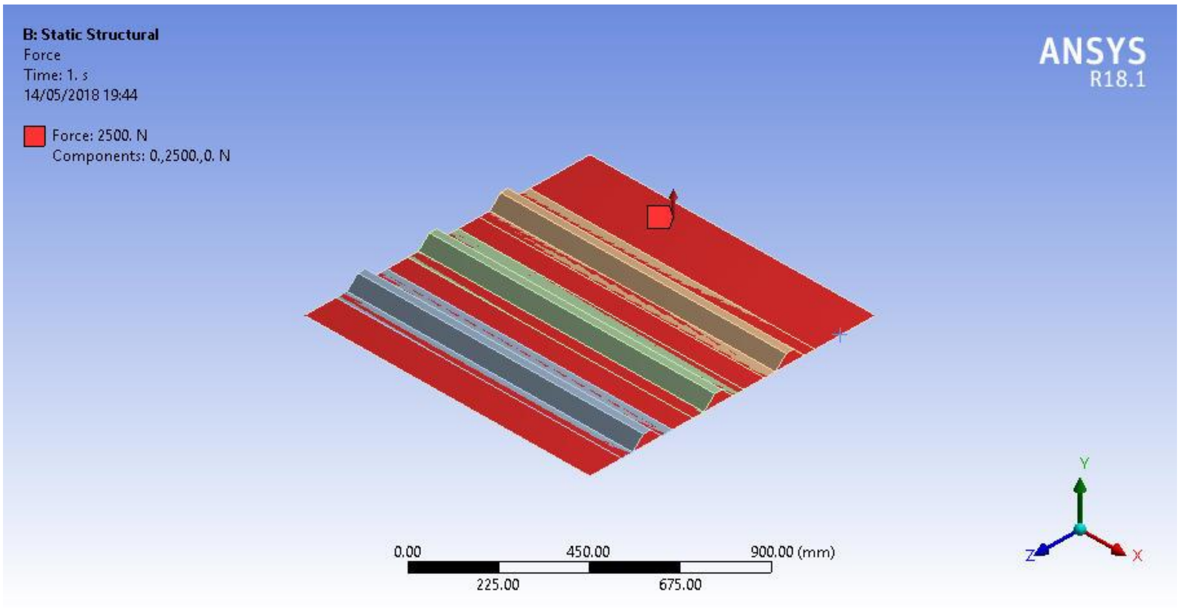


Figure 27. Load Application Direction

The test panel is aligned in such a way that the force had to be applied using the component method and therefore, the Y-axis is the axis along which the load needs to be applied in order to make a perpendicular impact, as shown in Figure 27. In this case, the highest value of load is applied. The values of the applied loads were calculated in section 5.1.

Meshing

The accuracy of the Finite Element Analysis is heavily dependent on the accuracy of the mesh used. A coarse mesh and the entire structure is meshed. Meshing requires a special understanding of the case being studied. Finely meshed components will require a huge amount of computational time. The meshed model is illustrated in Figure 28.

Table 3. Mesh Quality Details for Composite Panel

Number of Nodes	10235
Number of Elements	9590
Size Function	Adaptive
Relevant Centre	Coarse
Minimum Edge Length	1e-001 millimetres
Transition Ratio	0.272
Maximum Layers	5
Growth Rate	1.2

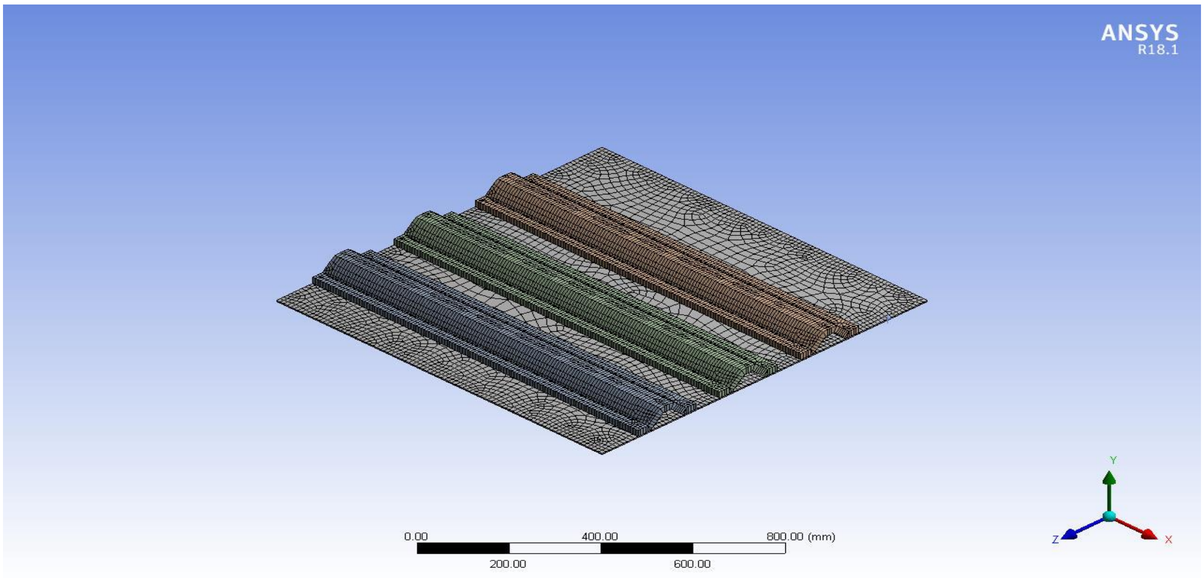


Figure 28. Mesh Quality

2.8.2. Static and Transient Test for Flat Plate with Window Cuts

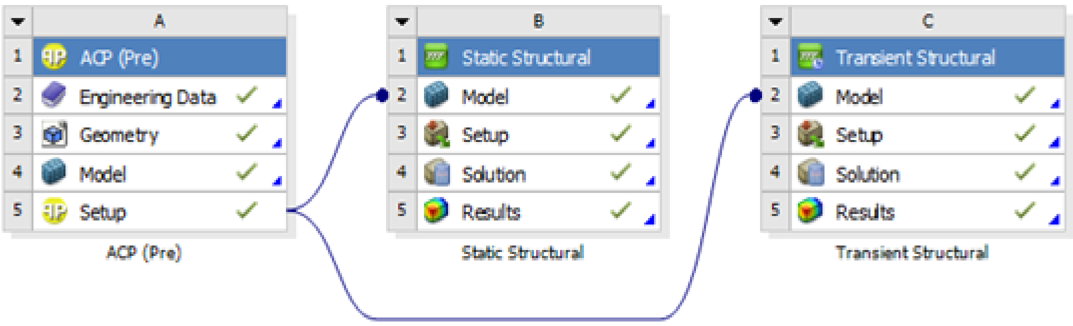


Figure 29. Project Schematic for Transient Structural Tests

For the second test, a simple skin panel, with no reinforcing stiffeners was modelled. Two extrusions were cut into the panel to replicate the window sections on fuselage. The thickness of the plate was increased to 40 millimetres, to counter for the lack of stiffeners. The test schematic can be viewed in the Figure 29.

Meshed Geometry

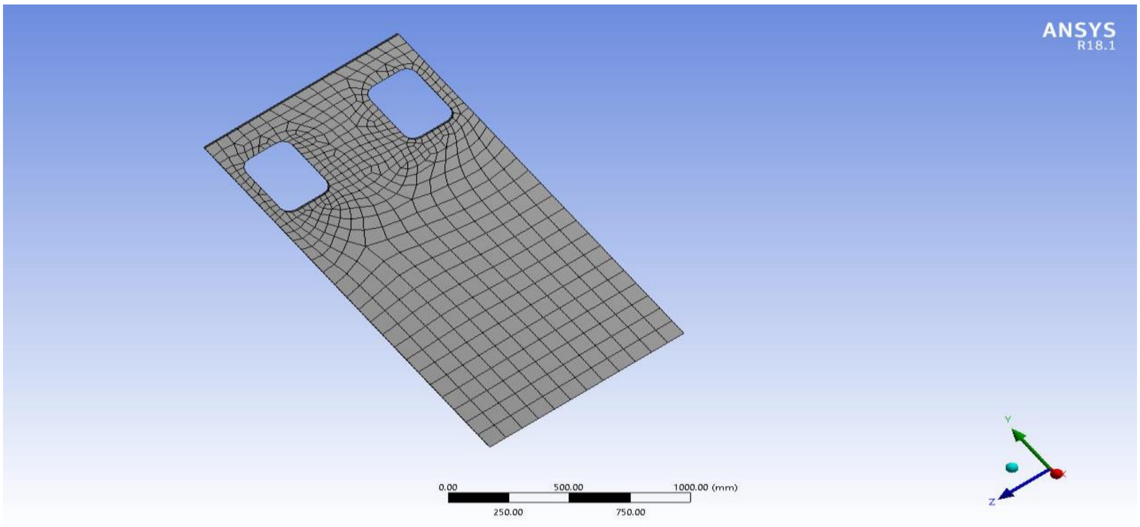


Figure 30. Meshed Geometry

Table 4. Mesh Quality Details for Flat Plate Fuselage Panel

Number of Nodes	604
Number of Elements	525
Size Function	Adaptive
Relevant Centre	Coarse

2.8.3. Boundary Conditions

Fixtures

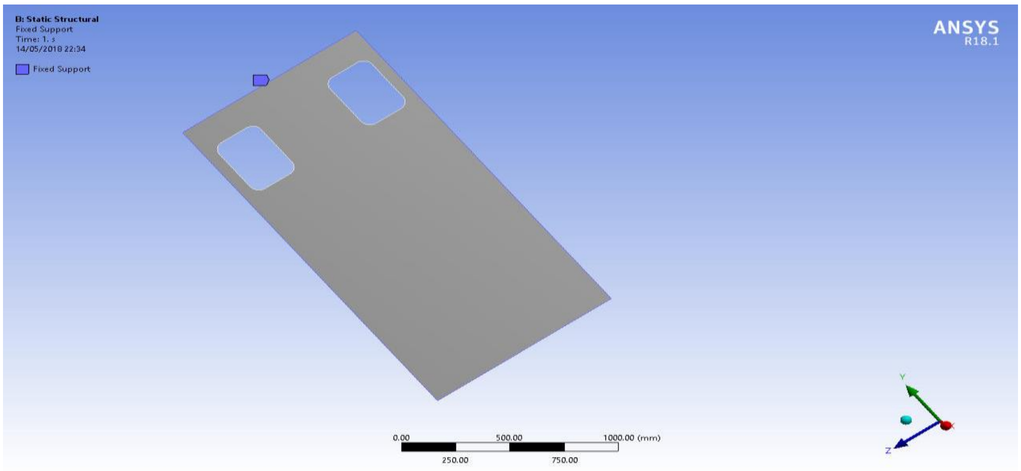


Figure 31. Fixed Support for fixtures

Load for Static Structural

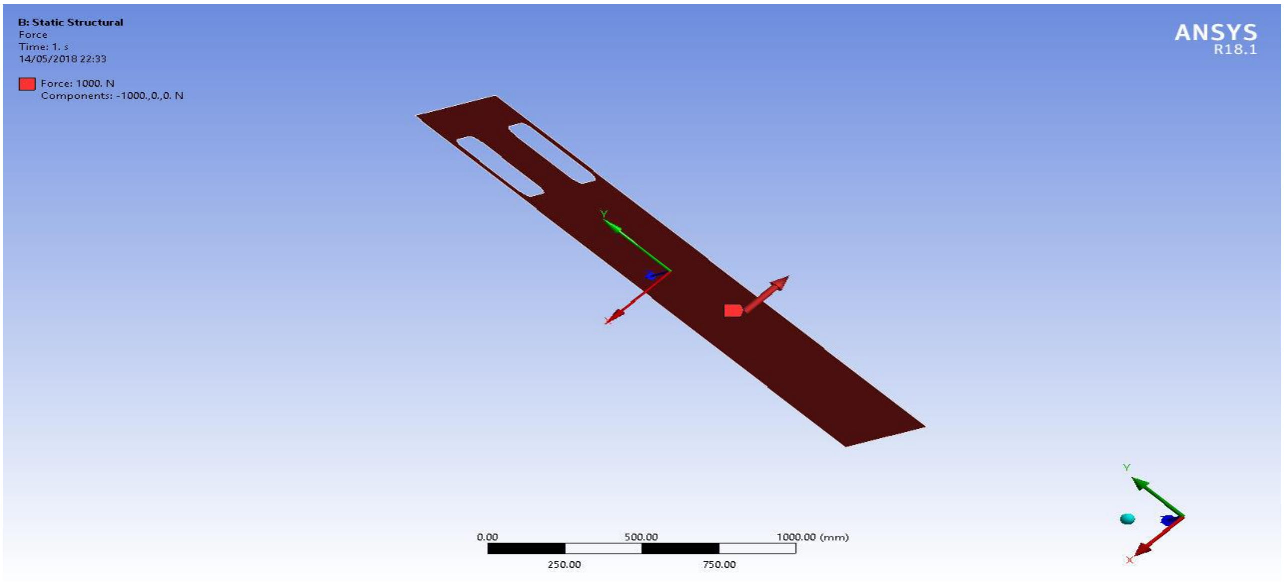
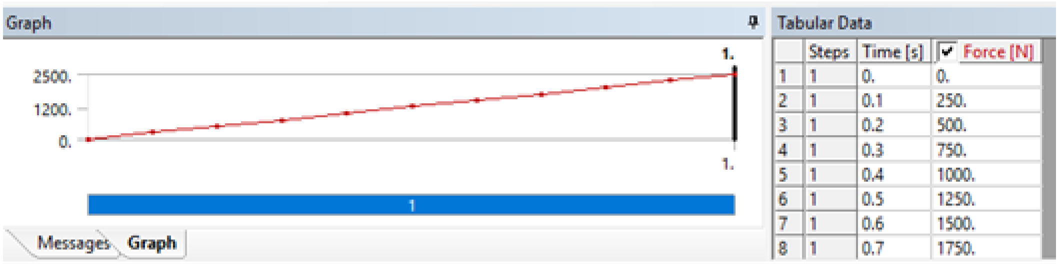


Figure 32. Load of 2500 Newtons applied

Load for Transient Analysis



	Steps	Time [s]	Force [N]
1	1	0.	0.
2	1	0.1	250.
3	1	0.2	500.
4	1	0.3	750.
5	1	0.4	1000.
6	1	0.5	1250.
7	1	0.6	1500.
8	1	0.7	1750.

Figure 33. Load Application Step for Transient Analysis using Tabular Data

3. Results Analysis of the Emperimentation for Aircraft Composite Structures

In this section, the discussions made to the values obtained after the simulations were carried out in ANSYS Inc. Workbench 18.1.

Finite Element Analysis Results

The different test panels were created and tested with the appropriate ply material properties to give the results. The values that are of the utmost importance with,

- Total Energy
- Deformation
- Stress developed

3.1. Static Analysis for Fuselage Panel at 2500N (Newtons)

Deformation **-** Deformation analysis if the basis of study for this thesis report. Understanding how the material behaves under certain load conditions is important. Appropriately modelling the geometry, along with the Boundary conditions will give results that should satisfy the query of the report.

3.1.1. Deformation analysis for 2500N

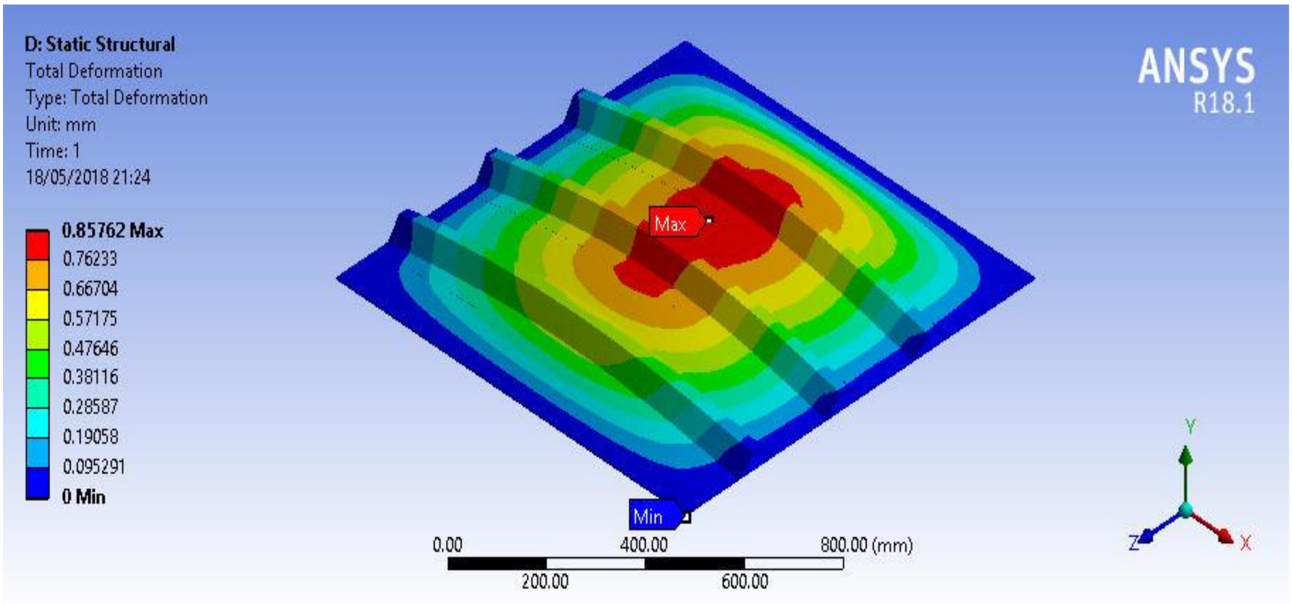


Figure 34. Deformation – Fuselage Plate with Stiffeners

3.1.2. Stress analysis for 2500N

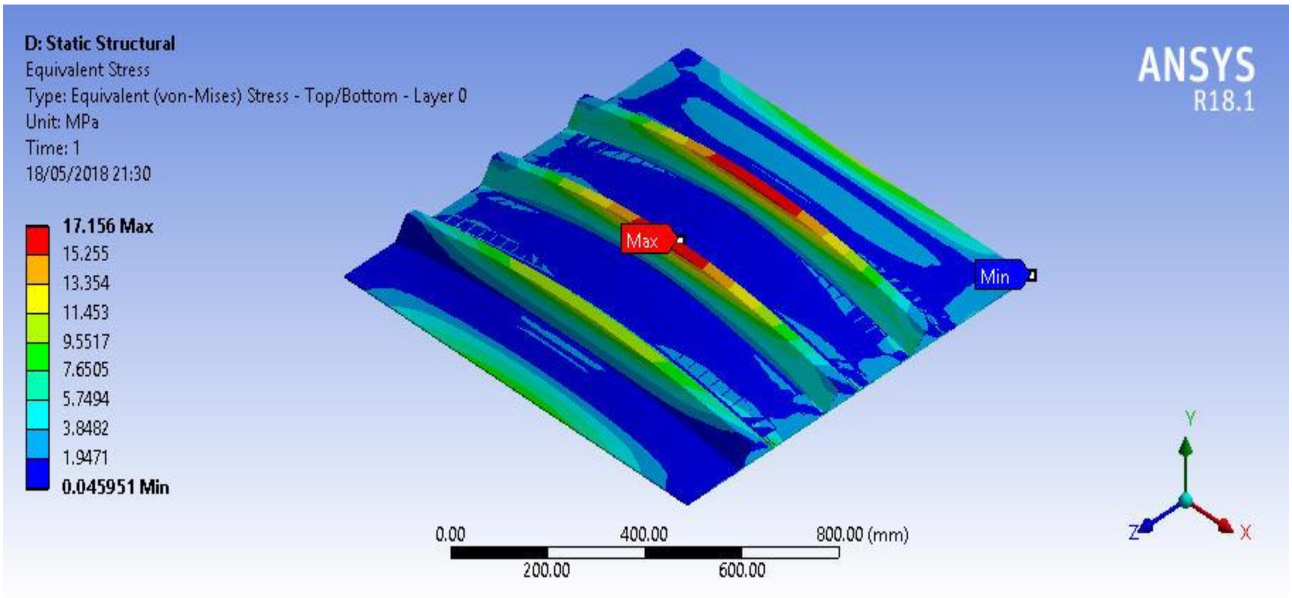


Figure 35. Stress generated at 2500N

The Static Structural Test for the Airbus A350 XWB was carried out using the ANSYS Static Structural Module. The above illustrated Figure 34 and Figure 35, from 3.1.1.1 and 3.1.1.2, illustrate the generated Deformations and Stresses. The values so derived show that having reinforcing has a positive effect on the structure. The stringers have been made of the same carbon fibre material. This has affected the deformation caused by the force of 2500N. The highest values are 0.86 millimetres for deformation and a value of 17.16 MPa (mega-pascals).

3.2. Static Analysis for Flat Plate Fuselage Panel

3.2.1. Total Deformation analysis of Flat Plate Fuselage Panel at 2500N

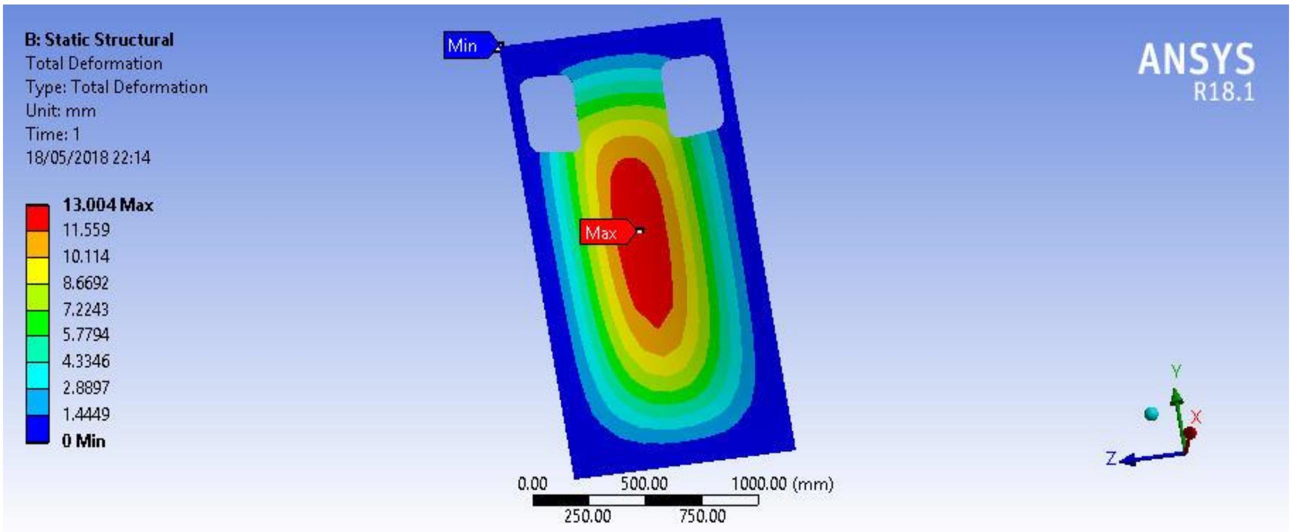


Figure 36. Maximum Deflection in Flat Plate at 2500N (Skin Thickness = 4 mm)

Equivalent Stress generated in Flat Plate at 2500N

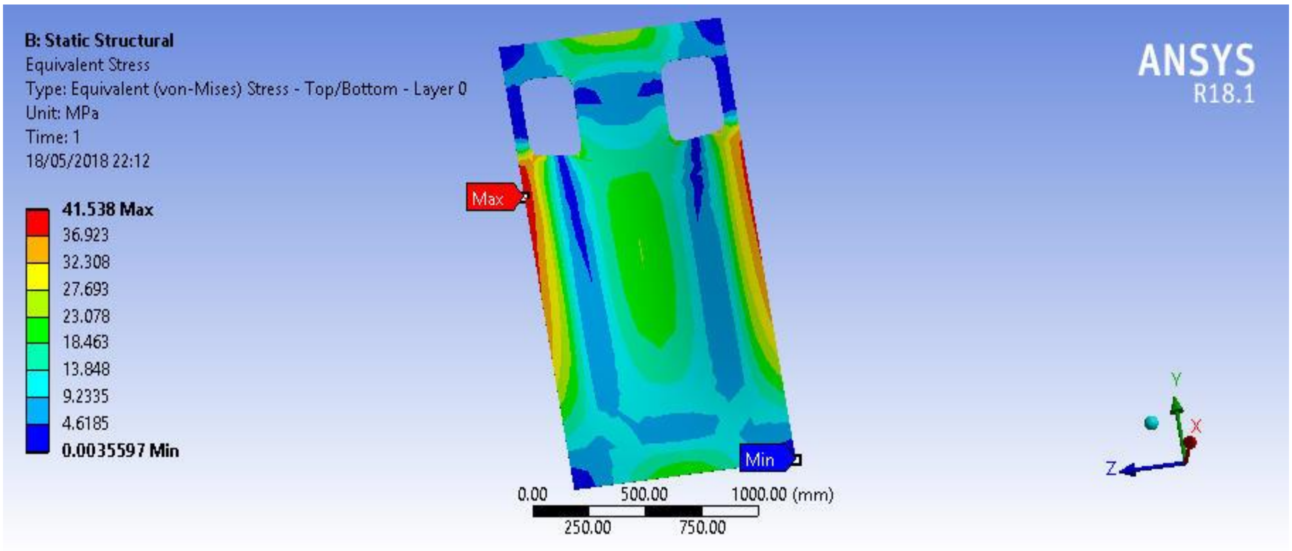


Figure 37. Maximum Stress generated and location

A similar static structure consisting of a Flat plate with cut-outs to resemble the window panels of the aircraft was considered. This test would show how the load can cause deformation in critical regions such as doors and windows. The Figure 36 and Figure 37, from 3.1.2.1 and 3.1.2.2, illustrate the results of the conducted test.

The fixed supports at the edges have shown how the stresses will concentrated towards such fixtures. In reality, the adjoining aircraft structure will provide more elasticity for the propagation of the stresses. The highest stress value here being the 41.54 MPa (mega-pascal). The highest value for deformation is 13.004 millimetres.

Fibre Failure (Core) generated in Flat Plate at 2500N – Inverse Safety Factor graph

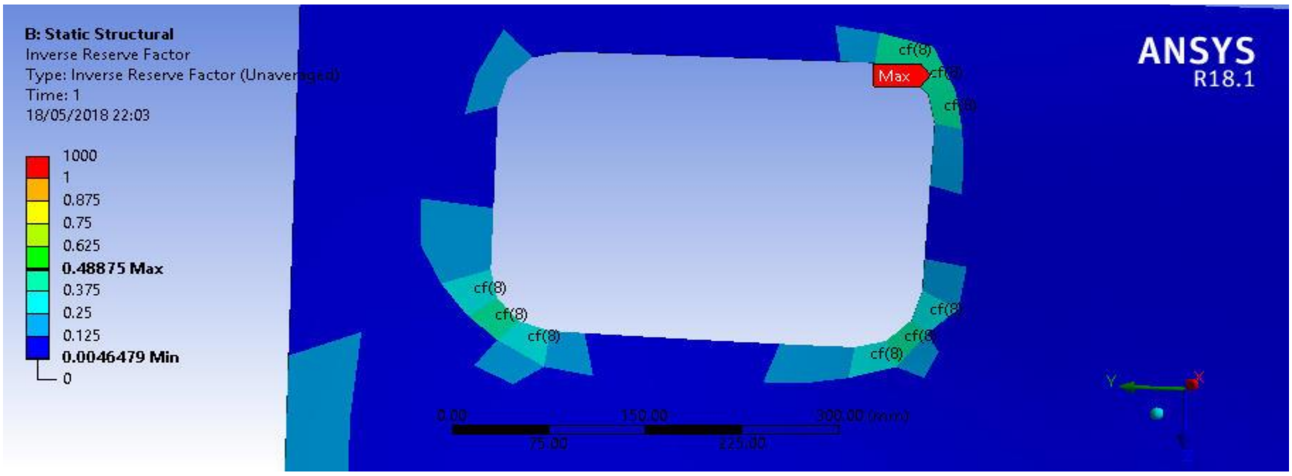


Figure 38. Failure criteria initiated to represent the failure modes

Fibre Failure Criteria for Safety Margin of the Composite Material

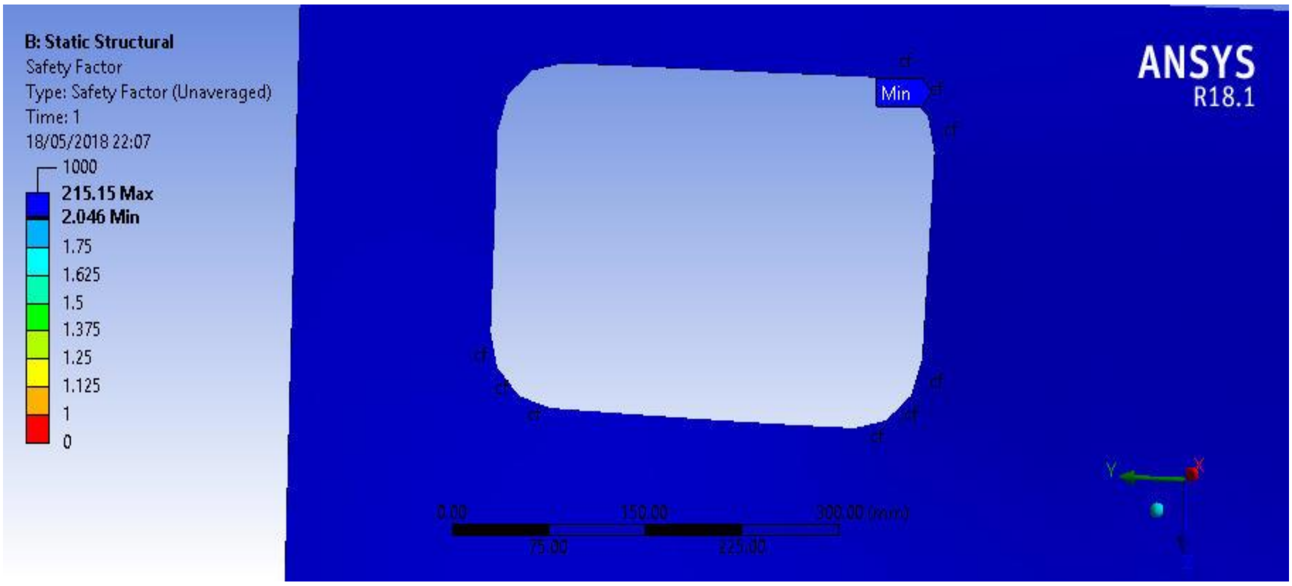


Figure 39. Safety Factor for Composite Flat Plate

After inserting the Composite Failure Tool, we can initiate the various criteria that we need to consider while working composites. The very first result to look at would be the Inverse Reserve Factor. This shows the failures that can occur when a load criterion is applied. Figure 38, from 3.1.2.3, depicts the failure: Core Failure will be generated, near the cut-out sections. The number in the bracket shows the layer where the failure will initiate from. Figure 39, from 3.1.2.4, shows the Safety Factor for the entire Flat Plate. The value is around 2.046 and would be generally considered a good number.

Such a test is needed to understand the behaviour of the composite structure and for the anticipation of failure modes. While designing, the methodology that is carried will need to be updated to fit the failure criteria within a suitable range or else the entire program could be scrapped if the product does not reflect the safety standards.

3.3. Transient Structural Analysis

Transient Analysis of the structure is done to give the effect of inertia when a force is applied on a body. This is study is carried with forces that are applied for very less periods of time. In this case, the study for conducted for an applied load of 2500N for 1 second.

3.3.1. Transient structural analysis with total deformation

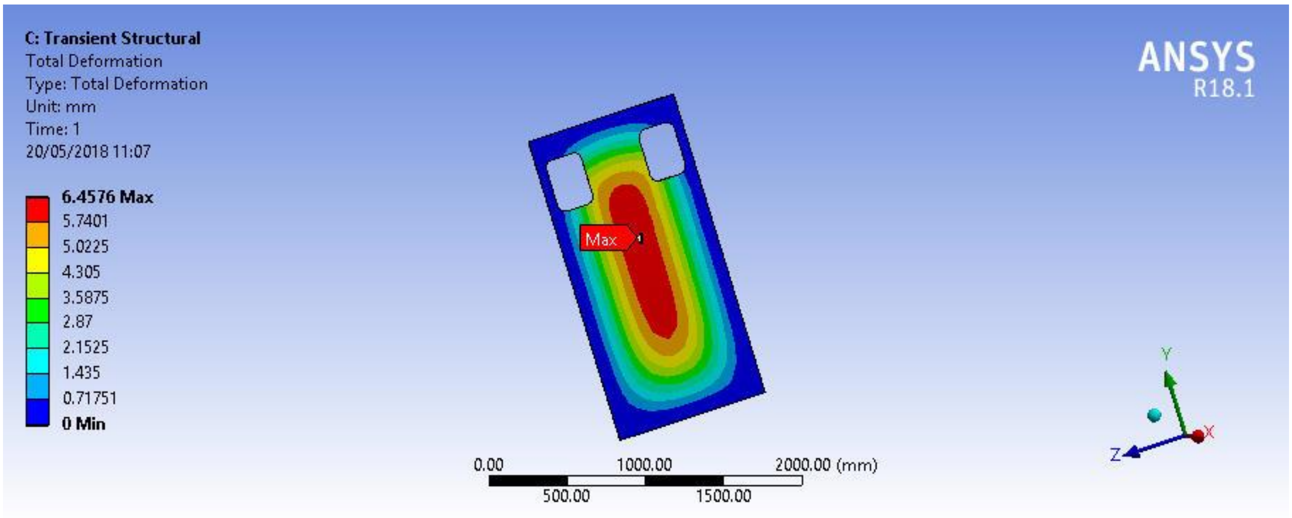


Figure 40. Deformation of Flat Plate section due to Transient Load

3.3.2. Transient structural analysis in equivalent stresses

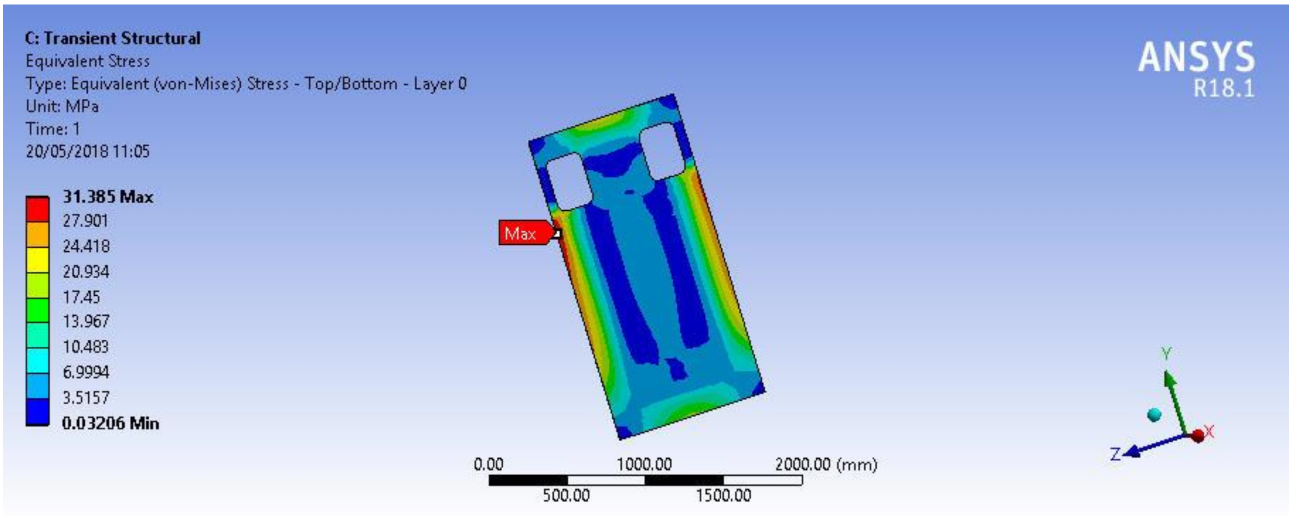


Figure 41. Stresses generated due to Transient Load Application

The EASA CS – 25 legislations have provisions for studying the effects of loads that are time dependent. The structure is expected to perform within the safety criteria for a minimum of three seconds when working under critical load criteria.

The Figure 40 and Figure 41, from 3.1.3.1 and 3.1.3.2, illustrate the deformation and stress values obtained due to the test carried out. The test was carried out for exactly one second with the time step split in 0.01.

3.3.3. Strain energy due to transient loading for flat plate fuseLage panel

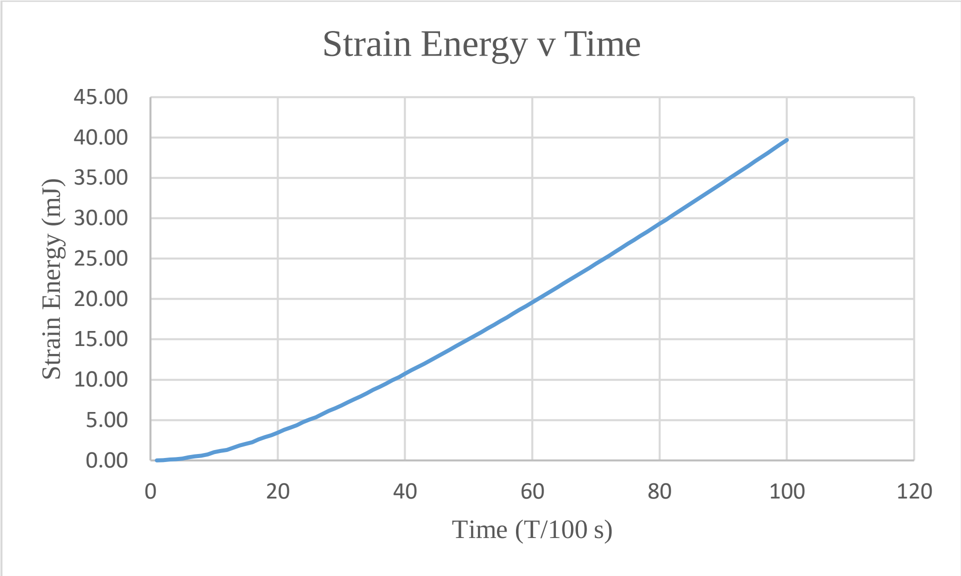


Figure 42. Strain Energy Response Graph

3.3.4. Strain Energy analysis due to Transient Load for Flat Plate Fuselage Panel

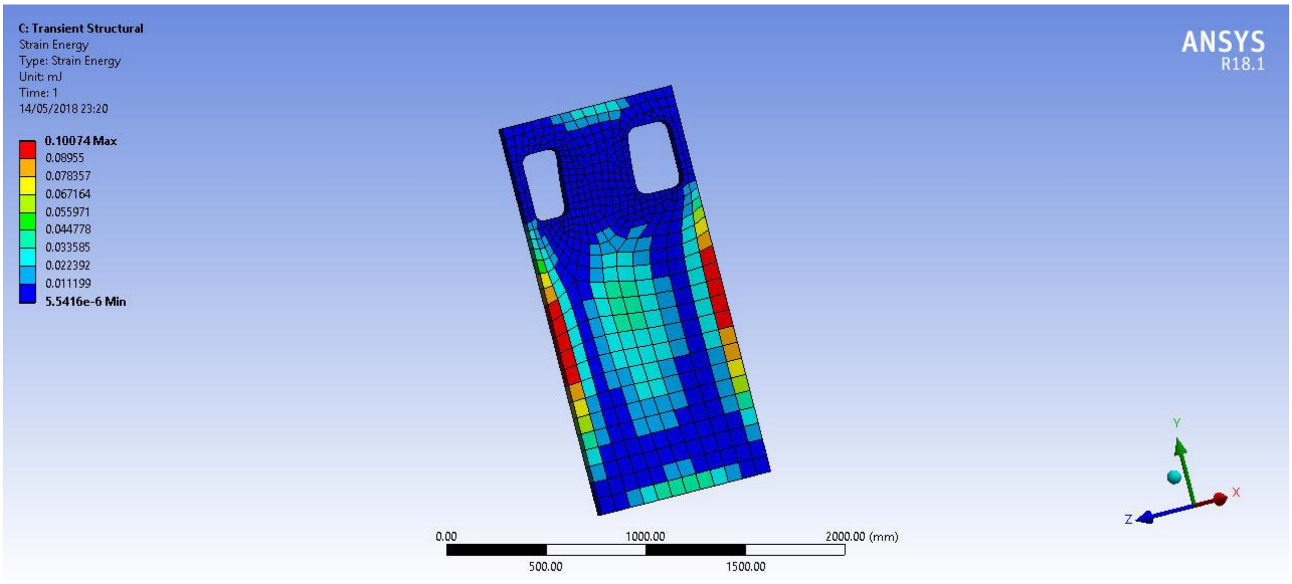


Figure 43. Strain Energy for Flat Plate Fuselage Panel

Strain Energy

Figure 42 depicts the graph of the Strain Energy v Time for the above conducted Transient Structural Analysis of the Composite Flat Plate. Strain Energy is the energy absorbed by a body when it subjected to a load, which causes a strain within the elastic limit of the body. The dispersion of the energy can be seen in the Figure 43.

3.4. Discussions with the maiden tests

The tests have been carried in accordance with the test carried out at University of California at San Diego (UCSD) and the provisions as laid down in the CS – 25 legislations, by the European Aviation Safety Agency. The model was tested in a Quasi – Static environment. This was done to study the effects the loads would have on the skin of the fuselage.

As the steps carried forward in the direction of completing the tests, it was seen that the structure would need further simplification to reduce computational time. Such a method needed to keep the essence of the original test while capturing the area of interest. The initial model was the entire Fuselage section of the aircraft under study but as time passed, the area of interest was identified and was subjected to further simplifications.

Working with the ANSYS ACP module required the understanding of how the software understands the inputs. Initial approach of using Solid body kept repeatedly failing as the ACP module is built to handle surface/shell bodies. Transferring of data from the ACP module to the testing system needs careful consideration.

It can be seen that the high energy/low velocity impacts do tend to affect the skin of the fuselage by causing deformations. The stresses generated do come into play when considering the failure criteria. Looking at the structure, and the way the plies have been stacked-up, any stress value that tends to exceed the pre-set value of limiting stress should be looked into.

The methodology to define the test was straight-forward and can be improved upon depending on the level of familiarity with software usage.

The main idea is to keep the aircraft fuselage section to be able to hold onto its CS – 25 certification, in case of a damage causing impact. The values so obtained have shown that it can keep aircraft safe enough for normal operations, until the routine check-up is carried out.

The methodology slightly varies in terms of the support being used. Considering that the skin panel is supported by a fixed support, the degree of freedom at the edge are constrained. The values of deformation obtained give us an idea about how the material is expected to behave.

The test sections showed variations with respect to the methodology used. The bending of the stringers in the static test was expected. The values that were obtained, were within the expected range for such load cases.

3.5. Recommendations and proposed changes

The relatively new nature of the composite fibre-based components means that testing on these materials will require newer and improved methods. The testing methodology for composites will undergo continuous changes up until there can be a certain way that defines a wide variety of working conditions. Working on a simulation software, it is easy to understand the different approaches each software uses. For someone, looking to work with these complex materials, he/she will need to have a strong understanding of the terms that govern the operations of these materials.

- Looking at the test section, there is possibility to include shear ties-to-skin interface. These can act as the required fastener criteria, to even further the behavior of the skin panel under loading conditions.
- Boundary conditions require changes in its nature. To imbibe the elastic nature of the fuselage section around it, the test panel needs to be fitted with elastic spring supports to enhance the elasticity effects.
- Use of Python scripting to generate the model so that the conditions of the body can be better manipulated via the User Defined function aspect.
- Following the EASA certification code for closeness in test model fabrication. The EASA, like any other organization does tend to make timely updates to the legislations it has passed previously. Following the trends of the market, a test model can be made as close as possible to the real case.
- As previously mentioned, composites provide a lot of combinations and permutations in order of stacking-up and layer orientation. Depending on the need, the material can be altered to match the performance requirement by changing the stack-up plots and/or the orientation of the ply. The behavior of the plies under orientation conditions can be studied using a broader spectrum of time and resources.

Conclusions

The ANSYS simulations were done to justify the objectives this thesis began with. The values so-obtained are based on the consideration of various conditions that the software took as input. The thesis was able to look into the methodology of simulating effects of loads on composite structures via the following way.

1. The literature review showed that most of the FE modelling has been on simple shapes such as rectangular shape cards. This is relatively easy to build and model with the material properties and the values obtained are easy to be understood. Complex models require careful consideration for the orientation with respect to the global coordinates. Choosing to model the test of actual aircraft like components has opened up possibilities of studies on objects that have different planes.
2. Formulation of the tests required careful understanding of what the end results need to look like. Understanding how the material behaves under different load criteria, a structural analysis for the behaviour-accounting was thought to be an ideal test. Explicit dynamics is another module that can be used to study the effects of accidental hits and be used for furthering the study in the field.
3. Building the models for tests was a challenge given how important it is to test the behaviour of the fuselage skin section. Working on the idea, many models were created to be tested along the way. Lack of resources such powerful computers that can handle higher-order calculations due to the fineness of the mesh used is one reason why complex natured CAD geometries were avoided. The test has kept the essence of the study alive while working on trimming the areas that do not need in-depth consideration at such a level. The values obtained were 0.86 millimetres of displacement for the composite fuselage section with stringer reinforcements and while a constant stress value on the skin could create a deformation of 13 millimetres. The simulation values are in the same range of value obtained through tests that were carried out by different organizations to check airworthiness of composite structures.

The Aerospace Industry is laden with people who are always looking to bring new products that are highly valuable in terms of performance of an aircraft. The new age composite is a direct result of such studies. But given how Aerospace Industry puts safety as its prime target, the materials that are used in the production of aero-vehicles should also have rigorous standards for performance. From the most sophisticated spacecraft to an everyday airliner, we are moving towards more and more composite and therefore, mitigation of problems involving composites should be a top priority.

List of references

1. Anoshkin, A., Zuiko, V., Tashkinov, M. and Silberschmidt, V. (2015). Repair of damage in aircraft composite sound-absorbing panels. Composite Structures, [online] 120, pp.153-166. Available at: https://www.researchgate.net/publication/268750681_Repair_of_damage_in_aircraft_composite_sound-absorbing_panels.
2. Katunin, A., Dragan, K. and Dziendzikowski, M. (2015). Damage identification in aircraft composite structures: A case study using various non-destructive testing techniques. Composite Structures, [online] 127, pp.1-9. Available at: https://www.infona.pl/resource/bwmeta1.element.elsevier-54cc02ba-c6ae-3eed-8c7c-3748c8438f24.
3. Soutis, C. (2009). Fibre reinforced composites in aircraft construction. Progress in Aerospace Sciences, [online] 41(2), pp.143-151. Available at: https://www.sciencedirect.com/science/article/pii/S0376042105000448.
4. Diamanti, K. and Soutis, C. (2010). Structural health monitoring techniques for aircraft composite structures. Progress in Aerospace Sciences, [online] 46(8), pp.342-352. Available at: http://www.scirp.org/(S(351jmbntvnsjt1aadkposzje))/reference/ReferencesPapers.aspx?ReferenceID=1980440.
5. Kashtalyan, M. and Soutis, C. (2007). Stiffness and fracture analysis of laminated composites with off-axis ply matrix cracking. *Composites Part A: Applied Science and Manufacturing*, [online] 38(4), pp.1262-1269. Available at: https://scinapse.io/papers/2000301072.
6. REIFSNIDER, K. and TALUG, A. (1980). Analysis of fatigue damage in composite laminates. *International Journal of Fatigue*, [online] 2(1), pp.3-11. Available at: https://www.sciencedirect.com/science/article/pii/0142112380900225.
7. Jollivet, T., Peyrac, C. and Lefebvre, F. (2013). Damage of Composite Materials. *Procedia Engineering*, [online] 66, pp.746-758. Available at: https://www.sciencedirect.com/science/article/pii/S1877705813019619.
8. Jagtap, K., Ghorpade, S., Lal, A. and Singh, B. (2017). Finite Element Simulation of Low Velocity Impact Damage in Composite Laminates. *Materials Today: Proceedings*, [online] 4(2), pp.2464-2469. Available at: https://www.sciencedirect.com/science/article/pii/S2214785317302961.

9. Bouvet, C., Rivallant, S. and Barrau, J. (2012). Low velocity impact modeling in composite laminates capturing permanent indentation. *Composites Science and Technology*, [online] 72(16), pp.1977-1988. Available at: http://oatao.univ-toulouse.fr/3324/2/Bouvet_3324.pdf.
10. Hyung Yun Choi and Chang, F. (2007). A Model for Predicting Damage in Graphite/Epoxy Laminated Composites Resulting from Low-Velocity Point Impact. *Journal of Composite Materials*, [online] 26(14), pp.2134-2169. Available at: http://journals.sagepub.com/doi/10.1177/002199839202601408.
11. Lessard, L. (1991). Damage tolerance of laminated composites containing an open hole and subjected to compressive loadings: Part I — Analysis. *Composites*, [online] 22(4), p.329. Available at: http://journals.sagepub.com/doi/abs/10.1177/002199839102500101.
12. Pagano, N. and Pipes, R. (2009). The Influence of Stacking Sequence on Laminate Strength. *Journal of Composite Materials*, [online] 5(1), pp.50-57. Available at: http://journals.sagepub.com/doi/10.1177/002199837100500105.
13. Hashin, Z. (1981). Fatigue Failure Criteria for Unidirectional Fiber Composites. *Journal of Applied Mechanics*, [online] 48(4), p.846. Available at: http://www.dtic.mil/dtic/tr/fulltext/u2/a088733.pdf.
14. Mikulik, Z. and Haase, P. (2018). *Damage tolerance and survivability of composite aircraft structures - Structural Integrity and Durability of Advanced Composites - 23*. [online] Sciencedirect.com. Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf
15. Balpreet S. Kukreja. 2018. Composites in the Aircraft Industry. [ONLINE] Available at: http://www.appropedia.org/Composites_in_the_Aircraft_Industry
16. Anon, Composites: Airbus continues to shape the future. *Airbus*. Available at: http://www.airbus.com/newsroom/news/en/2017/08/composites--airbus-continues-to-shape-the-future.html
17. Hamerton, 1998. Computer modelling and chemical kinetics of the cure of aerospace resin systems. *Emeraldinsight*. Available at: https://www.emeraldinsight.com/doi/full/10.1108/00022669910296918?fullSc=1&mbSc=1
18. Adam Quilter, 2017. Composites in Aerospace Applications. *ihsmarkit*. Available at: https://ihsmarkit.com/pdf/Composites-Aerospace-Applications-whitepaper_264558110913046532.pdf

19. Zoltan Mikulik & Peter Haase, 2012. CODAMEIN - Composite Damage Metrics and Inspection. *www.easa.europa.eu*. Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf.
20. Anon, 2005. *AIRCRAFT CHARACTERISTICS AIRPORT AND MAINTENANCE PLANNING*, Blagnac Cedex: AIRBUS S.A.S. Available at: www.airbus.com
21. ShuguangLi et al., 2017. The Tsai-Wu failure criterion rationalised in the context of UD composites. *www.sciencedirect.com*. Available at: https://www.sciencedirect.com/science/article/pii/S1359835X17303068
22. Hashin, Z., 1980. Failure Criteria for Unidirectional Fiber Composites. *Journal of Applied Mechanics*. Available at: http://appliedmechanics.asmedigitalcollection.asme.org/article.aspx?articleid=1405139
23. Anon, 2015. 2.1.5. Rosettes. *www.sharcnet.ca/Software/Ansys*. Available at: https://www.sharcnet.ca/Software/Ansys/17.0/en-us/help/acp_ug/acp_Rosettes.html
24. Tiwari, Nachiketa, 2012. Introduction to Composite Materials and Structures. *nptel.ac.in*. Available at: http://nptel.ac.in/courses/112104168/L32.pdf.
25. Load levels for different designs v severity of damage. (2012). [image] Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf
26. Ground Handling Operations Complexity (2012). [image] Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf
27. Location of the ground operations incidents. (2012). [image] Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf
28. ANALYSIS OF RAMP DAMAGE INCIDENTS AND IMPLICATIONS FOR FUTURE COMPOSITE AIRCRAFT STRUCTURE. (2012). [image] Available at: https://humansystems.arc.nasa.gov/publications/Kanki_Brasil_Analysis_Ramp_Damage.pdf
29. Patch Repair. (2012). [image] Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf
30. Approach Velocity of Ground Vehicle (2012). [image] Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf

31. Initial Design Lay-up of Airbus A350 XWB. (2012). [image] Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf
32. Design Revision of Airbus A350 XWB. (2012). [image] Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf
33. Airbus A350 Titanium usage. (2012). [image] Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf
34. Omega-profile stringers used in Airbus A350 XWB. (2012). [image] Available at: https://www.easa.europa.eu/sites/default/files/dfu/CODAMEIN_Report_20120312_highres.pdf

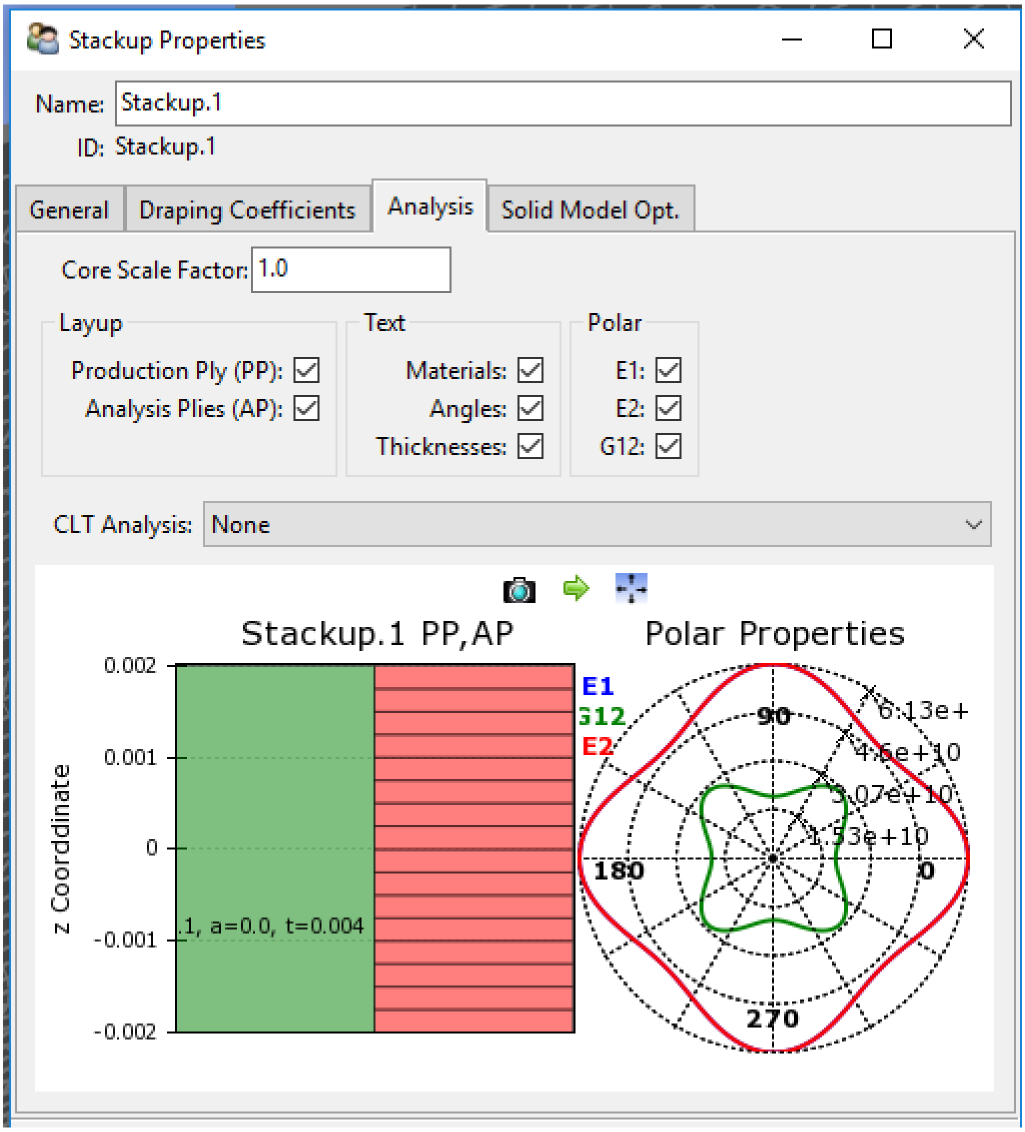
Appendices

Appendix I

The following table shows the physical properties of Flat Plate Composite structure with cut-outs.

Properties	
Volume	4.0628e+007 mm^3
Mass	57.692 kg
Surface Area(approx.)	1.6281e+006 mm^2

The below illustrated image is the representation of the Stackup properties of ACP module.



The following figure shows the Mechanical Properties of the chosen composite material.