Abstract

This thesis focuses on the subject of damage in composite materials and structures, in particular delaminations arising from an impact event and subsequent Mode I and Mode II loading and fatigue delamination growth. Interlaminar fracture toughness values have been calculated from an experimental study for DCB and ENF specimens. Specimens with artificial inserts at two different interfaces were used along with specimens with delaminations introduced from an impact event. The standard analysis method for both Mode I and Mode II has been adapted to account for the delamination away from the mid plane. For Mode I loading, the load to initiate delamination growth from experimental results is in good agreement with the predicted results from the adapted Mode I equation. For Mode II loading, crack migration did not appear obvious from the experimental study, and an adapted equation accounting for delaminations away from the mid plane has been successfully used.

A fatigue study on a structural element loaded both in-plane and out of plane has highlighted the complex nature of damage growth in composite structures. The study has highlighted the issues of delamination investigation using the ultrasonic NDT technique, whereby non-critical delamination growth is sometimes masked by the more dominant delamination and as such the complex growth of delaminations within a structure is difficult to quantify using this technique.
Dedication

To my wonderful husband Andy, who has supported me throughout my long study, making sure I persevered when life got in the way and keeping me well fed and watered at busy times of study.

To my two wonderful boys, Aidan and Finlay, without whom, I would have finished this thesis sooner, but who bring such joy to my life.

To my incredibly supportive parents, especially my Dad who inspired me to study engineering and sparked my interest in materials science and who has made me realise that we never stop learning, no matter how old we are!
Acknowledgements

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1. Introduction

Composite materials can be defined as materials comprising two or more constituents (chemically distinct) which, when combined, have properties which are greater than the properties of the individual constituents.

They occur naturally in forms such as bone and wood, however man-made composites were first used in aircraft construction in the 1960s and 1970s. The first generation of composites used in load-bearing safety critical aerostructures employed low toughness epoxy (thermoset) resin systems which resulted in laminated structures with a poor tolerance to low-energy impact. The low toughness of early composite aerostructures, combined with high probability of impact from threats such as runway debris lofted by aircraft wheels, or tool impacts during maintenance, necessitated overly conservative design and aircraft structures which were far heavier than theoretically possible. Whilst maintaining a focus on mass (and cost) reduction it therefore became paramount for material suppliers to develop composites with higher impact resistance and damage tolerance, as well as delivering a balanced set of other mechanical and physical properties, such as compression strength and wet Tg, and resistance to the harsh environmental conditions present in aerospace applications.

As the performance of composite materials, and associated design and test methodologies, have been improved so the use of composites within the aircraft industry has also increased. The A380 utilises 30 metric tonnes of structural composites in a total airframe weight of 170 metric tonnes (16%), whereas the A350XWB utilises 53% of composites in its airframe [Airbus (2015)]. Figure 1 illustrates the growth in the use of composites in Airbus and Boeing aircraft since 2009; showing an annual growth rate of over 10% per year.
The mechanisms involved in the initiation and growth of damage in composites are complex; this is due to the architectural complexity and intrinsic inhomogeneity of composites, as well as their anisotropic nature. Bulk structural failure of composites occurs due to a summation of a number of damage mechanisms, including matrix cracking, delamination, fibre / matrix de-bonding and fibre failure. The complexity of this failure process causes difficulty in the pursuit of robust numerical prediction techniques for use in safety critical aerostructures design, especially when compared to more traditional metallic materials. As a result, although a vast amount of academic and industrial studies have been published that aim to deliver robust composite failure predictions, designers are still forced to use a very conservative approach to design of composites, by regulatory authorities such as EASA (European Aviation Safety Authority). For civil aircraft, generally a safe life approach is adopted, in which the component has to remain defect free for the lifetime of the component, or more precisely that any non-detectable defect that exists, or may be undetectably formed whilst in service, must exist throughout the components life without growing (Attia et al (2001)). As a consequence, when designing with composites, a working strain of approximately $2700 \mu \varepsilon$ (design-limit strain...
level of 4000με is employed (Attia et al (2001)). The limit strain is the maximum strain that the structure is likely to experience during its design life, without experiencing any permanent deformation. The working strain is set by factors such as hot/wet notched compressive strength and static compressive strength after impact. At these low levels of working strain, composites have very good fatigue properties, allowing designs to be statically determined. In fact, at these low strain levels, any damage that is initiated does not usually grow under subsequent fatigue loading, therefore meeting the safe life design approach. The low strain levels used to date has led to the assumption that composites are highly fatigue damage tolerant. However, in order to meet the increasing demand for higher fuel efficiency, and as confidence in design increases, the overall structural weight of aerospace structures is likely to be reduced, with an associated reduction in thickness of some of the load bearing structures becoming a necessity. Reducing the thickness of such structures may lead to an increase in the operational strain, and as a consequence, fatigue may then become a more significant issue.

There has been considerable research conducted on delamination growth and damage mechanisms arising from an impact event; however, the focus has been on coupon specimens, which do not accurately reflect real damage processes in real structures. Very little work has been carried out on more representative structural specimens. Structures generally fail due to either oversights during design, construction or operation of the structure, or due to application of a new design or material, which produced an unexpected result (Anderson (2005)). Regarding oversights during design, construction or operation, the existing procedures should be sufficient to avoid failure; however if the procedures are not followed, failure may occur. When a new design or new material is introduced, the problem is more complex, and there are often factors that the designer is not expecting. As a result, a new design or material should only be placed into service once extensive testing and analysis has been carried out. The expected service performance must be assessed before the structure enters service. This assessment is in the form of structural testing, which will ensure and substantiate structural integrity as per certification criteria for either civil or military requirements.
(Niu (1992)). The basic “building block” approach (Figure 2) which builds in complexity from coupon tests to full scale tests, should be established in the early stages of development because the validation process is dependent on testing of all levels of the fabrication process (Niu (1995)).

![Figure 2 – Building Block Testing Approach (Niu (1995))](image)

The purpose of structural testing of all levels is to establish failure modes, demonstrate compliance with design criteria and correlate test results with theoretical predictions, therefore ensuring confidence that the part or overall structure, will perform satisfactorily throughout its service life.

1.1. Aims

The overall aim of this research project was twofold, firstly to investigate and carry out novel experimental testing techniques to understand damage behaviour at both the coupon level and the structural level in order to compare the results with standard analysis. Secondly to determine if standard analysis can be used and adapted to generate interlaminar fracture data from delaminations more representative of those seen in real structures.

1.2. Structure of thesis

This document consists of 5 chapters. The first introduces the thesis and outlines the aims of the study. Chapter two presents the literature review, chapter three outlines the experimental test methods for Double Cantilever Beam (DCB) specimens and End Notch Flexure (ENF) specimens which
contained either artificial inserts at the mid plane of the laminate or at 1/3 thickness of the laminate, or delaminations caused by a 50J impact event.

Chapter four presents the work carried out on the structural testing. Structural testing has been carried out in the form of fatigue tests for a structural panel with impact damage, loading both in plane and out of plane in order to evaluate the damage growth. Bringing these two novel test experiments together should provide an overview of the nature of damage behaviour at both the coupon level (from a non-standard test) and the structural level.

Chapter five is a summary of the thesis and includes recommendations for future work.
2. Literature Review

This section of the thesis starts by discussing the fundamentals of composite materials and summarises the key work carried out in the field of damage of composite materials, focussing on delamination and impact damage along with fatigue failure.

2.1. Fundamental aspects of composites – the fibre, matrix and interface

A composite material can be defined as a material comprising two or more constituents (chemically distinct) which, when combined, have properties which are greater than the properties of the individual constituents. Many composites are composed of just two phases, a matrix phase and a reinforcement phase. The properties of the composite are a function of the constituent phases, their type, amount and the geometry of the reinforcement. A flow chart (Figure 3) displays the classification of composites in terms of the reinforcement type (Matthews et al. (1995)).

![Flow Chart of Composite Classification](image)

**Figure 3 – Classification of reinforced polymer composite materials**

Within the three main headings; particulate, fibre and structural, of greatest interest to this study are the fibre and structural classifications, due to their relevance to the aeronautical industry.
2.1.1. The fibre

For the purpose of this thesis, when referring to the fibres, it is continuous fibres that are being discussed. The fibres perform the load carrying task and are primarily responsible for the structural properties, such as tensile and flexural strength and stiffness. Generally, the smaller the diameter the fibre, the greater the strength of the fibre, due to the larger surface area provided for the interface bond with the composite matrix (Matthews et al, (1995)). The fibres occupy the largest volume fraction in a composite laminate, with a typical aerospace fibre volume fraction being 65% to achieve the best balance of mechanical properties. The principal fibre materials of interest in aerospace components are glass, carbon or aramid fibres, however recently natural fibres such as flax, hemp, and jute are gaining increased importance in industrial applications due to their competitive specific tensile properties (Anandjiwala R (2015)). Glass fibres are low cost fibres which have good strength. They are typically only used for aircraft components that do not have to carry heavy loads or operate under large stresses, due to their relatively low tensile modulus, relatively low fatigue resistance and sensitivity to abrasion during handling (Mallick (2008)). Typical uses for glass fibres are for fuselage interior components as well as for wing fairings and wing fixed trailing edge panels. Aramid fibres have high toughness and energy absorbing capacity, tensile strength and stiffness with low density; however, the low compressive properties, as a result of fibre buckling at a low percentage of the tensile load, as well as poor environmental performance, has limited the use of aramids for major structural applications. When carbon fibres are combined with polymer matrix materials to create carbon fibre reinforced polymer composites the resulting composites are among the strongest and stiffest materials, which has led to their increasing use for high performance structures. With proper selection of carbon fibres and matrices, structures manufactured from composites can achieve greater strength and stiffness than equivalent metallic parts. Analysis by Mallik (2008) revealed that, an overall weight saving of approximately 40% was achieved in early application of composites compared to metallic (Table 1 and Table 2).
<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Component</th>
<th>Material</th>
<th>Overall weight saving over metal component (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>F-14 (1969)</td>
<td>Skin on the horizontal stabiliser box</td>
<td>Boron fibre - epoxy</td>
<td>19</td>
</tr>
<tr>
<td>F - 11</td>
<td>Under the wing fairings</td>
<td>Carbon- fibre - epoxy</td>
<td></td>
</tr>
<tr>
<td>F -16 (1977)</td>
<td>Skins on vertical fin box, fin leading edge</td>
<td>Carbon fibre - epoxy</td>
<td>23</td>
</tr>
<tr>
<td>F /A – 18 (1978)</td>
<td>Wing skins, horizontal and vertical tail</td>
<td>Carbon fibre - epoxy</td>
<td>35</td>
</tr>
<tr>
<td></td>
<td>boxes, wing and tail control surfaces, etc</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>horizontal stabiliser, flaps, ailerons</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

*Table 1 – Early applications of fibre-reinforced polymers in Military aircraft (Mallick (2008))*
<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Component</th>
<th>Weight reduction (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Boeing</td>
<td></td>
<td></td>
</tr>
<tr>
<td>727</td>
<td>Elevator face sheets</td>
<td>25</td>
</tr>
<tr>
<td>737</td>
<td>Horizontal stabiliser</td>
<td>22</td>
</tr>
<tr>
<td>737</td>
<td>Wing spoilers</td>
<td>37</td>
</tr>
<tr>
<td>756</td>
<td>Ailerons, rudders, elevators, fairings, etc</td>
<td>31</td>
</tr>
<tr>
<td>McDonnell-Douglas</td>
<td></td>
<td></td>
</tr>
<tr>
<td>DC-10</td>
<td>Upper rudder</td>
<td>26</td>
</tr>
<tr>
<td>DC-10</td>
<td>Vertical stabiliser</td>
<td>17</td>
</tr>
<tr>
<td>Lockheed</td>
<td></td>
<td></td>
</tr>
<tr>
<td>L-1011</td>
<td>Aileron</td>
<td>23</td>
</tr>
<tr>
<td>L-1011</td>
<td>Vertical stabiliser</td>
<td>25</td>
</tr>
</tbody>
</table>

*Table 2 – Early applications of fibre-reinforced polymers in commercial aircraft (Mallick, (2008))*

Carbon fibres are classified into three categories, polyacrylonitrile (PAN), pitch and rayon based fibres, which relate to the precursor material. Of these three, the PAN derived fibres are of greatest interest in aerospace applications because they offer the highest strength and best balance of mechanical properties. These fibres are available in standard (230GPa), intermediate (290GPa), high modulus (380GPa) and ultra high modulus (440GPa) grades (NetComposites (2016)). The standard modulus fibres, such as T300 and HTA, have the highest tensile strength, whereas the high modulus fibres have the lowest tensile strength, as well as lowest tensile strain to failure. The intermediate modulus, high strength fibres, such as T800 and IMS, have the highest strain to failure and are the most widely used fibres for primary structures due to the balance of good stiffness and strength.
2.1.2. Continuous reinforcing fibre

A wide range of continuous reinforcing fibre architectures are available for composites. Most forms are available either dry or pre-impregnated with the desired matrix. Pre-impregnated materials are traditionally known as “prepregs”. Dry fibre architectures require a method of applying the matrix during the lay-up process, typically using some form of liquid resin infusion techniques (e.g. by resin transfer moulding (RTM), or wet hand lay-up). One advantage of woven fibre reinforcements, or fabrics for reinforcement purposes is their ability to drape (conform) to curved surfaces without wrinkling. Dry fabrics can also be assembled using stitching techniques to produce fibre preforms tailored to the shape of the eventual component. Also the development and maturity of 3D woven textiles is accelerating rapidly. The toughness of simple unidirectional layered reinforcements can be improved when deployed in the form of a stitched non-crimped fabric (NCF) (Soutis (2005)). The prepreg form of NCF requires no additional matrix application, because the material is supplied from the manufacturer with the matrix preimpregnated, helping to maintain fibre alignment and allowing the material to be handled easily, in turn improving ease of manufacture.

Prepregs are available in a variety of forms; unidirectional thin layers (plies) of continuous fibres, typically with cured ply thickness of 0.127mm to 0.254mm and 2-D bidirectional fabric, typically 0.254mm to 0.508mm cured ply thickness. Prepreg materials remain the preferred choice of material form for primary structural applications due to enhanced control of the fibre and matrix spatial distribution and proportions (volume fraction), which in turn leads to a greater control over the mechanical properties compared to dry fibre reinforcements. In addition, the autoclave based manufacturing of prepregs generally creates structural composites of higher quality than those manufactured using resin infusion techniques. When manufacturing from prepreg materials, layers (plies) of the prepreg are stacked on top of one another in predetermined directions, until the desired laminate is produced. In order for the best combination of properties in all directions to be achieved, a quasi-isotropic lay-up may often be chosen.
The typical convention for describing the stacking sequence of a quasi-isotropic lay-up is as follows; \([([+45, 0, -45, 90]_s)^4]s\). In total within this lay-up there would be eight off +45° plies, eight off 0° plies, eight off -45° plies and eight off 90° plies. This lay-up is a balanced and symmetrical lay-up with 90 degree fibres at the mid plane.

Prepregs require curing to create structurally useful materials, which typically takes place in an autoclave at elevated temperature and pressure. In order to reduce energy costs, out-of-autoclave prepregs are becoming available, although currently their properties remain slightly lower than those cured in an autoclave. Out of autoclave resin infusion manufacturing techniques also offer lower cost production opportunities compared with autoclaves, but similarly the mechanical properties of the materials tend to be lower. Despite this many primary aerospace structural components are now manufactured using out of autoclave techniques. One example, the Airbus A330-300 spoiler assembly, manufactured by FACC using NCF and RTM is shown below in Figure 4.

![A330-300 Spoiler manufactured using NCF and RTM.](http://www.compositesworld.com/articles/composite-spoilers-brake-airbus-for-landing)
2.1.3. Matrix

The matrix (typically a polymer resin) provides support and alignment of the fibres, as well as acting as a load transfer medium into and out of the composite. The matrix only plays a minor role in the tensile load carrying capability of a structure, but has a major influence on the compressive strength, due to it providing lateral support against fibre buckling. The matrix also has an influence on the interlaminar shear strength; an important factor for components under bending loads, as well as in-plane shear properties (important for structures under torsional loads). The matrix also provides a barrier to impact damage, corrosion, moisture and elevated temperatures. The choice of fibre and matrix is extremely important when designing a structural composite component because the fibre, matrix and their interfaces dictate the final properties of the component (Mallick (2008)).

Typically, there are two classes of polymer matrices when considering aerospace structures; thermosetting and thermoplastic. Thermosetting resins solidify by forming chemical covalent cross-links to form a tightly bound three-dimensional network. In order to achieve optimum cross-linking and hence mechanical properties, it is necessary to cure the composite at one or more elevated temperatures for a pre-determined length of time, often with the application of pressure. The conventional epoxy aerospace resins are designed to cure at 120°C – 135°C or 180°C, in an autoclave at pressures up to 8 bar, occasionally with a post cure at higher temperatures (Soutis (2005)). Thermoplastics are not cross-linked, and instead of the strong covalent bonds seen in thermosets, have weaker Van der Waals bonds. On application of heat, the chains within the thermoplastic can flow past each other and the material changes from a rigid solid to a viscous liquid. Upon cooling the thermoplastic solidifies, and takes the desired shape (Matthews et al, (1995)).

The mechanical behaviour of thermoset composites and thermoplastic composites is different. Thermoplastics tend to have high strain to failure, good chemical resistance and generally good thermal stability, however,
the properties of thermoplastics are heavily dependent on both the temperature and strain rate of the test, and under constant applied load they will creep. Although the toughness of epoxy systems has improved since the first generation of composites were introduced to aircraft components in the 1960’s, they are still not as damage tolerant as thermoplastic materials (Soutis (2005)). Overall, due to their balanced set of properties (strength, toughness, cost and processability) thermosetting epoxy resins, are the most heavily utilised class of matrices used in aerospace structural composite applications.

2.1.4. Fibre / matrix interface

Improvements in fibre and matrix properties have been made over the years, with resulting improvements in laminate properties, however, these properties are heavily dependent upon the strength of the interface bond between the fibre and the matrix. The load acting on the matrix has to be transferred to the reinforcement via the interface, therefore the fibres must be strongly bonded to the matrix if their high stiffness and strength are to be imparted to the laminate (Soutis (2005)). The translaminar fracture behaviour is also affected by the interface bond, with a strong bond resulting in high stiffness and strength but often a low resistance to fracture (brittle behaviour), whereas a weak bond results in low stiffness and strength but a high resistance to fracture. It is therefore important to select the material which meets the design requirement, e.g. high strength or high fracture resistance (Soutis (2005)).

2.2. Damage

2.2.1. Definition

Damage is a very broad term, and this section of the report has been divided into sub sections which all relate to damage in composite materials. This includes the causes of damage such as impact and fatigue, through to the damage mechanisms exhibited by composite materials.
The majority of the work reviewed under this section has considered continuous fibre reinforced composites, because of their growing use in civil and military aerostructures, where the consequences of damage, specifically from an impact, are likely to be severe.

Damage in composites can be classified as initial / inherent flaws and in-service damage. Initial / inherent damage can be considered as damage that occurs during manufacture, such as contaminants, voids, porosity, de-bonding, fibre breakage, non-uniform fibre and matrix distribution, fibre misalignment, foreign inclusions, ply gaps, embedded defects, poor wetting of the fibres (leading to poor fibre-matrix bonding), or poor consolidation of the composite laminate, leading to air pockets. Typical in-service damage includes damage introduced during routine inspections, such as dropping of tools, along with damage resulting from thermal effects, such as exposure to temperature above the glass transition temperature (Tg), as well as foreign object damage (FOD) such as bird strike or runway debris lofted by aircraft tyres. Damage can also occur due to fluctuating loads, at levels lower than the ultimate stress of the material.

The most common failure of composites in aerospace arises from the damage resulting from an impact event, such as runway debris, bird strike, hail or ice impact. The effect of likely initial and in-service damage, and its effect on the residual performance of the component, must be considered in a damage tolerant design.

2.2.2. Impact damage – a major challenge for composites

Impact damage in aircraft composite structures can arise due to a variety of events, as discussed above. The damage resulting from these unexpected loads may remain undetected, in the form of barely visible impact damage (BVID), which could severely reduce the structural integrity of the structure. For this reason, current composite aerostructures must be designed so that non-detectable damage (BVID) can exist in the structure throughout its service life with no further growth.
When discussing impact damage, there are two terms that are of importance; impact damage tolerance, which defines the materials ability to withstand existing damage, and impact resistance which is the ability of the material to resist the formation of damage during impact.

The geometry of composite aerostructures is often typified by thin plate like shapes, which are susceptible to out-of-plane loading and therefore, their impact resistance is poor. To illustrate this Cantwell and Morton (1985) showed that impact energies as low as 4J were sufficient to reduce the load bearing capacity of a composite coupon by over 50%.

Impacts are generally classified as high velocity or low velocity although there is some confusion as to what these terms actually mean. Sjoblom et al (1988) and Shivakumar et al (1985), defined low-velocity impact as events which can be treated as quasi-static, the upper limit of which can vary from one to tens of meters per second depending upon the material stiffness and other material properties, as well as the impactor’s mass and stiffness and the boundary conditions. The researchers’ above defined high-velocity impact response as being dominated by stress wave propagation through the material, in which the structure does not have time to respond, leading to very localised damage. In this instance boundary condition effects are reduced because the impact event is over before the stress waves have reached the edge of the structure.

Cantwell and Morton (1991) used a rather different approach to classify the two impact classifications. In their work, impact techniques were used to define the classifications, with low velocity impact being defined as those in which instrumented falling weight impact testing, Charpy or Izod techniques are used, with velocities being typically 10 m/s. However, Abrate (1991) stated that low velocity impacts are in the order of 100 m/s and below.

A number of researchers’ (Joshi and Sun (1987) and Liu and Malvem (1987)), have, as an alternative approach, used the damage mechanism incurred in the composite laminate as a reference for the type of impact.
They defined low velocity impact as that in which delamination and matrix cracking is involved whereas high velocity impact has penetration induced fibre fracture. This appears to be a rather unusual method of defining the impact event because, for example, laminate thickness and toughness may have a dominating effect on damage mechanism formation rather than impact velocity. For this reason this approach will not be used within this particular study.

2.2.3. **Impact testing**

Ideally the impact test should represent, as near as possible, the loading conditions which a composite component is subjected to in service, and hence the failure mechanisms should be representative.

Generally, when characterising a material in terms of its impact resistance or impact damage tolerance for low velocity impact, a drop weight impact test is carried out. This involves a tower with rails supporting a known weight, which is dropped from a known height to impact the test specimen supported in the horizontal plane. By knowing the acceleration due to gravity, the engineer will know that the weight falling from the set height will contain a certain amount of impact energy at the point of impact. The material of interest is clamped / supported and the weight, attached to an impactor tup (striker) will hit the specimen and rebound or completely penetrate the specimen. The height of the rebound will be lower than the original height and the engineer can use the rebound height to calculate the energy that was absorbed during impact.

In more advanced testing, instrumented drop towers are used. In these instances, a piezo-electric load cell is attached to the falling weight and a data acquisition system interrogates the load cell over a given time interval during the impact event. Integration methods are then used to calculate specimen deflection and energy absorption from the measured force-time data.
2.3. The nature of impact damage

Many researchers have documented that damage initiation is shown on the load-time history from the impact test as a sudden load drop, due to loss of stiffness from unstable damage development. Following the drop in load, damage growth will stop, the composite laminate will be reloaded, and a cycle of damage propagation and arrest occurs until the impactor begins to rebound and the laminate is unloaded (Lee and Zahuta (1991) and Zhang (1998)).

Choi and Chang (1992) investigated the mechanisms of damage initiation and development in a carbon fibre reinforced epoxy laminate subjected to low velocity impact loading. They showed that there exists an impact velocity threshold below which no delamination occurs but above which significant damage is incurred. They also showed that matrix cracking represents the initial failure mode in these composites, a failure mechanism that subsequently triggers delamination at neighbouring interfaces.

Richardson and Wisheart (1996), and Abrate (1991, 1994 and 1998) both carried out comprehensive reviews on the low velocity impact properties of composite materials, summarising the work carried out by numerous researchers, most being concerned with low velocity impact and both the damage tolerance and damage resistance aspects of it. Richardson and Wisheart (1996), stated that the heterogeneous and anisotropic nature of fibre reinforced plastic (FRP) laminates gives rise to four major modes of failure; matrix mode (cracking occurring parallel to the fibres due to tension, compression or shear), delamination mode (produced by interlaminar stresses), fibre failure mode (in tension, fibre breakage and in compression, fibre buckling) and penetration (the impactor completely penetrates the laminate). Liu and Malvem (1987) reported that in order to understand damage initiation and propagation, it is not only the damage mechanism that is of importance, but also the interactions between all the failure modes. As can be seen, the majority of work reviewed dates from the 1990’s; this is due to the fact that a lot of work was carried out during
this period in order to gain a fundamental understanding of the formation of impact damage. Having reviewed the current literature in this field, it relates primarily to non-standard material types, such as a study by Sarasini et al (2016) on low velocity impact damage of carbon/flax hybrid composites, in which the natural flax fibres on the outside of a laminate guaranteed a higher impact damage tolerance, acting as hindrance to crack propagation in the laminate. A study by Selver et al (2015) investigated impact damage of non-crimp laminates, finding they absorbed more energy during low velocity impact compared to woven laminates, possibly due to extensive tow-level delaminations. On the other hand, a much larger dent depth was observed in the woven laminate after low energy impact. A recent study (Vieille et al, 2013), showed that carbon fabric laminates with different thermoplastic resins (PEEK and PPS) provided smaller delaminated areas than laminates with epoxy resin after low velocity impact tests; this result is due to tougher matrix system in thermoplastic composites. Olsson (2000) stated that the impact response of plates is governed by the impact / plate mass ratio, where small mass impactors (impactors weighing ¼ of the plate) result in a local response controlled by wave propagation with small deflections, larger impact loads and significantly larger damage for a given impact energy, whereas large mass impactors result in a quasi-static response.

2.3.1. Matrix failure

Richardson and Wisheart (1996) stated that matrix damage is the first type of failure induced by transverse low-velocity impact, and usually takes the form of matrix cracking as well as de-bonding between fibre and matrix. Matrix cracks are thought to occur due to property mismatching between the fibre and matrix, and are usually oriented in planes parallel to the fibre direction in unidirectional layers. A typical crack and delamination pattern was shown by Joshi and Sun (1985), Figure 5.
The matrix cracks in the upper layers (Figure 5a) and the middle layer (Figure 5b) start under the edges of the impactor. The shear cracks are a result of the very high transverse shear stress through the material, and are inclined at approximately 45°. The transverse shear stresses are related to the contact force and contact area (Choi et al., 1991). Richardson and Wisheart (1996) and Lee and Sun (1993), stated that the crack on the bottom layer of Figure 5a is termed a bending crack due to the fact that it is caused by high tensile bending stresses and is characteristically vertical.

Cantwell and Morton (1989) in their report on geometrical effects emphasised that the type of matrix cracking that occurs is dependent on the global structure of the impacted specimens. They reported that for long thin specimens, bending cracks in the lower layers occur due to excessive transverse deflection and subsequent membrane effects predominate, whereas short, thick specimens are stiffer and as a result higher peak contact forces induce transverse shear cracks under the impactor in the upper plies. An additional report by Cantwell and Morton (1990) stated that for thicker laminates, fracture usually initiated at the upper surface of the target, whereas initial failure in thinner laminates tended to occur in the lowermost ply, directly under the point of impact.

The review by Richardson and Wisheart (1996), highlighted the efforts from Chang, Choi and co-workers (1991, 1990, 1992, 1987), who concluded that the bending crack in the 90 degree layer is a result of stresses ($\sigma_{11}$, $\sigma_{13}$ and $\sigma_{33}$) for line loading impact damage. It was also concluded from their
research that $\sigma_{33}$ is very small in comparison to $\sigma_{13}$ and $\sigma_{11}$ during the impact event.

Sjoblem et al (1988) stated that the presence of matrix cracks do not dramatically affect the overall laminate stiffness during an impact event, whereas other forms of damage reduce the laminate stiffness as reported later.

2.3.2. Delamination

A delamination, or interfacial crack, is a crack that initiates and grows between the different plies of a composite material in the resin rich area. Cui and Wisnom (1993), Wang (1979), Wu and Springer, (1988) and Abrate (1991), all defined delaminations as a crack between plies of different fibre orientation and not between laminae in the same ply group. Liu and Malvem (1987), concluded that delamination was a result of the bending stiffness mismatch between adjacent layers, i.e. the different fibre orientations between the layers, a similar finding to Cui and Wisnom (1993), Wang (1979) and Wu (1988). However Hosur et al (1998) do not agree with this statement, along with others, (Preuss and Clark (1988), Kaczmerek (1995) and Smith et al (1989)) who have all reported that significant damage occurs between layers with the same fibre orientation. However, the severity of the delaminations depends upon the difference in the ply angles above and below the interface.

Delaminations can be classified into two types, interlayer and intralayer. An interlayer delamination is a crack that grows in the interface of two plies, without breaking the two plies, resulting in a crack following the same direction (Figure 6).
Figure 6 – Schematic showing interlayer and intralayer delaminations (Stratton and Pelegri (1999))

An intralayer delamination is a crack that grows in the interface but occasionally "jumps" to a neighbouring interface. As a result of the crack migration, it could fracture one, or more, of the plies and it also changes orientation (Figure 6).

In Liu and Malvem’s (1987) experimental work, it was discovered that delamination areas were generally oblong shaped, with their major axis being coincident with the fibre orientation of the layer below the interface. For 0/90 laminates, the shape became peanut shaped. These characteristics have been widely reported elsewhere (Joshi (1985), Wu (1988) and Chang et al, (1990)). In fact, it has been stated by numerous researchers, including Liu and Malvem (1987), that the stacking sequence 0/90 will cause the most detrimental effect and have the largest delamination when subjected to uniaxial loading only. This can be attributed to a greater stiffness mismatch, which links to different stresses being produced between the plies, leading to delaminations, compared to laminates with say a 0/+45 layup. The thickness of the laminate has also been reported to affect the delamination size (Finn et al, (1993)). Delamination is more likely to occur when the span lengths are small and when the laminate is thick, with low interlaminar shear, with Dorey (1986, 1987, 1988) deriving an equation (Equation 2.3.2-1) for the elastic strain energy absorbed at the point of delamination failure.
\[ \text{Energy} = \frac{2\tau^2 w L^3}{9E_f t} \]

*Equation 2.3.2-1*

where \( t \) = thickness, \( \tau \) = interlaminar shear strength (ILSS), \( w \) = width, \( L \) = unsupported length and \( E_f \) = flexural modulus.

Chang *et al* (1990), concluded that delamination was initiated as a mode I process, due to very high out-of-plane normal stresses caused by the presence of matrix cracks, and high interlaminar shear stresses along the interface. Liu *et al* (1993), from a fracture mechanics based analytical model, showed that both bending cracks and shear cracks could initiate delamination, but that delamination induced by shear cracks is unstable and that bending crack induced delaminations grow in a stable manner, proportional to the applied load. Choi and Chang (1992) reported that delamination growth was governed by interlaminar longitudinal shear stress (\( \sigma_{13} \)) and transverse in-plane stress (\( \sigma_{22} \)) in the layer below the delaminated interface and by interlaminar transverse shear stress (\( \sigma_{23} \)) in the layer above the interface. Razi and Kobayshi (1993), carried out a numerical simulation of impact induced delamination growth, concluding that mode II was the dominant failure mode for propagation.

An interesting paper by Wagih *et al* (2016) outlined the complex nature of delamination within composites after a quasi static impact event. The paper focussed on non-crimp fabric, thin-ply laminates (TP) and ultra thin ply (UTP) laminates, as these are reported to have benefits in terms of reducing intra-laminar, inter-laminar and splitting damage (Wagih *et al* (2016)). The authors combined microscope photographs and C-Scan results with the load displacement curves from a drop weight impact test and concluded that the complex mechanisms of the penetration process could be summarised in five stages. Stage I corresponded to the elastic response of the material, stage II, damage initiation where for the damage initiation was found to be dependent upon the thickness of the ply, thinner plies (UTP) showing delaminations as the first damage whereas slightly thicker plies (TP) showing matrix cracking. Delamination propagation was linked to stage III and stage IV showed fibre breakage with a large drop in the load-
displacement curves. Stage V was where fibres sheared out for TP laminates whereas for the UTP laminates, the fibres failed under tensile and compressive stresses. The damage onset was seen to occur earlier for the TP laminates compared to UTP, however fibre breakage occurred earlier in the UTP laminates. It was shown from this study (Waghir et al (2016)), that UTP laminates show better damage resistance than TP laminates however the maximum load capacity is larger for TP laminates.

An FEA study by Craven et al (2010) successfully modelled multiple delaminations (one at each ply interface) with realistic delamination shapes (peanut shaped as opposed to elliptical or circular) to determine the effect of delamination size and shape on buckling load. The boundary conditions applied to the model simulated compression after impact conditions with global buckling of the plate being prevented and only local buckling of the damage region allowed. The plots for the impact damaged regions showed linear behaviour up to buckling and then non-linear behaviour after buckling occurred. The authors Craven et al (2010) concluded that the peanut shaped delaminations buckle at a much lower strain and have a much lower stiffness compared to elliptical models. At 1% applied compressive strain the peanut shaped delamination showed a 65% stiffness reduction compared with the undamaged material. The effect of delamination size was also investigated in the study (Craven et al (2010)), concluding that the buckling of the damage region starts at increasing levels of strain the small the diameter of damage and the larger the delamination, the lower the residual stiffness compared with the undamaged material. This is thought to be due to the onset of global buckling buckling of the impact damage region occurring at lower strains for larger delaminations.

### 2.3.3. Fibre Failure

Fibre failure occurs later in the fracture process than matrix cracking and delamination. Fracture of the fibres typically only occurs with high energy impact, and when this fracture mode is present, it occurs directly under the impactor due to local stresses and indentation effects governed by shear
forces, and on the opposite face to the impact face, due to bending stresses. Dorey (1988) devised an equation (Equation 2.3.3-1) that relates the energy required for fibre failure due to back face surface flexure.

\[
\text{Energy} = \frac{\sigma^2 wtL}{18E_f}
\]

Equation 2.3.3-1

Where \(\sigma\) is the flexural strength, \(E_f\) is the flexural modulus, \(w\) is the width, \(L\) is the unsupported length and \(t\) is the specimen thickness.

The largest damage feature is usually delamination (Olsson et al (2003), which may cause significant reductions in flexural stiffness and buckling loads. However, studies demonstrate that impact damage zones frequently contain a smaller central region with fibre fracture (Sjogren, 1999). The resulting stress concentrations may cause premature failure both in tension and compression. Although the effects of delaminations has been a subject of extensive research (some of which has been outlined in section 2.3.2, methods for predicting delamination growth from an impact are still immature and are usually based on “equivalent” single delaminations neglecting the influence of fibre damage. An alternative approach used in a number of studies is to apply theories for notch failure and represent the impact damage by an “equivalent hole” or by a “soft inclusion”, an approach that Olsson et al (2003) state may be valid prior to local or global buckling of laminates with significant fibre damage.

### 2.3.4. Penetration

Penetration of the impactor into the laminate occurs when fibre fracture reaches a critical level. Cantwell and Morton (1989) showed that the impact penetration energy threshold increases with laminate thickness for carbon fibre reinforced composites. They also reported that the magnitude of the perforation threshold energy depends upon a large number of geometrical, projectile and material parameters, as well as on the incident velocity of the
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impinging projectile. Tracy et al (1985) reported that the indentation caused by the impact projectile absorbs a significant portion of the impact energy, this finding has been questioned by other authors, and therefore this area requires further investigation.

### 2.3.5. Factors affecting impact damage in composite materials

Richardson and Wisheart (1996) reported that damage due to impact loading in a specific composite structure is a function of the velocity, mass, modulus and shape of the impactor. Abrate (2001) stated that some impacts produce deformation in a small zone surrounding the point of impact, while others involve deformations of the entire structure. The most relevant parameter, in terms of resistance to low velocity impact, is the ability of the fibres to store energy elastically. This parameter can be determined by calculating the area under the stress-strain curve, which is governed by the fibre modulus and failure strain. Essentially, composites with large areas under the stress/strain curve are more effective energy absorbers, with materials containing fibres with a greater strain energy absorbing capacity offering improved impact energies (Cantwell and Morton (1991)). Cantwell and Morton also reported that the diameter of the fibre has a significant effect on the impact performance of the composite. The first generation carbon fibres, such as T300 and AS4 had diameters of 7-8 \( \mu \)m, whereas fibres, such as IM6, IM7 have diameters of approximately 5 \( \mu \)m, with an improved strain to failure, and hence an improved impact resistance (Cantwell and Morton (1991)).

The type of matrix also has a significant effect on the impact performance of the composite. In order to increase the toughness of materials (reduce the brittleness) developments in plasticising modifiers, a reduction in the cross linking density of epoxy resins and the inclusion of thin, tough layers at ply interfaces had to be made. Williams and Rhodes (1982) concluded that, for improved impact resistance, the strength of the matrix should exceed 69 MPa, and its strain to failure should exceed 4%.
The interphase region of composite materials also plays an important role in the damage mechanisms of composite materials. In general, improving the adhesion between the fibre and the matrix increases the incident impact energy required to initiate damage fourfold (Rogers et al (1971)). Bishop and Curtis (1983) drew attention to the possible improvement in impact tolerance of carbon fibre laminates, through the selective incorporation of woven fabric. Woven fabric was used to replace two opposing 45° plies, which normally provide sites for extensive delamination. This finding was also confirmed by Cantwell et al (1984).

2.3.6. Effect of geometry on impact damage

The majority of research that has been reviewed has been carried out on flat laminates, being either clamped or simply supported. By simplifying the geometry, structural effects are minimised and as a result more information on the material behaviour can be attained. However, in service, the structures that are likely to be subjected to an impact event will very rarely be a simple flat plate, and often will have structural additions such as stiffeners. Dorey (1986) investigated the impact response of stiffened panels, and reported that the energy to cause BVID dropped significantly near the stiffeners, where the structure was less compliant, and the stiffeners caused the damage to spread asymmetrically. Davies et al (1994) stated that impact forces will be higher in the stiffened regions, but that reduced deflections may lead to smaller strain and hence less strain induced failure. They also showed that at the edge of the stiffeners, delaminations were formed, whilst impacts directly over the stiffener caused debonding between plate and stiffener.

Stout et al (1999) reported that damage observed from an impact of an unsupported plate is very different from that observed in clamped plates. Within the unsupported structure, the damage was said to be a result of the elastic compressive and tensile waves generated by the impact event, rather than a structural bending.
2.4. Detection of damage in composites

2.4.1. Introduction

In order to gain a better understanding of damage mechanisms in novel coupons and structures, it is necessary to determine the extent of the damage that exists in the composite, and any subsequent growth of the pre-existing damage upon fatigue loading. The technology available for monitoring damage in composites has increased over the years, and this section of the review summarises the main methods.

The main types of non-destructive testing include tap testing, mechanical impedance, thermography, fibre bragg grating, laser shearography, x-ray radiography and IR thermography (Livingstone and Kilpatrick (1987), Henneke (1990), Bar-Cohen (1991), Burke et al (1994) Yang and He (2-16)).

2.4.2. Acoustic Impact

This is a simple technique which has been used for many years. Typically the technique is known as "coin tapping", the traditional method of testing aircraft flying surfaces. The vibrations are excited by use of a local impact (e.g. "tapping" with a hammer). The hammer is tapped onto the surface of the component by the tester, who then listens to the 'ringing' sounds generated. Any anomalies in the sound produced, due to the differences in the characteristic ringing sound are then recorded. When using an instrumented hammer to measure the force-time response, the results are more reliable, rather than relying on a subjective assessment of the audible noise. In the last few years, there have been rapid developments in this method, with automated computer based systems being used for the production-line testing of a variety of components.

Different defect types will cause different changes to the response of the structure. Instrumented acoustic impact systems have the potential to detect and characterise a variety of defects including; disbonds between the facesheet and honeycomb core, crushed core due to impact or overload, voids, inclusions and delaminations in composite repairs, core splice and
thickness change, potting, ply drop-off, doublers, ribs, spars location and composite repairs on fan cowling, flaps. The main application of this and other tap testing methods is in detection of disbonds and delaminations (Net Composites (2007)).

An interesting study by Bemment et al (2015), involved using acoustic emission sensors in order to reliably detect, locate and/or quantify the energy of impacts on a BAE systems HERTI UAV. It was found that using three sensors along the wingspan was enough to detect very low energy impacts (~0.5J), attribute them to a particular impact zone and indicate an estimate of the energy.

2.4.3. Laser Shearography

This technique is a non-contact technique that presents a visual qualitative map of the strain field of the surface of the structure in response to an applied stress. Subsurface features such as core splices, bulkheads and defective areas affect and distort the surface strain field and are therefore monitored. It is said that this technique is particularly well suited to detection of impact damage in CFRP components giving a unique defect signature and can even detect kissing bonds, which the majority of the other techniques can’t detect. Most shearographic images of CFRP defects appear and then stay uniform in size with respect to a constant stress (vacuum) or will reduce in size as the thermal loading effect dissipates in the material. With impact damage, the time taken for the thermal energy to penetrate, corresponds with the 'Xmas Tree' effect with the effect that defect images appear to enlarge with time (Net Composites (2007)).

2.4.4. Ultrasonic C-scan

In this technique components are placed in a bath of water and a combined emitter and receiver, piezoelectric transducer, also submerged in the bath, scans over the surface of the component. Ultrasonic pulses are retransmitted through the component and detected by the scanning head. The strength of the reflected ultrasonic pulse is then evaluated. If an ultrasonic pulse is completely reflected at an air gap, it cannot be
transmitted below it. Therefore, delaminations within laminates prevent the use of ultrasonic transmission due to air gaps (Saito and Kimpara (2006)).

There are two kinds of data processing in pulse echo C-scanning, echo amplitude (AMP) view and Time of Flight (TOF) view. The TOF view can be converted into a delamination depth view through the transverse sound velocity in CFRP. The TOF files together with image processing software allow the determination of the delaminated area (Net Composites (2007)).

Williams and Doll (1978) and Shoup et al (1982) were unable to detect fatigue damage in thin CFRP composites with through-transmission ultrasonics. However, Williams et al (1982), Bader and Boniface (1983), Kellas et al (1985), Nayeb-Hashemi et al (1986) and Scarponi and Briotti (1997) were all able, under certain fatigue conditions, to successfully detect the initiation and spread of fatigue cracks in CFRP’s with through-transmission ultrasonics. It was reported that small delamination cracks induced by fatigue loading could be detected, but only when the cracks grew in the direction transverse to the transmission path of the ultrasound waves. Other types of damage, such as fibre splitting (Kellas et al, 1985) or cracks parallel to the transmission path of the ultrasonic waves (Bader and Boniface, 1983, Cantwell and Morton (1985)) were harder to detect, this is thought to be because they do not offer a wide enough reflecting surface compared to delaminations. A number of researchers (Moran et al (1985), Wooh and Daniel (1990), Gorman (1991) and Steiner et al (1995), have shown that transverse cracks running parallel to the fibre direction can be detected, by orienting the transducer at an angle to the tested surface, so as to acquire the energy back scattered from damage. Kasap et al (1992) and Forsyth et al (1994) detected the debonding of fibres from the resin matrix as well as resin cracks by ultrasonics.

In a study by Hosur et al (1998) it was reported that due to the inherent inhomogeneous and orthotropic nature of composites, ultrasonic waves suffer high acoustic attenuation and scattering effect, therefore making data interpretation difficult. They did however suggest that these difficulties can be overcome by the proper selection of probe, probe parameter settings
like pulse width, pulse amplitude, pulse repetition rate, delay, blanking and gain and data processing including image processing. Within their study, they also reported that due to the inherent nature of delamination damage occurring at various interfaces throughout the thickness of the laminate, the normal practice of just determining the projected area is not sufficient, therefore a layerwise scanning technique, by locating the depth gate using time delay to determine the damage at each interface was adopted. A similar approach was also used by Moran et al (1988), whereby layer by layer images of delamination in an impacted composite panel were mapped, by utilising multiple gating, covering each interface, thereby collecting the information of all the interfaces in one scan.

2.4.5. Thermography

This is a technique based upon the analysis of thermal patterns, induced either by heating the specimen or by applying a mechanical oscillatory load. This technique is sensitive to delamination-type defects. The thermography NDE technique was used by Mitrevesi et al (2005) in order to determine the overall internal damage area and shape in impacted specimens. They reported that for specimens that contained permanent indentation, the thermo-scans were clear in depicting the damage area, however for BVID the results were not as clear. They did also state that although the technique is useful in determining the overall damage area induced by impacts, the technique cannot determine the various internal damage mechanisms present within the specimen. This technique is also unable to give information on the through thickness location of the flaw.

2.4.5.1. Selectively Heating Thermography

A study by Yang and He (2016) has shown that Infrared (IR) Thermography including pulsed thermography, lock in thermography and pulsed phase thermography shows great potential and advantages due to high inspection speeds, high resolution and sensitivity and detectability of inner defects due to heat conduction. Unlike ultrasonics, this method does not require a couplant and is a non-contact method. The principal of selective heating thermography involves inductive selective heating whereby an excitation signal is generated by the excitation module. A small period of high
frequency alternating current is driven to the inductive coil along the CFRP. The current induces eddy currents in the CFRP and heat is generated in the carbon fibres. The heat diffuses to surrounding non-conductive polymer matrix, and a time delay is seen until the heat is balanced in the CFRP. At the same time as induction heating, the temperature distribution on the surface of the CFRP is captured by IR camera (Yang and He, 2016).

2.4.6. X-radiography

This process involves the use of an X-ray opaque fluid to infiltrate the damaged area. Fracture is usually only detected when reasonably large and providing that some surface flaw is present to allow the infiltration of the enhancer. This technique is most suited to the detection of matrix cracks, where cracks frequently extend for 10mm or more. Detecting fibre fracture presents greater difficulty, since at low and intermediate impact energies this damage extends only a few millimetres and is frequently shielded by planes of delamination above and below the damage. The extent of delamination detection is poorer in X-radiography compared to the c-scanning technique, due to the difficulty in ensuring the complete penetration of the enhancing fluid and also diffraction of the x-rays (Cantwell and Morton (1985)). The technique can only detect damage at the surface, internal defects, impossible to fill with the dye, may remain undetected. When exact through the thickness positions of the defect is required, stereoscopic X-radiography techniques can be used. Two X-ray images are obtained from two different angles and then optically recombined to reconstruct a three dimensional view of the damage state. The interpretation of this technique is difficult, due to the difficulty in precisely locating the different delaminated and cracked layers (Aymerich and Meili (2000)).

2.4.6.1. X-ray microtomography (micro-CT)

A study by Symons (2000) outlines the x-ray microtomography method. This method is a non-destructive testing technique that allows 3-
dimensional imaging of the internal structure of a specimen. Symons (2000) states that it is a useful technique for investigating impact-damaged CFRP because it reveals the distribution of delaminations and matrix cracks without the need for physical sectioning and can provide clearer images of the distribution of damage than optical microscopy.

The method is basically a small scale form of CT scanning, used to create 2D and 3D X-ray attenuation maps of specimens up to a few centimetres in size, with a resolution typically of the order of tens of microns. For the X-ray energies used in CT, the dominant attenuation mechanism is Compton scattering, which is proportional to material density. Therefore, it is a reasonable approximation to use CT images as a density map of the object. The study by Symons (2000) concluded that a combination of ultrasonic C-scan and optical microscopy is a more flexible approach for rapid assessment. A more recent study by Penumadu et al (2016) has concluded that the ability to provide high and low resolution imaging non-invasively using x-ray tomography at multiple fields of view will be very useful for damage visualisation and quantification. Using selective ranges of attenuation, different damage features can be isolated in three dimensions.

A study by Schilling et al (2005) reported the benefit of using microtomography for fibre reinforced polymer composites. This method was reported to be able to characterise the internal geometry of flaws, including delamination, matrix cracking, and microcracking, in fibre-reinforced polymer laminates.

2.5. Fatigue

2.5.1. Introduction

The term fatigue, when relating to materials science, is used to describe the situation whereby under cyclic loading conditions, the load bearing capacity of the material falls with time, resulting in failures at stress levels below ordinary engineering static strength. The susceptibility or resistance of a given material to this cyclic loading is typically discussed in terms of the
number of cycles to failure, i.e., the life, \( N \), of the material under a given cyclic stress amplitude, \( S \), and is generally represented on an S-N curve.

Fatigue occurs generally in all engineering materials and there are common fatigue characteristics of all materials. The process of fatigue failures begins with microscopic cracks (often referred to as the initiation site), which, with each subsequent cycle, grow. This is a phenomenon that can be analysed by fracture mechanics, whereby the cracks grow under the influence of the fluctuating loads, until the cracks reach a critical length, exceeding the fracture toughness and hence failure occurs. The number of cycles required for failure can vary by, typically, a factor of ten, for nominally identical samples, due to an inherent statistical variability in fatigue lifetimes. However, in general, the greater the applied stress, the shorter the life. It is thought that unlike aluminium, composites exhibit an endurance limit or fatigue limit, a limit below which repeated stress does not induce failure, theoretically, for an infinite number of cycles of load.

Within metallic structures the stage of gradual and invisible degradation spans almost the lifetime and no significant reduction in stiffness is observed during the fatigue life. Although fibre-reinforced composites have a rather good rating in terms of fatigue life, the same cannot be said for the number of cycles to initiate damage or for the evolution of damage (Cavatorta et al (2006)). In composites, the damage growth often starts early on in the lifetime, typically during the first 10-20% of the life of the component.

2.5.2. Fatigue testing

There are various approaches for assessing composites in fatigue. Typically, S-N curves are developed (stress versus number of cycles to failure), if the number of cycles to failure at a specific stress amplitude is the parameter of interest. Interrupted fatigue testing is an alternative test regime in which a specimen (coupon or structure) is subjected to a predetermined number of cycles at a specified stress level and the damage within the specimen monitored through the use of non-destructive testing (NDT). Alternatively, a specimen could be subjected to a set number of fatigue cycles and then
a residual strength test carried out, to determine any degradation in the mechanical properties of the composite.

### 2.5.3. Damage in components under fatigue loading

The damage mechanisms during fatigue loading in multidirectional fibre reinforced polymer matrix composites are: matrix cracking, delamination between the plies and, finally, fibre fracture with associated fibre/matrix debonding. The first and most prolific form of damage observed is matrix cracking, which initiate and grow through the polymer matrix, transverse to the fibre direction in the longitudinal and off axis plies (Attia et al (2001)). Due to the mismatch between poisson’s ratios of adjacent plies, matrix cracking parallel to the fibres occurs in the 0° plies throughout the fatigue cyclic regime (Bailey et al (1979)). The primary factors which affect the fatigue damage mechanisms of composite materials are the type of matrix and fibre and the stacking sequence of the laminate, as these affect the strength and the stiffness. The mode of loading and the frequency of loading, along with environmental factors also influence the damage mechanisms and fatigue properties.

Numerous researchers (Reifsnider (1991), El Mahi et al (1995), Ogin et al (1985), Nairn and Hu (1992) and Tong et al (1997)) have stated that the importance of matrix cracking is that it is a principal source of stiffness reduction in composites, with it being the precursor of other damage mechanisms, such as delamination and fibre fracture. Degrieck and Paepegem (2001) agreed with this finding, stating that the damage developed early on in the fatigue life grows steadily, whilst the damage mechanism in the damage area may change. Within the damage areas, a loss of stiffness, accompanied by a gradual deterioration of the material, leading to a continuous redistribution of stress and a reduction of stress concentrations inside a structural component is reported to occur (Degrieck and Paepegem (2001)). Symons and Davis (2000) also reported that the progress of impact damage during fatigue loading involved both a gradual and larger discrete failure process. The gradual process involved a slow decrease in the stiffness of the impact damaged region and therefore a
reduction in the overall stiffness of the panel. The extent of the degradation in stiffness is dependent on a number of factors such as fibre and matrix type as well as the stacking sequence of the laminate.

It was observed by Symons and Davis (2000) that the back face blister created by the initial impact increased in size during the fatigue process. The large discrete events in the fatigue process were reported to include the growth of front face tensile cracks early on in the fatigue life, and delamination buckling of strips of surface ply around the damage zone.

Delamination usually initiates between plies of different orientations and grows perpendicular to the loading. Delamination occurs due to out of plane interlaminar shear as well as normal stresses which exist at free edges (Pagano and Byron (1971)). A study by Clarke (1989), reports that delaminations often occur in stacks with repeat intervals of four plies, when there is a four ply repeat in the laminate, indicating that the lay-up exerts a significant influence on damage distribution.

In most of the fatigue tests conducted by Symons and Davis (2000), rapid progression of the damage indicated that complete failure was imminent and was a result of fibre fracture. Debonding of the broken fibres, causes a load redistribution, leading to an increase in the probability of fracture in neighbouring fibres. Once a critical number of fibres have broken, fracture will occur. Jamison (1985) reported that a higher concentration of fibre breaks is observed at the intersection of transverse ply cracks with the longitudinal ply. Due to the fact that progression of damage indicates that failure is imminent, the initial onset of the growth of the impact damage under cyclic loading is of most interest, as opposed to predicting the steady, progressive growth of the damage with regards to fatigue damage prediction.

Although many researchers have concluded that there is a reduction in stiffness due to the fatigue failure process, the same cannot be said for the strength. The strength of some composite materials (e.g. glass fibre reinforced plastics (GFRP) and Aramid) is said to fall with number of cycles,
with failure occurring when the residual strength becomes equal to the applied cyclic stress (Adam et al (1986a)). This type of behaviour is termed ‘wear out’. However Carbon Fibre Reinforced Plastic (CFRP) laminates have not been reported to show this behaviour; instead a ‘sudden death’ situation occurs whereby instead of progressively decreasing the residual strength, a sudden decrease occurs very close to failure.

In a study by Pousartip et al (1986), the initial fatigue damage of CFRP laminates was delamination within which all the off axis plies were heavily cracked. The un-delaminated areas were reported to be lightly cracked (especially the 90° plies) but, by comparison, may be considered to retain their structural integrity. It was reported that, if the stresses are sufficiently low, then the first damage mechanism continues to completion, and further deterioration of the laminate was said to be due to fibre breakage and longitudinal splitting in the 0° plies. The axial stiffness of the laminate was reported to decrease linearly with increasing damage, with a 35% reduction in stiffness corresponding to complete delamination and cracking of the off axis plies.

For fatigue cycles, which incorporate a compressive load as part of the cycle, delaminations can exhibit local forms of buckling (Gaudenzi et al (1998)). For certain delamination sizes and locations along the laminate thickness, two main situations can arise. The first situation is where the thinner sub-laminate buckles locally while the thicker sub-laminate remain unaffected (Figure 7), within the second situation, both the sub-laminates buckle, exhibiting a sort of mixed buckling mode.

![Figure 7 – Schematic representing damage scenario – local buckling of the thinner sub laminate](image-url)
Such a mode can often be seen as a local mode with respect to the rest of the surrounding structure; however, it involves the whole delaminated area (Figure 8).

![Figure 8 – Schematic representing global buckling of the delaminated area](image)

The buckling of the thinner sub-laminate occurs first during the loading process, while the buckling in the thicker sub-laminate is dictated by the geometrical arrangements of the structure, such as delamination size and depth (Ricco and Gigliotti (2006)).

A more recent paper by Alderliesten (2015), outlined the improvement in damage tolerance of hybrid composite materials, in particular fibre metal laminate (FML) in relation to fatigue damage propagation, stating also their excellent resistance to impacts.

### 2.5.4. Delamination growth under fatigue loading

Delamination growth rate can be assumed to have three domains, sub-critical (slow), linear, and unstable growth. The growth rate depends on microscopic details of fibre architecture and resin properties in domain 1, on crack driving force (energy release rate) in domain 2, and on interlaminar fracture characteristics of the laminate in the unstable domain 3. Considerable research has been conducted on the delamination growth laws in domain 2, whereas very few studies have tried to model all three domains of delamination growth rate.

In typical aerospace applications, many structural components are curved, with tapered thickness containing plies with different orientations, which make delaminations grow with a mode mix that depends on the extent of
the crack; consequently delaminations generally grow in mixed mode (Blanco et al. (2004)). In the majority of studies investigating delamination growth, growth is investigated in mode I (opening), mode II (shear) as illustrated by (Figure 9) and mixed mode (mode I and II).

![Figure 9 - Basic fracture modes](image)

Typically, the mode III contribution in delamination growth is not considered as it is quite small for composite structures due to the constraints of adjacent plies, shown by Glaessgen et al. (2002) in laminated lap joints and by Jensen and Sheiman (2001) for laminated structures. Blanco et al. (2004) stated that delamination growth under fatigue loads in real composite components develop in a non-constant propagation mode.

Fatigue delamination growth is strongly affected by the relative location of the delamination through the plate thickness, the fatigue growth being slower when delaminations are located close to the impact surface. Delaminations located very near the impact surface will not grow, even after a very large number of applied compressive cycles. In a study by Saunders and Blaricum (1988), it was observed after C-scanning, that most of the delamination growth occurred in the ply layers close to the back face of the specimens, furthest away from the impact point, where the largest delaminations had been produced by the impact.

Jones et al. (1988) reported that compression-compression and tension-compression are the critical fatigue loading cases, with Ramkumar (1983) in agreement, stating that the least severe loading mode for growth of impact damage was tension-tension and most severe mode was compression-compression fatigue loading. Melin et al. (2002) also concluded
that the compressive part of the loading cycle was the most severe, and suggested that this was due to the compressive loading causing local buckling around the damaged region. Bennati and Valvo (2006) stated that under compression loading, the instability phenomena may promote further crack growth from a delamination and in some cases led to failure.

Mitrovic et al (1999) stated that impacted specimens subjected to fatigue loading need to have widths similar to the distance between two spars on an aircraft wing for example, if the results are to be readily transferable to in-service aircraft conditions. If the specimen width is too small, interaction between the impact damage and the edges might occur, with delaminations growing from the edge, which highlights the importance of geometry.

Flesher and Herakovich (2006) presented a method for predicting final failure of composite structures, and showed that delamination is the predominant mode of failure. Hayes and Lesko (2007) stated that failure of composites often occurs by buckling or delamination, as opposed to fibre failure, and commented on the fact that most durability studies have been limited to coupon level testing, and macro-level coupon studies may fail to predict ultimate failure at the structural level. Butler et al (2007) presented an analytical model for predicting the compressive fatigue limit strain of laminates containing BVID. The model provided the limit strain for the final rapid stages of growth of the delamination, when fatigue has led to delamination buckling and hence opening of delaminations at a significant depth within the laminate. In this model, the complex damage morphology was represented as a single, circular delamination, the area of delamination being determined by NDE methods. In this study, a fatigue limit strain of 3600μe was found, below which, propagation of the damage would not occur after 10⁶ cycles.

Hou et al (2001) developed a criterion that suggested that interlaminar shear stress always assists the initiation and propagation of delamination, whereas the effect of out-of-plane normal stress is very different. Out-of-plane tension was said to accelerate the development of delamination, whereas out-of-plane compression halts the development of delamination.
The effect of the compression was said to have the effect of the interlaminar shear stresses needing to be at least two times higher before any delamination would initiate, and that no delamination would occur once the compression exceeds a certain value.

Kardomateas et al (1995) outlined the significance of delamination buckling, in the sense that generally, the design for aircraft structures consists of thin skins, typically strengthened by longitudinal stiffeners. Aerodynamic loading induces lateral bending, with one side being subjected to compression. Any delaminations present on the compressive side within the skin, due to an impact for example, could incur local buckling, which effectively induces stress concentrations at the delamination front. Cyclic loading of compressed panels with delaminations (and hence repeated delamination buckling) causes a reduction of interlayer resistance as a result of damage accumulation at the tip. As a consequence, delaminations that would not propagate under static load, may grow and cause failure after a sufficient number of compression cycles.

2.6. Fracture mechanics and its application to delamination in composites

Fracture mechanics is based upon the initiation and growth of critical cracks that may cause premature failure in a structure. An interest in fracture mechanics arose as a consequence of the failure of the Liberty ships (Anderson (1994)). There are two alternative approaches to fracture analysis; the energy criterion and the stress intensity approach. This section outlines the principles of these two approaches.

2.6.1. The Energy Criterion approach

As mentioned in the introductory section of this study, the work by Griffith (1920) in the 1920’s was of major importance regarding fracture mechanics of materials. Griffiths’ work was a quantitative approach between fracture stress and flaw size, applying stress analysis of an elliptical hole, to the unstable propagation of a crack. The approach is based upon the first law of thermodynamics, a simple energy balance, whereby a flaw becomes
unstable and fracture occurs when the energy available for crack growth is large enough to overcome the resistance of the material (surface energy, plastic work and other energy dissipation associated with the propagating crack) (Anderson (2005)).

In the energy approach, the energy release rate is denoted as $G$, defined as the rate of change in potential energy with the crack area for a linear elastic material. When fracture occurs $G = G_c$, which is the critical energy release rate; a measure of interlaminar fracture toughness. For an infinite plate subject to a remote tensile stress, with a crack of length $2a$, the energy release rate is given by;

$$G = \frac{\pi \sigma^2 a}{E}$$  \hspace{1cm} \textit{Equation 2.6.1-1}.

At fracture this becomes

$$G_c = \frac{\pi \sigma_c^2 a_c}{E}$$  \hspace{1cm} \textit{Equation 2.6.1-2}

Here, $E$ is the Young's modulus, $\sigma_f$ is the remotely applied stress at fracture, and $a$ is half-crack length at fracture.

\begin{equation*}
2.6.2. \quad \textbf{The Stress intensity approach}
\end{equation*}

An alternative approach is the stress intensity approach, whereby the stress intensity factor characterises the crack tip stresses in a linear elastic material. If an assumption is made that a material fails locally at a critical combination of stress and strain, then fracture must occur at a critical stress intensity, $K_{IC}$. Therefore, $K_{IC}$ is an alternative measure of interlaminar fracture toughness. For the same conditions as stated for $G$, the stress intensity factor is given by;

$$K_I = \sigma \sqrt{\pi a}$$  \hspace{1cm} \textit{Equation 2.6.2-1}
Comparing Equation 2.7.1-2 and 2.7.2-1 results in a relationship between $K_I$ and $G$:

$$G = \frac{K_I^2}{E} \quad \text{Equation 2.6.2-2}$$

Therefore, the energy and stress-intensity approaches to fracture mechanics are essentially equivalent for linear elastic materials (Anderson (2005)). Irwin (1956) extended the Griffith approach to include energy dissipated by local plastic flow. The methods by Griffith and Irwin became known as linear elastic fracture mechanics (LEFM), with the methods becoming invalid when large plastic deformation precedes failure. Irwin’s work became known as the Irwin plastic zone correction, a relatively simple extension of LEFM (Anderson (2005)).

Interply delamination is considered to be the most critical failure mode limiting the long-term fatigue life of certain composite laminates (Wilkins et al (1980)). Delamination may occur in Mode I (opening mode), Mode II (sliding / in-plane shear mode), or Mode III (tearing mode) or a combination of these modes.

The compliance method is used in the calculation of the strain energy release rate for each mode, which is related to the specimen compliance by the following equation:

$$G_I or G_{II} = \frac{P^2}{2w} \frac{dC}{da} \quad \text{Equation 2.6.2-3}$$

Where

P = applied load
w = specimen width
a = crack length (measured during the test)
C = specimen compliance (slope of load-displacement curve for each crack length)
\[\frac{dC}{da} = \text{slope of the compliance vs. crack length curve.}\]

In mixed-mode linear elastic fracture mechanics, the fracture energy is expressed in terms of the Mode I and II stress intensity factors, \(K_I\) and \(K_{II}\) respectively.

2.6.3. Mode I and Mode II tests

Within this section a number of papers have been reviewed which focus on Mode I and Mode II testing, in particular novel test techniques and analysis.

A paper by Greenhalgh et al (2009), looking into fractographic observations on delamination growth and subsequent migration through the laminate, noted that migration of the crack dictates the delamination growth process and at a coupon level, by addressing migration, it is feasible to characterise the fracture toughness values of ply interfaces. It was reported in this paper that the toughness of 0/0 ply interfaces are not representative of those in structures but provide a conservative approach, as the toughness is increased at the 0/0 interface. A paper by Brunner et al (Brunner, 2008) reviewed the standardisation of test methods for fracture mechanics tests to determine delamination resistance or fracture toughness of fibre-reinforced, polymer-matrix composites. Developments leading towards new standardised test procedures were presented. Within the paper (Brunner, 2008), the limits to the ISO standard were outlined; namely the fact that the method is limited to quasi-static loading of unidirectional CFRP and GFRP. The applicability of the standard DCB specimen for delamination resistance testing of laminates with multidirectional lay-up was assessed by Choi et al (Choi, 1999) and de Morais et al (Morais, 2003, 2004). The problem that was highlighted within these papers was the issue with the delamination migrating from the mid plane, which is said to invalidate the analysis according to the ISO standard. As such, de Morais, 2004, reported that delamination resistance for multi-directional laminates can probably be quantified for initiation only, where no significant dependence on the
delamination interface was observed. Brunner et al (2008) summarised the report by stating that several test methods offer spreadsheets for data analysis, including various correction factors and offering the opportunity to compare different approaches for analysis (e.g. beam theory based versus experimental compliance). These corrected methods are investigated in some detail within the data analysis section of the Mode I and Mode II testing within this thesis.

The majority of the research discussed so far has concentrated on prepreg materials, however woven fabrics are being increasingly utilised because of the ease of handling and storage, simpler lay-up procedures and lower cost. Woven fabrics are also more damage tolerant in the presence of a delamination, Alif et al (1998). Alif et al report the damage tolerance of woven fabrics as being attributed to the non-planar interply structure of the fabric. The delamination being reported to interact with the matrix regions and the weave structure during its propagation, and as a result, experiencing substantial growth resistance. In the paper by Alif et al (1998), the data reduction method used was the empirical compliance method which is outlined in the ASTM D5528, a test method / data reduction method that appears to be the most widely used within the literature. A study by Blackman and Brunner (2013) investigated testing cross ply and woven fabric laminate materials, in the study it was found that for the woven materials the failure path of the crack did not deviate from between the central plies; however, slip stick behaviour was observed during crack propagation. Blackman and Brunner (2013) therefore highlight the complications with using the standard test methods in terms of identifying the initiation values, along with issues connected to recording crack growth in 1 mm intervals as specified in the test standard. The paper highlighted the importance of obtaining enough values to make the data analysis valid (all crack arrest points should be excluded from the analysis), as well as ensuring initiation occurs from a stable pre-crack.

A number of papers have also outlined a modified test method for specimens with enhanced fracture toughness by modifying the test specimen for Mode I testing. This can be achieved by bonding identical
reinforcing tabs made of conventional metals or composites to either side of the specimen. This was found to be especially important when dealing with non-crimp fabric composites due to the flexural modulus being significantly less than their UD counterparts (Wood et al (2007)). The study by Wood et al (2007) outlined the fact that for materials with enhanced mode II delamination toughness, such as those with stitching, a modified specimen would be needed to avoid the specimen failing in bending prior to crack propagation. A review by Blackman et al (2013), outlined the evolution of the ISO Standard test method for Mode I delamination toughness. The main conclusion was that, whereas the ASTM method requires testing using a film insert in a mode I test, the European Structural Integrity Society (ESIS) concluded that mode I pre-cracking was an essential part of the test as testing directly from the film insert could lead to unstable crack growth. Interestingly, re-initiating the crack from a mode I pre-crack yielded lower, conservative results, which led to the inclusion of this method within the ISO standard.

A recent study by Brunner et al (2016) emphasised the importance of developing a test methodology for multi-directional laminates for delamination fatigue of CFRP composites. The study emphasised the need for data analysis yielding rates or thresholds that can be transferred to engineering design in order to move to a damage tolerant design, so that certain amount of delamination or crack propagation is allowed.

Multiple delaminations / asymmetric thin film inserts
As mentioned in section 2.3, damage from an impact event is quite complex, with regards to modelling such damage, some models are only applicable to a damage process involving a single delamination. A number of investigation involving modelling multiple delaminations and transverse matrix cracks have been carried out by Zou et al (2001, 2002, 2003). Numerous publications in the literature (Yin (1998)) has reported results in which delamination propagation models have been formulated based on the total energy release rate. However, it is well known that the critical total energy release rate is not a material property, as it varies with the loading condition (Li et al (2006). As a result, models of this kind are only applicable
to problems in which the critical total energy release rates are measured under exactly the same condition. Their applications will be questionable in problems where the mode ratio changes as the delamination propagates (Li et al (2006)). This needs to be considered during the analysis of the DCB tests during this study and is why the emphasis has been placed on the initiation values as opposed to propagation. An investigation by Li et al (2006) looks at an improved formula from the standard virtual crack closure technique which takes into account sublaminates and the position of the delaminations in terms of sublaminates. The feature of the energy release rates derived by Li et al (2006) is that they apply equally to problems involving multiple delaminations as they were derived in the context of multiple delaminations.

2.7. Summary of literature Review

There are a very large number of papers available which discuss damage in composites; this review has only been able to summarise some of available work in this field. However, the core researchers in this field have published some interesting points, as highlighted in the preceding paragraphs.

The majority of the work discussed in this review has focussed upon carbon fibre reinforced polymer composites because these materials are most commonly used for structural applications in the aircraft industry; for this reason, these materials were selected as the main focus for the thesis. It was also noted that the most common matrix materials in aerostructures are the toughened epoxy thermosetting type, consequently, a toughened epoxy resin was selected as the material of choice for this study. Regarding the reinforcing fibre architecture and manufacturing technique considerations, it has been interesting to note that most of the work to date has concentrated upon UD prepreg materials, and hence, the structural study will focus upon these forms. However, also noted within the review, was the need for lower cost manufacturing methods and lower cost materials, hence non-crimp fabric will also be investigated within the study in the form of coupon specimens for Mode I and Mode II testing.
It has been reported that delamination is the predominant mode of failure, with failure often occurring by buckling or delamination, as opposed to fibre failure. It is for this reason that this study will concentrate on delamination behaviour in composites and how buckling can influence the initiation of growth within these delaminations.

Many reporters stated that delaminations exist as a crack between plies of different fibre orientation (due to the result of the bending stiffness mismatch between adjacent layers) and not between lamina in the same ply group. However, others have reported that significant damage can occur between layers with the same fibre orientation, with the intensity of the delaminations depending upon the difference in the ply angles above and below the interface. It is therefore important to determine the interface in which the critical delamination lies. This will be achieved here, using time of flight (TOF) ultrasonic C-scanning methods in order to map the through thickness delamination pattern.

Once a delamination is present, it has been reported that mode II is the dominant failure mode for propagation, with the mode mixity changing with the number of cycles. As a result of this, the experimental method for the structural testing was developed to provide high mode II levels during fatigue loading, in order for delamination to grow.

The final stages of delamination growth are reported to occur when fatigue has led to delamination buckling and hence opening of delaminations at a significant depth away from the impact point, within the laminate. Therefore, in order to achieve delamination growth, it is critical to ensure that the delaminations open, which can be achieved by local buckling of the composite. The critical depth of delamination occurs where the response undergoes a change in form, from a mode that opens the delamination, to one in which it is closed.

It is reported that interlaminar shear stress always assist the initiation and propagation of delaminations, along with out-of-plane tension, whereas out-of-plane compression halts the development of delamination. The effect
of the compression was said to have the consequence that the interlaminar shear stresses need to be at least two times higher before any delamination would initiate, and that no delamination would occur once the compression exceeds a certain value.

The majority of papers reviewed have concentrated on coupon or small-scale structural elements. However, in order to gain an insight into the damage mechanisms in real life structures a structure with components seen in service should be investigated, such as stringers. Loading of a plate with stringers in compression induces lateral bending, with one side being subjected to compression. Any delaminations present on the compressive side within the skin, could incur local buckling, which effectively induces stress concentrations at the delamination front. Cyclically loading of compressed panels with delaminations (and hence repeated delamination buckling) might cause a reduction of interlayer resistance as a result of damage accumulation at the tip. As a consequence, delaminations that would not propagate under static load may grow and cause failure after a sufficient number of compression cycles.

Damage growth is said to start early on in the fatigue lifetime, typically during the first 10 – 20% of the life of the component. This could mean that if no damage growth has occurred after this initial time, delamination growth will be unlikely throughout the lifetime. One study (Butler et al (2007)), reported that below a fatigue limit strain of 3600µε, damage would not grow after 10⁶ cycles, but this needs to be investigated further. It has also become apparent that fatigue delamination growth is strongly affected by the relative location of the delamination through the plate thickness, with growth being slower, or even a no growth situation, when delaminations are located close to the impact surface. Most delamination growth in thin laminates is reported to occur in the ply layers close to the back face of the specimens, where the largest delaminations had been produced by the impact.

The literature review highlighted the limited information available on delamination growth in structural components, and as such, an
understanding of the response of a sub-element under fatigue loading following an impact event seems a promising way forward, and is the approach adopted here.

Regarding woven fabrics, the literature has reported the complications with using the standard data analysis methods for mode I and mode II testing. Therefore tests involving (i) woven fabrics with delaminations from inserts at the mid plane, and at other locations throughout the laminate, (ii) delaminations caused from an impact event are areas that are investigated further within this dissertation. Hence, the objectives of the work are as follows:

- To investigate and carry out novel experimental testing techniques to understand damage behaviour at both the coupon level and the structural level in order to compare the results with standard analysis.

- To determine if standard analysis can be used and adapted to determine interlaminar fracture toughness values for delaminations more representative of those seen in real structures.
3. Interlaminar fracture toughness testing of Non Crimp Fabrics

3.1. Introduction

It is well known that composite materials, in particular laminated fibre-reinforced composites, have high specific strength and stiffness properties, and as such, these materials are being utilised to a much greater extent in airframe structures. The ‘Achilles’ heel of these composite materials is the through thickness property set. The through thickness tensile strength for CFRP has been reported as low as 50 MPa (Mespoulet S et al 1996) whilst the mode I interlaminar toughness may be less than 150 J / m² for a first generation epoxy system. It is also well known that airframe structures are susceptible to impact events, either due to runway debris, ice, hailstones or more severe impacts such as birdstrikes. These impacts cause through thickness stresses in a component, and if these stresses exceed the through thickness strength of the material, delaminations (separation of the plies) may initiate within the structure. The propagation of these delaminations is controlled by the interlaminar fracture toughness of the composite material (Robinson, 2000). There has therefore been extensive experimental work carried out to determine the interlaminar fracture toughness values of laminated composite materials with artificial inserts, usually expressed in terms of the critical energy release rate, \( G_c \), typically measured in mode I (opening mode, \( G_{I} \)) and mode II (shear mode, \( G_{II} \)).

Due to the extensive research and experimental work carried out in this area, it is possible to establish typical values for interlaminar fracture toughness for a wide variety of carbon fibre and matrix types with relative ease, however, the data tends to be restricted to unidirectional prepreg materials. These materials, although offering exceptional properties, are expensive to manufacture and as such, alternative material systems need to be investigated which can be manufactured using lower cost techniques. The aim of this section of the thesis was to use lower cost CFRP materials, specifically fabrics, containing natural damage caused from an impact event.
and from damage initiated from an artificial method and to compare the results in terms of the crack growth resistance.

3.2. Material and Preparation of Test Specimen

3.2.1. Introduction

This section of the investigation documents the materials and methods used in determining the Mode I and Mode II interlaminar fracture toughness values of non-crimp fabric carbon fibre composites. The nature of the materials tested is discussed in section 3.2.2. The sections which follow address the methods used to determine the interlaminar fracture toughness values. Section 3.3 discusses the double cantilever beam (DCB) method used for mode I testing and section 3.4 discusses the end notch flexure (ENF) method used to determine Mode II values.

3.2.2. Materials

The lay up selected for DCB and ENF testing is shown in Figure 10. The lay up was from 12 layers of intermediate modulus (IM) carbon non crimp fabric, having an uncured ply thickness of 0.508 mm, with single 5HS outer plies of 0.36 mm thickness.

The resin used was an epoxy resin which was infused using the liquid resin infusion method. The panel lay-up was [0/90], [+45/-45, 0/90, -45/+45, 0/90, +45/-45, 0/90], as illustrated in Figure 10. This lay-up was chosen as it was representative of the typical lay-ups seen in service for NCF materials.
The panel size was 1890 mm by 794 mm. One artificial insert was located at the mid plane (90° interface) of the panel and one located between the 0/90, +/- plies (Figure 10) during manufacture of the panel. After curing, the panel was impacted using 51 J, an energy level recommended to produce 50 mm diameter delamination area, a size selected to meet the requirements of delamination length recommended within the test specification (discussed in section 3.3 and 3.4). A schematic showing the location of the impact in relation to the artificial inserts can be seen in Figure 11.

Non-destructive testing, using the ultrasonic C-scanning technique was carried out by QinetiQ prior to sectioning, with 2D analysis being carried out on the entire panel (Figure 12).
In addition, through thickness, time of flight scanning was carried out on the impacted half of the panel (Figure 13). The time of flight scanning was carried out in order to determine the typical shape of the delamination caused by the impact event. On analysing the scan (Figure 13), the delamination pattern appeared to represent a barrel shape, a shape often observed in NCF’s. The largest delamination can be found to be at the mid plane of the specimen.
3.2.3. Specimen Preparation

The panel (as represented in Figure 11) was cut into sections, as identified in Figure 14. The section of the panel with the PTFE inserts was trimmed and thin strips from the edges pulled apart in order to identify the exact position of the film insert. The position of the delamination front was then marked on the panel.
In order to identify the delamination front in the impacted specimens, reference was made to the ultrasonic c-scan image (Figure 15). It was clear from the images that the delamination was larger than first intended, with an approximate delamination diameter of 90 mm. The specimens were therefore cut from the panel to ensure that the maximum extent of delamination extended 50 mm from the end of the coupon.

*Figure 14 – Specimen cutting plan*
Figure 15 – Example of delamination area for impacted specimens

The specimens were cut with the delamination front perpendicular to the specimen longitudinal direction. The ESIS ISO 15024 standard, recommended that the specimen length, L, should be at least 125 mm and as such 150 mm long specimens were cut. The width of the specimen was recommended to be 20 mm.
3.3. **Mode I: Double Cantilever Beam (DCB)**

3.3.1. **Introduction**

The double cantilever beam (DCB) test is commonly used to determine the mode I interlaminar fracture toughness values for composite laminates with test coupons manufactured to include a thin, non-adhesive film insert at least 50mm in length at one end of the specimen (Figure 16).

![Figure 16 – DCB test specimen geometry (BS ISO 15024 (2001))]()

The purpose of the embedded thin, artificial insert is to simulate an initial delamination. This is an imperfect process, not least because of the resin rich region generated around the finite-thickness insert.

3.3.2. **Test Method**

The test methods available are designed to be used when testing unidirectional laminates only. For this study, the European Structural Integrity Society (ESIS) test method ISO 15024 was used as a basis for the design of specimen and test specification for mode I testing. One reason that this test method recommends only using unidirectional lay-up is because these lay-ups exhibit very little anti-clastic bending (Poissons effect of beam subjected to bending), with the bending stiffness components satisfying the condition where the ratio \((D_{12})^2/(D_{11}D_{22})\) is much less than 1.
(D_{11}, D_{12} and D_{22} are components of the bending-stiffness matrix for the upper and lower arm of the double cantilever beam (BS ISO 15024 (2001))). It is thought that multidirectional lay-ups may not satisfy this condition, and hence may exhibit significant anti-clastic bending. Furthermore, multidirectional lay-ups typically exhibit crack branching away from the specimen mid-plane, and hence are not recommended within the test standard. However, a study by Fishpool (2013) on mode I testing of 3D woven composites concluded that some of the issues with woven materials is due to their enhanced toughness, they can suffer failure of the beam arms rather than through crack progression precluding toughness measurement. In this case the specimens can be reinforced with adhesively bonded tabs to prevent failure (Tanzawa et al, 1999, Dransfield et al, 1998, Tamuzs et al, 2003).

Within this study end blocks were lightly abraded along with the surface of the specimen and the end blocks were then bonded onto half of the specimens for each insert location (A and B shown in Figure 14) in order to provide a means of applying the load. Cyanoacrylate adhesive was used as the bond medium. In this way the specimen geometry for the mid-plane insert is consistent with that shown in Figure 16.

All specimen edges were coated with typewriter correction fluid to aid in the visual inspection of the crack tip and then, starting from the delamination front mark, 1 mm divisions were marked up to a total of 70 mm.

The length, l, width, B and thickness, 2h of each specimen were measured. The nominal average dimensions for the specimen can be seen in Table 3.
The parameters are defined below:

2h = total thickness of the specimen (each arm of the DCB specimen has thickness h)

H = thickness of load block

L₁ = distance from the centre of the loading pin to the midplane of the specimen beam to which the load block is attached

L₂ = distance between the centre of the pin-hole of the load-block and its edge, measured towards the tip of the insert

L₃ = total length of the load block

A = total length of the insert, distance between end of specimen on which load block are mounted and tip of the insert.

a₀ = initial delamination length, distance between the load line (intersection of the plane through the pin hole centres of the load blocks and the plane of delamination) and the tip of the insert on the edge of the specimen.

a = delamination length measured during the test

l = total length of the specimen

B = Width of the specimen

The specimens were mounted in the screw-driven Instron test machine with a 1 kN load cell. The test machine was equipped with a fixture to introduce the load to the pins inserted into the load blocks. The end of the specimen was supported in order to keep the beam orthogonal to the direction of the applied load.

Testing was carried out at a constant displacement rate of 1 mm / min, with load and displacement signals being recorded at regular intervals. The delamination growth was observed using a travelling microscope, with the specimen being illuminated to aid the visual observation.
The point of onset of delamination movement from either the insert or the impact delamination was recorded by noting down the crack length and the load and displacement values. After this, as many delamination length increments as possible were recorded in the first 5 mm and then subsequent delamination lengths were noted every 5 mm until the delamination had propagated at least 45 mm.

### 3.3.3. Data Reduction

#### 3.3.3.1. Introduction

Data reduction yields the critical energy release rates, $G_{IC}$, for initiation and propagation of a Mode I delamination, plotted as a function of delamination length. This plot is referred to as a delamination resistance cure, or R-curve. A typical R-Curve can be seen in *Figure 17*.

![Figure 17 - Schematic of an idealised R-curve from a DCB test](image)

Within the test method used (ESIS ISO 15024), it states that if the $G_{IC}$ initiation values measured from the insert are less than or equal to the propagation values plotted in the R-curve (as displayed in the schematic in *Figure 17*) then the minimum initiation value shall be considered a valid generic measure of the interlaminar fracture toughness of the composite. If the initiation values are greater than the propagation values, even though
the insert is thin and non-adhesive, then the DCB has to be repeated using wedge precracking techniques. This has to be assessed after the first test.

The data required for the analysis (see next section) were the initial delamination length, $a_0$, the delamination lengths, $a$, where $a = a_0 + |\Delta|$, measured delamination length increments, the corresponding loads, $P$ and displacements, $\delta$, and the width, $B$, of the specimen.

### 3.3.3.2. The Corrected beam theory data reduction method

The simple beam theory expression for the compliance of a perfectly built in DCB specimen with a clamped boundary condition will underestimate the compliance because the beam is not perfectly built in. A way to correct the effects of this is to treat the beam as if it contains a slightly longer delamination length $a + |\Delta|$, where $|\Delta|$ may be found experimentally by plotting the cube root of the normalised compliance, $(C/N)^{1/3}$ (where the load block correction, $N$, is described below) as a function of delamination length, $a$. The extrapolation of a linear fit through the data in the plot yields $\Delta$ as the $x$-intercept. If the $\Delta$ value from the fit is positive, a value of $\Delta = 0$ shall be used and this would be noted.

The critical energy release rate $G_{IC}$ is given by:

$$G_{IC} = \frac{3P\delta}{2B(a+|\Delta|)N} \quad \text{Equation 3.3.3.2-1}$$

Where $P$ is the load, $\delta$ is the displacement, $a$ is the delamination length, $B$ is the width of the specimen, $F$ is the large displacement correction, and $N$ is the load block correction. All initiation and propagation values are calculated.

The large displacement correction $F$ and the load block correction $N$ are calculated as follows:
\[ F = 1 - \frac{3}{10} \left( \frac{\delta}{a} \right)^2 - \frac{3}{2} \left( \frac{\delta l_1}{a^2} \right) \quad \text{Equation 3.3.3.2-2} \]

\[ N = 1 - \left( \frac{l_2}{a} \right)^3 - \frac{9}{8} \left[ 1 - \left( \frac{l_2}{a} \right)^2 \right] \delta l_1 a^2 - \frac{9}{35} \left( \frac{\delta}{a} \right)^2 \quad \text{Equation 3.3.3.2-3} \]

Where \( l_1 \) is the distance from the centre of the loading pin to the midplane of the specimen beam and \( l_2 \) the distance from the loading pin centre to its edge (Figure 16). The delamination resistance curve (R-curve), consisting of a plot of \( G_{IC} \) versus delamination length, \( a \), was determined for each specimen.

### 3.3.3.3. Modification for Crack path not at mid plane

In order to check if the results were to be expected, derivation of the initiation load and compliance from first principles were applied.

Insert at Mid plane

\[ I_1 = \frac{bh^3}{12} \quad I_1 \propto h^3 \quad \text{Equation 3.3.3.3-1} \]

\[ \delta_{\text{total}} = 2\delta = \frac{2pa^3}{3El_1} \quad \text{Equation 3.3.3.3-2} \]

Insert at 1/3 plane

\[ I_2 = \frac{b(2h)^3}{12} \quad \text{Equation 3.3.3.3-3} \]
\[ I_2 \propto \frac{bh^3}{27} \] \hspace{1cm} \text{Equation 3.3.3.3-4}

\[ I_3 = \frac{b(4h^3/3)^3}{12} \] \hspace{1cm} \text{Equation 3.3.3.3-5}

\[ I_3 \propto h^3 \frac{64}{27} \] \hspace{1cm} \text{Equation 3.3.3.3-6}

\[ \delta_{\text{total}} = \frac{pa^3}{3EI_2} + \frac{pa^3}{3EI_3} \] \hspace{1cm} \text{Equation 3.3.3.3-7}

\[ G = \frac{p^2}{2} \frac{dc}{da} \] \hspace{1cm} \text{Equation 3.3.3.3-8}

\[ G \propto \frac{dc}{da} \] \hspace{1cm} \text{Equation 3.3.3.3-9}

\[ \left( \frac{dc}{da} \right)_{\text{crack at mid plane}} \propto \frac{2}{l_1} = \frac{2}{h^3} \] \hspace{1cm} \text{Equation 3.3.3.3-10}

\[ \left( \frac{dc}{da} \right)_{\text{crack at } \frac{1}{3} \text{plane}} \propto \frac{1}{l_2} + \frac{1}{l_3} = \frac{27}{8h^3} + \frac{27}{64h^3} \propto \frac{243}{64h^3} \] \hspace{1cm} \text{Equation 3.3.3.3-11}

If \( G_c = \frac{p^2}{2} \frac{dc}{da} \), it follows that the load for crack initiation, \( P_i \):

\[ P_i \propto \frac{1}{\sqrt{\frac{dc}{da}}} \]

\[ \frac{P_i(\text{mid plane})}{P_i(\frac{1}{3}\text{plane})} = \sqrt{\frac{dc}{da}_{1/3}} \] \hspace{1cm} \text{Equation 3.3.3.3-12}

\[ \frac{P_i(\text{mid plane})}{P_i(\frac{1}{3}\text{plane})} = \sqrt{\frac{243}{64h^3}} \sqrt{\frac{243}{128}} = \sqrt{1.9} \approx 1.4 \] \hspace{1cm} \text{Equation 3.3.3.3-13}
From this derivation, it would be expected that the load for initiation of delamination at the mid plane would be 1.4 times the load of samples with the insert at a 1/3 plane.

### 3.4. Mode II: End Notch Flexure (ENF)

#### 3.4.1. Introduction

The same geometry of specimen was used for mode II testing as was used for the mode I tests outlined in 3.2.3. The end notch flexure test was used to determine the mode II fracture toughness values. The ASTM standard D5528 was followed.

#### 3.4.2. Test Method

Specimens were loaded in a standard three-point bend fixture, as shown in Figure 18 below.

![Figure 18 – End Notch Flexure test rig.](image)
The specimen edges were coated with typewriter correction fluid and the support location was marked to enable the initial crack length from the load point to be measured accurately. The loading roller had a radii of 2.5 mm.

In order to assist in crack initiation and to avoid any instability due to bonding through the release film insert, the specimens were pre-cracked 10 mm prior to the start of the test by placing the specimen in the three point bending rig attached to an Instron test machine applying the load. The specimens were located in the fixture so that the crack length was 40 mm. The specimen was loaded until the crack had grown under the mid point roller (crack growth of 10 mm). The crack tip of the pre crack was then marked and the specimens were mounted so that the ratio of $a/L = 0.5$ was obtained for the new crack (film + shear pre crack). The specimen was loaded using displacement control at a rate of 1 mm/min in order to enable slow stable crack propagation.

### 3.4.3. Data Reduction

A number of methods were looked at to interpret the results, the direct beam theory, modified beam theory and corrected modified beam due to asymmetric flexure.
3.4.3.1. The Direct Beam Theory

The equation used for the direct beam theory is given below:

\[ G_{IIC} = \frac{9a^2P\delta}{2B(2L^3+3a^3)} \]  

Equation 3.4.3.1-1

Where \( P = \) Peak load, \( \delta = \) displacement at peak load, \( B = \) Specimen width, \( L = \) half span and \( a = \) crack length.

3.4.3.2. The modified Beam theory

Russell and Street (Russell and Street, 1985) presented a beam theory solution for compliance and strain energy release rate for the ENF specimen.

\[ G_{IIC} = \frac{9a^2P^2}{16Eb^2h^3} \]  

Equation 3.4.3.2-1

In this equation, the value of \( E \) is modulus measured during compliance calibration for \( a = 0 \).

\[ E = \frac{l^3}{4Bch^3} \]  

Equation 3.4.3.2-2

Where \( h = \) half specimen thickness and \( 1/C = \) the initial slope of the load displacement plot, ignoring any initial non-linearity due to compliance within the fixture.

3.4.3.3. Modified beam with asymmetric correction

(Zhou & He (1993))

An interesting paper by Zhou and He (1993) stated that since the flexure of ENF specimens is asymmetric, due to the existence of the delamination, the slope of the centre of the ENF specimen is not zero and therefore the compliance calculation proposed by Russell and Street may be unreasonable.
Figure 20 is a photograph showing the side view of an ENF specimen under load. It is clear to see that the line of load introduction is not the line of symmetry. The point at which maximum deflection occurs is in the region of the cracked half segment from the central point.

![Figure 20 – Side view of an end-notched flexure test in progress (Zhou and He (1993))](image)

Zhou and He (1993) proposed therefore that the ENF specimen should be modelled as two cantilever beams DE and EA (Figure 21) instead of the beams BC and CD used in the Russell and Street derivation.

![Figure 21 – Bending of the ENF specimen (Zhou and He (1993))](image)

Having analysed the results by Zhou and He (1993) using Macaulays beam theory method, confidence in the approach was gained (see Appendix 1 for the analysis). Taking this same approach and considering the beam as two separate beams in terms of thickness, an alternative equation was derived to take into account delaminations away from the mid plane.
The resulting expression for the strain energy release rate (derivation can be found in Appendix 1) was:

\[ G = \frac{3P^2a^2}{16Eb^2h^3} \left( \frac{\frac{bh^3}{2(t_1 + t_2)^2} - 1}{t_1 + t_2} \right) \]

Equation 3.4.3.3-1

When \( t_1 = t_2 = h \), the term in the brackets tends to 3 and the standard result is then found.

\[ G = \frac{9p^2a^2}{16Eb^2h^3} \]

Equation 3.4.3.3-2

3.5. Mode I DCB Test results

3.5.1. Introduction

Results for specimens with PTFE inserts and impacts have been analysed and the data are presented in tables, along with graphs showing the response during test.

3.5.2. Standard Specimen with PTFE Inserts

Figure 22 shows the load versus displacement plots for each specimen.
The $\Delta$ value was determined by plotting the cube root of the normalised compliance, $(C/N)^{1/3}$ (Error! Reference source not found.). The $\Delta$ value from the fit was found to be positive in all specimens tested, the point at which the line of best fit crossed the x-axis was used in the modified G value. An example of the correction factor can be seen in Figure 23.
Figure 23 – Compliance check for a typical sample with artificial insert at the mid plane

The mode I interlaminar fracture toughness values calculated at each increment in crack length can be seen in Figure 24.

Figure 24 – $G_{IC}$ versus crack length for mid-plane artificial insert coupons
The overall results for the specimens with PTFE inserts at the mid-plane can be seen in Table 4. The initiation load, displacement and fracture toughness have been identified along with the maximum load for each specimen and corresponding fracture toughness at the maximum load. The maximum mode I fracture toughness value has also been included.

<table>
<thead>
<tr>
<th>Specimen Number</th>
<th>Load at A₀ (N)</th>
<th>Displacement at A₀ (mm)</th>
<th>G&lt;sub&gt;i&lt;/sub&gt; Initiation (J/m&lt;sup&gt;2&lt;/sup&gt;)</th>
<th>Max Load (N)</th>
<th>Crack length at max load (mm)</th>
<th>G&lt;sub&gt;i&lt;/sub&gt;C at max load (J/m&lt;sup&gt;2&lt;/sup&gt;)</th>
<th>Max G&lt;sub&gt;i&lt;/sub&gt;C (J/m&lt;sup&gt;2&lt;/sup&gt;)</th>
</tr>
</thead>
<tbody>
<tr>
<td>A01</td>
<td>95.26</td>
<td>1.48</td>
<td>392.6</td>
<td>139.89</td>
<td>47</td>
<td>1175.3</td>
<td>1175.3</td>
</tr>
<tr>
<td>A02</td>
<td>78.99</td>
<td>1.20</td>
<td>158.1</td>
<td>148.26</td>
<td>47</td>
<td>769.7</td>
<td>945.1</td>
</tr>
<tr>
<td>A03</td>
<td>58.80</td>
<td>1.10</td>
<td>230.5</td>
<td>134.46</td>
<td>47</td>
<td>1292.6</td>
<td>1312.2</td>
</tr>
<tr>
<td>A04</td>
<td>94.94</td>
<td>1.66</td>
<td>382.2</td>
<td>133.55</td>
<td>64</td>
<td>1401.3</td>
<td>1401.3</td>
</tr>
<tr>
<td>A05</td>
<td>90.78</td>
<td>1.55</td>
<td>319.6</td>
<td>140.58</td>
<td>47</td>
<td>1048.4</td>
<td>1048.4</td>
</tr>
<tr>
<td>Average</td>
<td>83.75</td>
<td>1.40</td>
<td>296.6</td>
<td>139.35</td>
<td>50.40</td>
<td>1137.5</td>
<td>940.26</td>
</tr>
<tr>
<td>S.D</td>
<td>15.43</td>
<td>0.24</td>
<td>100.78</td>
<td>5.89</td>
<td>7.60</td>
<td>244.0</td>
<td>483.24</td>
</tr>
<tr>
<td>C.V (%)</td>
<td>18.42</td>
<td>17.02</td>
<td>33.98</td>
<td>4.23</td>
<td>15.08</td>
<td>21.46</td>
<td>51.39</td>
</tr>
</tbody>
</table>

Table 4 – Mode I fracture toughness values of specimens with artificial insert at mid plane.

The results for the DCB tests for the specimens with PTFE inserts located at 1/3 thickness can be seen below. Figure 25 displays load versus displacement plot for each specimen and the fracture toughness value calculated per crack advance can be seen in Figure 26.
Figure 25 - Load versus displacement for artificial insert located between 0/90, +/- plies

Figure 26 - $G_{IC}$ versus crack length for artificial insert located between 0/90, +/- plies

The overall results for the DCB tests with the PTFE inserts located at 1/3 thickness can be seen in Table 5.
As with previous results, both initiation and maximum values have been provided.
3.5.3. Impact Specimens

The load was plotted as a function of displacement (Figure 27).

![Figure 27 - Load versus displacement plot for impacted coupon](image)

**Table 5** - Mode I fracture toughness values of specimens with artificial insert at between 0/90, +/- plane.

<table>
<thead>
<tr>
<th>Specimen Number</th>
<th>Load at A₀ (N)</th>
<th>Displacement at A₀ (mm)</th>
<th>Gᵢc Initiation (J/m²)</th>
<th>Max Load at max load (N)</th>
<th>Crack length at max load (mm)</th>
<th>Gᵢc at max load / Max Gᵢc (J/m²)</th>
</tr>
</thead>
<tbody>
<tr>
<td>B01</td>
<td>56.49</td>
<td>1.48</td>
<td>170</td>
<td>Block failed</td>
<td>N/A</td>
<td>N/A</td>
</tr>
<tr>
<td>B02</td>
<td>72.14</td>
<td>1.67</td>
<td>305.2</td>
<td>77.46</td>
<td>55</td>
<td>726.4</td>
</tr>
<tr>
<td>B03</td>
<td>62.85</td>
<td>1.90</td>
<td>248.4</td>
<td>65.78</td>
<td>58</td>
<td>463.1</td>
</tr>
<tr>
<td>B04</td>
<td>78.32</td>
<td>2.36</td>
<td>408.0</td>
<td>78.32</td>
<td>40</td>
<td>408.0</td>
</tr>
<tr>
<td>B05</td>
<td>57.92</td>
<td>1.78</td>
<td>234.8</td>
<td>59.71</td>
<td>59</td>
<td>503.2</td>
</tr>
<tr>
<td>Average</td>
<td>67.81</td>
<td>1.93</td>
<td>273.28</td>
<td>70.32</td>
<td>53.00</td>
<td>525.18</td>
</tr>
<tr>
<td>S.D</td>
<td>9.16</td>
<td>0.3</td>
<td>89.3</td>
<td>9.09</td>
<td>8.83</td>
<td>139.7</td>
</tr>
<tr>
<td>C.V (%)</td>
<td>13.51</td>
<td>15.8</td>
<td>32.7</td>
<td>12.93</td>
<td>16.66</td>
<td>26.6</td>
</tr>
</tbody>
</table>
The $\Delta$ value was determined by plotting the cube root of the normalised compliance, $(C/N)^{1/3}$, an example being provided in (Figure 28). The $\Delta$ value from the fit for all samples was found to be positive and as such a correction factor was not required in the results.

![Correction factor, $C/N^{1/3}$ vs $a$](image)

*Figure 28 – Compliance check for specimens with impact*

The mode I fracture toughness was plotted as a function of crack length (Figure 29). Only two of the four samples tested provided valid results to analyse, as two samples did not provide enough data points due to sudden crack growth or premature fracture.
The results for the specimens with impact damage can be seen in Table 6. Care should be taken when interpreting the results, as the analysis relied on the values of \( l_1 \) and \( a_0 \) being approximated for each specimen. It is known that more than one delamination plane exists in each specimen, however, in order to analyse the data, an approximation was made, as indicated in Table 6.

<table>
<thead>
<tr>
<th>Specimen Number</th>
<th>Load at ( A_0 ) (N)</th>
<th>Displacement at ( A_0 ) (mm)</th>
<th>( G_{IC} ) Initiation (J/m(^2))</th>
<th>Max Load (N)</th>
<th>Crack length at max load (mm)</th>
<th>( G_{IC} ) Max (J/m(^2))</th>
</tr>
</thead>
<tbody>
<tr>
<td>A1</td>
<td>161.96</td>
<td>0.37</td>
<td>76.04</td>
<td>161.96</td>
<td>59</td>
<td>956.70</td>
</tr>
<tr>
<td>A2</td>
<td>149.23</td>
<td>0.45</td>
<td>85.88</td>
<td>149.23</td>
<td>59</td>
<td>1366.29</td>
</tr>
<tr>
<td>Average</td>
<td>155.59</td>
<td>0.41</td>
<td>80.96</td>
<td>155.59</td>
<td>58.75</td>
<td>1161.50</td>
</tr>
<tr>
<td>S.D</td>
<td>9.00</td>
<td>0.06</td>
<td>6.96</td>
<td>9.00</td>
<td>0.35</td>
<td>289.62</td>
</tr>
<tr>
<td>C.V (%)</td>
<td>5.78</td>
<td>13.75</td>
<td>8.59</td>
<td>5.78</td>
<td>0.60</td>
<td>24.94</td>
</tr>
</tbody>
</table>

*Table 6 – Mode I fracture toughness values of specimens with impact damage*
Unfortunately it was not possible to inspect the specimens post failure using fractographic techniques as the customer required the specimens to be returned.

### 3.6. Mode II ENF Test Results

#### 3.6.1. Introduction

Within this section the results for the end notch flexure test can be seen. The maximum load and displacement has been provided along with the mode II fracture toughness.

#### 3.6.2. Standard Specimen with Artificial Insert

The load as a function of crosshead displacement can be seen in Figure 30.

![Figure 30 - Load versus displacement of mode II coupons with artificial inserts located at mid ply](image)

The results for the specimens with PTFE insert located at mid thickness can be seen in Table 7. Sample A16 has been highlighted red as this samples response under load was found to be inconsistent with the remaining batch of specimens.
The results for the mode II (ENF) test with the PTFE insert located at 1/3 thickness can be seen in Figure 31 and Table 8. As with the previous batch of samples, one sample has been highlighted red as the results are inconsistent with the remaining batch, this sample has not been included within the averages of the strain energy release rates.

**Table 7 – Mode II fracture toughness for artificial insert located at mid ply**

<table>
<thead>
<tr>
<th>Specimen Number</th>
<th>Width (m)</th>
<th>Max load (N)</th>
<th>Displacement at Max load (m)</th>
<th>$G_{IIc}$ (direct beam) (J/m$^2$)</th>
<th>$G_{IIc}$ (modified beam) (J/m$^2$)</th>
<th>$G_{IIc}$ (Corrected modified beam) (J/m$^2$)</th>
</tr>
</thead>
<tbody>
<tr>
<td>A16</td>
<td>0.02</td>
<td>953.7068</td>
<td>0.00169 0.050 0.025</td>
<td>764.09</td>
<td>369.81</td>
<td>369.81</td>
</tr>
<tr>
<td>A17</td>
<td>0.02</td>
<td>1218.46</td>
<td>0.00167 0.050 0.025</td>
<td>964.63</td>
<td>603.62</td>
<td>603.62</td>
</tr>
<tr>
<td>A18</td>
<td>0.02</td>
<td>1189.987</td>
<td>0.00167 0.050 0.025</td>
<td>939.08</td>
<td>575.74</td>
<td>575.74</td>
</tr>
<tr>
<td>A19</td>
<td>0.02</td>
<td>1356.998</td>
<td>0.00177 0.050 0.025</td>
<td>1138.63</td>
<td>748.69</td>
<td>748.69</td>
</tr>
<tr>
<td>A20</td>
<td>0.02</td>
<td>1410.162</td>
<td>0.00185 0.050 0.025</td>
<td>1232.95</td>
<td>808.50</td>
<td>808.50</td>
</tr>
<tr>
<td>Average</td>
<td></td>
<td>1225.86</td>
<td>0.00173</td>
<td>1007.88</td>
<td>621.27</td>
<td>621.27</td>
</tr>
<tr>
<td>S.D</td>
<td></td>
<td>177.89</td>
<td>0.0008</td>
<td>182.89</td>
<td>170.96</td>
<td>170.96</td>
</tr>
<tr>
<td>C.V (%)</td>
<td></td>
<td>14.51</td>
<td>4.49</td>
<td>18.15</td>
<td>27.52</td>
<td>27.52</td>
</tr>
</tbody>
</table>

*Figure 31 - Load versus displacement of mode II coupons with artificial inserts located between 0/90 and +/- plies.*
Table 8 - Mode II fracture toughness for artificial insert located between 0/90 and +/- plies.

### 3.6.3. Impact specimens

The specimens containing delaminations caused by a 50 J impact event were also tested under mode II loading. The load versus displacement plots for each specimen and the mode II fracture toughness values can be seen in Figure 32 and Table 9 respectively.

![ENF Impact Delamination](image-url)  
*Figure 32 – Load versus displacement plot for mode II loading of impacted coupon*
As with the DCB specimens, unfortunately fractographic analysis of the specimens was not possible due to the customer requiring the samples back.

### Table 9 - Mode II fracture toughness for impacted coupon

<table>
<thead>
<tr>
<th>Specimen Number</th>
<th>Width</th>
<th>Max load</th>
<th>Displacement at Max load</th>
<th>L</th>
<th>a</th>
<th>$G_{IIc}$ (Direct beam theory)</th>
<th>$G_{IIc}$ (Modified beam) Zhou</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>m</td>
<td>N</td>
<td>m</td>
<td>m</td>
<td>m</td>
<td>J/m²</td>
<td>J/m²</td>
</tr>
<tr>
<td>A6</td>
<td>0.02</td>
<td>1018.66</td>
<td>0.0016</td>
<td>0.050</td>
<td>0.025</td>
<td>782.31</td>
<td>421.89</td>
</tr>
<tr>
<td>A7</td>
<td>0.02</td>
<td>1182.47</td>
<td>0.0016</td>
<td>0.050</td>
<td>0.025</td>
<td>906.83</td>
<td>568.49</td>
</tr>
<tr>
<td>A8</td>
<td>0.02</td>
<td>1616.90</td>
<td>0.0020</td>
<td>0.050</td>
<td>0.025</td>
<td>1566.04</td>
<td>1062.94</td>
</tr>
<tr>
<td>A9</td>
<td>0.02</td>
<td>1591.10</td>
<td>0.0021</td>
<td>0.050</td>
<td>0.025</td>
<td>1590.63</td>
<td>1029.29</td>
</tr>
<tr>
<td>Average</td>
<td></td>
<td>1352.28</td>
<td>0.00185</td>
<td></td>
<td></td>
<td>1211.45</td>
<td>770.65</td>
</tr>
<tr>
<td>S.D</td>
<td></td>
<td>298.44</td>
<td>0.00027</td>
<td></td>
<td></td>
<td>426.80</td>
<td>323.95</td>
</tr>
<tr>
<td>C.V (%)</td>
<td></td>
<td>22.07</td>
<td>14.36</td>
<td></td>
<td></td>
<td>35.23</td>
<td>42.04</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Specimen Number</th>
<th>Width</th>
<th>Max load</th>
<th>Displacement at Max load</th>
<th>L</th>
<th>a</th>
<th>$G_{IIc}$ (Direct beam theory)</th>
<th>$G_{IIc}$ (Modified beam) Zhou</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>m</td>
<td>N</td>
<td>m</td>
<td>m</td>
<td>m</td>
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</tr>
<tr>
<td>A6</td>
<td>0.02</td>
<td>1018.66</td>
<td>0.0016</td>
<td>0.050</td>
<td>0.025</td>
<td>782.31</td>
<td>421.89</td>
</tr>
<tr>
<td>A7</td>
<td>0.02</td>
<td>1182.47</td>
<td>0.0016</td>
<td>0.050</td>
<td>0.025</td>
<td>906.83</td>
<td>568.49</td>
</tr>
<tr>
<td>A8</td>
<td>0.02</td>
<td>1616.90</td>
<td>0.0020</td>
<td>0.050</td>
<td>0.025</td>
<td>1566.04</td>
<td>1062.94</td>
</tr>
<tr>
<td>A9</td>
<td>0.02</td>
<td>1591.10</td>
<td>0.0021</td>
<td>0.050</td>
<td>0.025</td>
<td>1590.63</td>
<td>1029.29</td>
</tr>
<tr>
<td>Average</td>
<td></td>
<td>1352.28</td>
<td>0.00185</td>
<td></td>
<td></td>
<td>1211.45</td>
<td>770.65</td>
</tr>
<tr>
<td>S.D</td>
<td></td>
<td>298.44</td>
<td>0.00027</td>
<td></td>
<td></td>
<td>426.80</td>
<td>323.95</td>
</tr>
<tr>
<td>C.V (%)</td>
<td></td>
<td>22.07</td>
<td>14.36</td>
<td></td>
<td></td>
<td>35.23</td>
<td>42.04</td>
</tr>
</tbody>
</table>
3.7. Discussion

3.7.1. Mode I

The load-displacement curves for the specimens with an insert were very consistent and there was found to be very little scatter between the samples; both when the insert was at mid-plane or at 1/3 plane. This gave confidence in the results obtained. The load-displacement curves for all types of specimens tested show relatively abrupt load drops, this can be attributed to unstable crack growth. The goodness of fit for the compliance calibration meant that a $\Delta$ value (point of intersection of x axis from compliance plot) of zero could be used for the impacted specimens, however for the specimens with inserts a correction was applied to the results.

Unfortunately, only two of the impacted samples provided enough data points to be analysed. One sample failed prior to crack initiation (sample A4) and another sample had a large secondary crack (sample A3) and as such, monitoring crack growth became an issue. From the two remaining samples (Samples A1 & A2), sample A2 provided the more typical load displacement curve, with crack growth being more stable. Despite the novelty of using impact damaged specimens for mode I DCB testing, and the difficulties described above, the testing and analysis undertaken demonstrates the validity and usefulness of the experimental approach for future studies.

Table 10 below shows a comparison of the Mode I initiation values along with the maximum interlaminar fracture toughness value.
The results from test samples with film inserts clearly show a greater load at initiation with the insert located at the mid plane ($P_{\text{mid}} \sim 1.24P_{1/3}$). This shows very good agreement of experimental results with the result established from the derivation shown in equation 3.3.3.3-13, where $P_{\text{mid}} = 1.4P_{1/3}$. In terms of the initiation load for samples with an impact, the load to initiate a crack was 66% greater than with a starter crack at the mid plane. This implies that because there is more than one delamination arising from an impact event, the load is distributed over multiple delamination crack fronts and a greater load is therefore required to grow the critical delamination. The maximum $G_{IC}$ for the samples with insert at the mid plane and the samples with impact delamination were similar (within 10%), being 1137.5 J/m$^2$ and 1161.50 J/m$^2$ respectively however the maximum $G_{IC}$ for the sample with the insert at 1/3 plane was found to be 30% less than the insert at mid plane, this illustrates that although the load to initiate the delamination is greater for impacted samples, once the critical delamination grows, the energy required to propagate the crack is lower than the energy required to propagate a crack from an insert.

A very recent study by Stegschuster et al (2016) on mode I delamination of 3D woven composites reported Mode I initiation values as approximately 350 J / m$^2$ and propagation values of approximately 900 J / m$^2$, showing good agreement with the values obtained within this study.

<table>
<thead>
<tr>
<th>Sample Type</th>
<th>Average load at initiation (N)</th>
<th>Average $G_{IC}$ at initiation (J/m$^2$)</th>
<th>Max $G_{IC}$ (J/m$^2$)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Insert at mid plane</td>
<td>83.75</td>
<td>296.6</td>
<td>1137.5</td>
</tr>
<tr>
<td>Insert at 1/3 plane</td>
<td>67.81</td>
<td>273.28</td>
<td>525.18</td>
</tr>
<tr>
<td>Impact</td>
<td>155.59</td>
<td>80.96</td>
<td>1161.50</td>
</tr>
</tbody>
</table>

Table 10 – Comparison of initiation values and max $G_{IC}$
The shapes of the R-curve for the standard specimen with the insert at mid plane was typical (see Figure 17 and Figure 24), the initiation $G_{IC}$ values were found to be less than the propagation values, which complies with the requirements (ESIS ISO 15024) and as such the specimens were not required to be modified in terms of a wedge precrack. The propagation $G_{IC}$ values for the insert at 1/3 plane (between the +45/-45 plies) were found to be approximately half the propagation $G_{IC}$ values for the insert at the centre. The impacted samples appeared to have the same value for initiation and propagation, however these results have been rather difficult to interpret due to the presence of multiple delaminations and the associated challenge of accurately monitoring crack propagation and having accurate values to input into the analysis.

There have been reports that when characterising 0/0 ply interfaces, the plies tend to nest during processing which promotes the development of fibre bridging during testing, which increases the apparent toughness. Such nesting, and therefore fibre bridging does not develop in multidirectional ply interfaces (Greenhalgh et al, 2009) and as such the toughness value is not over estimated. Analysis of the fracture surface would be required to determine if fibre bridging had occurred in the NCF samples. The fact that the propagation value for the insert at the mid plane are greater than the propagation at 1/3 plane could be due to the fact that some fibre bridging occurred at the interface 0/90, whereas no bridging occurred between the +45/-45 interface.

### 3.7.2. Mode II

The load displacement plots for the samples with inserts at the mid plane and at 1/3 plane had very little scatter, giving confidence in the results. One sample with the insert at mid plane appeared to have a different compliance to the rest, also having a lower maximum load, however the variance between the maximum loads for all the samples was found to be relatively small, at 14%. All samples tested (with artificial insert and impact) demonstrated a linear response, followed by increasing non-linearities as the crack begins to propagate. All the samples exhibit a
maximum load level, after which the load drops as the crack begins to propagate. The crack propagated in a stable manner during the displacement controlled fracture testing for all the specimens, with the exception of those highlighted in red in the tables. Although the ENF test is designed for prepreg materials, the test was deemed successful due to the consistency of the results, and analysis was possible for all of the configurations included. Table 11 shows the comparison of results between different sample types as well as showing the results using different analysis methods.

<table>
<thead>
<tr>
<th>Specimen Type</th>
<th>$G_{IIc}$ (J/m$^2$)</th>
<th>$G_{IIc}$ (J/m$^2$)</th>
<th>$G_{IIc}$ (J/m$^2$)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Direct beam</td>
<td>Modified beam</td>
<td>Modified for asymmetric bending</td>
</tr>
<tr>
<td>Insert at mid plane</td>
<td>1007.88</td>
<td>621.27</td>
<td>621.27</td>
</tr>
<tr>
<td>Insert at 1/3 plane</td>
<td>1689.44</td>
<td>1132.42</td>
<td>485.32</td>
</tr>
<tr>
<td>Impacted</td>
<td>1211.45</td>
<td>770.65</td>
<td>770.65</td>
</tr>
</tbody>
</table>

*Table 11 – Comparison of mode II interlaminar fracture toughness*

Calculating results using the modified beam method is important as otherwise the interlaminar fracture toughness values would be overestimated. It should also be noted that if the standard modified beam data analysis is used for samples where the delamination is not at the centre then the interlaminar fracture toughness value would be overestimated considerably (233%).

A number of authors (Lee, 1999, Greenhalgh et al, 2009) have discussed the damage process of Mode II fracture, stating that cusp formation and deformation is thought to be the dominant energy absorbing process during mode II fracture. The authors have concluded that the cusp formation leads to an overall increase in fracture energy absorption and toughness with the mode II component, which is why mode II fracture exceeds Mode I fracture toughness. This has been validated in this study, with all mode II interlaminar fracture toughness values being greater than the mode I...
values. This could be studied further by looking at the fracture surfaces for NCF’s in future studies to determine if the cusp formation phenomenon is observed for NCF as well as prepregs.

Greenhalgh et al (2009) explained the mechanisms of crack migration in multidirectional laminates, stating that there is a natural tendency for a delamination to migrate upwards towards the compressive face of the laminate under bending until it reaches an interface in which the ply is orientated parallel to the driving force. Observing the lay-up of the NCF, the mid plane interface would be between a 90 / +45 ply and so the crack would likely to have migrated to the 0/90 ply interface, just one ply away (Figure 10). For the insert at 1/3 plane the ply interface was 90/90, however again the crack would only have migrated one ply away to the 0/0 interface. On observing the crack propagation, crack migration was not obvious during testing, which would support this assumption. Crack migration may have been limited due to the stitching technique used for NCF blanket production.
4. Experimental Work on Structural Element

4.1. Introduction

This section outlines the experimental fatigue study carried out on a series of impacted structural elements. The element characteristics are detailed, along with the results from the experimental tests and non-destructive testing. The results have been analysed and are outlined within the discussion.

As highlighted from the literature review, there is limited research available for the fatigue behaviour of structural components; the majority of work to date has concentrated upon coupon specimens, which although providing information on the material behaviour, structural effects are absent. Ideally, all testing should be carried out on a structure similar to that seen in service; however, large scale structural testing is extremely expensive and it is for this reason that this study has considered a structural element. In service, the structures that are likely to be subjected to an impact event often have structural additions such as stiffeners and as such, a single blade stiffened panel was identified as the structural element to test throughout this part of the study.

As reported in the literature review, Dorey (1986) investigated the impact response of stiffened panels, and reported that the energy to cause BVID dropped significantly near the stiffeners, where the structure was less compliant, and the stiffeners caused the damage to spread asymmetrically. Davies et al (1994) showed that at the edge of the stiffeners, delaminations were formed, whilst impacts directly over the stiffener caused debonding between plate and stiffener. It was for this reason that the location of the impact was chosen within this study to be in the centre of the length of the panel, adjacent to the foot of the stringer (Figure 33), therefore reducing the severity of edge effects, whilst maximising the chances of delaminations being formed, as opposed to debonding of the stringer.
Once a delamination is present, it has been reported (Rogers et al. (2008)), that mode II is the dominant failure mode for propagation and therefore the loading for this study was chosen to induce high levels of mode II loading, by loading both in-plane and out-of-plane.

### 4.2. Manufacture of the structure

Due to this study concentrating upon damage management in aerospace structures, it is carbon fibre reinforced epoxy resin that was selected for the material throughout the study. The material chosen was IMS - 977/2 which is a prepreg system containing continuous high strength carbon fibres in a toughened epoxy 977/2 resin system, supplied by Cytec, with a nominal fibre volume fraction of 55-wt%.

Structural elements were manufactured from Cytec’s IMS-977/2 prepreg. The skin lay-up for the elements was [+45, 0, -45, 0, 90, 0, -45, 0, +45], with each ply having a nominal thickness of 0.25 mm, resulting in a cured skin thickness of approximately 4.7 mm. The skin was approximately 290 mm long and 180 mm wide (Figure 33). The elements were cured in an autoclave at a temperature of 180°C, with a ramp up rate of 2°C / min and cured for 3 hours.
The blade was constructed from two back to back ‘L’ sections with a lay-up of [+45, -45, 0, 90]s [-45, +45, 0, 90]s, again, with a nominal ply thickness of 0.25 mm, giving a blade thickness of approximately 4.2 mm. This lay-up was chosen as it is representative of a typical aerospace lay-up. The blade height was 40 mm from the foot of the blade, with the blade foot having a width of 80 mm and thickness of 2.1 mm. The blade stringer was bonded onto the skin using Cytec FM73 film adhesive, typically 10 μm thick. A schematic showing the geometry and dimensions of the elements can be seen in Figure 33 above.

After trimming to size, and C-scanning, the elements were end potted using aluminium channelling, to a depth of 20 mm using Araldite AW-106 epoxy resin with HV953U hardener and HV997 accelerator, in order to prevent brooming of the panel ends and to ensure even loading across the panel width. The free length of the element was approximately 250 mm, as depicted in Figure 33.
4.3. Laminate quality

After curing, each of the elements was inspected using the ultrasonic C-Scanning technique, with a focused probe operating at 5MHz and scanning area of 60 mm by 60 mm. C-scanning was carried out in order to ensure the structures were of acceptable quality. All elements were well consolidated, with all panels passing quality control, showing no signs of porosity, delamination, inclusions, dry patches and good consolidation of the plies.

4.4. Impacting

The elements were all subjected to an impact event on the external face (Figure 34) from an instrumented drop weight impactor with an energy of 35J, using a 12.5 mm hemispherical steel tup. The impact was located on the skin of the panel half way between the edge of the foot of the blade, and the free edge of the panel, nominally 25 mm (Figure 34).

![Figure 34 – Schematic showing location of impact event and largest delamination caused by impact](image)

Second strike by the striker was avoided during impact. This particular impact energy was chosen because it has been reported (Davies (2002)) that at this impact energy significant damage should result, for a typical skin thickness of 5 mm.
4.5. **Non destructive testing**

After impact, and after each loading event, all elements were C-scanned with full waveform capture being carried out, in order to verify the ply in which the critical delamination occurred; Figure 35. A focused probe operating at 5 MHz was used. The resolution of the scan was set to 0.3 mm and the area of the scan limited to 60 mm by 60 mm, centred directly over the impact site, as this was sufficient to determine the delamination size. The only form of damage that could be detected by scanning was delamination; any matrix cracking or fibre fracture occurring could not be detected. However, as delamination growth is the growth mechanism of most concern, further analysis was not necessary. From the scan it was evident that the largest delamination was approximately 1 ply from the internal face of the panel, at the +45 / 0 ply interface (Figure 35).

![Image](image_url)

Largest delamination is coloured purple and is ~0.3mm from the internal face. Each ply is 0.27mm thick, therefore the delamination lies between the first and second ply from the internal face.

*Figure 35 – Through thickness C-Scan showing delamination pattern - Location of largest delamination between the -45 / 0 ply, one ply from the internal face*
4.6. Experimental arrangement

All structural elements were mounted on a 500 kN, four actuator, multi-axial test machine, with two parallel platens acting in the x-direction, and a third actuator, perpendicular to the x-direction. This third actuator was used in order to apply an out-of-plane displacement to the structural element, to induce bending of the panel. The bending was required in order to produce a shear across the delamination. The fourth actuator was disabled.

Prior to applying any significant loads to the panels, a layer of shimming epoxy was applied between the channelling on the panel and the platen, to remove any irregularities in the contact surfaces and to ensure an even load distribution across the panel. The platens, in the x-direction, were used to apply the in-plane load to the panel and the third actuator (z direction) was used to apply an out-of-plane displacement to the panel. A photograph of the set-up can be seen in Figure 36.

Figure 36 – Photograph of test set-up showing both in-plane actuator (x) and out-of-plane load application tup (z) with blade stiffened panel in position
A number of structural elements have been tested. The details of the strain gauge arrangement, loading arrangement and strain gauge results follows.

4.7. **Loading arrangement and results of element testing**

A description of the loading arrangement for each element tested is provided within this section. The table below provides details for the loading arrangement for each test.

<table>
<thead>
<tr>
<th>Element</th>
<th>Load application order</th>
<th>Position of out of plane application</th>
<th>Type of applicator head</th>
<th>In-plane strain – out of plane displacement range</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>In plane load then out of plane displacement</td>
<td>External face centre of panel</td>
<td>Single hemispherical head</td>
<td>1-4mm displacement, 2600µe</td>
</tr>
<tr>
<td>2</td>
<td>In plane load then out of plane displacement</td>
<td>External face, centre of panel</td>
<td>Single hemispherical head</td>
<td>0.5 – 2.5mm displacement, 4200 – 4400 µe</td>
</tr>
<tr>
<td>2</td>
<td>In plane load then out of plane displacement</td>
<td>External face, centre of panel</td>
<td>Spreader plate</td>
<td>1.5mm out of plane, 4400µe</td>
</tr>
<tr>
<td>2</td>
<td>Out of plane displacement then in plane load</td>
<td>External face, centre of panel</td>
<td>Spreader plate</td>
<td>3-5mm out of plane, 4000 – 5400 µe</td>
</tr>
<tr>
<td>2</td>
<td>Out of plane displacement then in plane load</td>
<td>Internal face</td>
<td>Centre of blade</td>
<td>2mm out of plane, 4300 µe</td>
</tr>
<tr>
<td></td>
<td>Loading arrangement</td>
<td>Displacement and Load Details</td>
<td></td>
<td></td>
</tr>
<tr>
<td>---</td>
<td>------------------------------------------------------------------------------------</td>
<td>-----------------------------------------------------------------------------------------------</td>
<td></td>
<td></td>
</tr>
<tr>
<td>3</td>
<td>Slight in plane load, then out of plane displacement followed by greater in plane load</td>
<td>Internal face 2 loading points either side of blade foot 2mm out of plane, 4400 με</td>
<td></td>
<td></td>
</tr>
<tr>
<td>3</td>
<td>Slight in plane load, then out of plane displacement followed by greater in plane load</td>
<td>Internal face 2 loading points either side of blade foot 2mm out of plane, 4400 με, fatigued to 10000 cycles</td>
<td></td>
<td></td>
</tr>
<tr>
<td>4</td>
<td>Out of plane displacement then in plane load</td>
<td>Internal face 1 loading point on blade foot on side of panel with impact 2 – 2.5mm out of plane, 4000 με in plane, fatigued to 1,100000 cycles</td>
<td></td>
<td></td>
</tr>
<tr>
<td>5</td>
<td>Out of plane displacement then in plane load</td>
<td>Internal face 1 loading point on blade foot on side of panel with impact 4200με in plane, 2.5mm out of plane, fatigued to 11,111 cycles</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

*Table 12 – Loading arrangement for each loading situation*

A selection of the load versus displacement responses for various loading arrangements along with the corresponding strain versus load response have also been reported. The strain responses reported were for strain in the x direction, with a negative strain indicating compressive strain and a positive strain indicating a tensile response. Where possible, a description of the buckle mode observed during the testing has also been described.
Element 1

A number of single cycle tests were carried out on the first structural element (element 1) with varying degrees of in-plane strain being applied first, followed by an out-of-plane displacement being applied to the external face, in the centre of the panel. The strain gauge arrangement for this element, for all loading situations can be seen in Figure 37. The strain gauges consisted of a single foil type gauge with a gauge length of 5 mm aligned to the x-axis of the panel. The location of gauge 1 was chosen to be 25 mm from the delamination centre. The approximate radius of the delamination was found to be 16 mm and therefore the strain gauge was positioned to capture the response of the material in close proximity to the delaminated region.

![Figure 37 – Schematic of strain gauge arrangement for Element 1](image)

The results for this loading arrangement can be seen in (Table 13).
### Table 13 – Loading results for Element 1

For this structural element test, an in-plane load was applied, until a strain of 2500 με was detected in gauge 1e, resulting in a load of approximately 180 kN. Following this, an out-of-plane displacement of 1.8 mm was applied. The load versus displacement graph for this element can be seen in Figure 38.

![Graph of load vs. displacement](image)

**Figure 38** – Load x versus displacement in y axis (load case 1)
The load versus strain graph for the test described above can be seen in Figure 39.

*Figure 39 – Load versus strain data after 2500με applied in-plane, followed by 1.8 mm out-of-plane displacement (load case 1)*

From the strain gauge responses of this element, it can be seen that up to point A (Figure 39), the element experienced uniform loading before globally buckling, as indicated by gauge number 1e located on the external face becoming more negative and gauge number 1i located on the internal face, becoming more positive.

*Figure 40 – Strain gauge response for Element 1 (Loading to 2500με in-plane, followed by 1.8 mm out-of-plane displacement applied to external face in centre of element)*
Following this initial test, further loading was applied to the same element, with again 2500 $\mu\varepsilon$ being applied in-plane, however an increase in out-of-plane displacement was required as no damage growth was detected after the initial loading case. The out-of-plane displacement was increased to 2.7 mm, as can be seen in Figure 41.

![Figure 41 – Load versus displacement in y axis (load case 2)](image)

The load versus strain graph for the second load case can be seen in Figure 42.

![Figure 42 – Load versus strain response after 2500 $\mu\varepsilon$ applied in-plane, followed by 2.7 mm out-of-plane displacement (load case 2)](image)
Again, it became apparent that no delamination growth had occurred and therefore the out-of-plane displacement was increased further, as can be seen in Figure 43.

![Figure 43 - Load versus displacement in y - axis (load case 3)](image)

Again, as with the previous loading cases, the load was plotted against strain (Figure 44).

![Figure 44 - Load versus strain response after 2500 µε applied in-plane, followed by 3.6 mm out-of-plane displacement (load case 3)](image)
The first significant observation under this loading situation was that in order to avoid fibre fracture at the location of the point loading, the maximum out-of-plane displacement would need to be limited to 3 mm. This was observed early on in the study, with fibre fracture on the internal face being detected at the radius between the blade and the foot.

Due to the first element experiencing fibre fracture at the loading point, further loading was applied to a second element.

Element 2
A considerable amount of testing was carried out on this element, however, not all the test data has been reported as no significant changes were detected from previous loading cases. The first loading arrangement entailed loading to approximately $4200 \, \mu \varepsilon$ in-plane, followed by an out-of-plane displacement being applied. The out-of-plane displacement ranged from 0.5 mm through to 2.5 mm, in increments of 0.5 mm (Table 14).

![Table 14 - Results for element 2 with approximately $4200 \, \mu \varepsilon$ in-plane strain being applied prior to out-of-plane displacement on external face in the centre of the panel](image)

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In order to show the strain gauge response for this loading arrangement, two of the loading cases have been selected for reference, the 1 mm out-of-plane and 2.5 mm out-of-plane case, as highlighted in red in Table 14.

The strain gauge arrangement for these elements can be seen in Figure 45.

![Figure 45 - Schematic of strain gauge arrangement for Element 2 – Strained to 4200 με in-plane](image)

As with element 1, the in-plane load was applied to a strain of 4200 με, an out-of-plane displacement of approximately 1 mm was then applied, as can be seen in Figure 46.

![Figure 46 - In-plane load response versus out-of-plane displacement of element](image)
The strain versus load graph for the first reported loading (highlighted red in Table 14) can be seen in Figure 47.

Upon examining the results, it appears that there is some local bending between points A and B in Figure 47, before it globally buckles at point B. The strain response indicates some shear deformation through the thickness, which seems to be the largest at point C. It was apparent that the buckling mode was identical to that observed by the first element. The diagram below (Figure 48) represents the strain response across the panel.
However upon scanning the element, it was apparent that no delamination growth had occurred and therefore the out-of-plane displacement that was applied to the external face was increased in increments of 0.5 mm until a maximum of 2.5 mm was applied, in an attempt to produce a greater shear load across the delamination. Figure 49 and Figure 50 show the loading and strain response of the element respectively for load case 2.5, where an out-of-plane displacement of 2.5 mm was applied.

*Figure 48 – Strain response of element 2, with 4200 $\mu\varepsilon$ in-plane, followed by 1 mm out-of-plane displacement*

*Figure 49 – Loading response of element 2 - loaded to 4200 $\mu\varepsilon$ in-plane, followed by 2.5 mm out-of-plane displacement being applied to the centre of the panel on the external face*
Figure 50 - Strain versus load response of element 2 – loaded to 4200 $\mu$ε in-plane, followed by 2.5 mm out-of-plane displacement being applied to the centre of the panel on the external face.

The response of the strain gauges after a 2.5 mm out-of-plane displacement was applied was the same as after a 1 mm out-of-plane displacement, as shown in Figure 51. However, the shear component closest to the delaminated region, as indicated by gauges 1i and 1e appeared to have reduced, which, as highlighted within the literature review would hinder the delamination growth.

Figure 51 - Strain response of element 2, with 4200 $\mu$ε in-plane, followed by 2.5 mm out-of-plane displacement.
Reviewing the C-scans revealed that as predicted, delamination growth had not occurred. As a result of this, the loading arrangement was altered, with an increase in in-plane strain being applied, prior to applying the out-of-plane displacement, in order to try to increase the shear component. Table 15 shows the loading applied with the increased in-plane strain.

<table>
<thead>
<tr>
<th>Element Number</th>
<th>Number of cycles</th>
<th>Max load (x)</th>
<th>In-plane strain (max)</th>
<th>In-plane strain other side of blade</th>
<th>Strain gauge diagram</th>
<th>Out of plane displacement (actuator)</th>
</tr>
</thead>
<tbody>
<tr>
<td>2</td>
<td>1</td>
<td>204</td>
<td>-4528</td>
<td>gauge 3</td>
<td>Figure 45</td>
<td>0.2</td>
</tr>
<tr>
<td></td>
<td></td>
<td>201</td>
<td>-4400</td>
<td>gauge 3 unknown unknown</td>
<td>Figure 45</td>
<td>0.5</td>
</tr>
<tr>
<td></td>
<td></td>
<td>205</td>
<td>-4400</td>
<td>gauge 3 unknown unknown</td>
<td>Figure 45</td>
<td>1</td>
</tr>
<tr>
<td></td>
<td></td>
<td>201</td>
<td>-4414</td>
<td>gauge 3</td>
<td>Figure 45</td>
<td>1.49</td>
</tr>
</tbody>
</table>

*Table 15 - Results for element 2 with approximately 4400 με in-plane being applied prior to out-of-plane displacement on external face in the centre of the panel*

As with the previous loading arrangement, two of the strain and load responses were analysed, highlighted in red in Table 15.

The load response for the first loading can be seen in Figure 52, with the strain response in the x-direction being applied in *Figure 53*.
Figure 52 – Loading to 4400 µε in-plane, followed by a 0.2 mm out-of-plane displacement being applied to the external face in the centre of the element.

Figure 53 - Strain versus load response of element 2 – loaded to 4400 kN in-plane, followed by 0.2 mm out-of-plane displacement being applied to the centre of the panel on the external face.
In comparison to Figure 47, point A on Figure 53 shows minor differences in strain from gauge 1i and 1e, therefore indicating that the shear element across the delamination was only small. As a result of this, delamination growth would be unexpected. Upon scanning the element at this stage no delamination growth was observed, however, it was apparent that in order to avoid point loading the panel and causing fibre fracture, a load spreader plate would be required, between the loading head and the element. Therefore all subsequent loading included the use of the spreader plate.

The final loading with this arrangement, had 4400 $\mu$e applied in-plane, followed by 1.5 mm out-of-plane displacement being applied to the external face of the element. The load verses out-of-plane displacement response for this arrangement can be seen in Figure 54, with the strain in the x-direction response being shown in Figure 55.

![Figure 54](image.png)

*Figure 54 - Loading to 4400 $\mu$e in-plane, followed by a 1.5 mm out-of-plane displacement being applied to the external face in the centre of the element*
Figure 55 - Strain versus load response of element 2 – loaded to 4400 με in-plane, followed by 1.5 mm out-of-plane displacement being applied to the centre of the panel on the external face.

As with previous loading cases for this element no delamination growth was observed; however, it was apparent that although the delamination size overall had not grown, full waveform capture NDE revealed that there were in fact some competing mechanisms occurring (Figure 56).

Figure 56 – Full waveform capture C-Scan, of element 2, after loading to 4400 με in-plane followed by 1.5 mm out-of-plane displacement.
It became apparent that as one delamination plane opened, under the influence of loading, the critical delamination plane closed, resulting in no obvious delamination growth. It was therefore necessary to change the loading arrangement to try to change the buckle shape in order to grow the critical delamination (one ply down from the internal face).

Further loading was carried out on Element 2, with the out-of-plane displacement being applied first and then the in-plane loading being applied (see Table 16).
<table>
<thead>
<tr>
<th>No of cycles</th>
<th>Max load (x)</th>
<th>In-plane strain (max)</th>
<th>In-plane strain other side of blade</th>
<th>Strain gauge diagram</th>
<th>Out-of-plane displacement (actuator)</th>
<th>Out-of-plane displacement (LVDT 1)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>kN</td>
<td>με</td>
<td>με</td>
<td></td>
<td>mm</td>
<td>mm</td>
</tr>
<tr>
<td>1</td>
<td>180 - 4054</td>
<td>gauge 3</td>
<td>-1855 gauge 1</td>
<td>Figure 45</td>
<td>2</td>
<td>3.4</td>
</tr>
<tr>
<td></td>
<td>198 - 4400</td>
<td>gauge 3</td>
<td>unknown unknown</td>
<td>Figure 45</td>
<td>2</td>
<td>3.76</td>
</tr>
<tr>
<td></td>
<td>200 - 4800</td>
<td>gauge 3</td>
<td>unknown unknown</td>
<td>Figure 45</td>
<td>2</td>
<td>4.64</td>
</tr>
<tr>
<td></td>
<td>208 - 5400</td>
<td>gauge 3</td>
<td>-2128 gauge 1</td>
<td>Figure 45</td>
<td>2</td>
<td>4.43</td>
</tr>
<tr>
<td></td>
<td>209 - 5381</td>
<td>gauge 3</td>
<td>-2494 gauge 1</td>
<td>Figure 45</td>
<td>2</td>
<td>5.53</td>
</tr>
</tbody>
</table>

*Table 16 – Loading of element 2 with out-of-plane displacement applied to external face in the centre of the panel prior to in-plane loading*

However, the buckle shape did not change, from that observed under the previous loading. The loading was therefore altered (Table 17), with the out-of-plane displacement being applied to the internal face, (representing aerodynamic loading of the wing). The loading was applied to the centre of the blade (*Figure 57*).
### Table 17 - Loading of Element 2 with out-of-plane displacement applied to internal face prior to in-plane loading

The strain gauge arrangement for this loading can be seen in Figure 57.

![Figure 57 - Schematic of strain gauge arrangement for Element 2 – final loading case](image-url)

<table>
<thead>
<tr>
<th>Element Number</th>
<th>Number of cycles</th>
<th>Max load (x)</th>
<th>In-plane strain (max)</th>
<th>In-plane strain other side of blade</th>
<th>Strain gauge diagram</th>
<th>Out-of-plane displacement (actuator)</th>
<th>Out-of-plane displacement (LVDT 1)</th>
</tr>
</thead>
<tbody>
<tr>
<td>2</td>
<td>1</td>
<td>210</td>
<td>-4105 gauge 6</td>
<td>-4312 gauge 8</td>
<td>Figure 57</td>
<td>1.99</td>
<td>4.6</td>
</tr>
</tbody>
</table>
The loading response to element 2, for this final loading situation can be seen in Figure 58.

![Displacement Vs Load Graph](image.png)

*Figure 58 – Applying a 2 mm out-of-plane displacement to the centre of the internal face, on the blade tip to element 2, before applying 4400 με in-plane strain.*

The strain versus load history for this final loading arrangement can be seen in Figure 59.
Applying a 2 mm out-of-plane displacement to the centre of the internal face, on the blade tip to element 2, before applying 4400 \( \mu \varepsilon \) in-plane strain.

Under this loading arrangement the buckle mode switched, as indicated by the strain gauges (Figure 60).

Figure 60 – Schematic of strain response after buckling of element 2 – loading on internal face on centre of blade.
In order to clarify the shear contribution through the thickness, Figure 61 highlights the gauge responses of interest.

On reviewing the strain response in Figure 61, it is important to first report that an offset in gauges 1i and 1e occurred, with localised buckling being indicated upon applying the out-of-plane displacement. Applying the in-plane load after this initial localised buckling had occurred, resulted in a large shear through the thickness, close to the delamination location. The loading arrangement appeared to be consistent with that required for delamination growth. Figure 61 indicates that maximum growth should occur at a load of approximately 170 kN, where the shear through the thickness was greatest.

Upon reviewing the ultrasonic C-scans taken after unloading, it became apparent that growth was observed at 4400 µε with 2 mm out-of-plane displacement being applied, as indicated in Figure 62. The delamination that appeared to have grown can be seen in purple in Figure 62, having grown from 32 mm in length to 40 mm in length. This delamination was found to be approximately 0.3 mm from the internal face of the element, at the interface of the +45° ply and the 0° ply.
However, damage to the blade occurred under this loading situation due to point loading the blade, therefore an alternative loading mechanism was manufactured, which had a dual loading head.

**Element 3**

In order to obtain a similar buckling response to the panel which enabled delamination growth to be observed (element 2), the third element was tested with two loading points, on the blade foot, either side of the blade. The strain gauge arrangement for this element can be seen in Figure 63.
The first loading arrangement for element 3 involved loading the panel with pure in-plane loading, to determine which direction the panel buckled. The second loading situation involved loading the element to 10 kN in-plane, followed by applying a 2 mm out-of-plane displacement and subsequently loading to 4400 µε in-plane (Figure 64 and Figure 65).
Figure 65 – Strain versus load response of element 3, with 2 mm out of plane displacement applied either side of blade, followed by 4400 $\mu$ε in -plane strain

As with the previous loading cases, the response of the strain gauges has been indicated on the schematic of the element in order to quantify the buckle response of the panel (Figure 66). In comparison to Figure 60, it became apparent that the buckling response of element 3 may not be appropriate to cause delamination growth.

Figure 66 – Strain gauge responses of element 3
Further loading was carried out on element 3 (Table 18), with fatigue cycles being applied in order to initiate delamination growth; however, no delamination growth was observed.

<table>
<thead>
<tr>
<th>Element Number</th>
<th>No of cycles</th>
<th>Max load (x)</th>
<th>In plane strain (max)</th>
<th>Strain gauge diagram</th>
<th>Out of plane displacement (actuator)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>kN</td>
<td>με</td>
<td></td>
<td>mm</td>
</tr>
<tr>
<td>3</td>
<td>1</td>
<td>212</td>
<td>4600</td>
<td>gauge 1e</td>
<td>Figure 46</td>
</tr>
<tr>
<td></td>
<td>1</td>
<td>264.25</td>
<td>4400</td>
<td>gauge 3i</td>
<td>Figure 46</td>
</tr>
<tr>
<td></td>
<td>1000</td>
<td>267.70</td>
<td>4400</td>
<td>gauge 5i + 5e</td>
<td>Figure 46</td>
</tr>
<tr>
<td></td>
<td>10000</td>
<td>250.0</td>
<td>4400</td>
<td>gauge 3i</td>
<td>Figure 46</td>
</tr>
<tr>
<td></td>
<td>1</td>
<td>143.75</td>
<td>4400</td>
<td>gauge 3i</td>
<td>Figure 46</td>
</tr>
</tbody>
</table>

*Table 18 – Loading of element 3*

It was suspected that loading the element either side of the blade resulted in a restriction of the localised buckling observed from the previous element and, as a result, the shear component through the thickness was not observed under this loading arrangement, resulting in a no-growth situation.

**Element 4**

A fourth element was tested with the loading arrangement modified to ensure the correct buckle shape, without point-loading the blade. This was achieved by applying the out-of-plane load to the blade foot, on the side with the impact. A loading head was designed to provide the out-of-plane displacement. The strain gauge arrangement for element 4 can be seen in Figure 67.
Once the loading arrangement was established to ensure that the buckle shape was appropriate to cause growth, various levels of loading were carried out on element 4 (Table 19), both for the single and multiple cycle situations. Under fatigue loading, the loading was purely compression, as this has been reported to be the most likely loading arrangement to cause delamination growth. An R-ratio of -10 was used. After each cyclic loading sequence, a compliance loading was carried out. This was used to determine the starting amplitude for the next fatigue loading.

<table>
<thead>
<tr>
<th>Element Number</th>
<th>No of cycles</th>
<th>Max load (x)</th>
<th>In plane strain (max)</th>
<th>Strain gauge diagram</th>
<th>Out of plane displacement (actuator)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>kN</td>
<td>με</td>
<td></td>
<td>mm</td>
</tr>
<tr>
<td>4</td>
<td>1</td>
<td>188</td>
<td>-4047</td>
<td>gauge 1i</td>
<td>Figure 65</td>
</tr>
<tr>
<td></td>
<td>18,078</td>
<td>169</td>
<td>-4054</td>
<td>gauge 1i</td>
<td>Figure 65</td>
</tr>
<tr>
<td></td>
<td>110,001</td>
<td>182</td>
<td>-4052</td>
<td>gauge 1i</td>
<td>Figure 65</td>
</tr>
<tr>
<td></td>
<td>1,100,001</td>
<td>208</td>
<td>-4029</td>
<td>gauge 1i</td>
<td>Figure 65</td>
</tr>
<tr>
<td></td>
<td>1</td>
<td>143.75</td>
<td>-4014</td>
<td>gauge 1i</td>
<td>Figure 65</td>
</tr>
</tbody>
</table>

*Table 19 – Loading results for element 4*
Figure 68 – Load versus out-of-plane displacement response of element 4, loaded 2 mm displacement out-of plane on blade foot (side with impact), followed by 4000 με in-plane strain application.

Figure 69 – Strain versus load response of element 4 - out-of plane displacement of 2 mm applied on blade foot (side with impact), followed by 4000 με in-plane load.
It should also be noted that on application of the out-of-plane displacement, a strain in excess of 1000 $\mu$ε was observed in gauge 1. The fact that the out-of-plane load was applied off the centre line (on the blade foot, on the impact side of the panel) created a slight twist in the panel, which resulted in the correct buckle shape, but as a result, the strain response either side of the blade was not the same. The gauges on the opposite side of the blade had a smaller initial strain, due to the out-of-plane load, as the panel did not buckle as significantly as the side with the impact. The element was fatigue cycled, and the load-strain response after 1,000,000 cycles can be seen in Figure 70.

Figure 70 – Strain versus load response of element 4 after 1,000,000 cycles

Figure 70 shows that the buckle shape was correct to cause delamination growth, buckling away from the web. Although the buckle shape was correct, the strain and out-of-plane displacement levels were not large enough to initiate the growth (Figure 71, Figure 72 and Figure 73).
After the cyclic loading had been carried out for 1,000,000 cycles, the aluminium channelling fractured, and the element had to be re-end-potted. The same element was then loaded to 2.5 mm out-of-plane, with 4000 με in-plane load being applied. It was observed during the test that the buckle shape appeared to be inconsistent with the previous loading case. Upon scanning, there were obvious signs of further damage being present within the existing damaged area. Figure 71, Figure 72 and Figure 73 show a region coloured white which indicates additional damage. However, on closer inspection, it became clear that the additional damage was in the form of back face splitting.

*Figure 71 – Element 4, full waveform capture after 35 J impact (scanned from the internal face). The white region indicates the largest critical delamination, 1 ply down from the internal face*
Figure 72 – Element 4, full waveform capture after 1,000,000 cycles at 4000 με in-plane 2 mm out-of-plane (scanned from the internal face)

Figure 73 – Element 4 after completing 1,000,000 cycles at 4000 με in-plane, 2 mm out-of-plane, re-potting and loading to 4000 με in plane, 2.5 mm out of plane
Element 5

The strain gauge arrangement for the final element tested can be seen in Figure 74.

![Strain gauge arrangement for element 5](image)

*Figure 74 – Strain gauge arrangement for element 5*

The loading conditions for element 5 can be seen in Table 20. This final element was loaded to approximately 4200 $\mu\varepsilon$ in-plane followed by 2.5 mm out-of-plane displacement and then fatigue cycled.

<table>
<thead>
<tr>
<th>Element Number</th>
<th>No of cycles</th>
<th>Max load (x)</th>
<th>In plane strain (max)</th>
<th>Strain gauge diagram</th>
<th>Out of plane displacement (actuator)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td>kN</td>
<td>$\mu\varepsilon$</td>
<td></td>
<td>mm</td>
</tr>
<tr>
<td>5</td>
<td>1</td>
<td>178</td>
<td>-4196</td>
<td>Gauge 1i</td>
<td>Figure 74</td>
</tr>
<tr>
<td></td>
<td>110</td>
<td>180</td>
<td>-4249</td>
<td>Gauge 1i</td>
<td>Figure 74</td>
</tr>
<tr>
<td></td>
<td>1,111</td>
<td>176</td>
<td>-4200</td>
<td>Gauge 1i</td>
<td>Figure 74</td>
</tr>
<tr>
<td></td>
<td>11,111</td>
<td>194</td>
<td>-4351</td>
<td>Gauge 1i</td>
<td>Figure 74</td>
</tr>
</tbody>
</table>

*Table 20 – Loading history of element 5*
The single cycle loading response and strain response can be seen in Figure 75 and Figure 76 respectively.

Figure 75 – Load versus out-of-plane displacement of element 5, with 2.5 mm out-of-plane displacement applied to blade foot (side with impact) followed by 4200 με in-plane strain

Figure 76 – Strain versus load profile for element 5, with 2.5 mm out-of-plane displacement applied followed by in-plane strain of 4200 με
On analysing the ultrasonic C-scans, immediately after impact (Figure 77), after 100 cycles (Figure 78), and after 11,111 cycles (Figure 79), it appeared that either delamination growth had occurred, or the critical delamination was opening up and becoming more significant, whilst the less significant delaminations closed.

**Figure 77** – Full-waveform capture of Element 5 after a 35 J impact

**Figure 78** - Full waveform capture of Element 5, after cyclic loading for 100 cycles at 4200 \(\mu\varepsilon\) in plane, 2.5 mm out-of-plane
Figure 79 - Full waveform capture of Element 5, after cyclic loading for 11,111 cycles at 4200 \( \mu \varepsilon \) in plane, 2.5 mm out-of-plane

In order to determine if delamination growth had occurred, the C-scans were analysed and delamination areas broken into sections as indicated by the areas identified in Figure 77, Figure 78 and Figure 79. The area of each section of the delamination was calculated and is shown in Table 21.
<table>
<thead>
<tr>
<th>Figure Number</th>
<th>Area number</th>
<th>Average area size (mm²)</th>
<th>Total area of critical delamination visible (mm²)</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Figure 77</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>1</td>
<td>555</td>
<td></td>
<td></td>
</tr>
<tr>
<td>2a</td>
<td>53.7</td>
<td></td>
<td>53.8</td>
</tr>
<tr>
<td><strong>Total area</strong></td>
<td></td>
<td></td>
<td>608.7</td>
</tr>
<tr>
<td><strong>Figure 78</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>1</td>
<td>547.3</td>
<td></td>
<td></td>
</tr>
<tr>
<td>2a</td>
<td>56.2</td>
<td></td>
<td></td>
</tr>
<tr>
<td>2b</td>
<td>1.88</td>
<td></td>
<td>60.52</td>
</tr>
<tr>
<td>2c</td>
<td>2.44</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Total area</strong></td>
<td></td>
<td></td>
<td>607.82</td>
</tr>
<tr>
<td><strong>Figure 79</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>1</td>
<td>513.3</td>
<td></td>
<td></td>
</tr>
<tr>
<td>2a</td>
<td>56.7</td>
<td></td>
<td></td>
</tr>
<tr>
<td>2b</td>
<td>5.05</td>
<td></td>
<td>67.18</td>
</tr>
<tr>
<td>2c</td>
<td>5.43</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Total area</strong></td>
<td></td>
<td></td>
<td>580.48</td>
</tr>
</tbody>
</table>

*Table 21 - Analysis of C-scans for Element 5, loaded to 4200 με with 2.5 mm out-of-plane displacement after 1 cycle, 100 cycles and 11,111 cycles*

This has shown that although on first inspection, the delamination appeared to have grown, the overall delamination area had not grown, and instead, as the less critical delaminations were closing and becoming less significant, the critical delamination, 1 ply down from the external face, became more critical.

### 4.8. Summary

A number of stiffened structural elements have been manufactured and tested. This component of the study has highlighted the complexity of the relationship between loading and deformation in CFRP structures and the need to consider both in-plane and out-of-plane loading. The specimen
geometry selected for this study permitted in-plane loading up to 4500 με whilst permitting out-of-plane displacements up to 3.5 mm. The impact event of 35 J provided representative damage; however, no in-plane load was applied during the impact event. The resulting damage was therefore a mixture of small delaminations, with the largest being located near to the back face, furthest from the impact site. This type of damage is typical of a flexible response where the greatest damage is created by shear. When this type of damage is placed under in-plane loading the largest delamination may not be at the critical interface and smaller delaminations may grow, often shielded from NDE inspection by the larger more dominant defect. This damage will continue to grow until it becomes larger than the damage shielding it. Consequently, care must be taken when interpreting NDE scans of damage, particularly with ultrasonic’s, as non-critical defects subjected to fatigue loading will be subject to fretting that will give the impression of the delamination becoming smaller. In fact, this is just because the resulting crack tip gives a better attenuation.

The table below summarises the loading conditions and observations of buckle shape and whether growth of delaminations was observed.

<table>
<thead>
<tr>
<th>Element Number</th>
<th>Out of plane loading position</th>
<th>Observations during testing</th>
<th>Delamination observation</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>In plane load followed by out of plane displacement to external face</td>
<td>Global buckling occurred during in-plane loading prior to out of plane displacement being applied.</td>
<td>No delamination growth occurred for all 3 tests up to 3mm out of plane displacement and 2600με in plane strain. Fibre fracture occurred on the surface at 3mm out of plane displacement</td>
</tr>
<tr>
<td>2</td>
<td>In plane load followed by out of plane</td>
<td>Local buckling before global buckling. Some</td>
<td>No delamination growth</td>
</tr>
<tr>
<td>2</td>
<td>Out of plane displacement followed by in-plane load on external face</td>
<td>Only small shear element seen from gauge response</td>
<td>No overall delamination growth occurred but review of C-Scans showed that small delaminations through the thickness grew whilst the larger delamination appeared to get smaller due to kissing bonds.</td>
</tr>
<tr>
<td>2</td>
<td>Out of plane displacement applied to internal face on centre of blade followed by in-plane load</td>
<td>Localised buckling occurred on applying out of plane displacement, applying an in plane load after initial localised buckling resulted in large shear through the Growth occurred at 4400 μe with 2mm out of plane displacement. Damage to blade occurred.</td>
<td></td>
</tr>
</tbody>
</table>
thickness close to the delamination region

| 3 | Slight in plane load followed by out of plane displacement applied either side of blade on blade foot on internal face followed by greater in plane load. Fatigue cycling followed static tests | No significant buckling occurred. Loading either side of blade may have restricted the localised buckling and shear component was not observed | No growth |

| 4 | Out of plane displacement applied on blade foot on side with impact on internal face followed by in plane load. Panel was fatigue loaded | Panel twisted due to off centre application of out of plane displacement – panel buckled away from the web. | Strain and displacement not large enough for growth – back face splitting occurred. |

| 5 | 2.5mm out of plane displacement applied on blade foot on side of impact followed by an in-plane | Shear through the thickness and global buckling | Growth observed. |
Table 22 shows that in order for delamination to grow the elements need to be subjected to the correct loading conditions in order for local buckling to induce shear and for global buckling to drive delamination growth. Growth of delaminations was observed and was captured using the C-scanning technique, however, the study emphasised the importance of reviewing the C-Scans carefully as some smaller delaminations resulting from the impact even grew whereas the larger delaminations appeared to reduce in size due to kissing bonds.

The experimental study demonstrated a method to provide vital information, which can be used to validate an engineering approach to delamination growth prediction. The specimens selected for this study have assisted in validating both the importance of out-of-plane displacement and the complexity of damage growth within composites structures.
5. Concluding Remarks and Recommendations

One of the primary aims of the study was to determine if standard analysis can be used and adapted to determine interlaminar fracture toughness values in fibre reinforced composite samples with layup and damage which are more representative of those seen in service than the traditional unidirectional coupon. The coupon testing conducted within this study has provided some interesting results and the following can be concluded:

- The limited scatter in data between samples tested in both mode I and mode II loading gives confidence that the results are valid.
- For mode I loading, the derivation for the off centre analysis for the initiation load for delamination growth was in good agreement with experimental results and as such, with further investigation, this method could be adapted to determine the initiation load for delaminations within any plane.
- For mode I loading, the initiation load to grow delaminations arising from an impact event is greater than that from an artificial insert. This finding is believed to be due to the applied load being distributed over multiple delamination crack fronts and as such a greater load is required to grow the critical delamination.
- For mode II loading, crack migration within the NCF sample was limited and the modified analysis, which takes into account the position of the delamination growth, could be used. As such, as long as the position of the crack was noted throughout the test, this analysis method could be used for delamination at any point throughout the sample thickness.

An additional aim of the study was to investigate and carry out novel experimental testing techniques to understand damage behaviour at the structural level. The significant findings from the structural study are as follows:

- There is a complex relationship between loading and deformation in CFRP structures and the consideration of both in-plane and out-of-
plane loading is required if deformation is to be fully understood at a structural level.

- Interpretation of the C scanning results is complex, as damage growth can be hidden, below a more prominent delamination, masking the damage mechanism occurring and in some instances, damage that is known to be present may appear smaller on the c-scan than in reality.

It is recommended that further testing of coupons in mode I and mode II is carried out on NCF materials so that fractographic studies can be carried out in order to analyse the fracture surface and confirm that limited crack migration occurs.

An additional recommendation is for further tests to be carried out on impacted coupons in order to obtain more results to validate the approach and to provide additional samples for fractographic investigation.
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Appendix 1

Derivation of Mode II Modified beam analysis

If the section of the Mode II sample from the point at which the deflection is taken to be zero and the slope is zero is considered, then

\[ M_x + \frac{p}{2} x = \frac{p}{2} (L - d) \]

\[ M_x = \frac{p}{2} (L - d - x) = 0 \]

\[ EI \frac{d^2 y}{dx^2} = -M \]

\[ EI \frac{d^2 y}{dx^2} = - \frac{p}{2} (L - d - x) \]

Integrating twice, we find
\[ EI \frac{dy}{dx} = \frac{P}{4} (L - d - x)^2 + A \]

and

\[ Ely = -\frac{P}{12} (L - d - x)^3 + Ax + B \]

Applying boundary conditions of \( x=0, y=0 \)

\[ 0 = -\frac{P}{12} (L - d)^3 + B \]

\[ B = \frac{P}{12} (L - d)^3 \]

Applying boundary conditions of \( x=0, dy/dx=0 \)

\[ 0 = \frac{P}{4} (L - d)^2 + A \]

\[ A = -\frac{P}{4} (L - d)^2 \]

\[ Ely = -\frac{P}{12} (L - d - x)^3 - \frac{P}{4} (L - d)^2 x + \frac{P}{12} (L - d)^3 \]

\[ l = \frac{bd^3}{12} \quad d = 2h \]

\[ Eb \frac{(2h)^3}{12} y = -\frac{P}{12} (L - d - x)^3 - \frac{P}{4} (L - d)^2 x + \frac{P}{12} (L - d)^3 \]

\[ Eb(2h)^3 y = -P(L - d - x)^3 - 3P(L - d)^2 x + P(L - d)^3 \]

\[ y = \frac{P}{8Eb} [-(L - d - x)^3 - 3(L - d)^2 x + (L - d)^3] \]

At B, \( x=L-d-a \)
\[ y = \frac{P}{8Ebh} [-L - d - (L - d - a)^3 - 3(L - d)^2(L - d - a) + (l - d)^3] \]

\[ y = \frac{P}{8Ebh} [-a^3 - 3(L - d)^2(L - d - a) + (l - d)^3] \]

\[ y = \frac{P}{8Ebh} [-2(L - d)^3 + 3a(L - d)^2 - a^3] \quad (1) \]

Working out the slope at the start of the split

\[ \left( \frac{dy}{dx} \right)_B = \frac{P}{EI} \left[ \frac{(L - d - x)^2}{4} - \frac{(L - d)^2}{4} \right] \]

\[ X = L - d - a \quad \text{where} \quad d = 2h \]

\[ \left( \frac{dy}{dx} \right)_B = \frac{3P}{8Ebh^3} [a^2 - (L - d)^2] \quad (2) \]

For both (1) and (2), \( y \) has been set as positive downwards

Therefore \( \Delta_{EB} = \frac{P}{8Ebh^3} [2(L - d)^3 - 3a (L - d)^2 + a^3] \)

\[ \theta_B = \frac{3P}{8Ebh^3} [(l - d)^2 - a^2] \]

In terms of \( \Delta_{AB} \) this is made up of \( \Delta_{AB1} (a \theta_B) + \Delta_{AB2} (tip \ deflections) \)

Consider the half of the split beam

\[ \Delta_{AB2} = \left( \frac{P}{4} \right) \frac{a^3}{3E} \left( \frac{bh^3}{12} \right) = \frac{Pa^3}{Eb^3} \]
\[ P_1 + P_2 = P / 2 \]
\[ \delta_1 = P_1 a^3 / 3EI_1, \delta_2 = P_2 a^3 / 3EI_2 \]
\[ \delta_1 = \delta_2 \Rightarrow P_2 = P_1 \left( t_2 / t_1 \right)^3 \]
Therefore \( P_1 \left[ 1 + \left( t_2 / t_1 \right)^3 \right] = p/2 \)

\[ \Delta A_{B_2} = \frac{P}{2} \left[ 1 + \left( \frac{t_2}{t_1} \right)^3 \right] \frac{a^3}{3Eb t_1^3} \]
therefore \[ \Delta A_{B_2} = \frac{2Pa^3}{Ebt_1^3 + t_2^3} \]

Ignoring shear deformation

\[ \Delta_{ED} = \frac{P}{4Ebh^3} \left[ L^3 + 3L^2d - d^3 \right] \] (1)

\[ \Delta_{EC} = \frac{P}{8Ebh^3} \left[ 3Ld^2 - 2d^3 \right] \] (2)

displacement under load is \( \delta \) where \( \delta = \Delta_{EB} - \Delta_{EC} \)

\[ \delta = \frac{P}{8Ebh^3} \left[ 2L^3 + 6L^2d - 3Ld^2 \right] \] (3)

\[ \Delta_{EA} = \frac{P}{8Ebh^3} \left[ 2(L - d)^3 - 3a (L - d)^2 + a^3 \right] \]
\[ + \frac{3P}{8Ebh^3} \left[ a(L - d)^2 - a^3 \right] + \frac{2Pa^3}{Eb[t_1^3 + t_2^3]} \]
\[ \Delta_{EA} = \frac{P}{8Ebh^3}[(L - d)^3 - a^3] + \frac{2Pa^3}{Eb[t_1^3 + t_2^3]} \]

but \[ \Delta_{ED} = \Delta_{EA} \]

\[ \frac{P}{4Ebh^3}[L^3 + 3L^2d - d^3] = \frac{P}{8Ebh^3}[(L - d)^3 - a^3] + \frac{2Pa^3}{Eb[t_1^3 + t_2^3]} \]

\[ [L^3 + 3L^2d - d^3] = (L - d)^3 - a^3 + \frac{8a^3}{[t_1^3 + t_2^3]} h^3 \]

\[ 3Ld^2 - 6L^2d - a^3 + \frac{8a^3h^3}{[t_1^3 + t_2^3]} = 0 (*) \]

Check, if \( t_1 = t_2 = h \), final term is +4a^3
And we find 3Ld^2 - 6L^2d - a^3= 0 or Ld^2 - 2L^2d - a^3= 0 - (a)

In equation (3) we want to lose (6L^2d - 3Ld^2) which we can do from (*) above

\[ 6L^2d - 3Ld^2 = \frac{8a^3h^3}{[t_1^3 + t_2^3]} - a^3 \]

Therefore

\[ \delta = \frac{P}{8Ebh^3} \left( 2L^3 + \frac{8a^3h^3}{[t_1^3 + t_2^3]} \right) - a^3 \]

therefore \[ C = \frac{\delta}{p} = \frac{2L^3 + \left[ \frac{8h^3}{t_1^3 + t_2^3} - 1 \right] a^3}{8Ebh^3} \]
\[ G = \frac{P^2 \, dc}{2b \, da} \]

\[ G = \frac{3P^2a^2}{16Eb^2h^3} \left\{ \frac{8h^3}{t_1^3 + t_2^3} - 1 \right\} \]

Note when \( t_1 = t_2 \), term in brackets = 3)

Standard result \( G = \frac{9P^2a^2}{16Eb^2h^3} \)