DAMAGE TOLERANCE OF FLUSH REPAIRS OF DISSIMILAR COMPOSITE MATERIAL SYSTEMS

A thesis submitted in fulfilment of the requirements for the degree of Doctor of Philosophy

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FEBRUARY 2015
DECLARATION

I certify that except where due acknowledgement has been made, the work is that of the author alone; the work has not been submitted previously, in whole or in part, to qualify for any other academic award; the content of the thesis is the result of work which has been carried out since the official commencement date of the approved research program; any editorial work, paid or unpaid, carried out by a third party is acknowledged; and, ethics procedures and guidelines have been followed.

Jun Yi Goh
ACKNOWLEDGEMENTS

I will like to thank my supervisors at RMIT University, Professor Chun Wang, Dr Adrian Orifici and Mr Steve Georgiadis. Their support and insight that they have provided to this PhD project have been invaluable. I will also like to thank Boeing Research and Technology Australia for the opportunity of undertaking this PhD project on the damage tolerance of scarf repairs. I will also like to acknowledge funding from the Australian Research Council Linkage Grant during the duration of my PhD.

Next, I will like to acknowledge the technical support team especially Robert “Bob” Ryan, Peter Tkatchy and Julian Bradler for their guidance and support in the area of manufacturing and material testing. I will also thank RMIT Microscopy and Microanalysis Facility for the use of their equipment. Special thanks are reserved to my colleagues at the Sir Lawrence Wackett Centre, namely Dr Khomkrit Pingkarawat, Dr Shuai He, Dr Matthew Donough, Dipesh Parekh, Mildred Lee and Eugene Chan. My time as a PhD candidate would not be as delightful without the time spent in supporting each other.

Finally, I dedicate this thesis to my parents. From Foundation Studies to the Bachelor Program and on to the PhD, it has been a long journey in my pursuit for education in RMIT University, Melbourne. I would not be able to travel so far without their support helping me through times of personal distress.
PUBLICATIONS

Journal Papers


Conference Papers

SUMMARY

Current repair techniques require the repair material to be identical to the parent structure in ply thickness, orientations and material. This poses a difficulty for repair engineers and manufacturers because of the lack of parent materials or the lack of certification of the parent material to be used along structural adhesives for field repair applications. In this case, the use of materials different from the parent structures in both ply thickness and mechanical properties presents new challenges, such as mismatch in local and global stiffness. Thus, it is important to address this new problem of scarf repairs with mismatched or dissimilar adherends.

The review of literature have identified a lack of detailed understanding on fracture behaviour of composite scarf joints under static and fatigue loads. The current body of literature does not characterise the fracture behaviour of joints along the plane of the scarf, such as adhesive, cohesive and composite matrix or composite fibre fracture. There is also a lack of understanding on the fracture behaviour of composite scarf joints with dissimilar adherends. On the strength of scarf joints, the effect of disbonds on scarf joints is not understood in literature. While there have been investigations on strength tests of composite materials, there is a lack of literature that quantifies the behaviour of disbonds in composite structures, particularly scarf repairs and joints. On scarf joints with disbonds, there is a lack of literature on the damage tolerance of composite scarf joints using commercially available damage modelling tools such as the virtual crack closure technique and cohesive zone model under static loads. Furthermore, there is a lack of numerical methods on the fatigue crack growth and endurance of composite scarf joints.

The strength of secondarily bonded composite scarf joints with different bondline flaw sizes were investigated through a series of experimental testing, analytical modelling and numerical simulation. Experimental results showed that the strength for complete fracture of scarf joints with flaws is dependent on the ply angle adjacent to the crack tip and the size of the flaw. Through fractographic analysis, it has been found that the fracture of composite scarf joints occurred in the composite adherend, at a distance that is a very small fraction of the ply thickness. This failure near the composite-adhesive interface was dominated mainly by matrix shear failure in the 0º and 45º plies and matrix peel failure in the 90º plies. Numerical analyses using composite material properties along the composite-adhesive interface gave better
predictions than when adhesive properties were used. This is consistent with the experimental observation that the fracture was within the composite, rather by cohesive failure of the adhesive. For scarf joints of pristine conditions or containing flaws, the cohesive zone model was capable of accurately predicting the ultimate strength. The virtual crack closure technique and the linear elastic fracture mechanics approaches were able to provide equally accurate predictions of the ultimate strength for flaws greater than around 3 mm. The predictive model using the cohesive zone model offers a robust technique to account for the effect of disbond on the ultimate strength of scarf joints and repairs.

A numerical modelling methodology has been developed to predict the strength and crack propagation of dissimilar adherend scarf joints with disbonds. Results obtained from the numerical models correlated well with experimental results. The two tiered modelling structure proved to be a robust methodology that ensures successful prediction of dissimilar adherend scarf joints with disbonds.

Finally, a linear elastic fracture mechanics methodology has been presented and proven to adhere to airworthiness requirements by determining states of slow growth and rapid growth in scarf joints with disbonds. At crack lengths larger than a minimum disbond size ($a/L > 0.125$), the LEFM methodology has shown to be capable of predicting the fatigue crack growth of joints accurately.
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Chapter 1.

INTRODUCTION

1.1. Context, Background and Aims

Aircrafts are designed for its light weight structures. In the past decade, fibre reinforced polymer composites have been preferred to metals, mainly due to its high specific strength and stiffness. As a result, composite materials have been applied widely on various secondary, and more recently, primary aircraft structures. Primary aircraft structures are critical load carrying structures that can cause catastrophic failure to the aircraft at failure. Wing skins and fuselage shells are primary aircraft structures that are based on monocoque or stressed skin designs. A major cause of damage on composite aircraft skins is induced by accidental impact damage. This causes delaminations, or disbonds, that are barely visible beneath the surface. This means that flush repairs, commonly known as scarf repairs, are required to maintain its external flush profile while recovering its load carrying capabilities. Repairs to such composite structures require the removal of the damaged area, followed by an adhesively bonded repair as shown in Figure 1. Due to the relatively low strength of structural adhesives to composite materials, the typical scarf angle vary between 1° and 6° to ensure that the bond is loaded under shear [1].

The design and prediction of the strength in adhesively bonded metallic repairs are traditionally based on the assumption that failure is cohesive, that is, cracking is entirely within the adhesive [2]. Due to the complexities of orthotropy and geometry discontinuities at ply interfaces in composite materials, there is a need to study the fracture behaviour of adhesively bonded repairs and the effect of disbonds on the fracture process.

Current repair techniques require the repair material to be identical to the parent structure in ply thickness, orientations and material [3]. This poses a difficulty for repair engineers and manufacturers because of the lack of parent materials or the lack of certification of the parent material to be used along structural adhesives for field repair applications. In this case, the use of materials different from the parent structures in both ply thickness and mechanical properties
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presents new challenges, such as mismatch in local and global stiffness. Thus, it is important to address this new problem of scarf repairs with mismatched or dissimilar adherends.

Airworthiness certification of adhesively bonded scarf repairs remains a significant challenge [4] due to the lack of non-destructive inspection techniques that can detect weak or kissing bonds. Existing airworthiness certification standards [3, 5] prescribe that safety-critical structures must meet safe flight without repair, and the design ultimate load in the presence of damage smaller than the detection limit [6]. In other words, bonded scarf repairs of safety-critical structures must be demonstrated, by experiments and analysis, to exceed the design ultimate load. Recent investigations have revealed that impact damage [7, 8] and pre-existing flaws [9] have a significant effect on a scarf joint’s load-carrying capacity and fatigue endurance [10]. However, current scarf repair design methodologies [1, 11-15] are exclusively based on the analysis of pristine joints that are free of any flaw or damage. There is a need to study the effects of defects or disbonds of the minimum detectable size under quasi static and fatigue loads.

![Figure 1: Schematic of a scarf repair](image)

Damage tolerance is a design philosophy used commonly in the aerospace industry. While the precise definitions of damage tolerance may vary [16], the aerospace industry commonly refers to the damage tolerance concept as the ability of a structure to withstand large, discrete damage and still maintain design limit strength, or retain its design ultimate strength in the presence of barely visible impact damage (BVID) during its service lifetime [11]. In the case of composite repairs such as scarf repairs, the requirement means that a bonded repair must be designed to sustain the design ultimate load in the presence of undetectable defects or disbonds under static
or fatigue loading conditions [17]. Therefore it is important to develop validated methodologies that can accurately predict the load carrying capacities of adhesively bonded scarf repairs under fatigue loads.

Boeing Aerostructures Australia (BAA) which is the internal customer of the partner organisation (Boeing Research and Technology Australia (BR&TA)) manufactures composite moveable trailing edges (such as flaps and ailerons) on the Boeing 787 Dreamliner. These components are manufactured using a unique resin-infusion technique. While standard repairs have been developed by Boeing for the 787 aircraft, there is a significant need to address the above mentioned deficiencies associated with the standard scarf repair method, in addition to a unique challenge of using dissimilar material systems. Because the development and certification of a resin-infusion repair material and the accompanying application process is prohibitively costly, repairs to these resin-infusion composite structures must use certified prepreg or wet-layup materials. These approved repair materials differ from the resin-infusion composites in both ply thickness and mechanical properties, presenting a major difficulty for Boeing.

Due to the significant differences in ply thickness and mechanical properties, preserving equal ply alignment will lead to repairs that violate the requirements for equal-stiffness [11]. It is now known that if the repair patch is stiffer than the parent structure, higher load will be attracted into the repair area [18], overloading the repaired region. On the other hand, misaligning the plies in an attempt to achieve equal stiffness will cause the adhesive to experience significantly higher stresses from that pertinent to the ply-by-ply replacement method [19]. Such a new capability will help to improve the competitiveness of BAA through waste reduction using in-factory composite repairs and will assist Boeing Defence Australia and other Australian aerospace companies for in-service repair and maintenance of composite structures in both commercial and military aircrafts. With more and more new composite structures entering into service for both commercial and military aircrafts, the demand for advanced composite repair concepts that overcome the aforementioned severe drawbacks of existing repair technologies will grow. With the airline industry’s strong demand on standardising on a reduced number of repair systems in an effort to reduce the cost of
maintenance, there is a generic requirement to develop composite repair concepts and validated methodologies for designing composite repairs using dissimilar composite materials.
Chapter 2.

LITERATURE REVIEW

2.1. Regulations on Airworthiness Certification and Damage Tolerance

The Federal Aviation Administration (FAA) states, in an Advisory Circular (AC) [20], acceptable means of adhering to the airworthiness certification requirements for composite aircraft structures. It mentions structural substantiation of a composite design into two categories: Static Strength and, Fatigue and Damage Tolerance.

Static strength substantiation of a composite structure is to be performed under appropriate environmental conditions, under critical loads that will be experienced by the structure during service. Furthermore, the strength of the composite structure is established at varying levels of complexity, known commonly as the “building block” approach. Tests and analyses are performed at the coupon, element, details, and subcomponent level, as shown in Figure 2. This build confidence in the material and the design of the structure [20].

Fatigue endurance and damage tolerance evaluation of composite structures must show that catastrophic failure due to fatigue, environmental effects, manufacturing defects, or accidental damage will be avoided throughout the operational life of the aircraft [20].

Damage tolerance evaluation begins with the identification of damage to the structure which would reduce the structural integrity of the aircraft, such as its location, damage type and size. Once assessed, the damage is placed into one of five categories, as shown in Figure 3. It states that bonded repairs need to firstly withstand the design ultimate load with damage up to a detectable threshold [17] and maintain continued safe flight with the complete disbond of the repairs. It is noted in the AC that there is a lack of standards to identify critical damage threats to various composite structures [20].
Figure 2: Example of a schematic of a building block approach to a fixed wing [20].

Figure 3: Schematic diagram of design load levels versus categories of damage severity [20].
Fatigue endurance evaluation mentioned in the AC focusses on the loads experienced during the service lifetime of the structure and the growth of damage with time. As shown in Figure 4 and Figure 5, the identified damage is characterised into three categories based on its fatigue crack growth behaviour; no-growth, slow-growth, and arrested-growth. The type of damage indicates the inspection interval, replacement or repair of the composite structure. This means that there needs a tool to predict the fatigue crack growth, supported by test evidence, to characterise the damage into one of three categories and extend its service life, while minimising the need for inspections [20].

The AC also mentions the continued airworthiness of repaired aircraft structures [20]. The composite aircraft should be designed for inspection and repair in a field maintenance environment. Damage during service and poor manufacturing processes may cause disbonds or defects which will weaken the structure. The AC mentions that damage may be classified into two categories, Repairable Damage Limits (RDL) and Allowable Damage Limits (ADL). The RDL outlines the details for damage to structural components that may be repaired based on existing data, while the ADL does not require repair. Both RDL and ADL must be based on sufficient analysis and test data to meet the appropriate structural substantiation requirements and other considerations outlined in the AC. Additional substantiation data will generally be needed for damage types and sizes not previously considered in design development [20]. This means that a methodology is required to determine the damage tolerance and fatigue endurance of bonded scarf repairs with damage for continued airworthiness.
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Figure 4: No-growth approach to fatigue crack growth [20].

Figure 5: Slow-growth and arrested-growth approaches to fatigue crack growth [20].
2.2. Failure Modes of Composite Materials and Adhesives

2.2.1. Interlaminar Delamination

Interlaminar delaminations of composite materials have been identified as a serious form of failure occurring between composite plies of a laminate and weaken the structure. Cracks form in the matrix due to the change in stresses in the through thickness direction of a quasi-isotropic laminate under the influence of BVIDs or out of plane loads. There are two common forms of interlaminar failure modes: Mode I – out of plane tensile and Mode II – in-plane shear. The double cantilever beam (DCB) [21] and end notched flexure (ENF) [22] tests are experimental methods in the determination of Mode I and Mode II interlaminar fracture toughness energy of composite materials respectively. The strain energy release rate (SERR) is the energy per unit plate width necessary to produce a unit crack growth at an interlaminar crack between two plies of a laminate. This effect reduces the mechanical strength of the structure and should be obtained experimentally to determine the tolerance of the composite material.

Figure 6: Mode I and Mode II Fracture

2.2.2. Net Tension Failure

Net tension failure is the fracture of composite laminates due to in-plane tension. The strength of composite laminates is dominated by fibre rupture for 0º plies and the matrix cracking for 90º plies with unidirectional fibres. Fibre rupture is caused by tensile stresses, $\sigma_{11}$, in the fibre direction. Failure is initiated first by fibre breakage which then causes voids within the laminate. The strength of the laminate is dominated by the fibre-matrix volume ratio and the number of 0º plies that are in the direction of the load [23]. Matrix cracking occurs when loaded under
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transverse tension, $\sigma_{22}$, or shearing, $\tau_{12}$, in a unidirectional laminate. This occurs when the fibres do not carry any tensional loads and the loads are carried purely by the matrix. The strength of the laminate is dependent on the tensile and shear strength of the matrix and the presence of a 90° ply in a multidirectional laminate [23].

2.2.3. Adhesive Failure

Fracture occurring in structural adhesives is classified as either adhesion or cohesion failure as shown in Figure 7. Adhesive failure is described as the failure of the adhesive to bond to the adherend. In this case, cracks run along the interface between the adhesive and the adherend. Cohesive cracking occurs when the cracks grow entirely within the adhesive layer. Cohesive cracking is due to a strong bond between the adhesive and the adherend and is the preferred form of crack propagation for adhesives [24]. In an aerospace repair aspect, both cohesive and substrate failures are acceptable for certification purposes.

![Figure 7: Types of failure associated with structural adhesives.](image)

2.2.4. First Ply Failure

First ply failure is a unique fracture behaviour that is caused by the interaction between composite laminates and structural adhesives. Commonly found in adhesively bonded lap-joints as shown in Figure 8, this behaviour occurs near the composite-adhesive interface but fails in composite laminate. The strength of such bonds was observed to be matrix dominated due to the presence of composite matrix on opposing fracture surfaces [25-28].
Figure 8: An example of first ply failure in composite lap joints.
2.3. Adhesively Bonded Scarf Repairs and Scarf Joints

2.3.1. Structural Analysis and Fracture Behaviour

Scarf repairs are an efficient method of repair to recover the load carrying capability of external aircraft skins. The damaged region is first removed and machined at angles between 1° and 6° to ensure that the bond is loaded under shear, while maintaining the external profile of the structure by being flush. Traditionally, scarf repairs were used to bond isotropic metallic adherends. The adhesive stresses were found to be constant along the bondline, except near the free edges [18]. In the case of composite scarf repairs, it has been found that the stress concentrations along the bondline depend on the ply angle and are highest at the terminations of load-carrying plies as shown in Figure 9 [1, 29-32]. Furthermore, stress concentrations were also found near the feathered ends of a scarf joint, which could lead to damage initiation in the bondline [33] or in-plane damage in the composite adherends [6]. The varying stiffness and stresses along the bondline due to the ply lay-up needs to be considered in the analysis of composite structures.

Figure 9: Stress concentrations along the plane of the scarf [1]

Scarf joints are two-dimensional representations of scarf repairs along the most highly loaded direction as shown in Figure 10. Scarf joints are single load path structures. Its strength is
dependent on the failure of adhesives along the bondline or laminate fracture, depending which is weaker. Scarf repairs, however, sheds load around the repair when the adhesive is loaded past the strength of the surrounding structure. Under similar loading conditions, materials and lay-up, the scarf repair is capable of carrying a higher load than scarf joints [1].

Scarfed holes are cut-outs that have been machined at small angles in preparation for bonded scarf repairs and were studied by Wang et. al. [6]. Experiments were performed on panels with straight holes and scarfed holes. While it was observed that the straight hole panels failed in a catastrophic behaviour, scarfed holes failed progressively. The failure of the scarfed hole began from the feathered edge of the scarf and propagated outwards. Finally, an analytical method was developed emphasising on the need to predict its progressive fracture behaviour [6].

A strain based method by Wang and Gunnion (2008) was developed for scarf joints with adhesive that experience an elastic-plastic behaviour. As the stress concentrations along the adhesives approaches the plastic strength, stresses are distributed along the plane of the scarf. It was deduced that the joint strength would be insensitive to the laminate stacking sequence, as shown in Figure 12 [1]. However, this was not the case experimentally. Three specimen configurations, consisting of two different lay ups, were manufactured. This creates a dissimilar adherend joint and two similar adherend joints. The strength of the dissimilar adherend joint was found to be between the strength of the other two similar adherend joints. Using the strain based method only failed to accurately predict the strength of the other two configurations. It was suggested that the stiffness of the composite adherend would have caused composite...
fracture ahead of cohesive failure in the adhesive, resulting in a loss of joint strength. There is a need to characterise fracture behaviour of adhesively bonded composite scarf joints.

Figure 11: Progressive damage of scarfed holes as seen in experiments and numerical analysis. [6]

Figure 12: Shear stress distribution of adhesively bonded scarf joints experiencing plastic behaviour [1].
Figure 13: Opposing fracture surfaces of the scarf joints showing cohesive and composite fracture [1].

2.3.2. Damage Tolerance Analysis

Damage tolerance is commonly known as the ability of a structure to withstand large, discrete damage and still maintain design limit strength. On the other hand, the composite structure needs to retain its design ultimate strength in the presence of barely visible impact damage (BVID) [11]. In the case of composite repairs such as scarf repairs, the requirement means that a bonded repair must be designed to sustain the design ultimate load in the presence of disbonds up to the minimum detectable size [17]. For the continued airworthiness of damaged scarf joints, a damage tolerance analysis needs to be able to predict the strength, or generically speaking, the resistance, and fracture behaviour of the joint in the presence of damage or disbonds.

Kim et. al. (2012) performed impact tests on scarf joints, subjected to in-plane tensile pre-strain levels [8]. Experiments showed that the joints failed catastrophically at high levels of impact energy and pre-strain levels. At lower pre-strain levels, the joints experienced adhesive disbonds and composite delaminations that were identified through C-scan. Numerical analysis showed the extent of the damage area in the adhesive and the composite. The cohesive zone model was used to predict failure in the adhesive and the composite delaminations [8]. Overall, the article demonstrated the damage tolerance of scarf joints under tensile pre-strain, and the ability to numerically predict the failure and fracture behaviour in composite scarf joints under dynamic conditions, as shown in Figure 14 and Table 1.
Table 1: Numerical prediction of the damage area and joint failure in scarf joints, with tensile pre-strain, under impact loads.

<table>
<thead>
<tr>
<th>Tensile pre-strain</th>
<th>Adhesive damage area without delamination (mm²)</th>
<th>Adhesive damage area with delamination (mm²)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>41</td>
<td>38</td>
</tr>
<tr>
<td>1000</td>
<td>90</td>
<td>60</td>
</tr>
<tr>
<td>2000</td>
<td>1090</td>
<td>134</td>
</tr>
<tr>
<td>3000</td>
<td>Failed</td>
<td>194</td>
</tr>
<tr>
<td>4000</td>
<td>Failed</td>
<td>Failed</td>
</tr>
</tbody>
</table>

Another aspect of damage tolerance was investigated by Harman (2006) [34]. In this article, the joints were impacted and subjected to compression and tension tests. It was identified by the authors that scarf joints are vulnerable to disbonds by impact to the feather end of the scarf. The feathered end is a region of local stress concentrations and will easily disbond under impact. The presence of disbonds near the feathered end of the scarf will significantly affect its load carrying efficiency. Results of the experimental tests have shown that the joints failed in the composite adherend at non-zero degree plies and cohesively, or adhesively in some cases, at the zero degree plies, as shown in Figure 15. It was also mentioned that scarf repairs do not
recover the strength of the parent structure and extra steps will be required to ensure that the repair is in accordance to damage tolerance principles [34].

As part of a damage tolerance analysis, the fatigue endurance aspect was explored by Alderliesten [35, 36]. The thesis developed a methodology to predict the fatigue crack growth in Glass Laminate Aluminium Reinforced Epoxy (GLARE) [35] and was applied on bonded doublers on GLARE parent structures with fairly good results [36]. Using linear elastic fracture mechanics principles, the articles presented an analytical method that predicted the crack growth rate of fibre metal laminate structures by accounting for the fracture behaviour of in the metal and the composite [35, 36].

Figure 15: Opposing fracture surfaces of undamaged scarf joints under tension [34].
2.4. Damage Modelling Techniques

The design and prediction of the strength in adhesively bonded repairs are traditionally based on the assumption that failure is cohesive, that is, cracking is entirely within the adhesive \[2\]. However, composite failure has been found as the major mechanism by which scarf repairs and joints fracture at room temperature \[9\]. With the increasing use of composite materials and structures, there is a need to accurately predict the fracture behaviour and strength of adhesively bonded repairs, in the presence of disbonds at the adhesive-composite interface on the fracture process. This section reviews on the analytical methods and commercially available numerical models that are applicable to this thesis.

2.4.1. Analytical Methods

An analytical method was presented by Harman and Wang (2006) as an optimisation tool for scarf joints \[19\]. The method takes governing equations for a smoothly tapered joint developed by Erdogan and Ratwani (1971) \[37\] and was solved using a finite difference method by Webber (1981) \[38\]. The analytical method derives the shear stresses in the adhesive, since shear is the primary loaded direction for scarf joints with a low taper angle. The method was compared against finite element models. Results show that the analytical model lacks the resolution of the finite element model as shown in Figure 16. However, the analytical model is deemed to correlate well to numerical results and requires less computational time \[19\]. This analytical method is a stress-based method which assumes that the adhesive is brittle and that failure initiates where the highest shear stresses occur, which is commonly at the termination of zero degree plies. The method holds promise in predicting the shear stresses along dissimilar adherend joints due its ability to integrate the stiffness of both adherends into the method. The formulation of the method can be found in reference \[19\].
2.4.2. Numerical Methods

Due to the complexities of orthotropy and geometry discontinuities at ply interfaces in composite materials, existing stress or strain based failure criteria require calibration of the characteristic distance \([1]\). Fracture mechanics based methods, such as the cohesive zone model (CZM) and the virtual crack closure technique (VCCT), overcome this difficulty. The propagation of cracks at bimaterial interfaces occurs when the structure is loaded past the threshold strain energy release rate (SERR) \([39]\). This method has been used widely to determine the loads required for crack propagation \([40-45]\). While the CZM is relatively new in its application to bimaterial interfaces, the VCCT model is now an industry-standard for performing damage tolerance analysis. One particular issue of applying VCCT to a bimaterial interface is the oscillatory singularity at the tip of a bimaterial interface crack \([46]\).
Numerical predictions of bonded scarf adherends have compared favourably to experiments in various publications using the CZM method [8, 29, 32, 47]. These publications explicitly modelled the crack path by embedding CZM elements that represented the unique mode of failure at that interface.

### 2.4.2.1. Virtual Crack Closure Technique

The VCCT is a well-established numerical technique to calculate the strain energy release rate at a crack tip. It is a linear elastic fracture mechanics analysis based on the product of the nodal forces, $F$, at the crack tip and nodal displacements, $v$, immediately behind the crack tip. Assuming the width and length of element are $b$ and $d$, respectively, the strain energy release rate (SERR), $G$ is given by the following expression,

$$ G = \frac{F \cdot v}{2 \cdot b \cdot d} \quad (1) $$

Disbond at the crack tip occurs when $G$ approaches the critical value, $G_C$, i.e.

$$ \frac{G}{G_C} \geq 1 \quad (2) $$

In a mixed mode loading condition, the critical SERR was defined by the Benzeggagh-Kenane (B-K) fracture criterion [48], which is given in Equation (3), where the exponent, $\eta$, is an empirical parameter. It was found that brittle, epoxy based composite materials correlated well with an exponent value of 1.75 [49].

$$ G_c = G_{IC} + (G_{IIc} - G_{IC}) \left( \frac{G_{II}}{G_{I} + G_{II}} \right)^\eta \quad (3) $$

Within Abaqus, the VCCT is incorporated into a progressive damage model that allows for automated modelling of crack propagation in a non-linear analysis. In this model, a damageable interface is defined as a bonded contact between two surfaces. A pre-existing disbanded region
is defined, which involves only a touching contact (free to open but cannot interpenetrate), and this is used to automatically define the crack tip. At the end of every increment in a non-linear analysis, Equation (2) is assessed at a crack tip node. If crack growth is deemed to occur, then the bonded contact at that node is converted to a touching contact for the next increment. In this way, automated crack progression can be captured, allowing for the simulation of stable crack growth, or crack growth occurring in a non-catastrophic manner.

2.4.2.2. Cohesive Zone Model

The critical Mode I and II strain energy release rate (SERR), $G_{IC}$ and $G_{IIc}$, obtained from fracture mechanics principles [50], are used to describe the force required to produce a crack in a structure. The energy, $G$, shown in Figure 17 as $d\Pi$, required to produce a crack area, $dA$, is found under the force-crack opening displacement curve in a Mode I test as seen in Figure 17. Using the theoretical calculations to obtain the SERR, the results are used in relation to a damage mechanics model that is implemented into finite element codes. A simple bilinear stress-strain curve is used to represent the elastic and plastic behaviour of the adhesive region or resin-rich region of composites that are governed by the critical SERR, $G_{IC}$ and $G_{IIc}$ (Figure 17) [51].

![Cohesive Zone Model Diagram](image)

Figure 17: Relationship between Fracture Mechanics and Damage Mechanics for CZM [51].
2.4.2.2.1. **Single Mode Delamination**

The cohesive element model is a finite element method in modelling the response of crack propagation between bonded interfaces [52]. Abaqus offers a library of cohesive elements to model the behaviour of adhesive joints, interfaces in composites, and other situations where the integrity and strength of interfaces may be of interest.

Based on the mechanical behaviour of the bond, a constitutive model specified directly in terms of traction versus separation is used to model the delamination of composite materials. Other models such as the continuum approach accounts for a finite thickness and the ability for lateral displacements when used to model gaskets.

As stated in Abaqus 6.10, the traction-separation model is intended for bonded interfaces where the interface thickness is negligibly small. Thus, it is used to model the resin rich region between the plies of a composite panel. This model is based on a linear elastic behaviour and a linear progressive degradation of the material stiffness. (Figure 18) [52].

![Figure 18: The Linear Elastic Traction-Separation Behaviour for Mode I and Mode II][53]

As shown in Figure 18, the linear elastic behaviour is unique between the two modes and is independent of each other. Both modes, however, require a mode specific input of the elastic stiffness of the material, $K$, the maximum stress for damage initiation, $\tau_1^{0}$ and $\tau_2^{0}$, and the
critical SERR, \( G_{IC} \) and \( G_{IIC} \) [52]. Using these values, the displacements due to the onset of damage, \( \Delta_{10} \) and \( \Delta_{20} \), (Equation (4)) and the final displacements, \( \Delta_{1f} \) and \( \Delta_{2f} \), (Equation (5)) can be found. The stiffness of the element will be reduced when the displacement value the onset value. At the final displacement, the stiffness of the material is zero and has completely failed.

\[
\Delta_{1,2}^0 = \frac{\tau_{1,2}^0}{K_{1,2}} \tag{4}
\]

\[
\Delta_{1,2}^f = 2 \cdot \frac{G_{IC,IIC}}{\tau_{1,2}^0} \tag{5}
\]

2.4.2.2.2. **Mixed Mode Delamination**

It is commonly found that a mixture of Mode I and II loading occurs in advanced composite structures. Under mixed-mode loading, damage initiation and softening behaviour may occur before the respective individual parameters are reached. Therefore, various criterion of mixed-mode loading is presented within Abaqus [52] dealing in delamination onset and propagation [54].
While Camanho & Davila (2002) presents three modes of loading [54], Mode III failure was excluded in literature by Greve & Pickett (2006) [51] and Barua & Bose (2007) [55]. However, in the literature by Camanho & Davila (2002), due to the lack of a mixed-mode test method incorporating Mode III loading, there is no reliable mixed-mode delamination failure criterion [54]. Therefore, most of the failure criteria proposed for delamination growth were established for mixed-Mode I and II loading only.

### Damage Initiation

Damage initiation under mixed-mode loading can be initiated by a quadratic interaction function based on the nominal stress or nominal strain ratio [54]. As shown in Equations (6) and (7), and also Figure 19, subscript ‘n’ represents Mode I component while subscripts ‘s’ and ‘t’ represents in-plane shear and torsion that is represented together as a Mode II component.

The nominal stress criterion is represented in Abaqus as:
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\[
\left\{ \frac{t_n}{t_n^o} \right\}^2 + \left\{ \frac{t_s}{t_s^o} \right\}^2 + \left\{ \frac{t_t}{t_t^o} \right\}^2 = 1 \tag{6}
\]

The nominal strain criterion is represented in Abaqus as:

\[
\left\{ \frac{\epsilon_n}{\epsilon_n^o} \right\}^2 + \left\{ \frac{\epsilon_s}{\epsilon_s^o} \right\}^2 + \left\{ \frac{\epsilon_t}{\epsilon_t^o} \right\}^2 = 1 \tag{7}
\]

The symbol $<$ used in the above equations represents the Macaulay bracket. It is used to signify that a pure compressive deformation or stress state does not initiate damage [54].

\subsection*{2.4.2.2.4. Damage Evolution}

Under the effects of mixed-mode loading, the fracture toughness on the mode ratio must be accounted for in the formulation of the cohesive elements. Two criterions exist in literature to predict delamination propagation under mixed-mode loading, the Power Law and the B-K criterion [54].

The most widely used criterion, the power law, is based on the ratio of individual modes and the interactions between them. It accounts for the applied energy on the specimen, $G$, the critical SERR, $G$ and the curve fitting parameter, $\alpha$.

\[
\left( \frac{G_I}{G_{IC}} \right)^\alpha + \left( \frac{G_{II}}{G_{IIIC}} \right)^\alpha = 1 \tag{8}
\]

However, Camanho & Davila (2002) stated that the power law criterion have did not accurately capture the mixed-mode fracture toughness in epoxy or thermoset composites.
The Benzeggagh-Kenane (B-K) fracture criterion is expressed as a function of the Mode I and Mode II fracture toughness and a parameter, $\eta$, obtained from mixed mode bending (MMB) tests at different mode ratios, as seen in Figure 21. The parameter can be obtained by solving using a least square fit or a curve fitting procedure to map out the progression of the fracture toughness energy to mode mixture [48].

$$G_{IC} + (G_{II} - G_{IC}) \cdot \left( \frac{G_{II}}{G_{I} + G_{II}} \right)^{\eta} = G_C$$  \hspace{1cm} (9)

Equation (9) describes a curve fitting equation used to describe the change in the critical SERR with respect to the mixture of the modes given that the Mode I and II SERRs were already obtained. In this equation, $G_I$ and $G_{II}$ are the energies at a given mode mixture as seen in Figure 19. [48]
2.4.2.2.5. Cohesive Zone Length

The length of the cohesive zone is defined as the distance from the crack tip to the point where the maximum cohesive traction is attained [53]. Cohesive elements in this region undergo separation and relative sliding due to normal and shear tractions at an interface. The tractions are applied till complete separation based on its critical SERR.

As presented by Turon et. al. (2007) [53], the following equation determines the mesh density required when modelling delaminations using cohesive elements. The cohesive zone length, $l_{cz}$, is a function of the through thickness Young’s modulus, $E$, the critical SERR, $G_C$, the stress to initiate delamination, $\sigma$, and a model dependent parameter, $M$. It is reported by Turon et. al. (2007), that the parameter, $M$, is commonly set to be either close or exactly close to unity ($= 1$) [53].

$$l_{cz} = M \cdot E \cdot \frac{G_C}{(\sigma^0)^2}$$

(10)

In order to obtain accurate results using cohesive elements, the tractions within the cohesive zone must be properly represented by sufficient discretization.

$$N_e = \frac{l_{cz}}{l_e}$$

(11)
The element length required, \( l_e \), is determined by the number of elements in the cohesive zone, \( N_e \), and the cohesive zone length, \( l_{CZ} \). It was suggested by Turon et. al. (2007) [53] that a minimum of 2 elements in the cohesive zone is required to accurately predict the propagation of delamination.

![Example of Mesh Size Effects on a DCB Model](image)

**Figure 22: Example of Mesh Size Effects on a DCB Model [53]**

### 2.4.2.2.6. Effects of Peak Stress on Mesh Density

Peak stress for damage initiation defines the value of stress at which damage initiates at a material point in the cohesive element [55]. The value of peak stress determines the interface stiffness, \( K \), of the cohesive element as shown in Figure 18.

### 2.4.2.2.7. Effects of Interfacial Stiffness

As stated in both Barua & Bose (2007) [55] and Meo & Thieulot (2005) [57], too low a value of interface stiffness leads to an inaccurate representation of the mechanical behaviour of the
interface, whereas high values will increase run-time and can promote numerical errors. Barua & Bose (2007) [55] have shown that the value for peak stress, \( \sigma \), to be a function of the interface stiffness, \( K \), and the opening displacement that results in the initiation of damage, \( \Delta \). (Equation (12))

\[
\sigma = K \cdot \Delta
\]  

(12)

It is then shown that the interfacial stiffness, \( K \), to be a function of the elastic modulus, \( E \), the thickness of the cohesive element, \( t \), and a parameter, \( \alpha \) (Equation (13)) Barua & Bose (2007), have stated that the value for \( \alpha \) to be much larger than 1 and recommends for \( \alpha \) to be at least 50. This causes a loss of stiffness to be less than 2% and is sufficiently accurate for most problems. This equation would provide a range of interfacial stiffness between \( 10^5 \) and \( 5 \times 10^6 \) N/mm\(^3\).

\[
K = \frac{\alpha \cdot E_3}{t}
\]  

(13)

However, literature by Meo & Thieulot (2005) [57] has shown that using an \( \alpha \) value of 1 in (13) and an assumed damage initiation displacement of \( 7 \times 10^{-6} \) m in Equation (12) has provided good agreement with experimental results. On the other hand, Camanho & Davila (2002) has set a constant interfacial stiffness of \( 10^6 \) for all interfacial thicknesses [56].
2.4.2.3. *In-Plane Continuum Damage Mechanics Model*

Abaqus offers a continuum damage mechanics (CDM) model that enables the prediction of the onset of damage and damage evolution behaviour for elastic-brittle materials with anisotropic behaviour. The model is primarily intended to be used with fibre-reinforced materials since they typically exhibit such behaviour [52]. The model assumes that the fibres in the fibre-reinforced composites are unidirectional as shown in Figure 23.

![Figure 23: Orientation of Fibres and the Local Coordinate System [52].](image)

The model is based on a bilinear behaviour that is composed of three stages as shown in Figure 24. The first stage is an undamaged response of the material that models the linear elastic response of the material. The second stage is a damage initiation criterion that determines the
onset of damage or the strength of the material. The third is a damage evolution response that models the softening behaviour of the composite material.

A methodology was presented by Wang et. al. in deriving the in-plane fracture energy of composite materials using open hole tensile (OHT) tests [6]. Experiments were performed on composite laminates with a unidirectional lay-up and an open hole of ¼ in. (6.35mm) in diameter. The strength to fracture the specimen at the hole was recorded. The experiment was represented in a numerical model. By knowing the in-plane strength of the material, the fracture energy was varied, and was calibrated so that the numerically derived OHT strength matched experimental values as shown in [6].

![Figure 25: Calibration of fracture energy values with OHT experimental results](image)

**2.4.2.3.1. Damage Initiation**

Damage initiation for fibre-reinforced materials is based on Hashin’s theory [58, 59]. This is further explored in literature by Matzenmiller et. al. (1995) [23] where four modes of composite failure are identified. The equations that govern these four modes are shown below.

Fibre Tension ( \( \sigma_{11} \geq 0 \)):

\[
\left( e_f^p \right)^2 = \left( \frac{\sigma_{11}}{x_f} \right)^2 - 1 \begin{cases} \geq 0 & \text{failed} \\ < 0 & \text{elastic} \end{cases}
\] (14)
Fibre Compression (\(\sigma_{11} < 0\)):
\[
(e_f^c)^2 = \left(\frac{\sigma_{11}}{X_C}\right)^2 - 1 \begin{cases} 
\geq 0 & \text{failed} \\
< 0 & \text{elastic}
\end{cases}
\] (15)

Matrix Tension (\(\sigma_{22} \geq 0\)):
\[
(e_m^T)^2 = \left(\frac{\sigma_{22}}{Y_T}\right)^2 + \left(\frac{T_{12}}{S}\right)^2 - 1 \begin{cases} 
\geq 0 & \text{failed} \\
< 0 & \text{elastic}
\end{cases}
\] (16)

Matrix Compression (\(\sigma_{22} < 0\)):
\[
(e_m^C)^2 = \left(\frac{\sigma_{22}}{Y_C}\right)^2 + \left(\frac{T_{12}}{S}\right)^2 - 1 \begin{cases} 
\geq 0 & \text{failed} \\
< 0 & \text{elastic}
\end{cases}
\] (17)

\(X_T\) Longitudinal tensile strength
\(X_C\) Longitudinal compressive strength
\(Y_T\) Transverse tensile strength
\(Y_C\) Transverse compressive strength
\(S\) In-plane shear strength
\(e_f^T, e_f^C, e_m^T, e_m^C\)
They are terms that represent the direction of loading. Subscript \(f\) and \(m\) represent fibre and matrix respectively. Superscript \(T\) and \(C\) represent tension and compression.

2.4.2.3.2. Damage Evolution

Abaqus introduces a characteristic length into the formulation to alleviate mesh dependency during material softening. Equation (18) shows the stress relation with strain, \(\varepsilon\), and the damaged elasticity matrix, \(C_d\). The matrix shown in Equation (19), contains the current state of the fibres, matrix and shear damage. It represents the remaining strength of the composite material after damage initiation.

\[
\sigma = C_d \cdot \varepsilon
\] (18)

\[
C_d = \frac{1}{D} \begin{bmatrix}
(1 - d_f)E_1 & (1 - d_f)(1 - d_m)\nu_{21}E_1 & 0 \\
(1 - d_f)(1 - d_m)\nu_{12}E_2 & (1 - d_m)E_2 & 0 \\
0 & 0 & (1 - d_s)GD_1
\end{bmatrix}
\] (19)

\[
D = 1 - (1 - d_f)(1 - d_m)\nu_{12}\nu_{21}
\] (20)
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\[ d_f \] State of fibre damage
\[ d_m \] State of matrix damage
\[ d_s \] State of shear damage
\[ E_1 \] Young’s modulus in the fibre direction
\[ E_2 \] Young’s modulus in the matrix direction
\[ G \] Shear modulus
\[ v_{12}, v_{21} \] Poisson’s ratio

The damage variables \( d_f, d_m, \) and \( d_s \) are derived from damage variables \( d_f^t, d_f^c, d_m^t, \) and \( d_m^c \) corresponding to the four failure modes previously mentioned.

\[
d_f = \begin{cases} 
  d_f^t & \text{if } \sigma_{11}^e \geq 0 \\
  d_f^c & \text{if } \sigma_{11}^e < 0 
\end{cases} \quad (21)
\]

\[
d_m = \begin{cases} 
  d_m^t & \text{if } \sigma_{22}^e \geq 0 \\
  d_m^c & \text{if } \sigma_{22}^e < 0 
\end{cases} \quad (22)
\]

\[
d_s = 1 - (1 - d_f^t)(1 - d_f^c)(1 - d_m^t)(1 - d_m^c) \quad (23)
\]

\( \sigma_{11}^e \) and \( \sigma_{22}^e \) are components of the effective stress tensor used to calculate the damage initiation criterion which is computed from Equation (24). The effective stress tensor is based on a damage operator, \( M \), shown in Equation (25).

\[
\hat{\sigma} = M \cdot \sigma \quad (24)
\]

\[
M = \begin{bmatrix}
  \frac{1}{1-d_f} & 0 & 0 \\
  0 & \frac{1}{1-d_m} & 0 \\
  0 & 0 & \frac{1}{1-d_s}
\end{bmatrix} \quad (25)
\]

Using the damaged elasticity matrix, the relevant stresses and strains are obtained below. These values of displacement and stress are thus able to monitor the four failure modes at the same time. This allows an exact prediction of the failure mode occurring. The equivalent displacement and stress for each of the four damage modes are defined as follows:
Fibre Tension ($\sigma_{11} \geq 0$):

\[
\delta_{eq}^{ft} = L c \sqrt{\langle \varepsilon_{11} \rangle^2 + \alpha \varepsilon_{12}^2}
\]

\[
\sigma_{eq}^{ft} = \frac{L c (\langle \sigma_{11} \rangle \langle \varepsilon_{11} \rangle + \alpha \tau_{12} \varepsilon_{12})}{\delta_{eq}^{ft}}
\]  

Fibre Compression ($\sigma_{11} < 0$):

\[
\delta_{eq}^{fc} = L c \langle -\varepsilon_{11} \rangle
\]

\[
\sigma_{eq}^{fc} = \frac{L c (\langle -\sigma_{11} \rangle \langle -\varepsilon_{11} \rangle - \langle -\sigma_{11} \rangle)}{\delta_{eq}^{fc}}
\]  

Matrix Tension ($\sigma_{22} \geq 0$):

\[
\delta_{eq}^{mt} = L c \sqrt{\langle \varepsilon_{22} \rangle^2 + \varepsilon_{12}^2}
\]

\[
\sigma_{eq}^{mt} = \frac{L c (\langle \sigma_{22} \rangle \langle \varepsilon_{22} \rangle + \tau_{12} \varepsilon_{12})}{\delta_{eq}^{mt}}
\]  

Matrix Compression ($\sigma_{22} < 0$):

\[
\delta_{eq}^{mc} = L c \sqrt{\langle -\varepsilon_{22} \rangle^2 + \varepsilon_{12}^2}
\]

\[
\sigma_{eq}^{mc} = \frac{L c (\langle -\sigma_{22} \rangle \langle -\varepsilon_{22} \rangle + \tau_{12} \varepsilon_{12})}{\delta_{eq}^{mc}}
\]  

The characteristic length, $L_c$, is based on the element geometry and formulation: it is a typical length of a line across an element for a first-order element; it is half of the same typical length for a second-order element. For membranes and shells it is a characteristic length in the reference surface, computed as the square root of the area. The term $\alpha$ is a user defined value that can only be either 0 or 1. This decides the contribution of shear in fibre tension [52].

As seen from Equations (26) to (33), the four modes each follow the bilinear damage evolution as seen in Figure 24. After damage initiation, the damage variable for any particular mode is shown in Equation (34). It is graphically represented in Figure 26.
\[ d = \frac{\delta_{eq}^f (\delta_{eq} - \delta_{eq}^0)}{\delta_{eq} (\delta_{eq}^f - \delta_{eq}^0)} \]  

(34)

\( \delta_{eq}^0 \) Initial equivalent displacement

\( \delta_{eq}^f \) Displacement which the material is completely damaged

Figure 26: Damage Variable as a Function of Equivalent Displacement [52].
2.5. Fatigue Crack Growth and Life Prediction Techniques

Composite scarf joints and repairs experience service loads that are less than the design limit loads of the structure. During service, this creates a fatigue load on the damage structure that leads to a stable crack growth over time. This section describes fracture mechanics methods in predicting the fatigue crack growth of damaged structures and design methodologies for fatigue tolerant structures.

2.5.1. Analytical Method

The VCCT has been applied by numerous authors in an analytical method to predict crack growth rates of damaged structures under fatigue [60-62]. Firstly, the structure is numerically modelled in a finite element analysis, with the VCCT embedded at the crack tip. The model is loaded by the maximum load and the SERR at the crack tip is determined using VCCT using Equation (1). The fatigue crack growth rate is determined using the Paris’ law formula as shown in Equation (35).

\[
\frac{da}{dN} = C(G)^m \tag{35}
\]

Results showed that the VCCT was capable of predicting the fatigue crack growth rates of the structure at an acceptable level of accuracy and precision. It was notably praised for its reliability and ease in the determination of the SERR at a crack tip in complex structures such as composite laminates or structural components.

2.5.2. Fatigue Threshold

A fatigue endurance methodology was proposed by Baker et. al. for scarf joints [63]. The strain threshold, \(e_{th}\), for the initiation of fatigue crack growth was derived empirically in Equation (36) by obtaining the threshold SERR, \(G_{th}\), for a scarf joint with disbond. The terms \(E, t, a\) and \(\theta\), represent the stiffness of the joints, thickness of the joint, disbond length, and angle of the
taper. From a damage tolerance perspective, the method was capable of predicting the strain required for the initiation of fatigue crack growth at a given flaw size.

\[ e_{th} = \frac{1}{E_t} \left[ 2 \Delta G_{th} \left( \frac{1}{E(t - a \sin \theta)} - \frac{1}{E_t} \right) \right]^{-1} 0.5 \] (36)

The empirical equation assumes that the crack will propagate along the plane of the scarf and is insensitive to the plies in the laminate. Firstly, this means that any crack bifurcation or deviation will not be considered. This results in an overprediction of the design limit load. Secondly, the variation in stiffness in the laminate will result in stress concentrations at stiff plies along the plane of the scarf. This means that damage initiation along the bondline, away from the crack tip, is ignored. Thirdly, the equation does not account for the ratio of mode mixity in scarf joints. It was assumed that the joints had high scarf ratios, where the joints were generally loaded in shear and the peel stresses were negligible. Nevertheless, the equation shows promise in providing a quick estimate on the resistance to fatigue crack initiation.

2.5.3. Mode Mixture

Fatigue mode mixture of composite structures has been thoroughly investigated by various authors. Review of their work have been performed and presented by Pascoe et. al. [64] and Blanco et. al. [65]. It was observed that, unlike static mode mixture, the behaviour of fatigue mode mixture is not well agreed upon as they merely predict the macroscopic behaviour of the structure without addressing the micro-mechanisms relating to delamination growth under fatigue loads [64].

The popular static mode mixture methodology proposed by Benzaggagh and Kenane (BK) has been adapted to describe fatigue behaviour [66]. It uses the Paris’s law formula in terms of \( \Delta G \) \( (G_{max} - G_{min}) \), but with \( C \) and \( m \) expressed as a function of the Mode I and II material constants, denoted by subscript \( I \) and \( II \), and empirically determined parameters, \( n_C \) and \( n_m \), as shown in Equations (37) and (38).

37
\[
\ln(C) = \ln(C_{II}) + \left[ \ln(C_i) - \ln(C_{II}) \right] \left(1 - \frac{G_{II}}{G_{tot}}\right)^{n_C} \tag{37}
\]

\[
m = m_i + (m_{II} - m_i) \left(\frac{G_{II}}{G_{tot}}\right)^{n_m} \tag{38}
\]

The BK methodology has also been adapted to describe the fatigue threshold of structures under mixed mode loading [67]. As shown in Equation (39), the equation requires the Mode I and II fatigue threshold SERR, the ratio of mode mixture and an empirically derived mode mixture constant for fatigue threshold, \( n_{th} \).

\[
\Delta G_{th} = \Delta G_{Ith} + (\Delta G_{IIth} - \Delta G_{Ith}) \left(\frac{G_{II}}{G_{T}}\right)^{n_{th}} \tag{39}
\]

2.5.4. Complete Fatigue SERR Regime

The Paris’ law for fatigue crack growth rate describes the linear portion of a fatigue regime, commonly known as the steady state regime. However, the behaviour of a structure is not complete without understanding the transition from threshold to steady state and steady state to rapid fracture. It also helps to analytically determine the nature of the fracture without cross referencing to the critical or threshold values at a particular disbond size.

A complete fatigue SERR regime was proposed by Martin and Murri to describe its sigmoidal shape, as shown in Equation (40) [68]. Various authors have modified the equation to predict a variety of structures [69-73] and fatigue behaviours [74, 75] to a reasonable level of agreement to experimental values.

\[
\frac{da}{dN} = C(G_{max})^m \left[ 1 - \left(\frac{G_{th}}{G_{max}}\right)^{n_1} \right] \frac{1 - \left(\frac{G_{max}}{G_{C}}\right)^{n_2}}{1 - \left(\frac{G_{max}}{G_{C}}\right)^{n_2}} \tag{40}
\]
2.6. Summary, Gaps and Research Questions

A comprehensive review of published research was performed and listed below:

- The certification regulation of damage tolerant composite structures;
- Structural characteristics of composite scarf repairs, scarfed holes and scarf joints;
- Damage tolerance of scarf repairs and joints;
- Damage tolerance analysis of scarf joints;
- Analytical and numerical models for damage tolerance analysis; and
- Fatigue endurance and crack growth prediction methodology of scarf joints.

The following gaps in literature have been identified:

- A lack of understanding on fracture behaviour of composite scarf joints under static and fatigue loads. Current body of literature does not characterise the fracture behaviour of joints along the plane of the scarf, such as adhesive, cohesive and composite matrix or composite fibre fracture.
- A lack of understanding on the fracture behaviour of composite scarf joints with dissimilar adherends.
- The effect of disbonds on scarf joints is not understood in literature. While there have been investigations on the pristine strength tests of composite materials, there is a lack of literature that quantifies the behaviour of disbonds in composite structures, particularly scarf repairs and joints.
- There is a lack of literature on the damage tolerance of composite scarf joints using commercially available damage modelling tools such as VCCT and CZM under static loads.
- There is a lack of numerical methods on the fatigue crack growth and endurance of composite scarf joints.

In summary, the following research questions are developed for this thesis:

- How do disbonds of varying sizes affect the fracture behaviour and static strength of scarf joints?
• How do scarf joints of dissimilar adherends differ from scarf joints of similar adherends, with and without the presence of disbonds, in terms of fracture behaviour and static strength?
• To explore how commercially available damage modelling tools (VCCT, CZM and continuum damage mechanics models) can be implemented to predict the strength and fracture behaviour of scarf joints, with disbonds, of similar and dissimilar adherends.
• How do disbonds of varying sizes affect the fatigue life of scarf joints?

2.7. Thesis Outline

This thesis is structured into six chapters, excluding the introduction and literature review, to address the research problems and objectives. A literature review on airworthiness standards of damage tolerant structures, analysis performed on composite scarf repairs and the current state-of-the-art in fracture modelling. The research problems were identified and the objectives are laid out into the following chapters. Chapter 3 describes experimental procedures that will be performed in this thesis. Chapter 4 presents experimental results on the damage tolerance and strength of composite joints, and material testing of its constituent materials. It identifies unique fracture behaviour in composite scarf joints, scarf joints with dissimilar adherends and scarf joints under cyclic loading. Chapter 5 investigates various numerical models that can predict the fracture behaviour of composite scarf joints with disbonds. Chapter 6 presents a predictive methodology used to determine the strength of scarf joints with dissimilar adherends. Chapter 7 presents a numerical methodology used to determine the fatigue crack growth rate of composite scarf joints.
Chapter 3.

EXPERIMENTAL SETUP

3.1. Introduction

Experiments were performed to understand the fracture behaviour of composite materials and scarf joints. Basic material tests were performed to measure the material properties of the composite prepregs and the adhesive that are required to model the scarf joints using finite element methods. The composite materials will be manufactured and tested, to test standards available in literature, to obtain the elastic properties, strength and strain energy release rates (SERR). This chapter describes the experiments performed in this thesis and its results will be published in the following chapter.
3.2. Material Properties and Testing

3.2.1. Materials Used

Three prepreg composite material systems were used: VTM264/T700 (VTM264) [76], Cycom970/T300 (Cycom970) [77] and HexPly914/T300 (HexPly914) [78] with a cured ply thickness of 0.22 mm, 0.22mm and 0.33mm respectively. The HexPly914 and the Cycom970 are woven fabrics with a curing temperature of 180ºC under vacuum for two hours. The HexPly914 has a satin weave and the Cycom970 has a plain weave. The VTM264 is a unidirectional prepreg with a curing temperature of 120ºC for 1 hour. For bonding, an epoxy-based structural adhesive VTA260 [79] was used, which has a curing temperature of 120ºC for 1 hour and a cured thickness of 0.2 mm.

3.2.2. In-plane Stiffness and Strengths – Tensile Test of Tapered Specimens

Composite materials were tested to the recommendation of ASTM D3039 [80] to measure their in-plane tensile properties. Specimens were manufactured using 8 plies in a unidirectional lay-up to measure the 0º and 90º ply properties of the VTM264, and the (0º) ply properties of the HexPly914 and Cycom970. Each specimen, with a constant rectangular cross-section of area, A, of 25 mm, was at least 250 mm long with bevelled tabs bonded at both ends of the specimen. Strain gauges were bonded to the specimen to obtain the longitudinal and transverse strain, $\varepsilon_l$ and $\varepsilon_t$. The specimens were loaded in a load cell at a rate of 1 mm/min up to failure and the peak load, $P$, was recorded. The following properties were obtained in the in-plane tensile test: Tensile Strength ($\sigma$), Modulus of Elasticity ($E$) and Poisson’s Ratio ($\nu$).

The ASTM D3518 describes tests to measure the in-plane shear properties of composite materials [81]. Specimens were manufactured using 8 plies in a unidirectional lay-up to measure the ±45º ply properties of the VTM264 and the (45º) ply properties of the HexPly914 and Cycom970. Each specimen, with a constant rectangular cross-section of area, A, of 25 mm, was at least 250 mm long with bevelled tabs bonded at both ends of the specimen. Strain gauges were bonded to the specimen to obtain the longitudinal and transverse strain, $\varepsilon_l$ and $\varepsilon_t$. The specimens were loaded in a load cell at a rate of 1 mm/min up to failure and the peak load, $P$,
was recorded. Similarly, the Shear Strength and In-plane Shear Modulus were obtained from this test.

3.2.3. In-plane Fracture Energy – Open Hole Tensile Test

To obtain the in-plane fracture energy of the composite materials, open hole tensile (OHT) tests were performed. As mentioned in Section 2.4.2.3, results from this test will be used to calibrate for the in-plane fracture energy for composite materials. Two sets of OHT specimens were manufactured using 8 plies in a unidirectional, (0º) and (45º), lay-up with a constant rectangular cross-section of area, $A$, of 30 mm, and length of 250 mm. Each set consisted 5 specimens. Electric drills drilled holes in the specimens at a diameter of ¼ in. (6.35 mm) and set at a low speed to prevent fracturing the surrounding material. The specimens were loaded at a rate of 1 mm/min up to failure. Particular attention was paid to ensure that the fracture occurred at the open hole. Finally, the strength of the specimens was determined similar to in-plane tensile tests.

3.2.4. Mode I – Double Cantilever Beam Test

Interlaminar Mode I critical SERR of the composite materials can be determined through double cantilever beam (DCB) tests performed in accordance to ASTM D5528 [21]. Specimens were manufactured to satisfy the minimum thickness criteria. The VTM264, Cycom970 and HexPly914 specimens had 16, 16 and 12 plies of unidirectional lamina, with fibres aligned along the length of the specimen. Each specimen had a width of 25 mm, and length of 150 mm, with a total of five specimens for each material system. A 5 µm thick non-stick polytetrafluoroethylene (PTFE) film of 50 mm in length was embedded at the mid-plane of the laminate to create a disbond. T-shaped tabs were adhered on both faces of the specimen at the disbonded end. The specimens were loaded at the T-shaped tabs at 1 mm/min. The test was stopped, unloaded and reloaded at crack growth intervals of 5 mm. The maximum load was used to calculate for the Mode I critical SERR. A travelling microscope measured the length of the crack. The modified beam theory (MBT) method mentioned in ASTM D5528 [21] as shown in Equation (41) and Figure 27, allows for the Mode I critical SERR to be calculated.
Chapter 3 - EXPERIMENTAL SETUP

\[ G_{IC} = \frac{3P\delta}{2b(a + |\Delta|)} \]  

(41)

Figure 27: Modified Beam Theory

\[
P, \delta, b, a, \Delta, C
\]

- \( P \): Load
- \( \delta \): Load point displacement
- \( b \): Specimen width
- \( a \): Delamination length
- \( C = \frac{\delta}{P} \): Compliance

3.2.5. Mode II – End Notched Flexure Test

Protocols by ESIS describes standards to determine interlaminar Mode II critical SERR of the composite materials using end notched flexure (ENF) tests [22]. The VTM264, Cycom970 and HexPly914 specimens had 16, 16 and 12 plies of unidirectional lamina, with fibres aligned along the length of the specimen. Each specimen had a width of 25 mm, and length of 150 mm, and five specimens were used for each test. A disbond of length 50 mm was embedded using a PTFE film at the mid-plane of the laminate for all three composite materials. The specimens were placed in a three point bend setup with the crack tip placed 25 mm from the roller. The crack front was measured using a travelling microscope. The test set up is as shown in Figure 28. The test was loaded at 0.5 mm/min until a load drop occurs. The Mode II critical SERR was determined using the corrected beam theory (CBT) method as shown in Equation (42).
Figure 28: Test set up of the ENF specimen.

\[
G_{IIc} = \frac{9 \, \frac{P^2}{a^2}}{16 \, b^2 \, E_f \, h^2}
\]  \hspace{1cm} (42)

- \(P\): Load
- \(a\): Delamination length
- \(b\): Specimen width
- \(E_f\): Flexural modulus
- \(h\): Half thickness of the specimen
- \(L\): Half span length
- \(C = \frac{\delta}{P}\): Compliance

3.2.6. Fracture Behaviour of Adhesively Bonded Composite Materials under Mode I and II Tests

Mode I and II tests were performed to understand the effect of disbonds at the composite-adhesive interface as identified in Sections 2.2.3 and 2.2.4. The VTM264, Cycom970 and HexPly914 specimens had two sets of 8, 6 and 6 plies of unidirectional lamina, with fibres aligned along the length of the specimen. After the two sets of laminate were cured, and the surfaces sanded and cleaned with distilled water. Before the laminates were adhesively bonded using the VTA260, a disbond of 50 mm was replicated by embedding a PTFE film at the composite-adhesive interface as shown in Figure 29.
Figure 29: Adhesively bonded composite materials with disbonds for Mode I and II tests.

Mode I DCB and Mode II ENF tests were performed in accordance to ASTM D5528 [21] and ESIS [22] respectively, as mentioned in Sections 3.2.4 and 3.2.5. The energy required for crack propagation along the composite-adhesive interface under Mode I and II loading was determined in equations (41) and (42) respectively.

3.2.7. Fatigue Crack Growth of Composite Materials under Mode I and II Tests

Mode I interlaminar fatigue tests were performed on DCB specimens to obtain the material constants, $C$ and $m$, as shown in Equation (43) [82, 83]. The composite material tested was the VTM264 carbon fibre/epoxy prepreg. The test was performed by applying a fatigue load at a frequency of 10 Hz. The manufactured specimens and the test setup were identical to the static DCB tests mentioned in Sections 3.2.4. Defined as the ratio between the minimum and maximum crack displacement, a displacement controlled loading cycle was kept constant at 0.1 over the entire stress intensity range. Delamination growth was measured at intervals of 2 to 10 mm of crack propagation to determine the crack growth length per load cycle ($da/dN$). Crack propagation was measured using a travelling microscope. The range of the cyclic stress intensity ($ΔG_i$) was varied from 30 to 2000 J/m$^2$ to obtain regions of no-growth, slow/stable growth and rapid growth in a Paris’ Law plot. The cyclic stress intensity can be determined using equation (44), adapted from Section 3.2.4.

$$\frac{da}{dN} = C(ΔG)^m = C\left(G_{max} (1 - R^2)^m \right)$$

$$G_{i,max} = \frac{3 P_{max} \delta_{max}}{2 b (a + |Δ|)}$$
There is a lack of test standard for conducting Mode II fatigue delamination characterisation. However, a widely cited literature by O’Brien et. al. has been adopted [84]. The Mode II interlaminar fatigue tests were performed in an ENF setup similar to the static tests mentioned in Section 3.2.5. A compliance calibration was first performed to obtain the bending stiffness of the laminate. The specimens were each statically loaded up to 10% of the static Mode II critical SERR. At least three loadings were performed at various crack lengths \(a\) of 30 to 40 mm away from the left of the span. The displacements, \(\delta\), and loads, \(P\), were recorded to plot the compliance \(C\) of the specimens as a linear function of the cube of the crack length \(a^3\), as shown in Figure 30, and the gradient \(m\) of the function was obtained. The calculated crack lengths were checked against the predicted crack lengths to prove the validity of this method.

![Compliance as a function of crack length](image)

**Figure 30:** An example of compliance as a function of crack length [84].

\[
C = A + ma^3
\]  

(45)

The maximum Mode II SERR \(G_{II,max}\) of the specimen under fatigue was determined using the compliance calibration relation as shown in Equation (46). Similarly, the range of the cyclic stress intensity \(\Delta G_{II}\) was varied from 50% to 10% of the static critical SERR to obtain regions of no-growth, slow/stable growth and rapid growth in a Paris’ Law plot. The initial and final crack length, and thus, the crack growth can be determined by solving Equation (45) for the crack length \(a\) using the compliance \(C\). This allows for an accurate measurement of the position crack front and the crack growth.
\[ G_{II,max} = \frac{3 \, m \, P_{max} \, \delta_{max}}{2 \, b} \]
3.3. Scarf Joints

3.3.1. Identical Adherends

This section presents the manufacturing procedure for scarf joints with identical adherends and disbands. The scarf joints had an embedded disbond length, \( a \), referring to Figure 31, of 3, 6, and 12 mm in length along the bondline. Joints without an initial flaw (\( a = 0 \)) were also manufactured to characterise the performance of the pristine joints. A lay-up of \([45^\circ/0^\circ/0^\circ/90^\circ/-45^\circ]_{2S}\) was used to manufacture two separate panels of VTM264/T700 composite. The cured panels were cut into coupons of 25 mm in width. Scarfing was carried out by tilting the coupons at 5\(^\circ\) and 3\(^\circ\) to a milling machine, producing a taper with a feathered end as shown in Figure 31. The scarfing process created a scarfed length of 50 mm and 84 mm in the 5\(^\circ\) and 3\(^\circ\) scarf joints respectively. It is important to note during the manufacturing stage to ensure that the loads were distributed equally at the gauge ends of the joint, a distance of approximately ten times the laminate thickness was maintained at both ends of the scarfed region. 5\(^\circ\) and 3\(^\circ\) scarf joint specimens had a total gauge length of 135 mm and 169 mm respectively. The scarfed surfaces were cleaned by light sanding and distilled water to replicate industry standards for field repairs. The scarfed adherends were then bonded with VTA260 adhesive with a PTFE film embedded between the adhesive and the feathered end of the adherend. As mentioned in Section 2.3.1, it was identified that the joint is the most susceptible to disbond at the feathered end. The joints were cured in accordance with the manufacturer’s recommended curing process. Scarf joints produced in this manner are representative of the situation where the damage material is removed through machining and a repair patch, of a similar material system and layup, is machined from a cured laminate. At least three specimens were manufactured for each set of embedded flaw condition.
3.3.1.1. **Static Strength Test**

Scarf joints were loaded to failure to determine the effect of disbonds on the fracture behaviour of scarf joints with identical adherends under static loads. The tests involved the 5° and 3° scarf joints. The procedure of this test is the same as tensile tests mentioned in Section 3.2.2. The scarf joints were loaded under static tension at a displacement rate of 1.0 mm/min until failure, where failure was determined as the complete loss of load-carrying capability. The maximum load and displacement at failure was recorded. A detailed inspection of the fracture surfaces was conducted, using visual inspection, optical microscopy and SEM.

3.3.1.2. **Fatigue Test**

The joints with a scarf angle of 5 degrees were tested under a tension-tension fatigue condition with a sinusoidal loading ratio (R) of 0.1, at a frequency of 10 Hz, and a constant maximum applied load that ranges from 25% to 50% of the static strength. A travelling microscope was used to measure crack lengths along the side of the scarf joint specimens. Correction fluid was applied to the side of the scarf joint to aid in the measuring the crack length. Measurements using the microscope required the test to be paused and the loading grips returned to its original position, where the amplitude of the sinusoidal loading behaviour is zero. The measurements were taken at regular intervals with respect to the size of the flaw and the maximum applied load. The compliance of the scarf joints were taken across the gauge length to take note of changes in stiffness during the test. To ensure that the compliance measured the change in stiffness due to disbonds on bonded scarf surfaces, the specimens were regularly inspected to ensure that the fracture occurred in the region of the scarf surface. The life of the scarf joints was recorded when the joints had completely failed.

3.3.2. **Dissimilar Adherend**

Dissimilar adherend scarf joint specimens were designed to achieve identical thickness as the original structure, to represent scarf repairs that are designed to maintain a flush profile with the structure. Both joint adherends were manufactured from two different woven carbon/epoxy
Chapter 3 - EXPERIMENTAL SETUP

prepreg composite material systems: Cycom970 and HexPly914. Both systems differ in resin matrices, weave and ply thicknesses. The HexPly914 and the Cycom970 have a curing temperature of 180ºC under vacuum for two hours. VTA260 structural adhesive was used to bond the adherends at a curing temperature of 120ºC under vacuum for one hour.

Scarf joints with dissimilar adherends were tested to understand its fracture behaviour in the presence of disbonds. Two sets of composite scarf joints were manufactured. In the first set of joints, the Cycom970 was used for the parent adherend, with HexPly914 being the repair adherend. In the second set of joints, the materials for the parent and repair adherends were swapped. The parent adherend was a [0º/45º]2s laminate that was cured first. Upon curing the total adherend thickness was 2.64 mm for the HexPly914 and 1.76 mm for the Cycom970. Each adherend was cut to a width of 25.4 mm and a length of 150 mm. The parent adherends were machined at a 3-degree angle to represent typical aircraft repairs. For the repair adherends, the layup was varied to match the thickness of the parent adherend. The Cycom970 repair used a [0º/45º 2/0º/45º 2/0º/45º]S ply sequence and the HexPly914 repair used a [0º/45º 0º/45º 0º/45º] ply sequence. The repair adherend plies were laid in an inverse stepped configuration using the scarfed parent adherend as a mould. It is noteworthy to mention that this manufacturing method creates small resin rich pocket at the ply ends along the scarf. The resulting scarf joint as described above is shown in Figure 32.

Figure 32: Manufacturing schematic of a scarf joint with dissimilar adherends.

Before curing the repair adherend, a layer of PTFE film was laid between the repair and parent adherends. This ensured that the adherends remained separate during the cure. The cured adherends were finally bonded together with an adhesive. PTFE films of three different lengths (3, 6 and 12 mm) were embedded at the feathered end of the repair adherend to simulate disbonds between the adhesive and the repair adherend. This disbonds replicated damage on the
repair due to manufacturing or accidental impact damage during service, to create a joint containing disbond. Joints without disbond, or “pristine” joints, were also manufactured as benchmark specimens. At least four samples of each disbond length were manufactured. The assembled joints were cured in autoclave in accordance with the manufacturer’s recommended curing process. The joints were loaded at both ends in quasi-static tension using a 50 kN Instron machine at a constant displacement rate of 0.5 mm/min until the joints broke apart. The load required to fracture the joint was recorded during the entire process.
Chapter 4.

EXPERIMENTAL RESULTS

4.1. Material Properties

The strength and fracture behaviour of composite scarf joints is dependent on the coupling of strengths and stiffnesses of its constituents, such as composite plies and its lay-up, and the adhesive. This can lead to fracture to occur the adhesive, composite interlaminar delaminations or fibre breakage. This section presents experimental findings and material properties of composite materials and adhesive used in this thesis.

4.1.1. In-plane Stiffness and Strength

During tensile tests, specimens showed linear behaviour followed by sudden brittle failure. There were no visible signs of interlaminar delamination, which suggests that the specimens fractured in net tension failure at the fracture surface. Specimens in the in-plane shear tests experienced non-linear loading behaviour before failing catastrophically. The fractured region was observed to be aligned to the direction of the fibres. This suggests that while the plies experienced in-plane shear, the fibres and matrix failed in tension in the direction of the orthogonal fibre direction at the instance of fracture. Results for the in-plane properties of the VTM264, HexPly914 and the Cycom970 are shown in Table 2.

<table>
<thead>
<tr>
<th></th>
<th>$E_{11}$ (GPa)</th>
<th>$G_{12}$ (GPa)</th>
<th>$\sigma_{11}$ (MPa)</th>
<th>$\tau_{12}$ (MPa)</th>
<th>$\gamma_{12}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>VTM264</td>
<td>120.2</td>
<td>3.94</td>
<td>2575</td>
<td>85.7</td>
<td>0.32</td>
</tr>
<tr>
<td>HexPly914</td>
<td>53.65</td>
<td>3.26</td>
<td>566</td>
<td>100</td>
<td>0.075</td>
</tr>
<tr>
<td>Cycom970</td>
<td>53.12</td>
<td>3.02</td>
<td>626</td>
<td>123</td>
<td>0.08</td>
</tr>
</tbody>
</table>
4.1.2. *Open Hole Tensile Strength*

Similar to Section 4.1.1, specimens in the (0°) OHT tests showed linear elastic behaviour and a sudden catastrophic fracture afterwards. Specimens in the (45°) OHT tests experienced non-linear loading behaviour before failing catastrophically. Results for the average OHT strength of the HexPly914 and the Cycom970 are shown in Table 3. It was observed that the strength of the OHT specimens in unidirectional (0°) and (45°) plies were fairly similar. This could be attributed to presence of T300 carbon fibres in both composite materials, Cycom970 and HexPly914, that carry load in both (0°) and (45°) ply directions.

Table 3: Average OHT strength of HexPly914 and Cycom970.

<table>
<thead>
<tr>
<th>Strength (MPa)</th>
<th>(0°)</th>
<th>(45°)</th>
</tr>
</thead>
<tbody>
<tr>
<td>HexPly914</td>
<td>284</td>
<td>153</td>
</tr>
<tr>
<td>Cycom970</td>
<td>279</td>
<td>177</td>
</tr>
</tbody>
</table>

4.1.3. *Mode I Fracture Toughness*

Crack was observed to propagate along the mid-plane of the DCB specimens. Although crack bridging was observed in the VTM264 specimens, there was an increase of less than 10% in the Mode I critical SERR. The Mode I critical SERR was averaged from four specimens, with five measurements taken each specimen, and presented in Table 4.

Table 4: Average Mode I critical SERR of the composite materials

<table>
<thead>
<tr>
<th>Composite material</th>
<th>Average Critical SERR (J/m²)</th>
<th>Standard Error</th>
</tr>
</thead>
<tbody>
<tr>
<td>HexPly914</td>
<td>256</td>
<td>11.2</td>
</tr>
<tr>
<td>Cycom970</td>
<td>329</td>
<td>8.8</td>
</tr>
<tr>
<td>VTM264</td>
<td>462</td>
<td>15.3</td>
</tr>
</tbody>
</table>
4.1.4. Mode II Fracture Toughness

Fractured surfaces of the ENF specimens showed that the crack propagated along the mid-plane. Results showed that the specimens behaved elastically during loading. A sudden loss of load was observed as the crack propagated. The Mode II critical SERR was averaged from four specimens and presented in Table 5.

<table>
<thead>
<tr>
<th>Composite material</th>
<th>Average Critical SERR (J/m²)</th>
<th>Standard Error</th>
</tr>
</thead>
<tbody>
<tr>
<td>HexPly914</td>
<td>1290</td>
<td>12.2</td>
</tr>
<tr>
<td>Cycom970</td>
<td>2378</td>
<td>24.7</td>
</tr>
<tr>
<td>VTM264</td>
<td>1603</td>
<td>77.2</td>
</tr>
</tbody>
</table>

4.1.5. Fracture Behaviour of Adhesively Bonded Composite Materials under Mode I and II Tests

Observation of the Mode I and II test specimens, through an optical microscope, showed that the opposing fracture surfaces of all three composite material systems contained fractured matrix surfaces and broken fibres on the opposing surfaces. Micrographs suggest that the fracture path propagated across the adhesive, breaking into the composite material and continued propagating in the composite matrix, near the composite-adhesive interface at a distance of approximately one fibre thick, along the plane of the interface. This behaviour is shown in Figure 33.
The measured critical SERR of the adhesively bonded composite specimens are presented in Table 6. Although the fracture occurred in the composite matrix, the SERR of the bonded composite specimens were significantly higher than the plain composite specimens. Thus, gains in the measured SERR of the specimens were attributed to the presence of the adhesive, but the damage remained localised in the composite material.

Table 6: Average Mode I and II critical SERR of the adhesively bonded composite materials

<table>
<thead>
<tr>
<th>Composite material</th>
<th>HexPly914</th>
<th>Cycom970</th>
<th>VTM264</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Mode I</td>
<td>Mode II</td>
<td>Mode I</td>
</tr>
<tr>
<td>Critical SERR (J/m^2)</td>
<td>692</td>
<td>1630</td>
<td>797</td>
</tr>
<tr>
<td>Standard Error</td>
<td>33</td>
<td>33</td>
<td>26</td>
</tr>
<tr>
<td>Failure mode</td>
<td>Composite</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

4.1.6. Fatigue Crack Growth of Composite Materials under Mode I and II Tests

The effect of the location of the crack front from the left span, or crack length, in an ENF test was briefly mentioned by O’Brien et al. [84]. It states that the crack length should be optimally
located 35 mm away from the left span. Mode II ENF tests were performed on a specimen with an initial crack length of approximately 25 mm. A constant cyclic displacement of 2 mm was applied to the mid span, with a load ratio of 0.1, which is approximately equal to a quarter of the critical Mode II SERR. The crack was allowed to propagate towards the mid span \((a = 50 \text{ mm})\) of the specimen. Results of the test, showing the Mode II SERR with respect to the crack length, are presented in Figure 34.

It was observed that the SERR of the ENF test specimen varied non-linearly over crack lengths of 25 to 50 mm. The SERR approached a maximum value at a crack length of 35 mm and the change in SERR at crack lengths of 30 to 40 mm was the least. Furthermore, it was observed that the load decreased as the crack length increased. This is due to the change in bending stiffness as the crack propagated from 25 to 50 mm. Thus, subsequent tests were performed at crack lengths of 30 to 40 mm. It is important to note that the specimens were pre-cracked before each test and that the crack fronts were manually shifted away from the embedded PTFE. For disbonds close to the mid span, through thickness damage to the laminate was inspected to ensure that the crack propagated along the mid-plane.
The fatigue delamination crack growth for unidirectional VTM264 laminates in Mode I and II can be found in Figure 35. The fatigue properties obtained from these tests are found in Table 7. It was observed that the Mode I and II fatigue threshold SERR ($\Delta G_{th}$) and the fatigue material constants ($C$ and $m$) of the VTM264 have very little differences. Unlike other composite materials in literature that have significantly different Mode I and II fatigue properties [85], this is a behaviour that is unique to VTM264.

Table 7: Mode I and II fatigue properties of VTM264

<table>
<thead>
<tr>
<th></th>
<th>Mode I</th>
<th>Mode II</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\Delta G_{th}$ (KJ/m$^2$)</td>
<td>0.033</td>
<td>0.033</td>
</tr>
<tr>
<td>$C$</td>
<td>1.76E-01</td>
<td>3.01E-02</td>
</tr>
<tr>
<td>$m$</td>
<td>4.697</td>
<td>4.034</td>
</tr>
</tbody>
</table>

Figure 35: Fatigue delamination crack growth for VTM264 laminates in Mode I and II.
4.2. Static Strength Tests of Scarf Joints

4.2.1. Similar Adherend

The experimental results of the 3º and 5º scarf joints, the strength and extension across the grips at failure, are summarised in Figure 36. During the tests, the specimens emitted minor cracking sounds, this was followed by catastrophic fracture that is typical of highly loaded brittle structures. It can be seen from Figure 36 that as the flaw size increases for the 5º scarf joint, the stress and extension at failure decreased. On the contrary, the 3º scarf allowed for a longer disbond without a significant loss of strength, due to a longer scarf surface than the 5º scarf.

Figure 37 presents a typical example of the opposing fracture surfaces of a 5º scarf joint with an embedded flaw (a = 12) and a pristine joint. The feathered ends of each adherend can be seen on both sets of opposing fracture surfaces. This indicates that for specimens with and without flaws, fracture occurred at the feathered end of one adherend, travelled along the bondline, then crossed the adhesive, and propagated along the bondline again towards the feathered end of the other adherend. Further visual inspection of the adherends showed that the bulk of the adhesive remained attached to one adherend. This initially suggested that adhesive failure, or failure along the composite-adhesive interface, was the principal damage mode. This behaviour was also observed in the 3º scarf joints. Further analysis was required to understand the fracture behaviour of adhesively bonded scarf joints.
Figure 36: Strength-extension at failure plot of 5º (Top Figure) and 3º (Bottom Figure) similar adherend scarf joints with disbonds.
Detailed SEM analysis revealed the presence of fibres on the opposing fracture surfaces. This is illustrated in Figure 38, where SEM images of both fracture surfaces are presented, in comparison with the adherend surface after machining but prior to bonding. As the scarf plane passes through each ply, the fracture surface characteristics changed with ply orientation, which is illustrated in Figure 38. On the 0º and 45º ply surfaces, the adhesive was covered with a thin layer of composite peeled off from the other adherend, which exhibited complete composite fracture without any sign of adhesive. These results showed that the fracture path was not at the composite-adhesive interface, but inside the composite adherend. Furthermore, it can be concluded that the distance between the fracture path and the composite-adhesive interface was comparable to the fibre diameter. The presence of matrix heckles on the majority of the fracture surface indicated that the fracture was largely driven by shear failure of the matrix. In addition, there were occasional instances observed on the fracture surfaces of pull-out of the adhesive carrier scrim; an example is shown in Figure 39. The adhesive carrier scrim serves to maintain a constant bondline thickness. During adhesive bonding, compression of the adherends led to a reduced amount of adhesive between the adhesive carrier scrim and the adherends. This means that there will be a low bonding strength between the adhesive and the polyester scrim. This contributed to the formation of imprints on the 0º ply terminations. For the 90º plies, opposing fracture surfaces showed loose fibres, which again suggests that the crack travelled through the composite ply. Heckles were not observed on the fracture surfaces, which indicates the failure was predominantly driven by tension.
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Figure 37: Fracture surfaces of VTM264 5° scarf joints with (a) a 12 mm flaw and (b) no flaw.

<table>
<thead>
<tr>
<th>Adhesive bulk surface</th>
<th>0° ply</th>
<th>45° ply</th>
<th>90° ply</th>
</tr>
</thead>
<tbody>
<tr>
<td>Matrix heckles</td>
<td>Fibres</td>
<td>Fibres</td>
<td>-45°</td>
</tr>
<tr>
<td>Fibres</td>
<td>-45°</td>
<td>90°</td>
<td></td>
</tr>
<tr>
<td>Adhesive carrier cloth</td>
<td>90°</td>
<td>0°</td>
<td></td>
</tr>
<tr>
<td></td>
<td>-45°</td>
<td>90°</td>
<td></td>
</tr>
</tbody>
</table>

Figure 38: SEM images of opposing faces of a fractured VTM264 scarf joint at various ply angles. Bottom row: Micrographs of the machined scarf surface before bonding.

A summary of the crack path observations is presented in Figure 40, which shows a schematic of the crack path through each ply, and a simplified schematic of the way in which the crack migrated through the adhesive at some point along the bondline. The flaw size was not found to affect the crack path, or the characteristics of the fracture plane on any of the plies. In general, it was concluded that the fracture occurred in the composite adherends, with tensile fracture in the 90° plies and interlaminar shear failure in the 0° and 45° plies. These results suggested that failure was controlled by the fracture properties of the composite material, instead of the adhesive. These experimental observations will be used to guide the development of FE models and the identification of the appropriate material parameters, which is described in the
following section. It can also be concluded that sufficient surface preparation was performed to prevent adhesion failure.

Figure 39: VTA260 Adhesive carrier material pull-out on the fracture surface of a 0° ply

Figure 40: (a) Overall crack propagation in the scarf joint. (b) Typical crack propagation path near ply terminations.

4.2.2. Dissimilar Adherend

The average strengths of the joints at varying disbond lengths were plotted against the size of the disbond $a$, normalised by the scarf length $L$, as shown in Figure 41. The three degree scarf
joints were observed to have fractured catastrophically. The ply orientation along the plane of the scarf is added in the figure. It was observed that the strengths of the scarf joints were fairly similar for small disbond lengths (less than 3 mm). As disbond size increased, the strength of the joint reduced dramatically in a non-linear rate. For the same disbond size, the strengths were similar between the HexPly914 and Cycom970 parent adherend configurations.

![Graph showing scarf joint strength vs. disbond size](image)

Figure 41: Strength of dissimilar adherend scarf joints with dis bonds superimposed with its respective ply orientations along the plane of the scarf. The data points represent dis bond lengths of 0 mm (Pristine), 3 mm, 6 mm and 12 mm.

The strength of a joint is determined by the strength of the load-carrying 0° plies, so that the loss of these plies as a result of the disbond results in a significant loss of strength in the joint. Furthermore, a disbond that places the crack tip at a ply with low elasticity reduces the fracture toughness of the joint. By matching the strength of the joint at a given disbond length against the superimposed ply orientation as shown in Figure 41, it was discovered that the loss in strength from a 3 mm to 6 mm disbond length coincided with the loss of 0° plies.

In scarf joints with dis bond lengths of 6 mm and 12 mm, examination using optical microscope initially suggested a cohesive failure close to the composite-adhesive interface. However, detailed analysis using a scanning electron microscope (SEM) revealed the presence of carbon fibres on both fracture surfaces as shown in Figure 42. On the repair adherend at the crack tip, adhesive carrier material was discovered protruding out of a composite fracture.
surface. The opposite surface showed no adhesive, indicating a predominant composite fracture. This behaviour was observed to occur for the entire fracture surface along the plane of the scarf. Therefore, it is concluded that the fracture path was neither at the composite-adhesive interface nor in the adhesive, but inside the composite adherend. This behaviour is identical to the fracture of similar adherend scarf joints as presented in Section 4.2.1. Figure 41 and Figure 42 suggests that the joint is weakened by the occurrence of bondline composite fracture behaviour and is of major concern in recovering the design strength of composite structures.

The mismatched adherend joints were inspected to understand the effect of different disbond sizes on the fracture propagation as shown in Figure 43. Fractographic analysis showed that for relatively small disbond sizes, of less than 3 mm, the adherends failed due to net tension failure across the laminate. This behaviour suggested that the adhesive bondline, mainly loaded in shear, had a higher strength than the tensile strength of the composite adherends. The fact that failure of joints, with disbond less than 3 mm, failed by net tension fracture of the HexPly914 laminate is consistent with its lower in-plane tensile strength than the Cycom970 laminate. For scarf joints and scarf repairs to recover the design strength of the structure, fracture in the repair adherend is undesirable. So it is important to consider laminate strengths for mismatched repairs by selecting repair materials with higher strengths than the parent structure, if the tolerable disbonds are small. In the present case, this is applicable for disbonds less than 3 mm in length.

Figure 42: Typical bondline composite fracture in the joint observed under the SEM.
For larger disbond sizes, the fracture propagated across the adhesive into the parent adherend. This was followed by bondline composite fracture and net tension failure at the other feathered end of the scarf adherend. This behaviour was fairly similar to the fracture path of similar adherend scarf joints in Section 4.2.1. This suggests that the failure was not related to the ply property and the lay-up of the adherends, but could be attributed to geometrical factors such as secondary bending. This behaviour requires further study to determine the mode of fracture at the crack tip and the change in adherend stiffness with respect to changes in disbond length in the following sections.

Figure 43: Observed crack propagation along the bondline of scarf joints with varying disbond lengths.
4.3. Fatigue Crack Growth Tests of Scarf Joints with Similar Adherends

Fatigue cracks in the three degree scarf joints with similar adherends were observed to first propagate across the adhesive, from the initial crack front to the other interface, and then continue travelling down the bondline, at a distance from the composite-adhesive interface comparable to a fibre diameter similar to the static fracture path reported in Section 4.2.1, shown in Figure 40.

![Graphs showing crack growth](image)

Figure 44: Observed crack lengths at various static residual strengths (RS) of three degree scarf joints with similar adherends
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The cracks were observed to propagate at a stable rate, for flaw sizes less than approximately a quarter \((a < 20 \text{ mm})\) of the overall scarf length, before rapid crack growth and failure, as shown in Figure 44. Static residual strengths (RS) of the three degree scarf joints can be found in Figure 36. This suggests that loads required to propagate the initial crack length are critical for flaws approximately greater than a quarter of the length of the three degree scarf, as represented in Equation (47).

\[
\text{If } a_0 < \frac{L}{4} \quad \Rightarrow \quad G (\sigma_{a_0}) = G_c (a > \frac{L}{4})
\]  

(47)

In scarf joints with disbonds of 6 mm, it was observed for crack lengths less than 10 mm, the fatigue crack growth rates were highly erratic, as shown in Figure 44 (c) and (d). At crack lengths longer than 10 mm, the crack growth rates were similar and showed similar trends in Figure 44 (c) and (d). This suggests that the life of the joint is not solely dependent on the strength of the joint. This suggests that scarf joints, experiencing cyclic loading with cracks at the bimaterial composite-adhesive interface, is not just dependent on the composite material but also dependent on the adhesive.

The compliance of the scarf joints with similar adherends were obtained to understand periods of no crack growth observed in Figure 44. It was observed that the compliance of the joint increased proportional to the number of cycles without the propagation of the crack front, as shown in Figure 46. This was followed by a jump in crack length and a sharp increase in compliance. This suggests that the joints experienced plasticity in the adhesive due to its low shear strength and high SERR.

Based on the evidence presented thus far, it is suggested that the fatigue life of composite scarf joints with disbonds at the composite-adhesive interface is dependent on the adhesive. It can be further deduced that at higher loads, the adhesive experiences plasticity. This is due to the lack of disbond growth and visible cracking away from the crack. Since the adhesive has a lower shear strength than the composite, the adhesive will experience plasticity and, thus, deformation and changes in compliance without visible crack growth.
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Figure 45: S-N curves of individual scarf joint specimens with various disbonds and loads

Figure 46: A sample of specimens that display a change in compliance with no visible crack growth.
Chapter 5.

ON MODELLING THE FRACTURE BEHAVIOUR OF SIMILAR ADHEREND JOINTS WITH DISBONDS

5.1. Introduction

In order to accurately predict the strength of composite scarf joints with disbonds at the composite-adhesive interface, an approach is required to represent the fracture behaviour observed in Section 4.2.1. Based on the fractographic observations described, the fracture occurred in the composite at a small distance comparable to the fibre diameter away from the composite-adhesive interface. Fracture mechanics based methods, such as the cohesive zone model (CZM) and the virtual crack closure technique (VCCT), overcome this difficulty. The propagation of cracks at bimaterial interfaces occurs when the structure is loaded past the threshold strain energy release rate (SERR) [39]. This method has been used widely to determine the loads required for crack propagation [40-45]. While the CZM is relatively new in its application to bimaterial interfaces, the VCCT model is now an industry-standard for performing damage tolerance analysis. One particular issue of applying VCCT to a bimaterial interface is the oscillatory singularity at the tip of a bimaterial interface crack [46]. This chapter explores the methodology in predicting interfacial fracture in scarf joints.
5.2. Model Development

5.2.1. Methodology

A two dimensional plane strain modelling approach was developed for the fracture of the composite matrix at the composite-adhesive interface. The approach ignored the extremely thin layer of resin-fibre material removed from the composite adherend, as the influence of this layer was considered negligible. To model the crack path, damageable interfaces were embedded along critical regions in the joint, as shown in Figure 47. Multiple models were created to investigate the nature and the path of the fracture by changing the damageable interface along the plane of the scarf. This is explained in later sections. The fracture properties of the composite-adhesive interface were taken from the composite properties in Table 4 and Table 5, and the use of adhesive properties in Table 6 and Table 8. Adhesive properties were obtained from manufacturer’s data sheet [86].

<p>| | | |</p>
<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>$E$ (GPa)</td>
<td>$v$</td>
<td>$G$ (GPa)</td>
</tr>
<tr>
<td>3 GPa</td>
<td>0.35</td>
<td>1.1 GPa</td>
</tr>
</tbody>
</table>

Table 8: Material properties of VTA260 adhesive

Figure 47: Damage models for composite scarf joint: (a) composite-adhesive interface, and (b) cohesive.
The virtual crack closure technique (VCCT) and the cohesive zone model (CZM) are numerical models that are dependent on the mesh density for numerical accuracy. The effect of the CZM interfacial strength and mesh density on the accuracy of numerical simulations has been discussed in [53, 87]. The mesh size of the cohesive zone elements needs to be sufficiently small to capture the effect of stress concentrations or cohesive damage zones at the crack tip. Given the magnitude of the interface stresses and the requirement to capture the cohesive zone with at least three elements, it was found that an element length of approximately 0.15 mm in the crack growth direction was required in the present investigation. Similarly, guidelines for applying the VCCT at a bimaterial interface are provided in literature [46]. Mathematical solutions of the SERR have been shown to oscillate at very small flaw sizes \((a \rightarrow 0)\). A range of element sizes were studied to ensure a converged FE solution with a mesh that was coarse enough to avoid oscillating results [46]. It was found that a bondline mesh length of 0.15 mm in the crack growth direction was suitable. Thus, a bondline mesh length of 0.15 mm was maintained for both CZM and VCCT models. The mesh scheme of the finite element model is shown in Figure 48.

![Figure 48: Ply-level modelling of the composite scarf joint and the mesh density near the bondline.](image)
5.3. **Adhesive Bondline Stress**

5.3.1. **Methodology**

Adhesive stresses along a scarf plane vary significantly as the ply stiffness is strongly affected by its orientation [1]. Traditionally, scarf joints are designed using identical isotropic adherends where the stresses along the adhesive are uniform. An elastic FE analysis was performed to determine the shear stress distribution of the current scarf joint lay-up in the adhesive along the bondline, and the results are shown in Figure 49. These results show that high shear stresses occur at the ends of 0° plies, which is similar to results of Wang and Gunnion [1].

An average stress failure criterion is based on using the average of all peel or shear stresses along the bondline. The average stress criterion is a very crude instrument. Since this technique is currently used in repair designs to size scarf repairs, it has been chosen as a comparison. Maximum stress may be more appropriate for brittle adhesives in the case of pristine joints. For joints containing disbond, the maximum stress is unbounded, due to the stress singularity at the crack tip. Consider a scarf joint between identical adherends of thickness, \( t \), and tapered at angle, \( \theta \). The average shear stress and the average peel stress for a given disbond \( a \) can be determined using the load-equilibrium method by Erdogan and Ratwani [37],

\[
\frac{\tau_{av}}{\sigma_{app}} = \frac{1}{\frac{1}{2} \sin(2\theta)} \left( \frac{a \sin \theta}{t} \right) \tag{48}
\]

\[
\frac{\sigma_{av}}{\sigma_{app}} = \frac{\sin^2 \theta}{\frac{1}{1} - \frac{a \sin \theta}{t}} \tag{49}
\]
Figure 49: Shear stress distribution in the adhesive mid-plane for a pristine scarf joint.

where the averaged peel stress, $\sigma_{av}$, and average shear stress, $\tau_{av}$, denote the averaged quantities along the mid-plane of the adhesive layer. These values are normalised using the applied (far-field) stress, $\sigma_{app}$. Figure 50 presents a comparison of these solutions with the results of a FE analysis, in which flaws of various sizes were embedded at the composite-adhesive interface. In the FE model, the composite-adhesive interface was set as a bonded contact, and setting a region to be disconnected with contact condition generated the required flaw. The model was loaded at a unit load of 1 MPa, which is within the elastic regime of the joint. It is seen that the numerical models and analytical equations for the average stresses in the adhesive are in good agreement.
Figure 50: Average bondline (a) peel and (b) shear stresses in a 5º scarf joint.
5.3.2. Results

A first order estimate of the load-carrying capacity of scarf joints containing a disbond can be estimated using the average shear stress criterion [1, 37]: fracture occurs when the average shear stress reaches the shear strength of the adhesive. From Equation (48), the ultimate strength $\sigma_{ult}$ of a scarf joint containing a disbond of length, $a$, can be expressed in terms of the shear strength as:

$$
\sigma_{ult} = \frac{\tau_f}{\frac{1}{2} \sin(2\theta)} \times \left(1 - \frac{a \sin \theta}{t}\right)
$$

(50)

shows a comparison of the predicted strength using the average stress criterion and the experimental results, plotted against the respective flaw size. It is clear that the strength of joints, measured in the experiments, decreased at a faster rate than that predicted by the average stress criterion. It can be concluded that the reduction in joint strength is greater than the expected reduction due to the loss of bond area, and that the average stress criterion is non-conservative and hence unsuitable as a design criterion.

Figure 51: Comparison between prediction by average stress criterion and experimental results of a $5^\circ$ scarf joint.
5.4. Linear Elastic Fracture Mechanics

5.4.1. Methodology

Linear elastic fracture mechanics (LEFM) considers the crack to grow once the SERR at the crack tip reaches a critical value. The application of LEFM at the composite-adhesive interface requires consideration of the bimaterial properties of this crack tip. The mismatch in elastic properties is commonly expressed in terms of the Dundurs second parameter, $\beta$, under plane strain conditions [88], which is given by:

$$\beta = \frac{1}{2} \left[ \frac{\mu_1 (1 - 2 \nu_2) - \mu_2 (1 - 2 \nu_1)}{\mu_1 (1 - \nu_2) + \mu_2 (1 - \nu_1)} \right]$$

(51)

where $\mu$ and $\nu$ are the shear modulus and Poisson’s ratio respectively, subscripts 1 and 2 refer to the two dissimilar materials surrounding the crack tip. From this, the crack tip singularity parameter, $\epsilon$ is given by [79]:

$$\epsilon = \frac{1}{2\pi} \ln \left( \frac{1 - \beta}{1 + \beta} \right)$$

(52)

The strain energy release rate, $G$, at the bimaterial interface is given by [43]:

$$G = \frac{K^2_i + K^2_{II}}{E_{eff} \cosh^2(\pi \epsilon)} = \frac{1 - \beta^2}{E_{eff}} (K^2_i + K^2_{II})$$

(53)

where $K$ denotes the stress intensity factors, subscripts $I$ and $II$ refer to parameters pertinent to the peel and shear loading modes, and $E_{eff}$ is the bimaterial effective modulus given by

$$\frac{1}{E_{eff}} = \frac{1}{2} \left( \frac{1 - \nu_1}{E_1} + \frac{1 - \nu_2}{E_2} \right)$$

(54)
Chapter 5 – ON MODELLING THE FRACTURE BEHAVIOUR OF SIMILAR ADHEREND JOINTS

The factors $K_I$ and $K_{II}$ can be expressed in terms of the basic solution for a crack in an infinite body [40], after introducing two factors $y_I$ and $y_{II}$ to characterise the geometric configuration. This gives the following equation:

$$K_I + iK_{II} = \left[ y_I \left( \frac{a}{L} \right) \sigma_{av} + i y_{II} \left( \frac{a}{L} \right) \tau_{av} \right] (1 + 2i\varepsilon)\sqrt{\pi a} (2a)^{-i\varepsilon}$$

(55)

where $a$ is the crack length, $\sigma_{av}$ and $\tau_{av}$ are the average peel and shear stresses. With regards to a loaded scarf joint, given by Equations (51) and (52), Equation (55) can be expressed in terms of the far-field applied stress, $\sigma_{app}$ [37]:

$$K_I + iK_{II} = \sigma_{app} \sqrt{\pi a} \left( Y_I + iY_{II} \right)$$

(56)

where $Y_I$ and $Y_{II}$ are the geometry factors of the scarf joint with a crack at the bimaterial interface. Now the strain energy release rate can be written as

$$G = \left( \frac{1 - \beta^2}{E_{eff}} \right) \left( Y_I^2 + Y_{II}^2 \right) \sigma_{app}^2 \pi a$$

(57)

This solution requires the geometry factors of the scarf joint to be known. Because of the complex geometries, inhomogeneous and anisotropic material properties, no analytical expressions are currently available. For each flaw size, a reference stress, $\sigma_{ref}$, of 1 MPa was applied, the Mode I and II SERR were determined using the finite element model, and the geometry factors were determined in Equation (57). Figure 52 presents the geometry factors as function of normalised flaw size. The largest values correspond to the cases when the material around the crack tip is surrounded by $0^\circ$ plies. It should be noted that these geometry factors are specific to the particular layup and geometry of the specimen and cannot be generalised. However, the factors can be determined for any specimen configuration using the approach outlined.

It is now possible to predict the onset of crack propagation, defined as when the SERR, $G$, approaches the critical value, $G_C$:  

79
\[
\frac{G}{G_c} \geq 1
\]  

(58)

Figure 52: Crack geometry factor of scarf joint with similar adherend.

In a mixed mode loading condition, the critical SERR was defined by the Benzeggagh-Kenane (B-K) fracture criterion \([48]\), which is shown in Equation (59) where the exponent, \(\eta\), is an empirical parameter that needs to be experimentally calibrated.

\[
G_c = G_{IC} + (G_{IIIc} - G_{IC}) \left( \frac{G_{II}}{G_{I} + G_{II}} \right)^\eta
\]  

(59)

The load-carrying capacity of a scarf joint containing a disbond can be determined as the stress that produces the critical SERR, \(G_c\). From Equations (57), (58) and (59), the joint strength is presented in Equation (60). The methodology to the LEFM method is shown in Figure 53.

\[
\sigma_{ult} = \sigma_{ref} \sqrt{\frac{G_c}{G}}
\]  

(60)
5.4.2. Results

Equation (60) was employed to determine the joint strength of 3° and 5° scarf joints at any flaw size. This approach assumes that the joint experiences catastrophic fracture as soon as the critical SERR is first reached. A comparison between this prediction and the experimental results is shown in Figure 54. The LEFM predictions are in fairly good correlation with the experimental results, and are conservative. In particular, the LEFM predictions are reliable for flaw sizes of over 3 mm, or greater than 5% of the overall length, but significantly over-predicted the joint strength for smaller flaws. This is due to the SERR approaching infinity as the flaw size approaches zero.
Figure 54: Comparison between LEFM prediction and experimental results
(Top - 5° scarf angle, Bottom 3° scarf angle).

From Figure 54, the LEFM predictions show that for a scarf joint with a 5° scarf angle, at a flaw size of 5.2 mm the joint strength starts to increase with increasing flaw size, until a flaw
size of about 7 mm is reached, after which the joint strength decreases with flaw size. The local peak of joint strength at 7 mm is roughly equal to the joint strength at 4 mm. This means that at flaw sizes from 4 mm to 7 mm, a stable crack growth region would be expected. Similarly, for the 3º scarf joints, a local minimum was observed at a flaw size of 8.5 mm. The strength of the joints were matched at 7 mm and 12 mm. In both curves, the minima was observed to be between two load carrying (0º) plies. Comparing the predictions against experimental values, the LEFM results would be too conservative, and a methodology that captures progressive crack growth would be required. This type of analysis is presented in the VCCT model and CZM results in the following sections.
5.5. Virtual Crack Closure Technique

5.5.1. Methodology

The VCCT model was employed to model crack propagation along the composite-adhesive interface. The numerical model was analysed under increasing displacement until failure and the ultimate strength of the joint was calculated from the maximum applied load. Disbonds of various lengths were embedded at the interface, by modifying the properties of the contact surface and the location of the crack tip. Two different sets of material properties were investigated to study whether using composite or composite-adhesive (adhesive) fracture properties presented in Table 4, Table 5 and Table 6, would yield better correlation with experimental results. The models using adhesive or composite fracture properties for the crack growth interface are labelled VCCT-adhesive and VCCT-composite, respectively.

5.5.2. Results

As shown in Figure 55, The VCCT model joint strength predictions using composite fracture properties provided excellent correlation with experimental results for flaw sizes greater than 3 mm in length. The better correlation using composite fracture properties over adhesive fracture properties is consistent with the SEM observations that fracture propagated in the composite adherends, and that the composite properties controlled the crack growth. However, as the flaw size decreases below 3.0 mm, the VCCT model predicted increasingly high strengths, exceeding the strength of pristine joints.

A comparison between the stress at initiation of cracking and the joint strength using the VCCT-composite model is given in Table 9, where the crack initiation was detected as the first instance of crack growth. From these results, it can be seen that for most flaw sizes the initiation of cracking occurred at stress levels very close to the joint strength. This agrees with the experimental observations, where only slight cracking noises were heard just before catastrophic failure for the flaw sizes investigated. The results in Table 9 also show that for flaw sizes of 5.2 mm and 6 mm there was a significant difference between the crack initiation and joint strength, indicating a larger stable period of crack growth. This agrees with the LEFM results presented previously that identified stable crack growth for flaw sizes between 4 mm
and 7 mm, and confirms the benefit of applying a progressive damage model for failure predictions.

Figure 55: Comparison between VCCT model predictions and experimental results. Filled symbols denote experimental results.

Table 9: Loads at the onset and propagation of flaw

<table>
<thead>
<tr>
<th>Flaw size (mm)</th>
<th>Crack initiation (MPa)</th>
<th>Joint strength (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.0</td>
<td>1396</td>
<td>1396</td>
</tr>
<tr>
<td>2.0</td>
<td>481</td>
<td>486</td>
</tr>
<tr>
<td>3.0</td>
<td>324</td>
<td>326</td>
</tr>
<tr>
<td>5.2</td>
<td>199</td>
<td>254</td>
</tr>
<tr>
<td>6.0</td>
<td>225</td>
<td>254</td>
</tr>
<tr>
<td>8.0</td>
<td>222</td>
<td>224</td>
</tr>
<tr>
<td>12.0</td>
<td>145</td>
<td>150</td>
</tr>
</tbody>
</table>
Chapter 5 – ON MODELLING THE FRACTURE BEHAVIOUR OF SIMILAR ADHEREND JOINTS

5.6. Cohesive Zone Model

5.6.1. Methodology

The CZM was applied to model progressive crack growth and failure in the scarf joints. To investigate the sensitivity of the location of the cohesive zone within the adhesive layer, models were created with the CZM elements embedded at the composite-adhesive interface and adhesive mid-plane as shown in Figure 47. Three models were investigated: “Composite” – failure along the composite-adhesive interface using composite properties; “Adhesive, interface” – failure along the composite-adhesive interface using adhesive properties, and; “Adhesive, mid-plane” – failure along the adhesive mid-plane using adhesive properties. The numerical models were analysed under displacement control until failure and the joint strength was computed as the total load applied at the loaded end of the joint at maximum load.

5.6.2. Results

The results of the CZM predictions for all three models in comparison with the experimental joint strengths are shown in Figure 56. The CZM predictions of all three models at flaw sizes of less than 2.5 mm were observed to be generally similar. It was found that experimental results for small flaw sizes (3 mm or less) showed better agreement with the "Adhesive, Interface" rather than "Composite" model predictions. This suggests that fracture in joints at flaw sizes of 3 mm or less is likely dominated by cohesive failure of the adhesive. At larger flaw sizes, the model with composite properties gives the best correlation with the experimental results, and followed the experimental results closely. This agrees with the results from the VCCT model, and confirms the conclusion that the composite properties controlled the joint strength in the experimental results. There was very little difference between the choices of interface for the two models using adhesive properties. Unlike the VCCT model, the CZM was able to provide reasonable predictions of joint strength at small flaw sizes and the pristine joint, though the predictions were slightly higher than the experimental joint strengths. However, the CZM requires more calibration efforts than the VCCT in general.
Chapter 5 – ON MODELLING THE FRACTURE BEHAVIOUR OF SIMILAR ADHEREND JOINTS

Figure 56: Comparison between CZM model predictions. Filled symbols denote experimental results.
5.7. Concluding Remarks

Experimental results have shown that the fracture of composite scarf joints would propagate into the composite laminate adherend. Fractographic analysis showed composite materials composed mainly of loose, broken fibres and sheared matrix on the fracture surfaces of both adherends. Numerical models, calibrated to mode I and II interlaminar critical SERR, predicted the strength of the scarf joint accurately. The successful application of these composite properties to the numerical models would suggest that the fibres did not play a significant part in resisting crack propagation. Furthermore, the application of adhesive properties along the composite-adhesive interface or the adhesive mid-plane provided poorer predictions of joint strength. This suggests that the strength of scarf joints was controlled by the properties of the composite matrix. Furthermore, this confirms observations that the fracture plane contained a matrix layer that was smaller than the ply thickness and of the order of the fibre diameter. This behaviour only holds true when the fracture is cohesive or if the SERR of the adhesive is lower than the matrix.

The damage tolerance and durability is critical for composite structures due to its susceptibility to delaminations and disbonds. Through linear elastic fracture mechanics approaches shown in previous sections, it was found that the geometry factors for a range of crack lengths was capable of identifying the state of crack propagation. By designing for steady state crack propagation, composite plies and structures can be made tolerant against catastrophic failure during operation by ensuring the crack size smaller than the detectable limit would not cause premature failure.

Experimental results showed that the loss of approximately a quarter of the bond length resulted in the loss of more than half the joint strength. In scarf repairs, as the disbond size increases, the stresses in the scarf are transferred to the surrounding adherend to cause disbond growth [1]. The analysis in this chapter has been focused on the interfacial fracture propagation along the bondline of the scarf joint using unidirectional composite plies. As scarf joints are designed with scarf angle aspect ratios between 1:20 and 1:40, fracture behaviour in the joints will likely involve interlaminar delaminations and in-plane ply fracture. Further research is needed to address the crack branching phenomenon and the effects of the load-carrying capability of scarf joints containing flaws.
Chapter 6.

STATIC STRENGTH AND FRACTURE BEHAVIOUR OF DISSIMILAR ADHEREND JOINTS

6.1. Introduction

Experimental results, observed in Section 4.2.2, have shown that scarf joints with dissimilar material systems exhibit fracture behaviour that is more complex in comparison to similar adherend joints in Section 4.2.1. An approach is developed to accurately predict the strength and fracture behaviour of composite scarf joints with dissimilar adherends. Fracture mechanics based methods, such as the cohesive zone model (CZM), the virtual crack closure technique (VCCT) and the continuum damage mechanics (CDM), were utilised to understand the fracture behaviour and predict its fracture path. This chapter presents a methodology in predicting complex fracture in adhesively bonded scarf joints.
6.2. Validation of Fracture Behaviour

6.2.1. Methodology

A preliminary study was performed to understand the fracture behaviour and path of composite fracture along the composite adhesive interface. A two dimensional plane strain numerical methodology was developed to validate experimental observations in Abaqus. Plane strain elements were used to model the adhesive and the composite adherends at the ply level, as a stack of individual plies rather a laminate. Material properties were applied based on values from Table 2 for the composite plies and adhesive. The fracture properties for the composite materials in Mode I and II delamination are shown in Table 4 and Table 5 respectively. Boundary conditions were applied on both ends of the scarf joint to replicate experimental testing constraints, and consisted of constrained displacements in all degrees of freedom, except for the loading displacement at one end. A non-linear implicit numerical analysis was performed using Abaqus/Standard.

Based on the fractographic observations described in the Section 4.2.2 (Figure 42 and Figure 43) that fracture occurred in the composite at a small distance, comparable to the fibre diameter, away from the composite-adhesive interface, the onset and propagation of cracks were assumed to take place along the composite-adhesive interface. Since the thin layer of resin-fibre material removed from the composite adherend was extremely small, its effect is not considered in this investigation.

The VCCT, as described in Section 2.4.2.1, was used to model damage propagation. Models corresponding to embedded disbond lengths of 3, 4, 6, 9, 12 mm were generated. Two models were employed to predict the strength of the scarf joints, as shown in Figure 57. The first numerical model, Model A, replicates experimentally embedded disbands at the repair-adhesive interface. The VCCT was assigned with repair fracture properties (Cycom970) and was embedded for the remaining length of the repair-adhesive interface. This model represents a disbond occurring along the repair-adhesive interface. In the second numerical model, Model B, the disbond was located at the parent-adhesive interface, thus, allowed the VCCT to be embedded along the parent-adhesive interface and was assigned with parent fracture properties.
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(HexPly914). The numerical model of the scarf joint was loaded in tension by increasing end displacement until the joint completely failed.

![Diagram showing Model A and Model B configurations]

Figure 57: Model A: Disbond embedded at the repair-adhesive interface. Model B: Disbond located at the parent-adhesive interface.

6.2.2. Results

Numerical results, shown in Figure 58 and Figure 59, from Models A and B provided much insight on the fracture behaviour of mismatched scarf joints. In this section, the terms “Cycom970 joint” and “HexPly914 joint” are, respectively, used to refer to joints with Cycom970 and HexPly914 as the parent adherend. It was observed that Model A predicted the strength of Cycom970 joints well but overpredicted the strength of HexPly914 joints. This is due to the difference in critical SERR between Cycom970 and HexPly914 composite plies, as shown in Table 4 and Table 5. Furthermore, Model A does not represent experimental observations of its fracture mode. The fracture mode in both joint designs belonged to the parent adherend, as illustrated in Fig. 4. Results from Model B showed that for disbond sizes greater than 6 mm, the model was fairly capable of predicting the strength of the joint. However, at disbond sizes of less than 6 mm, the results overpredicted the strength of the joint. It should be mentioned that such over prediction using fracture mechanics approach is consistent with the experimental results. Between the two designs, Model B showed consistency in correlating with experimental values, suggesting that Model B has captured the fracture mode of the joint.
However, Model B has failed to capture the fracture path as described in Figure 43, leading to an overprediction of the joint strength with decreasing disbond size.

Results shown in this section suggest that accurate simulation of the fracture mode and path is capable of accurately predicting the strength of mismatched joints. However, current methods are conservative in predicting the fracture behaviour, the material fractured and the location of the fracture. Current methods showed fracture only at one interface as it does not allow transitioning crack paths. There is a need to produce numerical models that is capable of predicting the strength of mismatched joints by selecting the path of least resistance based on the size of the flaw, loading conditions and the presence of geometric eccentricities.
Figure 58: Numerical prediction of scarf joints with Cycom970 parent adherend

Figure 59: Numerical prediction of scarf joints with HexPly914 parent adherend
6.3. An Approach to Predicting Fracture of Scarf Joints with Dissimilar Adherends

A modelling approach, as shown in Figure 60, was developed to capture the multiple fracture modes and complex crack bifurcation of adhesively bonded scarf joints with dissimilar adherends. The approach requires material properties that can be characterised into three groups; namely the elastic properties, the in-plane tensile fracture properties and the interlaminar fracture properties. Firstly, elastic material, in-plane tensile and interfacial fracture properties were obtained through experiments mentioned in Section 4.1. Then, the in-plane tensile and interfacial fracture experiments were modelled using the experimentally obtained material properties. The models were compared against the experiments to ensure its numerical accuracy. Finally, these material properties were applied to the scarf joint model to predict its strength and fracture behaviour.

Figure 60: A numerical methodology on the prediction of composite scarf joints with dissimilar adherends.
6.4. Identification of Material Properties

6.4.1. In-Plane Continuum Damage Mechanics Modelling

6.4.1.1. Modelling Methodology

A typical numerical model of the OHT specimen is shown in Figure 61. The gauge section of the specimen was modelled. It was defined to be the distance between the machine grips. The model was fixed on one end of the model and loaded in tension along the length of the specimen on the other end. Plane stress conditions were assumed due to the nature of the geometry and the fracture behaviour. The model was composed of three dimensional continuum shell elements that were paved parallel to the surface of the specimen. Each continuum shell represented one ply thickness. There were eight shells in total in the through thickness, similar to specimens in Section 3.2.3. A high mesh density was kept in the vicinity of the open hole and was gradually lowered as shown in Figure 61.

Material properties of the composite ply are found in Table 2. The Abaqus CDM (Hashin) model [52] for fibre reinforced composites, as explained in Section 2.4.2.3, was applied to the continuum shells to capture in-plane fracture behaviour observed in Section 4.1.2. For each composite material, OHT models were generated of unidirectional (0º) and (45º) plies. To generate (45º) plies, (0º) plies were rotated at 45º. Simulations were performed on OHT specimens in an explicit analysis due to the necessity of the Abaqus CDM model to run in Abaqus explicit.
The fracture energies in the $0^\circ$ and $90^\circ$ ply direction, $\Phi_{1,2}$, were varied over a range of values by manually changing the in-plane fracture energies and comparing the strength of the OHT specimens against experiment data in an iterative process. Since the composite materials were of fabric material, the fracture energies in the $0^\circ$ and $90^\circ$ fibre direction were kept identical. The solution of the in-plane fracture energy for each composite material would be obtained when the numerical strength of the OHT specimens in unidirectional ($0^\circ$) and ($45^\circ$) plies match experimental values.
6.4.1.2. Results

The numerical model and experimental OHT specimens are shown in Figure 62. In the numerical results shown in Figure 62, elements in red do not carry load and have fractured. Experimentally and numerically, composite material systems, Cycom970 and HexPly914, showed similar fracture behaviour when loaded in tension. In the unidirectional (0°) specimens, it was observed that fibres running along the 0° experienced rupture and the specimens failed perpendicular to the direction of the load. Specimens with a unidirectional (45°) layup experienced fracture at a 45° angle. Fibres perpendicular to the fracture were observed to have ruptured. Overall, numerical modelling of OHT specimens composed of fabric composite material systems showed excellent resemblance in fracture behaviour with experiments.

![Figure 62: Comparison between experimental and numerical fracture behaviour of OHT specimens](image)

The results of the iterative method on the OHT specimens are shown in Figure 63. It was observed that the in-plane fracture energy of the Cycom970 and HexPly914 in the 0° and 90° ply directions for the Abaqus CDM model for composite materials were approximately 10
KJ/m². This coincides with experimental observations that the strength of both composite material systems was similar due to the presence of T300 fibres in both systems.
Figure 63: Correlation of numerically derived OHT strength of HexPly914 and Cycom970 against experiments.
6.4.2. Interlaminar Propagation Modelling

6.4.2.1. Modelling Methodology

DCB and ENF test specimens were modelled, as shown in Figure 64. Both models were composed of solid, eight noded elements. Each element represented one ply thickness. There were 16 and 12 plies for the Cycom970 and HexPly914 specimens respectively. Cohesive elements were embedded in the mid plane of the model to replicate a bonded interface. Pre-embedded disbonds were modelled by removing cohesive elements. A disbond length of 50 mm was modelled from the edge of the DCB specimen. In the ENF specimen, a disbond was modelled from the edge of the specimen to a length of 25 mm from one of the bottom rollers. Since the interfacial thickness of the adhesive was much smaller than the width of the actual specimen, plain strain conditions were applied to the model to reduce computational efforts. The models were only 1 mm wide, with only one element in the width direction. The elements were constrained in displacement and rotation along the width of the specimen to enforce plane strain conditions.

The elements were assigned with linear elastic material properties of the Cycom970 and HexPly914 fabric composite material from Table 2. The cohesive elements were assigned with interfacial fracture properties of the Cycom970 and HexPly914 from Table 4 and Table 5. The cohesive zone length, as mentioned in Section 2.4.2.2.5, introduced by Turon et. al., was implemented in the models to ensure that the cohesive zone can be sufficiently represented with elements [53].

The DCB specimens were assigned boundary conditions at the disbonded end of the specimen, where a displacement was set on the top surface and fixed from moving on the bottom surface, as shown in Figure 64. The support rollers in the ENF tests were modelled. Each roller had a diameter of 2.5 mm and they were spaced 50 mm apart. The rollers were modelled using shell elements. The elements were assigned to be rigid during the analysis. The bottom rollers were designated to be fixed and the top roller was set to a downward displacement of 5 and 7 mm for the HexPly914 and the Cycom970 specimens respectively, as shown in Figure 64. The rollers and the disbonded surfaces along the mid-plane were assigned general contact.
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definitions. The mid-plane surfaces and the bottom rollers were designated to be frictionless. The middle roller was assigned to be rough, where the two contact surfaces do not slide.

Explicit analyses were performed on the models for consistency with the simulations in the previous section. The force-displacement data were extracted from the edge of the DCB specimen and the top roller in the ENF model. Numerical values were compared against experimental results to validate the accuracy of the modelling technique.

![Figure 64: DCB and ENF test specimens modelled in Abaqus](image)
6.4.2.2. Results

Force-displacement curves of the experimental tests and numerical models are shown in Figure 65. It is noteworthy to mention that the nature of the explicit analysis causes spikes or fluctuations in the force displacement curves. This numerical artefact was observed during crack initiation and propagation, and was attributed to the balance of forces during each time step in an explicit analysis. All damage states were observed to occur in the middle of the laminate.

Experimentally, it was observed that the composite materials, HexPly914 and Cycom970, had similar stiffnesses, as shown in Figure 65. It was observed than the numerically predicted stiffnesses of the HexPly914 were fairly accurate but the numerically modelled stiffness of the Cycom970 were lower than experiments by approximately 25%. The flexural modulus of both material systems were derived using the Corrected Beam Theory for mode I and II, as shown in (61) and (62) respectively [22]. The averaged flexural modulus of the HexPly914 and Cycom970 are shown in Table 10. Despite both material systems having similar in-plane tensile stiffness, the Cycom970 has a higher bending stiffness than the HexPly914. This behaviour validates statements made by various authors that a satin weave is more pliable than a plain weave [89, 90].

\[
E_f^I = \frac{8 (a + |\Delta|)^3}{CBh^3} \quad (61)
\]

\[
E_f^{II} = \frac{I^3}{4BCh^3} \quad (62)
\]

Table 10: Average flexural modulus of HexPly914 and Cycom970.

<table>
<thead>
<tr>
<th>Flexural Modulus</th>
<th>$E_f$ (GPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>HexPly914</td>
<td>37.4</td>
</tr>
<tr>
<td>Cycom970</td>
<td>53.8</td>
</tr>
</tbody>
</table>

The discrepancies observed between numerical and experimental results is due to the modelling nature of the material systems. The numerical models were generated at a ply-by-ply level that
Chapter 6 – STATIC STRENGTH AND FRACTURE BEHAVIOUR OF DISSIMILAR ADHEREND JOINTS

ignores the weave within each ply. While both material systems have similar in-plane stiffnesses, it is important to note that the HexPly914 specimens (3.96 mm) were thicker than the Cycom970 specimens (3.52 mm) which suggests that the Hexply914 specimens would have a higher bending stiffness. However, experimental and numerical results suggests that the Cycom970 plain woven composite material provides higher flexural modulus per unit thickness than the satin weave HexPly914. Since the composite laminate was modelled using solid, eight noded elements with linear elastic material properties, the weave of the composite material was not modelled and resulted in a lower Cycom970 bending stiffness than the experimental results as shown in Figure 65. Since there is a loss of bending stiffness, the strength and fracture behaviour of later models will be affected.
Figure 65: Numerical prediction of Mode I and II force-displacement curves of HexPly914 and Cycom970.
6.5. Predicting Fracture of Scarf Joints with Dissimilar Adherends

6.5.1.1. Modelling Methodology

The dissimilar adherend joints were modelled in Abaqus. A typical schematic of the model is shown in Figure 66. Two sets of dissimilar adherend joints were modelled and labelled identical to experiments in Section 3.3.2. The entire gauge length of the scarf joints was modelled. The composite plies and film adhesive were modelled using three dimensional continuum shell elements, with one element per ply or adhesive thickness. The continuum shell elements were aligned either parallel to the plane of the ply for composite materials related elements or parallel to the plane of the scarf for adhesive related elements. It is important to note that due to the nature of the geometry, wedge like regions will occur throughout the model. Since it is not possible to align the continuum shell elements parallel to the plane of the ply, six-noded, solid, wedge elements will be required to fill up these regions. Cohesive elements were embedded between each continuum shell element in the composite laminates and at the composite-adhesive interfaces along the plane of the scarf. With conditions similar to Section 6.4.2, plain strain settings were applied to the model.

Continuum shell and wedge elements belonging to the composite material, Cycom970 and HexPly914, and the adhesive, VTA260, were assigned linear elastic material properties from Table 2 [86]. The Abaqus CDM model for in-plane damage [52] was applied to the continuum shells to capture in-plane fracture behaviour of the composite material and the film adhesive. Fracture properties of the composite material were taken from the results of the iterative method mentioned in Section 6.4.1. The fracture properties of the film adhesive were obtained from manufacturer’s data sheet [86]. Cohesive elements located in the Cycom970 and HexPly914 laminates were assigned its interfacial fracture properties from Table 4 and Table 5. Cohesive elements along the composite-adhesive interface were assigned interfacial fracture properties belonging to the adjacent composite material. Due to the manufacturing process of the dissimilar adherend scarf joints, mentioned in Section 3.3.2, resin rich regions were designated in the model. The regions were assigned with an isotropic composite resin material property from the manufacturer’s data sheet [77, 78].
Boundary conditions were assigned at both ends of the scarf joints. The models were loaded in tension on one end and fixed on the other. Explicit analyses were performed on the models and the strength of the joints were recorded and correlated against experimental data to validate the reliability of the modelling methodology for dissimilar adherend scarf joints.
6.5.1.2. Results

Figure 67 and Figure 68 show the fracture behaviour of dissimilar scarf joints. Fractures that have completely failed along the plane of the figures are shown in red. Interfacial fractures are shown as voids in between coloured elements. The mesh of the numerical model was removed from the figures to provide clarity. Results show that the dissimilar adherend joint model is capable of coupling the in-plane continuum damage mechanics model with the interfacial fracture model. This allows for the crack to initiate anywhere in the scarf joint and propagate either between interfaces, due to disbonding and delaminations, or due to net tension failure.

In comparison with experimental results in Figure 43, Figure 67 presented good correlation. The pristine scarf joint and the joint with a 3 mm disbond, failed in net tension failure in the parent material, composed of the HexPly914. Results showed that delaminations occurred in the HexPly914 laminate to bridge cracks through the thickness of the joint. This coincides with the non-consistent sawtooth behaviour seen commonly in composite materials failing in net tension fracture. For joints with disbonds of 6 mm and 12 mm in size, the fractures were observed to occur at the crack tip, where the crack travelled across the adhesive and along the plane of the scarf, before failing in net tension failure at the tip of the joint. Similarly, numerical fracture paths were identical to experimental observations.

Figure 68 presents crack propagation for dissimilar scarf joints using Cycom970 for the parent adherend. The pristine condition scarf joint model showed good correlation with experimental results. The joint failed in net tension failure at the repair HexPly914 adherend. For disbond lengths of 3 mm, 6 mm and 12 mm, the joints exhibited different behaviour to the experimental results. At 3 mm, the fracture occurred at the Cycom970 parent adherend instead of the HexPly914 repair adherend. At 6 mm and 12 mm, the fracture occurred at the crack tip. The crack travelled across the adhesive and the Cycom970 adherend, indicating fracture in net tension failure.
Figure 67: Fracture behaviour of HexPly914 parent joints.

Figure 68: Fracture behaviour of Cycom970 parent joints.
The strength of the dissimilar adherend joints are presented in Figure 69. An additional model with a disbond length of 9 mm was generated to provide a better understanding of the change in strength to disbond length. Both HexPly914 and Cycom970 parent scarf joint models provide good correlation with experimental values.

Despite inaccuracies in crack propagation of the Cycom970 parent with disbond lengths of 3 mm, 6 mm and 12 mm, the strength of the joints correlated well with experimental values. It was previously identified in Section 6.4.2, that the bending stiffness of the Cycom970 was lower in the numerical models due to the improved bending stiffness of a plain weave composite material. This suggests that Cycom970 laminates in the dissimilar adherend models would experience greater deformation. This change in geometry resulted in changes in load paths and stress concentrations. Thus, the failure of the joint has produced a crack path that is different to specimens in the experiments.
Figure 69: Numerical prediction of HexPly914 parent joints.
6.6. Concluding Remarks

A numerical modelling methodology has been developed to predict the strength and crack propagation of dissimilar adherend scarf joints with disbonds. Results obtained from the numerical models correlated well with experimental results. The two tiered modelling structure proved to be a robust methodology that ensures successful prediction of dissimilar adherend scarf joints with disbonds. Modelling of woven fabrics proved to be difficult due to variations in bending stiffnesses caused by a difference in weave. It was found that the plain weave provided higher bending stiffness than the satin weave. The in-plane damage mechanics model method requires an iterative method to derive the in-plane fracture energy for crack propagation.
Chapter 7.

FATIGUE LIFE PREDICTION METHODOLOGY

7.1. Introduction

A key airworthiness certification requirement for adhesively bonded scarf repairs of aircraft composite structures is to demonstrate through analysis or tests that catastrophic failure due to fatigue, environmental effects, manufacturing defects, or accidental damage will be avoided throughout the operational life of the aircraft [20]. While methods for predicting the strength of composite structures, without repairs [91] and with repairs [92, 93], are available in literature and demonstrated in previous sections, there is a lack of analytical methods that can predict the growth rates of disbonds in scarf repairs. While the fatigue properties of a composite material can be determined using double cantilever beam (DCB) and end notched flexure (ENF) tests, it is not clear how the fatigue endurance of scarf repairs can be estimated. This section aims to present a fracture mechanics based approach for predicting the disbond growth rate and fatigue life of scarf joints containing pre-existing flaws. The approach will be validated against experimental results from Section 4.3.
7.2. Methodology

For a scarf joint containing an embedded disbond, it was explained in Section 5.4.1, that the energy release rate for a given load, $G_{\text{max}}$, at the crack tip can be determined using a reference stress method. The key equation, Equation (60), is presented again in Equation (63), where $G_{\text{I,ref}}$ and $G_{\text{II,ref}}$ are the mode I and mode II strain energy release rates when the scarf joint with a flaw of length $a$ is subjected to a reference stress, $\sigma_{\text{ref}}$.

\[
G_{\text{I, max}} = \left(\frac{\sigma_{\text{max}}}{\sigma_{\text{ref}}}\right)^2 G_{\text{I, ref}}
\]

\[
G_{\text{II, max}} = \left(\frac{\sigma_{\text{max}}}{\sigma_{\text{ref}}}\right)^2 G_{\text{II, ref}}
\]  

(63)

Under mixed-mode loading, the two individual strain energy release rates can be combined to using the Benzeggagh-Kenane (B-K) fracture criteria to determine the SERR of the applied load ($G_{\text{max}}$), critical SERR ($G_C$), threshold SERR ($G_{\text{TH}}$) and Paris’ law material constants ($C$ and $m$), which is given by Equations (64), (65), (66), (67) and (68) [48, 66, 67].

\[
G_{\text{max}} = G_{\text{I, max}} + (G_{\text{II, max}} - G_{\text{I, max}}) \left(\frac{G_{\text{II, max}}}{G_{\text{I, max}} + G_{\text{II, max}}}\right)^\eta
\]

\[
G_C = G_{\text{IC}} + (G_{\text{IIc}} - G_{\text{IC}}) \left(\frac{G_{\text{II, ref}}}{G_{\text{I, ref}} + G_{\text{II, ref}}}\right)^\eta
\]

\[
\Delta G_{\text{th}} = \Delta G_{\text{Ith}} + (\Delta G_{\text{IIth}} - \Delta G_{\text{Ith}}) \left(\frac{G_{\text{II}}}{G_T}\right)^{n_{\text{th}}}
\]

\[
\ln(C) = \ln(C_{\text{II}}) + [\ln(C_{\text{I}}) - \ln(C_{\text{II}})] \left(1 - \frac{G_{\text{II}}}{G_{\text{tot}}}\right)^{n_C}
\]

\[
m = m_I + (m_{\text{II}} - m_I) \left(\frac{G_{\text{II}}}{G_{\text{tot}}}\right)^{n_m}
\]  

(64)

(65)

(66)

(67)

(68)

For the VTM264, a value of $\eta = 1.75$, $n_{\text{th}} = 1.96$, $n_C = 0.35$ and $n_m = 1.85$ was used in the B-K fracture criteria. These values were empirically determined from literature with good agreement to experiments [48, 66, 67].

At this point, the SERR of the structure is checked against the limiting conditions of the fatigue regime, namely the threshold and critical SERRs. If $G_{\text{max}} \geq G_C$, the structure has failed. If $\Delta G \leq \Delta G_{\text{th}}$, the crack has stopped propagating.
The growth rate of disbonds, as shown in Equation (69), in composite structures under fatigue loading is commonly expressed in terms of the cyclic strain energy release rates, $G_{\text{max}}$ and $G_{\text{min}}$, as shown in Equation (70), where the parameters $C$ and $m$ depend on the material and the loading ratio, $R$, as shown in Equation (71).

$$\frac{da}{dN} = C(\Delta G)^m \left[ 1 - \left( \frac{\Delta G_{th}}{\Delta G} \right)^{n_1} \right] \left[ 1 - \left( \frac{G_{\text{max}}}{G_C} \right)^{n_2} \right]$$ \hspace{1cm} (69)

$$\Delta G = G_{\text{max}} - G_{\text{min}} = G_{\text{max}}(1 - R^2)$$ \hspace{1cm} (70)

$$R = \frac{\sigma_{\text{min}}}{\sigma_{\text{max}}}$$ \hspace{1cm} (71)

The number of cycles spent in growing a disbond of unit length, $\Delta a$, can be determined using Equation (72). The number of cycles required for catastrophic fracture of the joint can be determined by the sum of unit cycles.

$$\Delta N = \left( \frac{da}{dN} \right)^{-1} \Delta a$$ \hspace{1cm} (72)

Figure 70 illustrates the above mentioned analytical methodology used in the determination of fatigue crack growth and life of composite scarf joints. Subsequent sections aims to validate this methodology.
Chapter 7 – Fatigue Life Prediction Methodology

Figure 7: An analytical method in the determination of fatigue crack growth and life of composite scarf joints.

\[ \frac{dN}{dP} = N \]

Determine life of structure \( N \)

Determine number of cycles \( N \) for crack growth

\[ N = N + \frac{dN}{dP} \]

Calculate mixed mode values of \( G_{	ext{mix}} \)

\[ G_{\text{mix}} = G_{\text{mix}} \]

\[ G_{\text{mix}} = G_{\text{max}} \]

\[ G_{\text{mix}} = G_{\text{max}} \]

Next analysis

Proceed to life

Decision:

\[ \begin{align*}
\text{If } \frac{dN}{dP} &< N \text{ then \text{Yes}} \text{ to next analysis} \\
\text{else if } \frac{dN}{dP} &> N \text{ then \text{No}} \text{ to next analysis} \\
\text{end analysis}
\end{align*} \]
7.3. Model Development

Finite element models were developed in Abaqus 6.10 [52] for this analysis. Plane strain four-node orthogonal (CPE4) and three-node triangular (CPE3) elements were used to model the adhesive and the composite adherends, with ply-level mesh refinement. Material properties for the composite plies are listed in Table 2, Table 4, Table 5 and Table 7 and the adhesive in Table 6 and Table 8. The properties of ±45° plies were derived using ply coordinate transformation equations, considering only the terms in the plane of the model [1]. Boundary conditions were applied on both ends of the scarf joint to replicate experimental testing constraints, and consisted of constrained displacements in all degrees of freedom, except for the loading displacement at one end. A non-linear implicit numerical analysis was performed using Abaqus/Standard to account for the effects of secondary bending.

Based on the fractographic observations described in the previous section that the fracture occurred in the composite at a small distance comparable to a fibre diameter away from the composite-adhesive interface, the onset and propagation of cracks was assumed to be within the composite, adjacent to the composite-adhesive interface. The effect of the extremely thin layer of resin-fibre material is ignored for simplicity; the crack path was assumed to be along the adherend-adhesive interface. With reference to Figure 71, Interface 1 represents disbonds and crack paths propagating between the feathered end of a scarfed adherend and the adhesive. Interface 2 represents disbonds and crack paths between the blunt end of the scarfed adherend and the adhesive. The distance (Δa) between the nodes along the bondline was refined to approximately 0.264 mm. Using the virtual crack closure technique (VCCT), the strain energy release rates (\(G_I\) and \(G_{II}\)) were determined. The results are displayed in Figure 72 for a range of flaw sizes (a), normalised by the length of the scarf (L), under an applied load of 1.0 MPa. From these results, the appropriate strain energy release rates at any given applied load and crack length can be readily computed from Equation (2). Drops in SERR (\(G_I\) and \(G_{II}\)) at large flaw sizes show crack propagation at unit load. This means that for loads greater than 1 MPa, the crack propagates and the joint fails at these flaw sizes.
Figure 71: Critical regions of the scarf joint embedded with the VCCT model. Two different numerical models were generated. Crack paths propagate in the same direction for both interfaces.

Figure 72: Numerically derived Mode I and II strain energy release rate ($G_I$ and $G_{II}$) curves of the scarf joint, under unit load (1 MPa), at a range of flaw sizes ($a$), normalised by the length of the scarf ($L$).
7.4. Analysis

7.4.1. Analytical Prediction on the Fatigue Life of Composite Scarf Joints

The fatigue crack growth rate of three degree scarf joints were analytically derived using the methodology mentioned in Section 7.2. The numerical predictions were obtained from Interface 1 and Interface 2 for scarf joints with an initial disbond of 12 mm and presented in Figure 73. It was observed that results from Interface 2 correlated well with experimental data and results from Interface 1 predicted a more rapid fatigue crack growth rate. Both curves predicted the trend of well, with a rapid rise in crack growth rate at approximately a quarter of the scarf length ($a/L = 1/4$).

Figure 73: Predicted fatigue crack growth of scarf joints with an initial disbond length of 12 mm.

Numerical predictions for scarf joints with disbonds of 6 mm are shown in Figure 74 in the form of an S-N curve. It was observed that the numerical results had greatly under predicted the life of the joints in Interface 1 and over predicted the life of the joints in Interface 2. It is important to note in Figure 74, that dashed lines represent predicted life of joints with disbonds in Interface 2 that were higher than a million ($1 \times 10^6$) cycles. This suggests that there are other fracture mechanisms are influencing the fatigue crack growth rate and life of scarf joints with a 6 mm disbond. It was also observed that the effects of small cracks in fracture mechanics
becomes prevalent at approximately $a/L = 0.125$ ($a = 10$ mm), and can be observed from experimental data in Figure 44 (c) and (d), where the fatigue crack growth rates varied greatly for crack lengths less than 10 mm. This suggests that the scarf joints experienced plasticity during cyclic loading which does not adhere to LEFM principles. Thus, the fatigue life prediction of crack lengths $a/L < 0.125$ can be conservatively predicted using LEFM methodology at Interface 1.

The fatigue life predictions of the scarf joint at varying disbond lengths and maximum fatigue loads at Interface 1 and 2 are presented in Figure 75 and Figure 76 respectively. Predictions for disbond lengths less than 10 mm ($a/L = 0.125$) were excluded due to over predictions by the numerical model.

It was observed that the methodology, mentioned in Section 7.2, predicted conditions for the fatigue threshold ($N = 1 \times 10^6$) and catastrophic failure ($N = 1$) of the joint. The methodology had predicted a rapid loss of load carrying capability and recovery at 125 MPa as shown in Figure 75 and Figure 76. Thus, the methodology provides an engineering tool to design damage tolerant composite structures and predict the criticality of disbonds during service.
Figure 75: Fatigue life predictions of scarf joints with varying disbond lengths and loads at Interface 1.
Figure 76: Fatigue life predictions of scarf joints with varying disbond lengths and loads at Interface 2
7.5. Concluding Remarks

The methodology has proven to adhere to airworthiness requirements by determining states of slow growth and rapid growth in scarf joints with disbonds. For three degree scarf joints, at crack lengths larger than a minimum disbond size \((a/L > 0.125)\), the LEFM methodology has shown to be capable of predicting the fatigue crack growth of joints accurately. At small disbond sizes \((a/L < 0.125)\), the methodology over predicted that the three degree scarf joint would withstand cyclic loads past the fatigue threshold typically set at a million cycles. This is due to the inability of the LEFM methodology to predict failure unrelated to disbond lengths such as net tension failure and plasticity. Overall, this methodology allows for engineers and technicians to better gauge the life of scarf joints and repairs by minimising the need for scheduled checks.

It was observed in experiments that the crack propagated from Interface 1 to Interface 2 and in numerical predictions that Interface 2 predicted the fatigue crack growth rate more accurately. However, numerical predictions for Interface 1 predicted a higher fatigue crack growth rate. This suggests that while it required a certain SERR to propagate across the adhesive at the crack tip, less SERR was required to propagate along Interface 2 than to propagate back across the adhesive to Interface 1. The fracture based modelling did not allow crack paths which were shown in experiments.
Chapter 8.

CONCLUSION AND FUTURE WORK

8.1. Conclusion

Damage tolerance and strength of scarf repairs and joints are important design consideration due to the brittle nature of composite materials and the susceptibility of the repairs to defects and damage. Despite advancements in numerical methods and application of composite repairs to critical aircraft structures, there is a lack of understanding of composite-composite bonded repairs and modelling methodology to predict the damage tolerance of composite scarf repairs. This research project aims to develop a set of modelling methodologies that accounts for the fracture behaviour of bonded scarf repairs under static and fatigue loading and, thus, allow for dissimilar composite material systems to be applied for repairs.

A review of current literature on the airworthiness standards for aircraft structures has identified the following requirements to predict the damage tolerance of composite structures:

- Bonded repairs need to withstand the design ultimate load with damage up to a detectable threshold, and;
- Maintain continued safe flight with the complete disbond of the repairs.
- The identified damage is characterised into three categories based on its fatigue crack growth behaviour; no-growth, slow-growth, and arrested-growth.

The modelling methodology would need to adhere to these standards using the current state-of-the-art in predictive modelling tools. In achieving this, experiments and numerical models were developed to understand the fracture behaviour of adhesively bonded composite scarf joints under static and fatigue loads.
8.1.1. Fracture Behaviour of Adhesively Bonded Scarf Joints

Experimental results have shown that the fracture of composite scarf joints at room temperature would propagate into the composite laminate adherend near the composite-adhesive interface along the plane of the scarf. Fractographic analysis of the fracture surface showed loose, broken fibres and sheared matrix on the fracture surfaces of both adherends. This shows that there is a need to consider fracture at the composite-adhesive interface in adhesively bonded joints, a behaviour that does not occur in metal joints.

Numerical models have been generated to consider this fracture behaviour. Fracture models that were embedded with mode I and II interlaminar critical SERR of the composite matrix, predicted the strength of the scarf joint accurately. The application of adhesive properties along the composite-adhesive interface or the adhesive mid-plane provided poor predictions of joint strength. This suggests that the strength of scarf joints was controlled by the properties of the composite matrix. Furthermore, this confirms observations that the fracture plane contained a matrix layer that was smaller than the ply thickness and of the order of the fibre diameter. Results show that VCCT and CZM with composite properties should be applied to bonded composite joints.

8.1.2. Fracture Behaviour of Dissimilar Adherend Scarf Joints

Experiments performed on dissimilar adherend scarf joints showed that fracture occurred at the composite-adhesive interface, identical to similar adherend scarf joints. Fracture behaviour in the dissimilar adherend scarf joints was found to be dependent on the size of the flaw size, the strength, and the stiffness of the composite material systems.

A numerical modelling methodology has been developed to predict the strength and crack propagation of dissimilar adherend scarf joints with disbonds. Results obtained from the numerical models correlated well with experimental results. The modelling methodology has been shown to be a robust methodology that ensures successful prediction of dissimilar adherend scarf joints with disbonds. The in-plane damage mechanics model method requires an independent method to derive the in-plane fracture energy for crack propagation.
8.1.3. Scarf Joints under Cyclic Loading

Scarf joints were placed under cyclic loading to determine its fatigue crack growth rates at various crack lengths. It has been observed that joints with small disbonds ($a/L < 0.125$) showed large variation in time taken for the flaw to grow a small increment. Changes in compliance were observed to increase during periods of no crack growth, followed by a sudden increase in crack length and compliance. Since the adhesive has lower shear strength than the composite, the adhesive will experience plasticity and, thus, deformation and changes in compliance without visible crack growth.

An analytical model has been developed to predict fatigue crack growth rates of scarf joints with disbonds. The disbonds need to be greater than a minimum size ($a/L > 0.125$) for the methodology to provide accurate results.

8.1.4. Comments on the Dominant Fracture Behaviour of Bonded Scarf Repairs

Scarf repairs are traditionally bonded with a tough adhesive. This ensures that the repair recovers the design ultimate load of the structure, and in the case of failure, fracture occurring away from the bondline of the adhesive. However, composite structures consist of materials with different physical properties. This means that the fracture behaviour in adhesively bonded composite scarf repairs follows a path of least resistance. Results have shown that the fracture occurs in the matrix of the composite near the composite-adhesive interface and not in the adhesive. In some cases, the strength of the matrix along the bondline was observed to be stronger than the net section, leading to net tension failure. It is important to note that this thesis has applied an adhesive that is tougher than the matrix of three different composite material systems. Thus, other forms of fracture behaviour are possible in other bonded repairs when the adhesive is weaker than the matrix. This could be due to changes in environmental conditions or applications of non-standard adhesives.

8.1.5. Comments on Similar and Dissimilar Bonded Scarf Repairs

Studies on dissimilar bonded scarf repairs were performed to develop analysis methods for dissimilar bonded scarf repairs. The first study investigated on the fracture behaviour of similar
adherends. The second study investigated on the fracture behaviour of dissimilar adherends, a relatively more complex structure than the subjects in the first study. It was observed that the strength of the structure was governed by the material with the lowest fracture toughness. In dissimilar repairs, joint strength was controlled by adherends with the lower fracture toughness. This behaviour allows better predictability of fracture paths over joints with similar adherends.
8.2. Future Research

This thesis has provided a basic framework in predicting the damage tolerance and strength of scarf joints with disbonds. Numerical modelling methodologies that are capable of predicting various failure behaviours have been identified. The work in this thesis can be expanded to investigate the fracture behaviour of scarf repairs of composite panels with dissimilar adherends and, different forms of loading and environmental conditions.

8.2.1. Fracture Behaviour of Scarf Repairs with Dissimilar Adherends and Co-Bonded Substrates

Scarf joints are two-dimensional representations of scarf repairs along the principle loading direction. Scarf joints are single load path structures. Its strength is dependent on the failure of adhesives along the bondline or laminate fracture, depending which is weaker. Scarf repairs of composite panels, however, sheds load around the repair when disbond occurs. Scarf repairs of composite panels have been reported to carry higher load than an equivalent scarf joint [1].

Due to the radial nature of the scarf repair, scarf repairs potentially presents more fracture behaviours than scarf joints. Since this thesis was set out to predict the damage tolerance of scarf repairs for aircraft structures such as wing skins and fuselages, investigations need to be performed on scarf repairs to identify fracture behaviours not observed in scarf joints. Furthermore, scarf repairs on aircraft structures are designed to be multiple load path structures. Wing and fuselage skins are designed to experience tension and compression due to bending, and shear due to torsion. This means that scarf repairs need to be loaded under bi-axial loading and constraints. Effects of Mode III interlaminar fracture failure should also be investigated for repairs. This mode was avoided due to the nature of scarf joints.

It is also important to note that scarf repairs of aircraft structures maybe co-bonded to reduce turnaround time. This thesis performed tests on secondarily bonded specimens to ensure consistency of the scarf angle along the bondline and avoids chemical incompatibility of the repair to the adhesive during curing. However, further work should be performed to understand the fracture behaviour of co-bonded repairs in the presence of disbonds.
8.2.2. Variable Loading and Environmental Conditions

Two types of loading conditions for composite aircraft structures have been identified in literature: 1) Compression after impact and; 2) variable amplitude loading. Compression After Impact (CAI) is a critical component on the damage tolerance of aircraft structures [94, 95]. Composite materials were identified to be weaker under compression than tension. This means that a there is a need to predict the damage tolerance and strength of scarf joints under compression to prevent composite structures from failing during service. Variable amplitude loading should also be considered since constant amplitude rarely occurs during service. During service, aircraft structures experience variable tension-tension or tension-compression loading states. In building up confidence in the predicting the damage tolerance of repairs, failure modes associated with variable amplitude loading needs to be thoroughly understood. Such factors need to be considered to understand changes in its load carrying capabilities and fracture behaviour.

Aircraft structures experience changes in environmental conditions such as temperature and humidity during service. Bonded repairs are susceptible to a loss of performance at extreme environmental conditions. Under hot-wet conditions, adhesives would increase in ductility which results in a loss of its fracture toughness. On the other hand, at cold temperatures, the adhesive would become brittle, resulting in the inability to distribute load along the bondline. It is worthwhile to investigate the fracture behaviour of joints with defects under extreme environmental conditions.

8.2.3. Fatigue Behaviour of Scarf Repairs with Dissimilar Adherends

This thesis has investigated on the fatigue crack propagation of scarf joints of similar adherends. Results showed that the joints experienced changes in compliance without visible crack growth, suggesting that the joints experienced plasticity in the adhesives. Further tests can be performed on the adhesives to identify plastic behaviour under static and fatigue loads using simple coupon tests. The changes in compliance may also be due to crack initiation and propagation occurring away from the disbond. Non-destructive inspection can be performed on the scarf joints at various number of cycles to identify this occurrence. Lastly, tests should also be
performed on scarf joints and repairs with dissimilar adherends to identify a larger range of fracture behaviours and satisfy the scope of this thesis.
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