Evaluation of 3D failure modes in CFRP composite laminates

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**EVALUATION OF 3D FAILURE MODES IN CFRP
COMPOSITE LAMINATES**

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EVALUATION OF 3D FAILURE MODES IN CFRP COMPOSITE LAMINATES

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University in partial fulfilment of the requirement for
the degree of
Doctor of Philosophy**

2016

Statement of Originality

I hereby certify that the work embodied in this thesis is the result of original research and has not been submitted for a higher degree to any other University or Institution.

.. 25-November-2016. . . .
Date

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Abstract

Wind power is the most successful and cost effective technology for clean energy production. Currently most of wind turbine blades are made of fiber reinforced polymer (FRP) composites. In order to provide useful information to researchers for optimized design of multidirectional FRP composite structures, it is vital to investigate the static and fatigue behaviors of FRP composites associated with failure mechanisms, which is the aim of this research. Specifically, the contents of this thesis are as follows:

Firstly, the tensile and fatigue behaviors of three kinds of carbon fiber reinforced polymer (CFRP) laminates with stacking sequences of $[0/+45/-45]_s$, $[+45/-45/0]_s$ and $[+45/-45/0]_{2s}$ that have been extensively used by wind blade manufacturers were investigated. Results show that the fatigue performance of the $[+45/-45/0]_s$ laminate is more excellent than $[0/+45/-45]_s$ laminate. Comparing $[+45/-45/0]_s$ laminate and $[+45/-45/0]_{2s}$ laminate, the predicted fatigue strength at 1 million cycles for the latter is just a little higher than that for the former. The effect of embedded delamination location on the flexural behaviors of CFRP laminates with symmetric layup sequence of $[+45/-45/0]_{2s}$ were investigated. Results show that the embedded delamination with the same size at different positions or interfaces has different extent of negative effects on the flexural strength of $[+45/-45/0]_{2s}$ CFRP laminates, but exerts no effects on the initial flexural modulus of elasticity of the laminates. After failure, all the specimens were reloaded to obtain the residual properties. It might be concluded that the reduction ratio of the initial flexural modulus is positively correlated with the areas of parallel damage (whose plane is parallel to the loading direction) and the reduction ratio of the flexural strength is almost positively correlated with the area of all the damage. After investigation of static flexural behaviors, the flexural fatigue behaviors of the $[+45/-45/0]_{2s}$ CFRP laminate composites were studied. Results show that the flexural fatigue life at failure can be quantitatively determined by the cycle when the tracked secant stiffness degrades by 21%. The two-parameter Weibull distribution well

represents the statistical nature of the fatigue life data of the CFRP laminate. Combined with the Sigmoidal model, the Weibull analysis method reliably predicts the fatigue life as a function of the applied deflection level. Lastly, the effects of fiber orientation of adjacent plies on the mode I interlaminar fracture behaviors were studied. Results show that computed tomography (CT) scans enabled an insight into the real crack fronts of the three kinds of specimens after DCB tests. The mode I interlaminar fracture toughness at the initial stage (G_{init}) is almost the same for 0//0, 0//45 and 45//−45 interfaces. While the toughness at the crack propagation stage (G_{prop}) in specimens with 0//0, 0//45 and 45//−45 interface continuously increases. Besides, the numerical model with variable fracture toughness is applicable to represent the mode I interlaminar fracture toughness of the FRP specimens and in good agreements with the experiments.

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Abbreviations

FRP fiber reinforced polymer
CFRP carbon fiber reinforced polymer
GFRP glass fiber reinforced polymer
CT computed tomography
PMC polymer matrix composites
RTM resin transfer molding
DCB double cantilever beam
ENF end-notched flexure
MMB mixed mode bending
FRMM fixed ratio mixed mode
ADCB asymmetric double cantilever beam
VCCT virtual crack closure technique

Publications

Zhe Liu, Peifeng Li, Narasimalu Srikanth, Tong Liu, Gin Boay Chai. Quantification of flexural fatigue life and 3D damage in carbon fibre reinforced polymer laminates. Composites Part A: Applied Science and Manufacturing. 2016, 90, 778-785

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Chapter 1 Introduction

In this chapter, firstly the background of this research was introduced, secondly the objectives to be achieved in this research were indicated, then the layout of this dissertation was displayed, finally the findings and outcomes were concluded.

1.1 Background

With accelerating consumption of fossil fuels, recently the world is shifting the need of energy resources towards sustainable and renewable energy like wind energy. Wind power is the most successful and cost effective technology for clean energy production.

As the electricity generators of wind power, wind turbines are getting more and more widespread use because of the advantages of fast installation, free pollution, low operation and maintenance cost. The wind turbine blade, which is one of the main components of a wind turbine, is designed referring to aerodynamics for capturing the maximum energy from wind. Innovative materials with light weight and excellent mechanical properties are developed for wind blades. Among all kinds of materials, fiber reinforced polymer composites have been widely used for wind blades because of the advantages of light weight, high specific stiffness and excellent fatigue performances [1]. Nowadays most blades are made of glass fiber reinforced polymer composites (GFRP), but more and more researchers and designers are moving towards employing carbon fiber reinforced polymer composites (CFRP) because of its higher stiffness.

1.2 Problem statements and objectives

The FRP composites developed for wind turbine blades are subjected to static or fatigue loading in service. And wind turbines are expected to sustain its service for around 20-30 years as required because the installation and replacement is time-consuming and inconvenient. So it is vital to investigate the

static and fatigue behaviors of FRP composites associated with failure mechanisms.

Firstly, different stacking sequences are considered for different components of the wind blades, and different stacking sequences will lead to different mechanical performances of fiber reinforced polymer composites. But seldom researchers have investigated the effects on the fatigue performance of FRP composites. So the effects of layup sequence on the static and fatigue behaviors of FRP composites are necessary to be investigated.

Secondly, many factors, such as manufacturing errors, operational, or maintenance activities, will bring about damage, which could decrease substantially the strength and stiffness of the FRP composites. But some wind turbine should continue to operate even there are slight damage in the blades due to manufacturing errors or in-service operation to save the cost of installation or replacement. On the other hand, wind turbine blades in service are subjected to highly variable loads such as aerodynamic, gravitational and inertial forces. The aerodynamic and gravitational forces will bring about the existence of bending stress in the blades [2]. So it is vital to study the flexural behaviors of FRP composites with existing damage. Delamination is a typical failure mechanism of FRP composites, many researchers investigated the effects of delamination on the mechanical behaviors of FRP composites. But most of the researchers focused on the effects of delamination on the buckling and compression behaviors of FRP composites. Seldom researchers investigated the effects of delamination on the flexural behaviors of FRP composites.

Thirdly, the prediction of fatigue life and strength has been primarily based on experimental investigations. But fatigue life/strength results of fiber reinforced polymer (FRP) composites are usually variable because of the inhomogeneous and anisotropic features of FRP composites, even under carefully controlled testing conditions. So the fatigue life could be predicted based on the analysis of the dispersed fatigue results with statistical methods.

Global damage in CFRP composites subjected to fatigue loads is usually associated with stiffness degradation. Stiffness can reflect the laminate's condition and resistance capacity to fatigue loads, which can be helpful for life prediction or ranking residual strength data. While the stiffness degradation levels can change depending on composite component, loading type, loading level and layup sequence [3]. For displacement-controlled fatigue tests, the most widely used fatigue failure criterion is when the stiffness falls to a specified level. As there is no constant standard of defining fatigue failure criterion regarding stiffness degradation, a greater understanding of stiffness degradation for a CFRP composite laminate subjected to different fatigue loads is worthwhile. In addition, the fatigue failure mechanisms of FRP composites are complex and generally in the forms of delamination, debonding, matrix crack, and fiber failure [4]. But it is an unresolved problem on which is the major mechanism responsible for fatigue failure of FRP composites [5-7]. So it is necessary to observe the damage progression patterns during fatigue tests and determine how the accumulated damage lead to properties degradation of the FRP composites. Among sorts of non-destructive techniques, Computed Tomography (CT) technology has been proven to be a powerful tool for characterizing the damage of materials in recent years. Many researchers have used CT scanning technology to investigate the failure mechanisms of fiber FRP composites, which proved its strong practicability in this area. The use of X-ray CT technology associated with reconstruction and visualization software for analysing qualitatively and quantificationally the damage progression in CFRP laminates subjected to flexural fatigue loads has not yet been reported.

Lastly, the interlaminar fracture behaviors are necessary to be investigated. As delamination is a typical failure mechanism in FRP composites, which will lead to the degradation in stiffness, strength and fatigue life [8]. And delamination is only kind of failure mechanism during interlaminar fracture tests. Numerous researchers have investigated the interlaminar fracture toughness of FRP composites, but mostly unidirectional FRP composites.

However, the FRP laminates extensively employed in industrial applications are multidirectional (MD) and delamination propagation is more common between plies with different fiber directions in real composite structures [9]. Not much research has investigated the interlaminar toughness at the interface between plies with different fiber directions in multidirectional composites in comparison with unidirectional composites. The lack of research on this subject is mainly due to the difficulty that crack jumping may occur when delamination propagates, which might lead to inaccurate toughness values [10, 11]. But the selection on the layup of the FRP laminates for interlaminar fracture tests need to be considered to inhibit crack jumping. Due to the difficulty of determining the interlaminar fracture toughness at the interface between plies with different fiber directions in multidirectional composites, most of current engineering practices regard the interlaminar toughness at any interface of FRP composite structures as the same with that at 0//0 interface in unidirectional laminates. And most researchers use cohesive finite elements to model all the interfaces in multidirectional FRP laminates with interlaminar toughness at 0//0 interface in unidirectional FRP laminates [12-15]. However, Zhao Libin indicated that the parameters of cohesive fracture model applicable for the unidirectional laminates may not be necessarily suitable for multidirectional composite laminates [16]. More importantly, it is a controversial issue on whether the mode I interlaminar fracture toughness depends on fiber directions at the interface. So it is very important to investigate the interlaminar toughness values for interfaces between plies with different directions for more accurate prediction of delamination development in real FRP composite structures [17, 18]. And more and more research on the interlaminar toughness of multidirectional FRP laminates should be carried out for optimized design of real FRP composite structures. In addition, the delamination simulation in multidirectional FRP laminates may be influenced by the curved delamination front [19]. Many researchers modelled the crack front shape of the FRP composites under DCB tests [20-22]. But seldom researchers observed the

crack front shape by experiments. In most literatures focusing on interlaminar toughness of multidirectional specimens, the crack front is regarded as straight for the measurement of toughness. This assumption may not be reasonable, especially for multidirectional specimens [21-23].

This thesis focuses on the investigation of the mechanical behavior and the associated failure mechanisms of FRP composites subjected to different kinds of loads. Specifically, the objectives of this thesis are as follows:

- (a) Compare the static as well as fatigue behaviors of FRP composites with different layup sequences, which will expect to provide useful information to researchers for improving in-service properties of FRP composites.
- (b) Investigate the effects of delamination location on the flexural behaviors of FRP laminates, which will expect to provide useful information to researchers for predicting the mechanical properties and failure mechanisms of FRP composites with existing delamination.
- (c) Get further investigation of static and fatigue flexural behaviors of FRP composites which will expect to provide useful information to researchers for prediction of residual properties and fatigue life of FRP composites.
- (d) Investigate the dependency of the fiber orientation of the two adjacent plies on the mode I interlaminar toughness of FRP composite laminates, which will expect to provide useful information to researchers for predicting delamination propagation and optimized design of multidirectional FRP composite structures.

1.3 Dissertation overview and layout

The layout of the dissertation includes:

Chapter 1 provides a brief introduction including background, problem statement and objectives.

Chapter 2 gives a detailed literature review on the constitutions, manufacturing techniques, failure mechanisms and mechanical behaviors of FRP composites. Then the applications of X-ray computed tomography (CT) on FRP composites are also discussed.

Chapter 3 presents the principals and methods of experiments including material fabrication, mechanical tests and damage characterization.

Chapter 4 elaborates the first major set of results. The tensile and fatigue behaviors of three kinds of CFRP laminates with layup sequences of $[0/+45/-45]_s$, $[+45/-45/0]_s$ and $[+45/-45/0]_s$ that have been extensively used by wind blade manufacturers were investigated. CT scanning technology was employed to observe the damage patterns during the fatigue tests. Two models were used to represent the trends of fatigue life for the two kinds of laminates. And the differences of stiffness degradation were characterized for the three laminates. Thus this research will expect to provide useful information to researchers for improving in-service properties of fiber reinforced polymer composites.

Chapter 5 elaborates the second major set of results. The effects of delamination location on the flexural behaviors of CFRP laminates with symmetric layup sequence of $[+45/-45/0]_{2s}$ extensively used by wind blade manufacturers were investigated. The artificial pre-delamination was created at the first $(+45/-45)$, second $(-45/0)$, third $(0/+45)$, ninth $(+45/0)$, tenth $(0/-45)$ and eleventh $(-45/+45)$ interface of the laminates, respectively. Then the six kinds of specimens and the specimens without artificial pre-delamination were subjected to three-point bending until failure and subsequently reloading. The flexural strength and modulus of the seven kinds of specimens during flexural tests were compared. Moreover, the damage including delamination, fiber failure and transverse crack in the seven kinds of specimens after failure was analysed qualitatively and quantitatively with CT technology, which have never been done by other researchers in the author's knowledge. Besides, numerical modelling with virtual crack closure technique (VCCT) was conducted to analyse the effects of embedded pre-delamination on the stress distribution in the laminates. In addition, the fracture surfaces at the interfaces inside the specimen were investigated with Scanning Electron Microscope (SEM) to get a further understanding of the failure mechanisms. Thus, this research will expect to provide useful information to researchers for predicting the mechanical

properties and failure mechanisms of FRP composites with existing delamination.

Chapter 6 elaborates the third major set of results. Static three-point bending tests were performed to determine the material elastic characteristics and strength of CFRP laminates with symmetric layup sequence of $[+45/-45/0]_{2s}$ extensively used by wind blade manufacturers. Then displacement-controlled flexural fatigue tests were performed on CFRP specimens at different displacement levels. After that, Weibull distribution function associated with a Sigmoidal model was employed for the fatigue life prediction and the stiffness degradation trend during cycling was also investigated. Lastly, the damage progression in the CFRP specimens subjected to fatigue bending loads was analysed qualitatively and quantificationally by means of CT technology. Thus, this systematic research focused on investigating the fatigue life prediction, stiffness degradation and damage progression of CFRP laminates subjected to flexural fatigue loads will expect to provide useful information to researchers for predicting the fatigue life and residual properties of FRP composites.

Chapter 7 elaborates the fourth major set of results. Double cantilever beam (DCB) tests were conducted to investigate the mode I interlaminar fracture behaviors for 0//0, 0//45 and 45//-45 interfaces of CFRP composite laminates. Then the crack front shapes of the three kinds of laminates were observed by computed tomography (CT) technique for more accurate calculation of the toughness. After that, the crack resistance curves (R curve) for the three laminate configurations were fitted by a model. After that, a numerical model with variable fracture toughness was employed to represent the mode I interlaminar fracture behaviors. Lastly, Mode I interlaminar fracture behaviors characterized by the load-displacement curves, edge views of the specimens and SEM images of the fracture surfaces for the three laminate configurations were compared. Thus, this systematic research focused on investigating the dependency of the fiber orientation of the two adjacent plies on the mode I interlaminar toughness of FRP composite laminates will expect to provide

useful information to researchers for optimized design of multidirectional FRP composite structures.

Chapter 8 concludes all the results in this thesis and recommended future work.

1.4 Originality

The originalities of this research are:

1. Investigated the effects of pre-delamination embedded at different interfaces in symmetric CFRP laminates on the flexural behaviors. Specifically, the mechanical properties of symmetric CFRP laminates with pre-delamination embedded at different interface were compared. And the damage including delamination, fiber failure and transverse crack in the specimens after failure was analysed qualitatively and quantitatively with CT technology. The numerical modelling with VCCT was conducted to analyse the effects of embedded pre-delamination on the stress distribution in the CFRP laminates.
2. Predicted the fatigue life of symmetric CFRP laminates using Weibull distribution function associated with a Sigmoidal model. The damage progression in CFRP laminates subjected to flexural fatigue loads using X-ray CT technology associated with reconstruction and visualization software was analyzed qualitatively and quantitatively.
3. Compared the mode I interlaminar fracture behaviors at 0//0, 0//45 and 45//45 interfaces of CFRP composite laminates. The crack front shapes at the three interfaces were observed by computed tomography (CT) technique for more accurate calculation of the toughness. The interlaminar fracture behaviors at the three different interfaces were well represented by a numerical model with variable fracture toughness.

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Chapter 2 Literature review

In this chapter, the previous research work related to the current topic was reviewed. Firstly, the constitutions and manufacturing technologies of the FRP composites are described. Then the mechanics of composites employed in this topic were introduced. After that, the main damage patterns in FRP composites were presented, followed by several main mechanical behaviors. Finally, the principal theories and typical application examples of Computed Tomography technology are indicated.

2.1 Constitutions of FRP composites

Fiber reinforced polymer composite is a material with strong continuous fibers or non-continuous fibers surrounded by polymer matrix. The polymer matrix is to spread the fibers and transmit the load to the fibers. The fiber/matrix bonding formed in the fabrication period exerts critical effects on the properties of the material.

2.1.1 Fiber

Fiber is composed of many filaments and the diameter of each filament is between 5 and 15 μm [1]. As Figure 2.1 shows, the fibers are normally in short or long fibers. Short fibers with length from fractions of millimetres to dozens millimetres are usually in the forms of felts and mats. And long fibers that cut during fabrication period are normally employed as is or woven.

Figure 2. 1 Different fiber forms [1]

Main fiber materials are glass, aramid, Kevlar, carbon, boron, silicon carbide and so on. In fabricating composite materials, the fibers can take the unidirectional, bidimensional and tridimensional forms. Specifically, unidimensional form includes unidirectional tows, yarns and tapes. Woven or nonwoven fabrics (felts or mats) belong to bidimensional form. And fabrics with variously oriented fibers belong to tridimensional form.

Before the formation of the reinforcements, a surface treatment is conducted on the fibers to reduce the abrasion of fibers when passing through the forming equipments and enhancing the bonding strength with the polymer matrix.

2.1.1 Polymer matrix

The resistance of the polymer matrix composites (PMC) to the degradative activities, such as impact, water absorption, chemical attack, is determined by the matrix properties. That means, the matrix is usually the weak component in the PMC material.

The polymer matrix in the PMCs can be mainly divided into thermoset and thermoplastic. The characteristics of the two types of polymer matrix are show in Table 2.1, which are compared by Darrel R. Tenney from NASA Langley Research Center.

Table 2. 1 General characteristics of different polymer matrix

Resin type	Process temperature	Process time	Use temperature	Solvent resistance	Toughness
Thermoset	Low	High	High	High	Low
Toughened thermoset
LT* thermoplastic					
Thermoplastic	high	Low	Low	Low	High

*lightly crosslinked

Thermosets include polyesters, vinylesters, epoxies, bismaleimides, and polyamides. Epoxy is the most extensively used thermoset for advanced composites. When thermosets are heated for chemical reactions (curing), the polymer chains are crosslinked so that the whole matrix is connected as a three-dimensional network, which can be shown on the left side of Figure 2.2. Cross-

linking is irreversible, so thermosets could not be reprocessed (re-melt). If the cured thermoset polymer is heated to high temperature, the chains break up and the polymer degrades. Due to the three-dimensional network structure, the thermoset presents high dimensional stability and can resist high temperature.

Figure 2. 2 Cross-linking chain of thermoset [2]

Thermoplastics, sometimes called engineering plastics, include some polyesters, polyetherimide, polyamide imide, polyphenylene sulphide, polyether-etherketone (PEEK) and so on, which are composed of long, discrete molecules. Normally the thermoplastics melt to a viscous liquid at around 260 °C to 3710 °C, and, after forming, are cooled to an amorphous, semi-crystalline, or crystalline solid. Figure 2.3 shows the processing of thermoplastics. Different with the thermosets, the processing of thermoplastics is reversible. Most of thermoplastics perform better in high-temperature strength and chemical stability than thermosets, but have better resistance to impact. However, some high-performance thermoplastics recently developed, such as PEEK, present remarkable strength at high temperature and resistance to solvents. Thermoplastics have attracted much attention from industries, because the thermoplastics can be shaped as desired just by reheating them to the process temperature. Currently, thermoplastics are employed mainly with discontinuous fibers. However, high-performance thermoplastics reinforced with continuous fibers are getting more and more attention from the industries.

Figure 2. 3 Linear chain of thermoplastic [2]

2.2 Mechanics of composite structures

2.2.1 Classic lamination theory

Composites are frequently made of plies bonded together to form a laminate. An x, y, z orthogonal coordinate system is employed to analyze laminates, which is shown in Figure 2.4. The orientations of continuous, unidirectional plies are specified by the angle θ (in degree) with respect to x -axis. The angle θ is positive in the counterclockwise direction.

Figure 2. 4 The x, y, z global coordinate and the x_1, x_2, x_3 local coordinate [3]

Thin laminates are characterized by three stiffness matrices denoted by $[A]$, $[B]$ and $[D]$. In this section these matrices for thin, flat laminates undergoing small deformations. The analyses are based on the laminate plate theory and are formulated using the approximations that the strains are linearly across the laminate, (out of plane) shear deformations are negligible, and the out-of-plane normal stress σ_z and the shear stresses τ_{xz} , τ_{yz} are small compared with the in-plane σ_x , σ_y and τ_{xy} stresses. The x, y, z refer to a coordinate system with the x and y axes in a suitably chosen reference plane, and z is perpendicular to this

reference plane. Frequently, though not always, for convenience the reference plane is taken to be the midplane of the laminate. The strains in the reference plane are represented by Equation 2.1.

$$\varepsilon_x^0 = \frac{\partial u^0}{\partial x}; \quad \varepsilon_y^0 = \frac{\partial v^0}{\partial y}; \quad \gamma_{xy}^0 = \frac{\partial u^0}{\partial y} + \frac{\partial v^0}{\partial x} \quad 2.1$$

Where u and v are the x , y components of the displacement and the superscript 0 refers to the reference plane.

The Kirchhoff hypothesis, that normal to the reference surface remain normal and straight, is adopted. In Figure 2.5, for small deflections the angle of rotation of the normal of the reference plane χ_{xz} is represented by Equation 2.2.

$$\chi_{xz} = \frac{\partial w^0}{\partial x} \quad 2.2$$

Where w^0 is the out-of-plane displacement of the reference plane. The total displacement in the x direction is represented by Equation 2.3.

$$u = u^0 - z\chi_{xz} = u^0 - z \frac{\partial w^0}{\partial x} \quad 2.3$$

The total displacement in the y direction is represented by Equation 2.4.

$$v = v^0 - z \frac{\partial w^0}{\partial y} \quad 2.4$$

Figure 2. 5 Deformation of a plate in the x - z plane [3]

By definition, the strains are represented by Equation 2.5.

$$\varepsilon_x = \frac{\partial u}{\partial x}; \quad \varepsilon_y = \frac{\partial v}{\partial y}; \quad \gamma_{xy} = \frac{\partial u}{\partial y} + \frac{\partial v}{\partial x} \quad 2.5$$

By substituting Equations 2.3 and 2.4 into these expressions, Equation 2.6 can be obtained.

$$\begin{aligned} \varepsilon_x &= \frac{\partial u^0}{\partial x} - z \frac{\partial^2 w^0}{\partial x^2} \\ \varepsilon_y &= \frac{\partial v^0}{\partial y} - z \frac{\partial^2 w^0}{\partial y^2} \\ \gamma_{xy} &= \frac{\partial u^0}{\partial y} + \frac{\partial v^0}{\partial x} - z \frac{2\partial^2 w^0}{\partial x \partial y} \end{aligned} \quad 2.6$$

These equations can be written in the Equation 2.7.

$$\begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{Bmatrix} = \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + z \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix} \quad 2.7$$

Where $\varepsilon_x^0, \varepsilon_y^0, \gamma_{xy}^0$ are the strains in the reference plane, and k_x, k_y and k_{xy} are the curvatures of the reference plane of the plate defined as Equation 2.8.

$$k_x = -\frac{\partial^2 w^0}{\partial x^2}; \quad \varepsilon_y = -z \frac{\partial^2 w^0}{\partial y^2}; \quad \gamma_{xy} = -\frac{2\partial^2 w^0}{\partial x \partial y} \quad 2.8$$

in-plane forces and moments acting on a small element (Figure 2.6) are represented by Equation 2.9.

$$\begin{aligned} N_x &= \int_{-h_b}^{h_t} \sigma_x dz; \quad N_y = \int_{-h_b}^{h_t} \sigma_y dz; \quad N_{xy} = \int_{-h_b}^{h_t} \tau_{xy} dz \\ M_x &= \int_{-h_b}^{h_t} z \sigma_x dz; \quad M_y = \int_{-h_b}^{h_t} z \sigma_y dz; \quad M_{xy} = \int_{-h_b}^{h_t} z \tau_{xy} dz \end{aligned} \quad 2.9$$

Where N and M are the in-plane forces and moments (per unit length), and h_t and h_b are the distances from the reference plane to the plate's surfaces (Figure 2.6 and 2.7). The transverse shear forces (per unit length) are represented by Equation 2.10.

$$V_x = \int_{-h_b}^{h_t} \tau_{xz} dz; \quad V_y = \int_{-h_b}^{h_t} \tau_{yz} dz \quad 2.10$$

Figure 2. 6 The forces and moments [3]

Figure 2. 7 Distances from the reference plane [3]

For plane-stress condition the stress-strain relationships for each ply are represented by Equation 2.11.

$$\begin{aligned} \begin{Bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{Bmatrix} &= [\bar{Q}] \begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{Bmatrix} = \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{Bmatrix}; \\ \begin{Bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{Bmatrix} &= [Q] \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix} = \begin{bmatrix} Q_{11} & Q_{12} & Q_{16} \\ Q_{12} & Q_{22} & Q_{26} \\ Q_{16} & Q_{26} & Q_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{Bmatrix} \end{aligned} \quad 2.11$$

Where $[\bar{Q}]$ is the stiffness matrix of the ply in the x-y coordinate system. The elements of this matrix are obtained from the elements of the stiffness matrix $[Q]$ in the x_1, x_2 coordinate system by the transformation Equation 2.12.

$$\begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix} = [T_\sigma]^{-1} \begin{bmatrix} Q_{11} & Q_{12} & Q_{16} \\ Q_{12} & Q_{22} & Q_{26} \\ Q_{16} & Q_{26} & Q_{66} \end{bmatrix} = [T_\varepsilon] \quad 2.12$$

Where $[T_\sigma]$ and $[T_\varepsilon]$ are given by Equation 2.13.

$$[T_\sigma] = \begin{bmatrix} c^2 & s^2 & 2cs \\ s & c^2 & -2cs \\ -cs & cs & c^2 - s^2 \end{bmatrix}; [T_\varepsilon] = \begin{bmatrix} c^2 & s^2 & 2cs \\ s & c^2 & -2cs \\ -cs & cs & c^2 - s^2 \end{bmatrix} \quad 2.13$$

Where $c=\cos\theta$ and $s=\sin\theta$. For an orthotropic ply the local coordinates x_1, x_2 are in the orthotropy directions. For transversely isotropic plies these local coordinates are parallel or perpendicular to the fibers. The elements of the stiffness matrix $[Q]$ are represented by Equation 2.14.

$$[Q] = \begin{bmatrix} E_1/(1 - \nu_{12}\nu_{21}) & \nu_{12}E_2/(1 - \nu_{12}\nu_{21}) & 0 \\ \nu_{12}E_2/(1 - \nu_{12}\nu_{21}) & E_2/(1 - \nu_{12}\nu_{21}) & 0 \\ 0 & 0 & G_{12} \end{bmatrix} \quad 2.14$$

By substituting Equation 2.7 and 2.11 into Equation 2.9, the Equation 2.15 and 2.16 can be obtained.

$$\begin{aligned} \begin{Bmatrix} N_x \\ N_y \\ N_{xy} \end{Bmatrix} &= \int_{-h_b}^{h_t} \left\{ [\bar{Q}] \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + [\bar{Q}]_z \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix} \right\} dz = \int_{-h_b}^{h_t} [\bar{Q}] dz \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + \\ &\int_{-h_b}^{h_t} z [\bar{Q}] dz \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix} \end{aligned} \quad 2.15$$

$$\begin{aligned} \begin{Bmatrix} M_x \\ M_y \\ M_{xy} \end{Bmatrix} &= \int_{-h_b}^{h_t} z \left\{ [\bar{Q}] \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + [\bar{Q}]_z \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix} \right\} dz = \int_{-h_b}^{h_t} z [\bar{Q}] dz \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \end{Bmatrix} + \\ &\int_{-h_b}^{h_t} z^2 [\bar{Q}] dz \begin{Bmatrix} k_x \\ k_y \\ k_{xy} \end{Bmatrix} \end{aligned} \quad 2.16$$

The stiffness matrices of the laminate are defined as Equation 2.17

$$[A] = \int_{-h_b}^{h_t} [\bar{Q}] dz; [B] = \int_{-h_b}^{h_t} z [\bar{Q}] dz; [D] = \int_{-h_b}^{h_t} z^2 [\bar{Q}] dz \quad 2.17$$

The elements of these matrices are (i, j=1, 2, 6) represented by Equation 2.18.

$$A_{ij} = \int_{-h_b}^{h_t} \bar{Q}_{ij} dz; B_{ij} = \int_{-h_b}^{h_t} z \bar{Q}_{ij} dz; D_{ij} = \int_{-h_b}^{h_t} z^2 \bar{Q}_{ij} dz \quad 2.18$$

The [A], [B] and [D] matrices are stiffness matrices of laminate, and $[\bar{Q}]$ is the stiffness matrix of the ply. Since $[\bar{Q}]$ is constant across each ply, the integrals in the equations above may be replaced by Equation 2.19 (i, j=1, 2, 6).

$$\begin{aligned} A_{ij} &= \sum_{k=1}^K (\bar{Q}_{ij})_k \cdot (z_k - z_{k-1}) \\ B_{ij} &= \frac{1}{2} \sum_{k=1}^K (\bar{Q}_{ij})_k \cdot (z_k^2 - z_{k-1}^2) \\ D_{ij} &= \frac{1}{3} \sum_{k=1}^K (\bar{Q}_{ij})_k \cdot (z_k^3 - z_{k-1}^3) \end{aligned} \quad 2.19$$

Where K is the number of plies in laminate. z_k and z_{k-1} are the distances from the reference plane to the two surfaces of the kth ply. And $(\bar{Q}_{ij})_k$ are the elements of the stiffness matrix of the kth ply.

With the preceding definitions of the stiffness matrices, the expressions for the in-plane forces and moments become Equation 2.20.

$$\begin{Bmatrix} N_x \\ N_y \\ N_{xy} \\ M_x \\ M_y \\ M_{xy} \end{Bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} & B_{11} & B_{11} & B_{11} \\ A_{12} & A_{22} & A_{26} & B_{11} & B_{11} & B_{11} \\ A_{16} & A_{26} & A_{66} & B_{11} & B_{11} & B_{11} \\ B_{11} & B_{12} & B_{16} & D_{11} & D_{12} & D_{16} \\ B_{12} & B_{22} & B_{26} & D_{12} & D_{22} & D_{26} \\ B_{16} & B_{26} & B_{66} & D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \\ k_x \\ k_y \\ k_{xy} \end{Bmatrix} \quad 2.20$$

By inverting Equation 2.20, Equation 2.21 can be obtained.

$$\begin{Bmatrix} \varepsilon_x^0 \\ \varepsilon_y^0 \\ \gamma_{xy}^0 \\ k_x \\ k_y \\ k_{xy} \end{Bmatrix} = \begin{bmatrix} \alpha_{11} & \alpha_{12} & \alpha_{16} & \beta_{11} & \beta_{11} & \beta_{11} \\ \alpha_{12} & \alpha_{22} & \alpha_{26} & \beta_{11} & \beta_{11} & \beta_{11} \\ \alpha_{16} & \alpha_{26} & \alpha_{66} & \beta_{11} & \beta_{11} & \beta_{11} \\ \beta_{11} & \beta_{12} & \beta_{16} & \delta_{11} & \delta_{12} & \delta_{16} \\ \beta_{12} & \beta_{22} & \beta_{26} & \delta_{12} & \delta_{22} & \delta_{26} \\ \beta_{16} & \beta_{26} & \beta_{66} & \delta_{16} & \delta_{26} & \delta_{66} \end{bmatrix} \begin{Bmatrix} N_x \\ N_y \\ N_{xy} \\ M_x \\ M_y \\ M_{xy} \end{Bmatrix} \quad 2.21$$

2.2.2 Determination of out-of-plane stresses

Though the classic lamination theory is normally based on the assumption that the out-of-plane normal stress σ_z and the shear stresses τ_{xz} , τ_{yz} are small compared with the in-plane σ_x , σ_y and τ_{xy} stresses (Figure 2.8). Sometimes the

out-of-plane stresses should be calculated for a plate under lateral loading. According to [4, 5], the equilibrium of forces on the plate gives Equation 2.22.

$$\frac{\partial V_x}{\partial x} + \frac{\partial V_y}{\partial y} + p = 0 \quad 2.22$$

Equilibrium of moments on the plate gives Equation 2.23.

$$\frac{\partial M_{xy}}{\partial x} - \frac{\partial M_y}{\partial y} + V_y = 0; \quad \frac{\partial M_{yx}}{\partial y} + \frac{\partial M_x}{\partial x} - V_x = 0 \quad 2.23$$

When only forces in the x-direction are considered without gradients the y-direction ($Q_y=0$ and $\partial/\partial y=0$ for all response variables) this reduces to Equation 2.24.

$$\frac{\partial V_x}{\partial x} = -p; \quad \frac{\partial M_x}{\partial x} = V_x \quad 2.24$$

(a) vertical shearing forces (b) bending and twisting moments

Figure 2. 8 Plate under lateral loading

The forces and moments acting on the plate cause internal stresses. Consider an element with dimensions dx, dy, dz . According to [3], the equilibrium of forces gives Equation 2.25.

$$\begin{aligned} \frac{\partial \sigma_x}{\partial x} + \frac{\partial \tau_{yx}}{\partial y} + \frac{\partial \tau_{zx}}{\partial z} + f_x &= 0 \\ \frac{\partial \tau_{xy}}{\partial x} + \frac{\partial \sigma_y}{\partial y} + \frac{\partial \tau_{zy}}{\partial z} + f_y &= 0 \\ \frac{\partial \tau_{xz}}{\partial x} + \frac{\partial \tau_{yz}}{\partial y} + \frac{\partial \sigma_z}{\partial z} + f_z &= 0 \end{aligned} \quad 2.25$$

Where f_x, f_y and f_z are the body forces per unit volume in x, y and z directions, respectively. From equilibrium equations of moments neglecting the

higher order terms, the equations $\tau_{xy}=\tau_{yx}$, $\tau_{yz}=\tau_{zy}$ and $\tau_{zx}=\tau_{xz}$ can be obtained. When the stress gradients in the y direction is neglected ($\partial/\partial y=0$ for all response variables) and all the body forces neglected, Equation 2.26 can be obtained.

$$\begin{aligned}\frac{\partial \sigma_x}{\partial x} + \frac{\partial \tau_{xz}}{\partial z} &= 0 \\ \frac{\partial \tau_{xy}}{\partial x} + \frac{\partial \tau_{yz}}{\partial z} &= 0 \\ \frac{\partial \tau_{xz}}{\partial x} + \frac{\partial \sigma_z}{\partial z} &= 0\end{aligned}\tag{2.26}$$

As just moment M_x is acting on the plate, Equation 2.27 can be obtained by combining Equations 2.7 and 2.21.

$$\begin{Bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{yx} \end{Bmatrix} = \begin{Bmatrix} \beta_{11} + \delta_{11} \cdot z \\ \beta_{21} + \delta_{21} \cdot z \\ \beta_{61} + \delta_{61} \cdot z \end{Bmatrix} M_x\tag{2.27}$$

By combining Equations 2.11, 2.24 and 2.26, Equation 2.28 can be obtained.

$$\begin{aligned}\left(\frac{\partial \tau_{xz}}{\partial z} \right)_{(k)} &= - \left(\bar{Q}_{11(k)} \cdot (\beta_{11} + \delta_{11} \cdot z) + \bar{Q}_{12(k)} \cdot (\beta_{21} + \delta_{21} \cdot z) + \bar{Q}_{16(k)} \cdot (\beta_{61} + \delta_{61} \cdot z) \right) \cdot V_x \\ \left(\frac{\partial \tau_{yz}}{\partial z} \right)_{(k)} &= - \left(\bar{Q}_{61(k)} \cdot (\beta_{11} + \delta_{11} \cdot z) + \bar{Q}_{62(k)} \cdot (\beta_{21} + \delta_{21} \cdot z) + \bar{Q}_{66(k)} \cdot (\beta_{61} + \delta_{61} \cdot z) \right) \cdot V_x \\ \left(\frac{\partial \sigma_z}{\partial z} \right)_{(k)} &= - \left(\frac{\partial \tau_{xz}}{\partial x} \right)_{(k)}\end{aligned}\tag{2.28}$$

With the boundary condition of zero shear stresses at the outer surface, Equation 2.29 can be deduced.

$$\begin{aligned}\tau_{xz}^{(k)} &= \sum_{j=1}^{k-1} \int_{z_{j-1}}^{z_j} \left(\frac{\partial \tau_{xz}}{\partial z} \right)_{(j)} \cdot dz + \int_{z_{k-1}}^z \left(\frac{\partial \tau_{xz}}{\partial z} \right)_{(k)} \cdot dz \\ \tau_{yz}^{(k)} &= \sum_{j=1}^{k-1} \int_{z_{j-1}}^{z_j} \left(\frac{\partial \tau_{yz}}{\partial z} \right)_{(j)} \cdot dz + \int_{z_{k-1}}^z \left(\frac{\partial \tau_{yz}}{\partial z} \right)_{(k)} \cdot dz\end{aligned}$$

$$\sigma_z^{(k)} = - \sum_{j=1}^{k-1} \int_{z_{j-1}}^{z_j} \left(\frac{\partial \tau_{xz}}{\partial x} \right)_{(j)} \cdot dz - \int_{z_{k-1}}^z \left(\frac{\partial \tau_{xz}}{\partial x} \right)_{(k)} \cdot dz \quad 2.29$$

Integration of these equations results in Equation 2.30.

$$\begin{aligned} \tau_{xz}^{(k)} &= - \sum_{j=1}^{k-1} \left\{ \left(\bar{Q}_{11(j)} \cdot b_{11} + \bar{Q}_{12(j)} \cdot b_{21} + \bar{Q}_{16(j)} \cdot b_{61} \right) \cdot (z_j - z_{j-1}) + \frac{1}{2} \left(\bar{Q}_{11(j)} \cdot d_{11} + \bar{Q}_{12(j)} \cdot d_{21} + \bar{Q}_{16(j)} \cdot d_{61} \right) \cdot (z_j^2 - z_{j-1}^2) \right\} \cdot V_x - \left\{ \left(\bar{Q}_{11(k)} \cdot b_{11} + \bar{Q}_{12(k)} \cdot b_{21} + \bar{Q}_{16(k)} \cdot b_{61} \right) \cdot (z - z_{k-1}) + \frac{1}{2} \left(\bar{Q}_{11(k)} \cdot d_{11} + \bar{Q}_{12(k)} \cdot d_{21} + \bar{Q}_{16(k)} \cdot d_{61} \right) \cdot (z^2 - z_{k-1}^2) \right\} \cdot V_x; \\ \tau_{yz}^{(k)} &= - \sum_{j=1}^{k-1} \left\{ \left(\bar{Q}_{61(j)} \cdot b_{11} + \bar{Q}_{62(j)} \cdot b_{21} + \bar{Q}_{66(j)} \cdot b_{61} \right) \cdot (z_j - z_{j-1}) + \frac{1}{2} \left(\bar{Q}_{61(j)} \cdot d_{11} + \bar{Q}_{62(j)} \cdot d_{21} + \bar{Q}_{66(j)} \cdot d_{61} \right) \cdot (z_j^2 - z_{j-1}^2) \right\} \cdot V_x - \left\{ \left(\bar{Q}_{61(k)} \cdot b_{11} + \bar{Q}_{62(k)} \cdot b_{21} + \bar{Q}_{66(k)} \cdot b_{61} \right) \cdot (z - z_{k-1}) + \frac{1}{2} \left(\bar{Q}_{61(k)} \cdot d_{11} + \bar{Q}_{62(k)} \cdot d_{21} + \bar{Q}_{66(k)} \cdot d_{61} \right) \cdot (z^2 - z_{k-1}^2) \right\} \cdot V_x; \\ \sigma_z^{(k)} &= \sigma_z^{(k-1)}(z = z_{k-1}) - \sum_{j=1}^{k-1} \left\{ \left(\bar{Q}_{11(j)} \cdot b_{11} + \bar{Q}_{12(j)} \cdot b_{21} + \bar{Q}_{16(j)} \cdot b_{61} \right) \cdot (z_j - z_{j-1}) + \frac{1}{2} \left(\bar{Q}_{11(j)} \cdot d_{11} + \bar{Q}_{12(j)} \cdot d_{21} + \bar{Q}_{16(j)} \cdot d_{61} \right) \cdot (z_j^2 - z_{j-1}^2) \right\} \cdot (z - z_{k-1}) \cdot p - \left\{ \frac{1}{2} \cdot \left(\bar{Q}_{11(k)} \cdot b_{11} + \bar{Q}_{12(k)} \cdot b_{21} + \bar{Q}_{16(k)} \cdot b_{61} \right) \cdot (z - z_{k-1})^2 + \frac{1}{6} \left(\bar{Q}_{11(k)} \cdot d_{11} + \bar{Q}_{12(k)} \cdot d_{21} + \bar{Q}_{16(k)} \cdot d_{61} \right) \cdot (z - z_{k-1})^2 \cdot (z + 2 \cdot z_{k-1}) \right\} \cdot p \end{aligned} \quad 2.30$$

While the local interlaminar shear stress in the kth ply (along fiber orientation) is evaluated as Equation 2.31 according to its orientation.

$$\begin{aligned} \tau_{13}^{(k)} &= \tau_{xz}^{(k)} \cos \theta_{(k)} + \tau_{yz}^{(k)} \sin \theta_{(k)} \\ \tau_{23}^{(k)} &= -\tau_{xz}^{(k)} \sin \theta_{(k)} + \tau_{yz}^{(k)} \cos \theta_{(k)} \end{aligned} \quad 2.31$$

2.3 Failure mechanisms in FRP composites

2.3.1 Fiber failure

Fiber breakage is the failure mechanism in the FRP composites in tensile stress state or compressive state. Figure 2.9 shows the fiber breakage in cross-ply open-holed carbon fiber reinforced PEEK-titanium composites subjected to tensile loads [6].

Figure 2. 9 SEM images of the cross-ply open-holed carbon fiber reinforced PEEK-titanium samples after 65% ultimate load [6]

Fiber microbuckling may occur when the FRP composites subjected to compressive or bending loads fails. Figure 2.10 shows fiber microbuckling of CFRP composites subjected to compressive loads [7]. It is obvious that the band between the two positions of fiber breakage is inclined at $\beta \sim 15^\circ$ to the fiber longitudinal direction. The microbuckling, shown as broken fiber bands, keeps its inclination angle. Berg and Salama [8] indicated that the band inclined to enable the buckled fibers to bear compression and shear deformation.

Figure 2. 10 SEM image showing fiber microbuckling in UD CFRP composites[7]

Fiber crushing is another phenomenon when FRP composites fail under compressive loading. Failure occurs at the fiber level of scale due to a shear instability such as buckling within the fiber. Figure 2.11 shows the fiber crushing in FRP composites [9, 10]. It is often associated with the fact that the fibers themselves are microcomposites comprising wavy fibers embedded in a soft matrix [9]. When the matrix is sufficiently stiff and strong, alternative

failure modes such as fiber crushing intervene. Fiber crushes once the uniaxial strain in the material equals the intrinsic crushing strain of the fibers. In the case of fiber crushing, glass fibers normally fail in compression in the form of splitting. For carbon, Kevlar and wood fibers, microbuckling forms within each fiber, and kinking bands are observed in the fibers of width less than the radius [9, 11-14].

(a) schematic [9] (b) braid CFRP rod [10] (c) unidirectional rod [10]

Figure 2. 11 Fiber crushing

When a fiber breaks, equilibrium requires that the net load on the cross section containing the broken fiber be unchanged. Therefore, the average stress in the remaining fibers must increase. The increase in the average stress over a fiber cross-section next to the broken one raises the probability that one or more of them will break. So the propagation of fiber breaks can also be regarded as a mechanism of failure. The probability of occurrence of this mode of failure increases with the average fiber stress because of the increasing number of scattered fiber breaks and the increasing stress level in overstressed fibers.

2.3.2 Matrix failure

As matrix in FRP composites, polymer exhibits much weaker tensile and compressive strength comparing with unidirectional fibers. There are three typical matrix-dominated failure modes in FRP composites, which are transverse tension (tension normal to the fibers), shear and peel [15]. This section will describe the features of matrix cracks in CFRP composites subjected to static loads using SEM and apply accurately descriptive terminology.

A typical transverse tension failure in a composite is shown in Figure 2.12. It is obvious that rivers cross the fibers and converges in the direction of overall crack propagation [16-20]. As Figure 2.12 shows, the rivers flow to both sides of the fiber because the fibers act as local stress concentrations. It should be noted that voids with dimensions smaller than 10 μm exert no significant effect on the fracture [20].

(a) Rivers in direction of crack propagation (arrowed) ($\times 500$) (b) Rivers angled to fibers (F) ($\times 1000$).

Figure 2. 12 Rivers in the transverse tension failure (Crack propagate from left to right) [15]

(a) Cusps at an angle to ± 45 fibers ($\times 1000$) (b) Shear surface

Figure 2. 13 Typical shear failure [15]

On the macroscopic perspective, transverse tension failures are often planar with a dark, semi-mat appearance. As the fracture propagates, the failure mode

transforms from transverse tension to peel step by step. Peel is a combination of transverse tension and shear, which will be described in the next paragraph.

One of the significant characteristics of shear failure is cusps, which are not applicable for transverse tension failure. Interlaminar shear failures usually occur at the interface with different fiber directions, one of which is in, or close to, the direction of shear stress [17]. Where the shear stress direction occurs at an angle to the fibers, the cusps still tend to be aligned perpendicular to the shear stress, but preferentially follow the fibers a-a and b-b in Figure 2.13(a). From Figure 2.13(b), the macroscopic morphology of shear failure is dull and albescent.

The two mating surfaces of a typical peel are shown in Figure 2.14. It should be noted that there are ribs such as at A, B, C, D and those more intensively occurred along X-X, where the ridges in Figure 2.14(a) corresponds to the valleys in Figure 2.14(b). The peels grow in the fiber direction with the ribs usually perpendicular to the fiber directions, but, when the alignment is different, the ribs are normal to the crack front.

Figure 2. 14 Mating surfaces of a peel ($\times 150$) [15]

2.3.3 Fiber/matrix debonding

It is assumed that various failure modes initiate from the micro cracks. The FRP composites are assumed to present excellent fiber/matrix bond performance [21]. When cracks propagate, the transverse tensile stress and interfacial shear stress can lead to fiber/matrix debonding, which is separation of fibers from matrix. The fiber/matrix interfacial strength can be affected by the factors such

as moisture, temperature, or humidity. If the fiber/matrix interface is strong enough, the presence of fiber/matrix debonding is rare before the fiber failures occur and propagate, which is usually regarded as brittle failure. If the interfacial strength is low, the fiber/matrix debonding will extensively occurs before fiber failures. The SEM micrographs in Figure 2.15 displays the clean fibers and extensive fiber/matrix debonding on the fracture surfaces of glass fiber/vinyl ester composite specimen subjected to short-beam shear test [22].

Figure 2. 15 Extensive fiber/matrix debonding in glass fiber/vinyl ester composites [22]

2.3.4 Delamination

Pagano and Schoeppner [23] divided the causes leading to delamination into two groups in terms of the shape of composite components or structures. The first group includes the curved component such as pressure vessels, fan-shaped, tubular, cylindrical, and spherical components. The other group is variable-thickness components including joint region, free edge, bolt junction and so on. In these two groups of practical components, delamination is prone to occur between the adjacent plies caused by the normal and shear stress.

Moreover, hydrothermal effects, layup fabrication and in-service conditions are also the reasons for the delamination. Because the coefficient of thermal expansion as well as moisture rate of the fiber and polymer is different, so different plies shrink at different extent during the curing procedure and expand at different extent after moisture absorption, which is one of the critical cause for residual stress [24, 25]. Besides, rich resin regions are prone to appear

during the curing procedure, which will cause the curing time of the adjacent plies different. This will decrease the properties at the interface, which leads to occurrence of delamination [26, 27]. In addition, transverse load due to low-velocity impact is also one of critical reasons for delamination.

V. V. Bolotin [26, 27] divided delamination into internal delamination and near-surface delamination. Internal delamination is caused by the micro cracks at the ply interface in the composite components, which will reduce the load capacity of the components. Especially when the composite component is subjected to compressive loads, the flexural behaviors of the component will be seriously impacted, which can be shown in Figure 2.16. Though the delamination splits the composite component into two parts, global buckling occurs because the two parts exhibit the similar flexural behaviors.

(a) disposition across the laminate and (b) effect on the overall stability

Figure 2. 16 Internal delamination [26]

Near-surface delamination is much more complicated than internal delamination. The deformation of the delamination region is affected more by the thin laminate part than the thick laminate part. So for the near-surface delamination in the practical composite components, not only the delamination propagation should be avoided, but the local stability of the composite laminates also need to be considered. The near-surface delamination can be divided into six types as shown in Figure 2.17.

Figure 2. 17 Types of near-surface delamination [26]

When the internal delamination or near-surface delamination occurs in the composite components, they may propagate if the composite components are subjected to static or cyclic loads. This will reduce the strength and stability of the components and even cause catastrophic failure. Thus, a deep insight into the delamination initiation and propagation is important.

On micro level, there is damaged region in the front of the delamination. Based on the toughness of the resin and stress state, there are totally four modes of delamination propagation including crack opening mode (mode I), shear modes (mode II and mode III) and mixed-modes, which are shown in Figure 2.18. For mode I delamination propagation, the micro cracks near the crack front tip will merge and grow and fiber/matrix debonding will occur, which will lead to the delamination propagation. And fiber bridging and fiber failure will occur associated with fiber/matrix debonding. While for mode II and mode III, the micro cracks near the crack front tip will merge and propagate along the direction at 45° angle relatively to the ply orientation, until the rich-resin region. The micro cracks merge at the interface and exhibit zigzag appearance, which is shown in Figure 2.19.

Figure 2. 18 Mode I, mode II and mode III crack propagation modes [28]

(a) microcrack formation (b) microcrack growth (c) microcrack connection

Figure 2. 19 Initiation and propagation of a mode II delamination at the ply interface [29]

2.4 Mechanical behaviors of FRP composites

Fiber-reinforced polymer (FRP) composites have been widely employed in aerospace structures, automobile components and wind turbines to reduce weight with high strength and stiffness. The FRP composite structures in service are subjected to different static or cyclic loads, so a deep investigation into the mechanical behaviors of FRP composites is vital. This will expect to provide useful information to researchers for understanding the failure mechanisms and predicting the mechanical properties such as residual strength, residual stiffness and service life of FRP composites.

2.4.1 Effects of pre-delamination on the properties of FRP composites

Theoretically, the defects in the FRP composite structures developed in engineering practices should be minimized to achieve perfect performance. However, many common activities, such as manufacturing errors, operational,

or maintenance activities, will lead to damage, which could decrease tremendously the strength and stiffness of the FRP composites. Under these conditions the seven common damage patterns like delamination, transverse crack, fiber failure and fiber/matrix debonding may occur, which may result in the degradation in stiffness, strength and fatigue life or even structural failure [30, 31].

Wind turbines are expected to sustain its service for around 20-30 years as required because the installation and replacement is time-consuming and inconvenient. However, even there is slight damage in the blades due to manufacturing errors or in-service operations, some wind turbine should continue to operate to save the cost of installation or replacement.

Delamination is a typical failure mechanism of FRP composites, lots of researchers investigated the effects of delamination on the mechanical behaviors of FRP composites [31-35]. Z. Aslan and M. Şahin [31] studied the effects of the multiple delamination caused by low velocity impact on the properties of E-glass/epoxy composite laminates subjected to compression loads, but not affected by the longest and near-surface delamination size. They also found that the size of beneath delamination has no significant effect on the buckling load and compressive failure load of composite laminates. X. W. Wang et al. [32] investigated the effects of single and multiple embedded delamination on the failure mechanisms of GFRP specimens subjected compression loads experimentally and analytically using finite element method. They found that the strength of laminates with multiple delaminations reduces at the most extent when the delaminations split the laminate as similar thickness, which almost coincide with the finite element analysis. F. Cappello and D. Tumino [33] also studied the effects of embedded multiple delamination on the buckling behaviors of unidirectional and cross-ply FRP specimens subjected to uniaxial compressive loads with finite element method. They indicated that the delamination length and distribution as well as layup sequence of the laminates affects the critical load of the FRP composite laminate. G. J. Short et al. [35]

investigated the effects of delamination size and distribution on the compressive behaviors of multidirectional GFRP specimens and found that the failure load is positively related with the size and through-thickness position of the delamination.

Obviously most of the researchers focused on the influences of delamination on the buckling and compression behaviors of FRP laminates. Seldom researchers investigated the effects of delamination on the flexural behaviors of FRP composites. Wenran Gong [36] investigated the four-point bending behaviors of delaminated crossply composite panels with damage created from low velocity impact test or artificial embedded delamination. Maozhou Meng [37] investigated the four-point bending fatigue behaviors of unidirectional and crossply laminates with moisture ingress and found that the buckling driven delamination near the loading roller is the mode I dominated failure at the initial stage while the fatigue crack propagation is mode II dominated. Moustafa Kinawy [38] employed an analytical model to investigate the crack propagation in a laminated composite laminates subjected to pure bending. However, they all have not compared the effects of pre-delamination location on the bending behaviors of FRP composites.

2.4.2 Fatigue behaviors and damage progressions of FRP composites

Wind turbine blades in service are subjected to highly variable loads such as aerodynamic, gravitational and inertial forces. The aerodynamic and gravitational forces will bring about the existence of bending stress in the blades [39]. As wind is variable and chaotic, so the most common failure mode of the wind blades is mechanical fatigue [40, 41]. Therefore, it is essential to investigate the flexural fatigue behaviors of CFRP composites in order to predict the service life of the blades subjected to variable loads.

The prediction of fatigue life and strength has been primarily based on experimental investigations. But fatigue life/strength results of fiber reinforced polymer (FRP) composites are usually variable because of the inhomogeneous and anisotropic features of FRP composites, even under carefully controlled

testing conditions. So many researchers analysed the dispersed fatigue results using statistical methods [42-44], which show that Weibull distribution function has reasonable experimental verification and well developed statistics.

Global damage in CFRP composites subjected to fatigue loads is usually associated with stiffness degradation. Stiffness can reflect the laminate's condition and resistance capacity to fatigue loads, which can be helpful for life prediction or ranking residual strength data. While the stiffness degradation levels can change depending on composite components, loading type, loading level and layup sequences [45]. Philippidis, T. and A. Vassilopoulos [46] conducted static and fatigue tests on multidirectional glass fiber reinforced polyester composite specimens. They reported that the S-N curves corresponding to 95% reliability level are quite similar to those for 5-20% stiffness degradation. Liu, B. and L. B. Lessard [47] considered fatigue failure as the point when the load bearing capacity declined to the level of the applied stress in the fatigue cycle, which is most extensively used for force-controlled fatigue tests. S. M. Jessen et al. [48] defined 20% as the stiffness degradation level for stress-controlled tension-tension fatigue failure criterion of pultruded glass composites by comparing the elastic modulus of damaged material and undamaged material. C. J. Jones [49] adopted a failure criteria of 15% reduction in stiffness on [0/90] epoxy-based laminates during force-controlled axial fatigue tests. G. C. Shih and L. J. Ebert [50] regarded 10% load drop as failure criteria of unidirectional FRP composites under displacement-controlled fatigue tests. The standard ISO 13003 defines fatigue failure as a stiffness reduction level between 5% to 20%. N. K. Kar et al. [51] chose a 20% drop in cyclic stiffness as the failure criteria of hybrid composite rods under displacement-controlled flexural fatigue tests. As there is no constant standard of defining fatigue failure criterion regarding stiffness degradation, a greater understanding of stiffness degradation for a CFRP composite laminate subjected to different fatigue loads is worthwhile.

In the past 20 years, many studies have been reported regarding investigations on damage progression in cross-ply and multi-directional fiber reinforced polymer composites subjected to fatigue loading. Talreja R. [52] concluded that there were four major fatigue mechanisms: matrix crack, delamination, fiber/matrix debonding and fiber failure.

It has been found that different damage mechanisms are dominated at different stress levels. Matrix crack is the major mechanism when composites are subjected to low loads, and delamination would occur at medium loads, while fiber failure can be observed at high loads [52]. Wind turbine blades are in service under low load and high cycle fatigue conditions, so matrix cracking has been extensively studied by many researchers [53-56].

S. L. Ogin [57] examined a glass fiber reinforced polymer (GFRP) laminate with layup sequence of $[0/90]_s$ and found the relationship between stiffness degradation and transverse crack of matrix. A.W. Wharmby [58] found that the relationship between normalized stiffness and crack density is linear for the fiber reinforced polymer laminates subjected to cyclic loading, and he determined the life for transverse crack initiation, delamination initiation, and final failure.

Tong et al. [59] investigated the evolution of crack in $[0/90/-45/+45]_s$ glass fiber reinforced polymer laminates subjected to tension-tension fatigue load. They found that transverse cracks occurred first, then cracks in -45 ply formed from the edges of existing 90° cracks, followed by +45 matrix cracks. Masters and Reifsnider [60] investigated the fatigue crack growth of laminates with two kinds of layup $[0/90/\pm 45]_s$ and $[0/\pm 45/90]_s$. Although transverse cracks occurred at initial stage for both kinds of laminates, each kind of laminates presented different damage development patterns. This confirmed that the damage progression is dependent on laminate layup.

Matrix cracking in the initial stage does not directly lead to the final failure. However, it can be an incentive to other damage patterns. Kashtalyan M et al. [61] found that matrix cracks could cause delamination onset, because there

will be local stress concentrations when matrix cracks grow at the interfaces between plies. Delamination often occurs at the laminate edge due to the highest interlaminar stresses at this position, which is caused by the elastic property mismatch between plies with different fiber orientations. O' Brien TK [62] compared the local delamination within the specimen bulk and the edge delamination. He found that local delamination brought about stress concentrations which caused premature laminate failure, whereas the edge delamination grew in a more stable pattern.

The fatigue failure mechanisms of FRP composites are complex and generally in the forms of delamination, debonding, matrix crack, and fiber failure [52]. But it is an unresolved problem on which is the major mechanism responsible for fatigue failure of FRP composites [51]. Some reported that matrix crack were the dominant mechanism, while many others suggested fiber failure be primarily responsible [63, 64]. So it is vital to observe the crack progression patterns during fatigue tests and determine how the accumulated damage lead to properties degradation of the FRP composites.

2.4.3 Interlaminar fracture behaviors of FRP composites

Delamination is one of the most typical failure mechanisms of FRP composite materials, which will lead to the degradation in stiffness, strength and fatigue life [30]. As delamination is only kind of failure mechanism during interlaminar fracture tests, so interlaminar fracture test is one of the most extensively employed method to investigate the initiation and propagation of delamination.

Numerous researchers have investigated the interlaminar fracture toughness of FRP composite laminates especially the unidirectional laminates. P. Coronano et al. [65, 66] conducted static and fatigue mode I interlaminar fracture tests on composites made of a 3501-6 epoxy matrix with AS4 unidirectional carbon fibers at different temperatures, concluded that temperature can affects the resistance to delamination of the unidirectional CFRP composites. M. F. S. F. de Moura et al. [67] conducted double cantilever beam tests on unidirectional CFRP composites with cutting fibers during crack

propagation to investigate the influences of bridging on the measured mode I interlaminar fracture energy. They indicated that the appearance of bridging accounts for 60% of energy being dissipated during crack propagation. T. Kevin O` Brien et al. [68] conducted ENF test on unidirectional IM7/8552 graphite epoxy composites and drafted ASTM protocol, which is useful for developing mode II fatigue standards. M. M. Shokrieh et al. [69] investigated the effects of initial crack length and DCB specimen thickness on the mode I delamination curve (R-curve) behaviors of unidirectional GFRP materials and introduced a mathematical relationship to predict the mode I delamination behaviors (from the initiation to propagation) of E-glass/epoxy DCB specimens. K. Pingkarawat and A. P. Mouritz investigated the mode I interlaminar and fatigue behaviors of CFRP laminates with z-pins reinforced and found that the volume content, diameter and length of z-pins affects the interlaminar toughness, fatigue resistance. M. Heidari-Rarani et al. [70] proposed a three-linear cohesive zone model (CZM) to determine the mode I interlaminar fracture behavior of unidirectional DCB specimens and verified the agreement between simulations and experiments. P. F. Liu and M. M. Islam [71] proposed an anisotropic exponential cohesive model to predict the mixed-mode interlaminar fracture behaviors of composites and verified the good agreement with other researchers by numerical simulation of DCB, ENF, mixed-mode bending fracture (MMB) and fixed ratio mixed mode (FRMM) procedures. R. Jones et al. [72] used a modified Hartman-Schijve equation to compute the growth of mode I, II and mixed mode I/II fracture delaminations in unidirectional CFRP composites and found it is accurate, and they also discussed the potential for this equation to be used to overcome the no growth philosophy associated with current composite designs. Stefano Bennati et al. [73] proposed a model of the asymmetric double cantilever beam (ADCB) test, which can be adopted to determine the MMB toughness of composites.

However, the FRP laminates extensively employed in industrial applications are multidirectional (MD) and delamination propagation is more common

between plies with different fiber orientations in real composite structures [74]. Not much research has investigated the interlaminar toughness at the interface between plies with different fiber orientations in multidirectional composites in comparison with unidirectional composites. The lack of research on this subject is mainly due to the difficulty that crack jumping may occur when delamination propagates, which might lead to inaccurate toughness values [75, 76]. However, the interlaminar toughness values can also be determined for comparison even if crack jumping is observed in the laminates under DCB tests [77-79]. De Moraes et al. [79] regarded bending stiffness of the laminated beams as major cause leading to crack jumping for DCB tests. So the selection on the layup of the FRP laminates for DCB tests need to be considered to inhibit crack jumping. Due to the difficulty of determining the interlaminar fracture toughness at the interface between plies with different fiber orientations in multidirectional composites, most of current engineering practices regard the interlaminar toughness at any interface of FRP composite structures as the same with that at $0^\circ//0^\circ$ interface in unidirectional laminates. And most researchers use cohesive finite elements to model all the interfaces in multidirectional FRP laminates with interlaminar toughness at $0^\circ//0^\circ$ interface in unidirectional FRP laminates [80-83]. However, Zhao Libin indicated that the parameters of cohesive fracture model suitable for unidirectional laminates are not necessarily suitable for multidirectional composite laminates [84].

Double cantilever beam (DCB) test is usually employed to investigate the mode I interlaminar toughness. The most common used standard methods, ISO 15024 [85] and ASTM D5528 [86], have been established for the mode I interlaminar toughness of unidirectional (UD) FRP composites. Many researchers employed the test standards developed for unidirectional FRP composites to investigate the interlaminar toughness of multidirectional FRP composites [69, 87]. H. Chai [88] reported that the mode I interlaminar toughness is independent of the interface angles for T300/BP-907 and T300/5208 laminates. A. B. Pereira and A. B. de Moraes [77] conducted DCB

tests on the multidirectional CFRP specimens with $0^\circ/\theta$ delamination interfaces and performed finite element analysis, then concluded that the mode I interlaminar toughness G_{Ic} was not affected by the fiber orientations at the delamination interface. However, T. A. Sebaey et al. [89] investigated the mode I interlaminar toughness and crack growth in multidirectional CFRP laminates and found the toughness values depended on both the degree of fiber bridging and the fiber orientations at the delamination interface. P. Naghipour et al. [90] studied the effect ply orientation on mixed mode delamination behaviors in unidirectional and multidirectional composites numerically and experimentally. Based on the investigation he found that mixed mode interlaminar toughness of multidirectional laminates increases remarkably comparing with unidirectional ones due to appearance of toughening mechanisms such as curved crack fronts and intralaminar energy absorption. A. Laksimi et al. [91] conducted DCB tests on glass/epoxy composites with stacking sequence of $[\pm\theta_{2s}]_s$ and concluded that the initial energy release rate (not pure mode I, but rather a mixed mode I+II or even I+II+III because of the intralaminar fractures) does not depend on the ply angle θ at the delamination interface, but the energy release rate during delamination propagation depends primarily on the ply angle θ . Robinson and Song [92] compared mode I interlaminar toughness for $0^\circ//0^\circ$ and $45^\circ//{-}45^\circ$ interface of a kind of carbon fiber epoxy composite (XAS-913C), and found that the latter is higher by 24% than the former. While Ozdil and Carlsson [93] investigated the mode I interlaminar toughness of glass/polyester for $0^\circ//0^\circ$, $30^\circ//{-}30^\circ$ and $45^\circ//{-}45^\circ$ interfaces, and indicated that the initiation toughness decreased with the increased off-axis angle in the laminates. N. S. Choi et al. [94] conducted DCB, end-loaded split (ELS) and FRMM tests on $[-45^\circ/0^\circ/+45^\circ]_{2s}[+45^\circ/0^\circ/{-}45^\circ]_{2s}$ carbon-fiber/epoxy specimens with initial delamination at the $+45^\circ/{-}45^\circ$ interface to assess the mode I, mode II and mixed-mode interlaminar toughness, respectively. They just clearly defined the initial interlaminar fracture energy G_c but could not define the G_c during crack propagation due to its variation caused by crack jumping. But it was concluded

that the initial G_c is always significantly greater than unidirectional laminates. Thus it is a controversial issue on whether the interface angle affects the interlaminar fracture toughness or how it affects, though many researchers have studied the interlaminar fracture behaviors of multidirectional composites. So the consideration on the interlaminar behaviors for interfaces between plies with different orientations is important for more accurate prediction of delamination initiation and propagation in real FRP composite structures [95, 96]. And more and more research on the interlaminar toughness of multidirectional FRP laminates should be carried out for optimized design of real FRP composite structures.

In addition, the delamination simulation in multidirectional FRP laminates may be influenced by the curved delamination front [97]. Many researchers modelled the crack front shape of the FRP composites under DCB tests [76, 78, 98, 99]. But seldom researchers observed the crack front shape by experiments. In most literatures focused on interlaminar toughness of multidirectional specimens, the delamination length is determined from the specimen edges and the crack front is regarded as straight for the measurement of toughness. This assumption may not be reasonable, especially for multidirectional specimens [98-100].

2.5 X-ray CT applied for FRP composites

In the recent decade, many researchers have used CT to investigate the failure mechanisms of fiber reinforced polymer composites, which proved its strong practicability in this area [101-108]. Digby D. Symons [101] characterized the damage of carbon fiber reinforced polymer laminated plates during impact tests and quasi-static indentation tests using optical microscopy, C-scan and CT. He found that CT is the most accurate method to characterise the exact distribution of damage. But it should be also noted that C-scan and optical microscopy is a relatively flexible and convenient method. Yuri Nikishkov et al. [109] observed the voids and delamination in the CFRP specimens under curved-beam tests using X5000 X-ray CT system. And structural models were created by

converting the defect automatically to predict failure. For more microscopic investigation, Paul J. Schilling et al. [102] measured the interior geometry of microcrack in fiber reinforced polymer laminates with a SkyScan 1072 desk-top X-ray microscanner. But he found that the sample size which can be observed is limited to around 1.5 mm, a point at which the crack opening is round 20% of the reconstructed pixel width. And he facilitated investigating larger specimens by using dye penetrant (a mixture of isopropyl alcohol 5 ml, zinc iodide 30 g, Kodak photo solution 5 ml and distilled water 5 ml) to magnify the contrast, which enable the detection of microcracks in samples up to 15 mm in width, with the crack opening <5% of the pixel width. Figure 2.20 shows the volumetric reconstruction of three kinds of specimens with microcracks with the usage of dye penetrant. Yuri Nikishkov et al. [110] proposed a new technique to quantify porosity/void dimensions in composites, which employs a sub-pixel contour generation for the average of the air and material grey values obtained in CT scans. Whilst D. Raz-Ben Aroush et al. [105] used European Synchrotron Radiation Facility (ESRF in Grenoble) to observe the damage development in unidirectional bundle-reinforced 3D composites. And they proposed a simple fracture mechanic to relate the critical fiber cluster size to the composite strength. G. P. McCombe et al. [103] used Henry Moseley X-ray Imaging Facility associated with Nikon-Xtek 320kV electron beam machine to scan the quasi-isotropic CFRP composites subjected to low velocity impacts and visualize the 3D damage inside using VG studio max 2.1 and Avizo 6.0 software packages. Figure 2.21 shows the segmented damage volumes at resolution of 35 μm from a laminate under impact test. Apart from the visualization, they also calculate the damage volumes in each ply of unmodified CFRP laminates, laminates with hollow glass fibers (HGF) added at the 3rd or 13th ply interface, which are shown in Figure 2.22. Similarly, Sinan Fidan et al. [111] characterized the single and repeated low velocity impact damage behaviors of GFRP armor steel composite with cone beam computed tomography (CBCT) technique and D. J. Bull et al. [112] segmented 3D

damage of particle toughened CFRP composite laminates under low velocity impact load.

(a) the 1.8 mm×1.4 mm graphite/epoxy specimen (b) the 3.9 mm×1.4 mm glass/epoxy specimen (c) the 5.0 mm×1.4 mm graphite/epoxy sample

Figure 2. 20 Volumetric images of the specimen (grey) and the damage (black) [102]

(a) angled top view

(b) side view

Figure 2. 21 Segmented damage in a laminate impacted at 4 J [103]

(a) unmodified specimens, (b) the HGF@IF13 and (c) the HGF@IF3 specimens impacted at 3, 4, 5 J on ply 16

Figure 2. 22 Mean damage volume per ply [103]

Many researchers have investigated the damage progression during the whole loading process using computed tomography [106, 113-117]. A. J. Moffat et al. [106] employed ESRF associated with an X-ray-transparent loading rig to scan the damage in a $[90/0]_s$ CFRP composite samples 4 mm wide with symmetric 1 mm edge notches under in situ loading. To get greater visualization, the surrounding composites were partially removed using software to reveal the damage. In addition to this, he also quantified the crack opening displacements via in situ loading, which can be shown in Figure 2.23. To get a relatively high accuracy for damage assessment of composites, W. Hufenbach [115] designed a new test device combining an integrated testing machine with a CT-system V|tome|x-L 450 (GE Phonenix X-ray). A. E. Scott [117] employed ESRF in Grenoble to capture the in situ measurement of the fiber damage development in a carbon-epoxy notched $[90/0]_s$ laminates subjected to tensile loading. Figure 2.24 shows the 3D damage in specimen at 20% to 80% of failure load by using the VG studio Max v2.1. Apparently they distinguished the different types of damage by different colours in the Figure 2.24, which can provide more information on the damage distribution in the laminates. Subsequently, A. E. Scott [118] compared the experimental results with a multi-scale model which is used to predict damage propagation in composites subjected to tensile loads and concluded that the model resulted in accurate predictions. To characterize the damage progression and the effects of voids on the fatigue performance of FRP composites, J. Lambert [107] prepared several $[+45/-45/0]_{3s}$ glass/epoxy laminates with voids inside. Then eight specimens cut from the laminates were tested to different fractions of their estimated fatigue life, and CT scanned respectively to observe the damage progression during fatigue test. A three dimensional CT image of the damage in stage two as an example can be seen in Figure 2.25, in which the delamination is represented in red and the voids in yellow. R. Brault et al. [119] developed an in situ loading device in the chamber of micro CT facility for investigation of mechanical behaviors of FRP materials subjected to three-point bending loads. They used the CT facility associated

with the in situ rig and Digital Volume Correlation (DVC) to measure the displacements and deformations in whole volume of CFRP composite specimen. R. Böhm et al. [120] compared the damage analyses of composites subjected to tensile and compression loads using in situ CT technique and normal CT technique (scans taken after unloading). He found that the crack length and delamination areas observed by in situ CT are larger, which is due to the crack closure effects during unloading.

(a) the shape and COD of the split at 40% fracture strength, (b) CT section taken at 140 μm in the y direction. (c, d) and (e, f) are the corresponding images at 50% and 60% fracture strength, respectively

Figure 2. 23 The evolution and crack opening displacement (COD) of a 0° split at increasing loads [106]

Figure 2. 24 Extraction and segmentation of matrix damage: in situ loading from 20% to 80% of failure load, composite has been made partially transparent [117]

(a) taken 0.2 mm from the surface of the specimen. (b) taken 0.6 mm from the surface of the specimen

Figure 2. 25 CT images of damage in a $[+45/-45/0]_{3s}$ glass/epoxy laminate [107]

But only static tensile or compressive loading could be achieved for in-situ computed tomography system. The use of X-ray CT technology associated with reconstruction and visualization software for analysing qualitatively and quantitatively the damage progression in CFRP laminates under flexural fatigue loads has not yet been researched.

2.6 Summary

In this chapter, the constitutions of FRP composites were introduced firstly. Secondly the manufacturing technologies including wet hand layup process, autoclave process, resin transfer molding process, thermostamping, pultrusion and thermoplastic filament winding were introduced. The material and fabrication method in this research were selected based on the knowledge and the availability of the equipment in the laboratory. Thirdly the classic lamination theory and determination of out-of-plane stresses were described, which was employed to analyse the distributions of the stress components in the FRP composites subjected to loads. Then the main failure mechanics in FRP composites including fiber failure, matrix failure, fiber/matrix debonding and delamination were presented. Finally, previous research work on static bending, fatigue bending and interlaminar fracture behaviors and X-ray CT applications for FRP composites were reviewed, which proved the necessity of this research.

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Chapter 3 Experimental methodology

In this chapter, the procedures and principals of all the experimental methodologies including material fabrication, mechanical tests and microstructure characterization were presented.

3.1 Fabrication

The aim of the study is to observe the damage progression and investigate the mechanical behaviors of CFRP developed for wind turbine blades. Considering the standards for design of wind turbine blade, an off-the-shelf flame retardant unidirectional (UD) modified epoxy prepreg L-930HT (GT700) supplied by J.D. Lincoln Inc. of United States was considered for specimen fabrication. The physical properties of the prepreg are introduced in Table 3.1.

Table 3. 1 Physical properties of the prepreg L-930HT

Physical properties	Values
Fiber areal weight	150 g/m ²
Standard resin content	40% by weight
Volatile content	2% max
Standard track	Medium
Cured ply thickness	0.165-0.178 mm
Shelf life	6 months at 4°C or below 14 days at room temperature
Cure cycles	60 minutes at 121-135°C with 50 PSI minimum

Figure 3.1 shows the procedure of the material fabrication. Firstly the prepreg was removed from freezer at temperature -40°C and defrosted in lab at temperature 20°C with relative humidity 50% for 24 hours, which can ensure water droplets not form on the epoxy prepreg during cutting and stacking process.

After defrosting process, layers with different fiber orientations were cut from the prepreg according to the layup sequences of the laminates. All the tools for cutting are shown in Figure 3.1(a), which includes rulers, scissor, knife and roller. And Figure 3.1(b) shows the methods how to cut 0°, 45 and 90° plies

from the roll of unidirectional prepreg. The dimensions of all the prepreg cut for the present study are all 190 mm $\times$ 250 mm. Then the layers were stacked one by one and compact it with the roller to avoid the air trapped between each layer.

After the prepreg layup process, a typical autoclave mould was made, which consists of breather, aluminum caul plate, porous peel ply, bleeder and non-porous peel ply. Figure 3.1(d) shows the moulding arrangements of all the components. Two aluminum caul plates with different sizes were required, one of which was used as base of the mould to hold the whole mould. The second plate with smaller size was put on the laminate to distribute the pressure force evenly during curing. Bleeders were used to absorb excess resin in the laminates with a porous peel ply located between laminate and bleeder to allow the transfer of resin and make the laminate surface smooth. To avoid sticking of resin to plates, a non-porous peel ply was placed between the bleeder and caul plates. While breather was employed to create uniform vacuum over the laminate and remove air and volatiles from the whole laminate. Then a thermocouple was placed and stick to the edges of the laminates to measure the temperature during curing process and the whole mould was sealed tightly with thermal insulation plastic bag, which can be shown in Figure 3.1(e). Finally, a vacuum valve was placed on top of the breather to connect to autoclave vacuum pump, which is shown in Figure 3.1(f).

(a) Tools for cutting

(b) Prepreg cutting

Figure 3. 1 Procedure of the material fabrication

An autoclave is a large rigid pressure vessel equipped with temperature and pressure control system. The sealed mould was then transferred into the autoclave implanted in Aerospace Structures Laboratory of NTU for the curing of the laminates, which is shown in Figure 3.1(f). Referring to the optimal cure cycle obtained from the manufacturer J.D. Lincoln Inc. for the L930HT (GT700) Prepreg, the temperature inside the autoclave chamber was set to rise at $3^{\circ}\text{C} / \text{min}$ to 127°C and hold for 60 min. The pressure inside the chamber was maintained at 0.41 MPa and the vacuum pressure in the sealed mould pressure should be kept at -0.069 MPa (i.e., 0.069 MPa lower than the atmospheric pressure) during the whole curing process.

The cure of epoxy resins is complicated [1]. Initially there is reaction between epoxy and hardener reactive groups so that somewhat larger molecules are formed. As cure proceeds, larger and larger molecules are formed but it

should be noted that the average molecular size is still small even when half the reactive groups have reacted. When the molecular size increases as cure progresses, some very highly branched molecules are formed and then more and more highly branched structures develop. The critical point is gelation when the branched structures extend throughout the whole sample. At the gel point small and branched molecules are present which are soluble, hence the curing sample contains sol as well as gel fractions. As cure proceeds, most of the sample is connected into the three-dimensional network to produce a structural material so that the sol fraction becomes small. To provide a framework of understanding the cure process of thermosets, a isothermal time-temperature transition (TTT) diagram introduced by J. K. Gillham [2]. Figure 3.2 is a simplified version which illustrates the dominant effect of the onset of vitrification as T_g increases to the cure temperature. T_{g0} is the glass transition temperature of the mixture of epoxy prepolymer-hardener-additive (uncured resin). $_{gel}T_g$ is the temperature at which gelation coincides with vitrification and $T_{g\infty}$ is the maximum glass transition temperature of the system (fully cured resin). The useful temperature range for cure is between T_{g0} and $T_{g\infty}$. Within this range, curing behavior is critically dependent on $_{gel}T_g$. If cure temperature is higher than $_{gel}T_g$, gelation precedes vitrification, if cure temperature is lower than $_{gel}T_g$, vitrification occurs first.

Figure 3. 2 Simplified time-temperature-transition diagram (TTT) for resin cure

So when the temperature in the chamber was increasing, the resin viscosity decreases and the chemical reaction of the resin began. The excess resin was

squeezed out from the laminates attributed to the constant pressure in the chamber. At the cooling stage, the prepreg layers consolidated together and transformed to glassy state. Meanwhile, the vacuum pressure was applied to draw out all volatiles emerging during curing process.

Totally six kinds of layup sequences of CFRP laminates were considered in the present research. The layup sequences $[0/-45/+45]_s$, $[+45/-45/0]_s$ and $[+45/-45/0]_{2s}$ were considered for comparison of static and fatigue tensile behaviors. Then layup sequence $[+45/-45/0]_{2s}$ was considered for investigating the static and fatigue flexural behaviors of CFRP laminates. The layup sequences $[0_{10}/0_{10}]$, $[0_8/-45/0//+45/0_9]$ and $[0_8/-45/45// -45/45/0_8]$ were considered for investigating the mode I interlaminar fracture behaviors. Due to the different layup sequences, the thickness of the laminates is different. The width and length of all the laminates fabricated for the present study are 190mm and 250 mm respectively. Based on the aim of the present study, low-speed water-jet cutting is selected to avoid defects which may encounter during cutting process. After curing, excess resin was absorbed by the bleeder. The fiber content weight fraction of all the laminates after curing in the present study is around 70%. The dimensions of the specimens are displayed in Table 3.2.

Table 3. 2 The dimensions of all specimens in the present study

Test	Layup sequence	Thickness (mm)	Length (mm)	Width (mm)	Referring standard
Tensile	$[0/-45/+45]_s$	1.15	160	12	ASTM D3039/3479
Tensile	$[+45/-45/0]_s$	1.15	160	12	ASTM D3039/3479
Tensile	$[+45/-45/0]_{2s}$	2.13	160	12	ASTM D3039/3479
Bending	$[+45/-45/0]_{2s}$	2.13	150	12	ASTM D790
Mode I	$[0_{10}/0_{10}]$	3.7	170	20	ASTM D5528
Mode I	$[0_8/-45/0//+45/0_9]$	3.7	170	20	ASTM D5528
Mode I	$[0_8/-45/45// -45/45/0_8]$	3.7	170	20	ASTM D5528
Mode II	$[0_{10}/0_{10}]$	3.7	180	20	ASTM D7905
Mode II	$[0_8/-45/0//+45/0_9]$	3.7	180	20	ASTM D7905
Mode II	$[0_8/-45/45// -45/45/0_8]$	3.7	180	20	ASTM D7905

The specimens for the present study can be split into five groups.

To investigate the static and fatigue tensile behaviors of the CFRP composites, dozens of $[0/-45/+45]_s$, $[+45/-45/0]_s$ and $[+45/-45/0]_{2s}$ specimens with dimensions shown in Table 3.2 were cut from the laminates.

To investigate the static and fatigue behaviors flexural of the CFRP composites, forty-eight $[+45/-45/0]_{2s}$ specimens with dimensions of 150 mm×12 mm×2.13 mm were cut from the laminates according to the cutting plan shown in Figure 3.3 (note that Teflon layer haven't been inserted inside).

To investigate the effect of pre-delamination on the flexural behaviors of symmetric CFRP composites. Seven laminates with symmetric layup sequences of $[+45/-45/0]_{2s}$ were fabricated. One of the seven laminates is normal without any embedded pre-delamination, which is named as NPD. For the other six laminates in which pre-delamination was created by embedding a Teflon layer with thickness of around 20 μm at the first (+45/-45), second (-45/0), third (0/+45), ninth (+45/0), tenth (0/-45) and eleventh (-45/+45) interface are named as PD1, PD2, PD3, PD9, PD10 and PD11 respectively. Apparently, the positions of pre-delamination embedded in the PD1, PD2 and PD3 laminates are symmetric about the neutral plane with PD11, PD10 and PD9 laminates, respectively. Figure 3.3 shows the preparation of specimens with pre-delamination. It is shown that the Teflon layer was embedded in the central part of the laminates. Eight specimens were cut from each laminate and the dimensions of the Teflon in the specimens are 12 mm×12 mm. The seven kinds of laminate configurations are list in Table 3.3, in which PD represents the embedded pre-delamination.

Figure 3. 3 Preparation of the CFRP specimens for bending tests (unit: mm)

Table 3. 3 Configurations of laminates NPD, PD1, PD2, PD3, PD9, PD10 and PD11

Laminate name	Configuration
NPD	[+45/-45/0/+45/-45/0/0/-45/+45/0/-45/+45]
PD1	[+45/PD/-45/0/+45/-45/0/0/-45/+45/0/-45/+45]
PD2	[+45/-45/PD/0/+45/-45/0/0/-45/+45/0/-45/+45]
PD3	[+45/-45/0/PD/+45/-45/0/0/-45/+45/0/-45/+45]
PD9	[+45/-45/0/+45/-45/0/0/-45/+45/PD/0/-45/+45]
PD10	[+45/-45/0/+45/-45/0/0/-45/+45/0/PD/-45/+45]
PD11	[+45/-45/0/+45/-45/0/0/-45/+45/0/-45/PD/+45]

As for the layup sequence $[+45/-45/0]_{2s}$ investigated in the present study, there are totally three kinds of interface 0/0, 0/45 and +45/-45. So subsequently the interlaminar fracture behaviors at the three kinds of interfaces were studied. As the literature is reviewed previously, DCB test can be conducted to investigate the mode I interlaminar fracture behaviors of FRP composites. But the layup sequence of the DCB specimens should be considered carefully to avoid inaccurate values of interlaminar toughness. To obtain reasonable layup sequences, a specimen for DCB test can be regarded as composed of three “arms”. Arm 1 and arm 2 are in the cracked region, and the un-cracked section is regarded as arm 3. Take the laminate configuration $[0_8/-45/45// -45/45/0_8]$ for example, arm 1, arm 2 and arm 3 is $[0_8/-45/45]$, $[-45/45/0_8]$ and $[0_8/-45/45// -45/45/0_8]$, respectively (the symbol // stands for delamination plane). The layup

sequences of the laminates should satisfy criteria on the two parameters D_c and B_t , which have been extensively employed to characterize the strain energy release rate (SERR) components along the width of the specimen [3-5]. Davidson et al. [3] stated that significant errors in interlaminar toughness measurement can be eliminated for the laminates with a small enough D_c , which can be determined by Equation 3.1. Usually the value of D_c is recommended to be smaller than 0.25 in all the three arms [5-7].

$$D_c = \left(\frac{D_{12}^2}{D_{11}D_{22}} \right) \quad 3.1$$

Where D_{ij} ($i, j=1, 2$) is the bending stiffness. The parameter B_t defined in Equation 3.2 characterizes the skewness of the crack front shape caused by bending-twisting coupling of the three arms. B_t is required to be as small as possible to minimize the skewness of the SERR distribution [4].

$$B_t = \left| \frac{D_{16}}{D_{11}} \right| \quad 3.2$$

To satisfy the criteria presented above, three kinds of layup sequences: $[0_{10}/0_{10}]$, $[0_8/-45/0//+45/0_9]$ and $[0_8/-45/45// -45/45/0_8]$ were considered, where the symbol “//” represents the delamination embedded during the fabrication process. Table 3.4 shows the values of the parameters in the three arms of the laminates, which indicates that the three layup sequences are reasonable for the current research. In addition, the plies far away from the interface for the two multidirectional laminates were chosen as 0° plies, which can minimize the misleading effects of the stacking sequence of the arms and highlight the effects of the two adjacent plies at the interface.

Table 3. 4 Parameters of the three specimen configurations used for DCB test (// stands for delamination plane)

Parameter	$[0_{10}/0_{10}]$			$[0_8/-45/0//+45/0_9]$			$[0_8/-45/45// -45/45/0_8]$		
	Arm 1	Arm 2	Arm 3	Arm 1	Arm 2	Arm 3	Arm 1	Arm 2	Arm 3
D_c	0.0026	0.0026	0.0026	0.023	0.043	0.077	0.083	0.083	0.0033
B_t	0	0	0	0.039	0.069	0.00071	0.031	0.031	0

To investigate the mode I interlaminar fracture behaviors, $[0_{10}/0_{10}]$, $[0_8/-45/0/+45/0_9]$ and $[0_8/-45/45/-45/45/0_8]$ laminates were fabricated. And 170 mm×20 mm×3.7 mm specimens with 70 insert length were cut from the laminates, which is shown in Figure 3.4. The thickness of all the laminates was measured as around 3.7 mm, which is reasonable as thick specimens are able to avoid large displacements and plastic deformations during DCB tests.

Figure 3. 4 Preparation of the CFRP specimens for mode I interlaminar fracture tests
(unit: mm)

3.2 Mechanical tests

3.2.1 Determine mechanical properties of unidirectional lamina

CFRP unidirectional laminates were fabricated for static tests to obtain properties at the longitudinal and transverse directions. To obtain the tensile properties, $[0]_8$ and $[90]_8$ specimens with dimensions of 160 mm×12 mm×1.45 mm were cut from the unidirectional laminates for static tensile tests, which were conducted with MTS 810 Material Testing System at speed of 0.5 mm/min according to ASTM D3039. It should be noted that CFRP tabs with length of 50 mm were bonded on the two ends of the specimen to prevent gripping damage. Two one-element strain gauges GFLA-3-350-70 manufactured by Tokyo Sokki Kenkyujo Co., Ltd. were bonded on central region of each specimen to measure actual strain in the longitudinal and

transverse direction. The Digital Data logger TDS- 530 was connected to the strain gauges to record actual strain at a frequency of 1 Hz during mechanical test. The data logger and testing machine was started simultaneously, in which way the actual stress values obtained from testing machine and actual strain values obtained from data logger can match by the time. To obtain the compressive properties, $[0]_8$ and $[90]_8$ specimens with dimensions of 150 mm×12 mm×1.45 mm were cut from the unidirectional laminates for static compressive tests, which were conducted with Instron 5569 at speed of 1.5 mm/min according to ASTM 3410. It should be noted that CFRP tabs with length of 69mm were bonded on the two ends of the specimen to prevent gripping damage, which result in gauge length of 12 mm. Figure 3.5(a) and 3.5(b) shows the specimen subjected to tensile loading and compressive loading respectively. The in-plane shear properties were obtained with $[0/90]_{2s}$ laminates according to ASTM D7078. The v-notched specimens with notch width of 32 mm and thickness of 1.45 mm were cut from the laminates. All the shear tests were conducted with MTS 810 Material Testing System. More than four specimens were tested for each layup and the results show good repeatability.

(a) Tensile test

(b) Compressive test

Figure 3. 5 Static tensile and compressive tests

3.2.2 Static and fatigue tensile tests

All the static and fatigue tensile tests were conducted using MTS 810 Material Testing System according to ASTM D3039 standard and ASTM D3479

standard. To measure actual strain in the longitudinal direction, a one-element strain gauge GFLA-3-350-70 manufactured by Tokyo Sokki Kenkyujo Co., Ltd. were bonded on central region of each specimen. The Digital Data logger TDS-530 was connected to the strain gauges to record actual strain at a frequency of 1 Hz during mechanical test. The data logger and testing machine was started simultaneously, in which way the actual stress values obtained from testing machine and actual strain values obtained from data logger can match by the time. The loading speed for static tensile tests was 0.5 mm/min. The fatigue tests were performed under force control mode at a frequency of 5 Hz and R ratio ($R = \text{minimum stress} / \text{maximum stress in cycles}$) of 0.1.

3.2.3 Static and fatigue bending tests

All the bending tests were performed with MTS 810 Material Testing System. A three-point bending fixture was designed and manufactured, which can be shown in Figure 3.6. Figure 3.6(a) shows the assembly drawings of the fixture. It is obvious the whole fixture is clamped by hydraulic wedge grips with clamping pressure of 3 MPa. The loading nose and supports are made of stainless steel 304, which is stiff enough to make sure the elastic deformation of them has negligible effects on the results. Because the surfaces of the stainless steel rollers are smooth, so specimens may slide out during fatigue bending tests. Hence two stainless steel cylinders were stucked with tape on each support roller to prevent the specimen sliding out during bending tests, which is shown in Figure 3.6(b) and 3.6(c). And a paperboard was stucked on each side of the fixture to prevent the specimens sliding along the longitudinal direction during bending tests. During fatigue bending tests, the bottom clamp of the machine may rotate when it moves up and down cyclically, which will result in specimen sliding or unacceptable results. Four plastic rollers were mounted on the fixture to prevent the bottom clamp from sliding. Before the fatigue bending tests, the positions of the four rollers were adjusted to make sure there is almost no gap between the roller and the chamber wall. Thus, the rollers will roll on the wall during fatigue bending tests, in which case the bottom clamp won't

rotate. The rolling friction force between the roller and chamber has negligible effects on the bending loads.

(a) Design drawing (b) Fixture clamped in loading machine

(c) Fixture

Figure 3. 6 Setup of static and fatigue three-point bending test

To investigate the static and fatigue flexural behaviors, static bending tests were performed on eight specimens and fatigue bending tests were conducted on the other forty specimens. For static bending tests, the crosshead velocity was set as 2 mm/min and the span length was 75 mm for static bending tests based on the specimen geometry and ASTM D790. For fatigue bending tests, displacement-controlled flexural fatigue tests with the sinusoidal waveform (ASTM D7774) were performed under five different cyclic deflection levels (CDL). The CDL normalised by the static failure deflection (SFD) was $\gamma_c = \text{CDL}/\text{SFD} = 0.58, 0.65, 0.71, 0.84$ and 0.91 . Eight specimens were repeated in the flexural fatigue tests under each CDL. The ratio of the minimum and maximum displacement was constant at $R = 0.1$ for all the fatigue experiments.

The frequency was limited to 2 Hz given the large midspan deflection. The stiffness was measured and monitored in the fatigue tests.

To investigate the effects of pre-delamination on the flexural behaviors of symmetric CFRP laminates. Static flexural tests were conducted on all the PD1, PD2, PD3, PD9, PD10 and PD11 specimens. The crosshead velocity was also set as 2 mm/min and the span length was 75 mm.

As for the calculation of flexural stress or strength on the specimen subjected to three-point bending. The following paragraphs are to describe how the calculation formula is derived.

Beam represents structure that loads act transversely to the longitudinal axis. The lateral loads acting on a beam cause the beam to bend, thereby deforming the axis of the beam into curve line, which is called as the deflection curve of the beam [8].

Figure 3.7 shows a portion AB of a beam in pure bending produced by a positive bending moment M . The cross section may be of any shape provided it is symmetric about y-axis. Under the moment M , its axis is bent into a circular curve, cross section mn and pq remain plane and normal to longitudinal lines. The surface ss in which longitudinal lines do not change in length is called the neutral surface, its intersection with the cross-sectional plane is called neutral axis, for instance, the z axis is the neutral axis of the cross section. The radius of curvature is represented by ρ , and the curvature κ is defined as

$$\kappa = \frac{1}{\rho} \quad 3.3$$

The stress in the longitudinal direction σ_x are positive below neutral surface and negative above neutral surface, which can be shown in Figure 3.8. The longitudinal stress σ_x can be expressed as:

$$\sigma_x = E\varepsilon_x = -E\kappa y \quad 3.4$$

As no axial force acts on the cross section, the only resultant is M , thus force equilibrium condition should be satisfied as:

$$\sum F_x = \int \sigma dA = -\int E\kappa y dA = 0 \quad 3.5$$

Figure 3. 7 A portion of a beam in pure bending

Figure 3. 8 Distribution of longitudinal stress σ_x in a beam

As E and κ are constants at the cross section, so the equation can be obtained:

$$\int y dA = 0 \quad 3.6$$

It can be concluded that the neutral axis passes through the centroid of the cross section, also for the symmetrical condition in y axis, the y axis must pass through the centroid, hence, the origin of coordinates O is located at the centroid of the cross section.

The moment resultant of stress σ_x is expressed as:

$$dM = -\sigma_x y dA \quad 3.7$$

Then

$$M = -\int \sigma_x y dA = \int E\kappa y^2 dA = E\kappa \int y^2 dA \quad 3.8$$

$$M = E\kappa I \quad 3.9$$

Where I is the moment of inertia of the cross-sectional area, which for rectangular cross section can be expressed as:

$$I = \frac{bd^3}{12} \quad 3.10$$

So it can be concluded that:

$$\kappa = \frac{1}{\rho} = \frac{M}{EI} \quad 3.11$$

Which is the moment-curvature equation. And EI is called as flexural rigidity.

So the normal stress in the longitudinal direction is:

$$\sigma_x = -E\kappa y = -Ey\left(\frac{M}{EI}\right) = -\frac{My}{I} \quad 3.12$$

Which is called as the flexure formula, the stress σ_x is called as bending stress or flexural stress.

For three-point bending beam, the midpoint bears the maximum bending moment along the whole beam. The midpoint bending moment is the force reacted from any of the support ($P/2$) multiplied by the half length of the beam ($L/2$). While in the equation, M_x represents the bending moment in the unit width. So the bending moment for three-point bending load can be determined by:

$$M = \left(\frac{P}{2}\right) * \left(\frac{L}{2}\right) \quad 3.13$$

Thus the flexural stress on the outer surface at the midpoint is determined by:

$$\sigma_{x(\text{midpoint})} = -\left(\frac{P}{2}\right) * \left(\frac{L}{2}\right) * \left(\frac{-d}{2}\right) * \left(\frac{12}{bd^3}\right) = \frac{3PL}{2bd^2} \quad 3.14$$

3.2.4 Mode I interlaminar fracture tests

ASTM D5528 was established to determine mode I interlaminar fracture toughness of unidirectional FRP composites. But many researchers used this standard to determine the toughness of multidirectional FRP composites [5, 9]. In the present research, using the same standard ASTM D5528 for the three kinds of laminates is reasonable for the purpose of comparison. As shown previously, the dimensions of specimens for mode I interlaminar fracture tests are 170 mm×20 mm×3.7 mm with 70 mm insert length. This insert length

equals to an initial delamination length of 50 mm plus the length required for bonding the hinges. Five specimens were tested for each kind of layup sequence. As Figure 3.9 shows, a pair of hinges was used in the DCB test to apply the loads on the end of the specimen. A high-resolution camera JAI BM-500GE and Nikon Coolpix L310 were used to observe the crack growth. Both edges of specimen were coated with white correction fluids for clear observation of the crack tips. A measurement scale with 1 mm divisions was bonded on each edge of the specimen to measure the crack length from the photos taken by each camera. The DCB tests were performed on an Instron 5569 machine with a load cell of 1 kN at a displacement loading rate of 5 mm/min.

Figure 3. 9 Setup of the DCB test

Firstly, initial loading is conducted to determine the initial mode I interlaminar toughness. During loading, the values of load and displacement were recorded at every 1 mm delamination length increments. If unstable delamination growth from the insert is observed, note in the results if the delamination length increment is beyond 5 mm. The mode I initial interlaminar toughness G_{IC} can be determined using the load and displacement values at the beginning of nonlinearity on the plot (NL), or at the point at which delamination is visually observed with a microscope (VIS), or at the point at which the compliance has increased by 5% (5%/Max). The NL G_{IC} value, which is usually the lowest G_{IC} initiation values, is normally for generating delamination failure criteria and analysing damage tolerance of composite structures. Physical evidence indicates that the beginning point of nonlinearity on the load-displacement plot corresponds to the physical initiation of

delamination from the insert in the interior of the specimen width. All three initiation values can be used to analyse quantitatively the effect of constituent features and environmental conditions on G_{IC} of composites.

After initial loading, the specimen was unloaded at 25 mm/min and then reloaded again. The load and displacement data at every 5 mm delamination length is recorded, until the delamination crack has propagated at least 45 mm from the tip of the precrack, and again at every 1 mm increment of delamination growth of the last 5 mm of delamination propagation. Finally, the specimen is unloaded again at 25 mm/min.

3.2.5 Mode II interlaminar fracture tests

Normally the end-notched flexure (ENF) test is adopted to measure the mode II interlaminar fracture toughness, G_{IIC} , of FRP composite laminates under mode II shear loading. And ASTM D7905 was established to determine mode II interlaminar toughness of FRP composites. As shown previously, the dimensions of specimens for mode II interlaminar fracture tests are 180 mm*20 mm*3.7 mm with 50 mm insert length. Figure 3.10 shows the setup of ENF test, in which an Instron 5569 machine with a load cell of 1 kN was employed in the present study.

Figure 3. 10 Setup of ENF test

Firstly, a light white correction fluids were coated on the specimen edges, which enable clear visual observation of the delamination tip and in making compliance calibration (CC) markings. Once the paint is dry, the tip of the

insert was marked with a thin vertical pencil line. The edges are then marked within the cracked region at the distances of 20, 30 and 40 mm from the tip of the insert. Then non-precacked compliance calibration (NPC-CC) tests and non-precacked fracture test were conducted, which can be shown in Figure 3.11.

Figure 3. 11 Configuration of specimen for NPC tests

As in the Figure 3.11, h represents the specimen half-thickness, a_1 (=20 mm) and a_2 (=40 mm) represents the crack lengths used during NPC-CC tests. a_0 is the delamination length used in NPC fracture test. L represents the specimen half-span, which is 50 mm. r represents the radius of the roller, which is 5 mm.

During the NPC-CC, the specimen was placed in the fixture so that the NPC-CC mark is aligned with the center of the support roller. The first NPC-CC test is then performed by loading the specimen with a crack length a_1 to a specified force, which can be estimated according to ASTM D7905. Then the specimen was repositioned to make sure the mark for NPC-CC (a_2) is aligned with the center of the support roller. After that, the NPC-CC test for a_2 was conducted.

After the two NPC-CC tests, the specimen was repositioned in the fixture so that the NPC fracture test mark was over the center of the support roller. Then the specimen is then loaded until the delamination advances, as seen by visual assessment on the specimen or by a drop in force on the force versus displacement plot. Subsequently the specimen was unloaded at a nominal rate of 0.5 mm/min.

After the two NPC-CC tests and NPC-CC fracture test, the three compliances versus crack length cubed were plotted, which are from the two NPC-CC tests (a_1 = 20 mm and a_2 = 40 mm) and from the NPC fracture test (a_0 = 30 mm). At

each crack length, the compliance is determined by a linear least squares regression analysis to obtain the slope of the displacement versus force (δ/P) data. Then the NPC-CC coefficients, A and m , are to be determined using a linear least squares linear regression analysis of the compliance, C , versus crack length cubed (a^3) data of the form as shown in Equation 3.15.

$$C = A + ma^3 \quad 3.15$$

While the unloading data from the NPC fracture test is used to calculate a_{calc} , which represents the crack length measured from the existing NPC-CC fracture test mark. The compliance of the NPC fracture test unloading line, C_u , is used in the Equation 3.16.

$$a_{calc} = \left(\frac{C_u - A}{m} \right)^{1/3} \quad 3.16$$

Subsequently, a new set of marks were placed on the specimen edges for precracked (PC) tests based on the value of a_{calc} . Figure 3.12 shows the configuration of the PC tests.

Figure 3. 12 Configuration of specimen for PC tests

Like the NPC-CC tests, the first PC-CC test was performed with $a_1=20$ mm, followed by PC-CC test with $a_2=40$ mm. Subsequently, the PC-CC fracture test was conducted on the specimen until the delamination advances, as seen by visual assessment on the specimen or by a drop in force on the force versus displacement plot. Then the specimen was unloaded. Similarly, the PC-CC coefficients, A and m were also calculated according to equation.

Finally, the NPC and PC toughness is determined using Equation 3.17.

$$G_{IIc} = \frac{3mP_{max}^2 a_0^2}{2B} \quad 3.17$$

Where B is the specimen width, which is 20 mm in the present study. $P_{\max}$ is the maximum force from the fracture test. When computing the NPC G_{IIC} , these parameters were taken from the NPC-CC and fracture tests. When computing the PC G_{IIC} , these parameters were taken from the PC-CC and fracture tests.

3.3 Characterization

3.3.1 Scanning electron microscope (SEM)

The scanning electron microscope (SEM) is an equipment to characterize heterogeneous organic and inorganic materials. In the equipment, the object for analysis is irradiated with a finely focused electron beam. The kinds of signals when electron beams impinges on a specimen surface include secondary electrons, backscattered electrons, Auger electrons, characteristic x rays, and photons of various energies. These signals can reflect the features of the sample (composition, surface topography, crystallography, etc.). Among all types of signals, secondary electron is the most extensively used because it can reflect the surface topography of the specimen at relatively high resolution [10].

In the present study, JEOL JSM-5600LV Scanning Electron Microscope was adopted to observe the fracture morphologies inside the specimen after failure. This will expect to obtain a further understanding of the failure mechanisms for this material.

3.3.2 X-ray computed tomography

Failure analysis is essential for every single engineered product. The failure analysis is to understand the causes of any undesirable defects, investigate the failure models and even predict the in-service life. Among sorts of non-destructive techniques, CT technology has been proven to be an excellent tool for characterizing the damage of materials in recent years.

An industrial CT system normally consists of x-ray source, detector array, mechanical systems to position the component, a computer system for reconstruction, a display to view the results, and data storage media.

Figure 3.10 shows a typical schematic of the a CT scanning configuration [11]. The single explanation for operational principals is as follow (literatures

[12, 13] show the further details). As shown in Figure 3.10, the object is fixed on a rotation stage, which is located between the gun and the detector. A cone of X-rays is emitted from the gun and passes through the specimen, which will be received by the detector panel. During the scanning procedure, the object is incrementally rotated through 360° about the Z-axis with an X-ray radiograph (projection) taken. It should be noted that the object should always be kept within the X-ray cone when the stage is rotating. Moreover, the object should be positioned as closer to the target as possible to achieve a higher spatial resolution.

In the present study, Yxlon X-ray CT system was applied to conduct the tomography experiments. X-ray source was operated at 55KV and 46μA. The detector was a CCD camera with 1848 x 1480 sensitive elements. A set of 720 projections was taken within 360° during scanning. Then the projection images were reconstructed into a three dimensional volume of the specimen by Power Recon, a software designed by Simtech. Subsequently, AVIZO/FIRE software packages were used to segment and quantify the damage (e.g. delamination, transverse crack and fiber failure) from the surrounding composite.

3.4 Summary

In this chapter, the procedures and principals of all the experimental methodologies were presented. Firstly, the fabrication of FRP composite laminates and specimens were described. Then several types of mechanical tests conducted for this topic were indicated. Finally, material characterization methods including SEM and X-ray scanning were presented.

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Chapter 4 Static and fatigue tensile behaviors of CFRP composite laminates

In this chapter, the tensile and fatigue behaviors of three kinds of CFRP laminates with stacking sequences of $[0/+45/-45]_s$, $[+45/-45/0]_s$ and $[+45/-45/0]_{2s}$ that have been extensively used by wind blade manufacturers were investigated. CT scanning technology was employed to observe the damage patterns during the fatigue tests. Two models were used to represent the trends of fatigue life for the two kinds of laminates. And the differences of stiffness degradation were characterized for the three laminates.

4.1 Introduction

Different stacking sequences are considered for different components of the wind blades, and different stacking sequences will lead to different mechanical performances of fiber reinforced polymer composites. But seldom researchers have investigated the effects on the fatigue behaviors of FRP laminates. So the effects of layup sequence on the static and fatigue behaviors of FRP composites are necessary to be investigated.

4.2 Experimental methods

Three laminates with symmetric layup sequences of $[0/+45/-45]_s$, $[+45/-45/0]_s$ and $[+45/-45/0]_{2s}$ were fabricated, which are extensively used by wind blade manufacturers. Then the prepreg laminates with dimensions of 190 mm×250 mm were cured in an autoclave at 127°C for 60 min under the vacuum pressure of -0.034 MPa (0.034 MPa lower than Atmosphere Pressure). The pressure inside the chamber was kept at 0.41 MPa during the whole curing process.

From each laminate, dozens of specimens were cut down for tensile and fatigue test according to ASTM D3039 standard. The dimensions of the specimens can be found in Table 3.2. Tensile and fatigue tests were conducted using MTS 810 Material Testing System according to ASTM D3039 standard

and ASTM D3479 standard. For static tensile tests, a one-element strain gauge GFLA-3-350-70 manufactured by Tokyo Sokki Kenkyujo Co., Ltd. were bonded on central region of each specimen to measure actual strain in the longitudinal direction. The Digital Data logger TDS- 530 was connected to the strain gauges to record actual strain at a frequency of 1 Hz during mechanical test. The data logger and testing machine was started simultaneously, in which way the actual stress values obtained from testing machine and actual strain values obtained from data logger can match by the time. The fatigue tests were conducted under force control mode at a frequency of 5 Hz and R ratio ($R = \text{minimum stress} / \text{maximum stress in cycles}$) of 0.1.

Yxlon X-ray CT system was applied to conduct the tomography experiments in this study. X-ray source was operated at 55 KV and 46 μA . During tensile and fatigue tests, several specimens were interrupted at different time intervals for CT scanning.

4.3 Comparison of static performance

For each of the three CFRP laminates, three specimens were tested statically to failure. Figure 4.1 shows stress-strain curves obtained in static tensile test for the six specimens. The average values of ultimate strength, Young's modulus and failure strain are reported in Tables 4.1. In Table 4.1 the ratio represents the ratio of value of specified property (ultimate strength, Young's modulus or failure strain) divided by that of $[+45/-45/0]_{2s}$ laminates. According to ASTM D3039 standard, the modulus E is the initial tangent modulus on the stress-strain diagram, which is an appropriate method for the fiber-dominated composites.

The curves starts from the first linear stage which represents the elastic part of the specimens, followed by the second stage which is characterized by the initiation of damage and the crack propagation until failure. For the $[+45/-45/0]_{2s}$ laminates, the stress-strain curve show a little drop at the intermedia stage because of 'first ply failure'. But subsequently, the curve does not show as much of a change in slope after the drop as the theoretical curve does. This

may be due to the fact that the actual ply failure occurs gradually over a finite strain range, whereas instantaneous ply failure at a single strain level is assumed in the analysis. The same reasoning may explain the absence of jumps in the stress-strain curve after ply failure [1].

Figure 4. 1 Stress-strain curves in static tensile test

Although the three kinds of CFRP laminates have same number of plies with layers oriented at 0° , $+45^\circ$ and -45° , they presents substantially different mechanical properties due to the different layup sequences. It is can be clearly seen from Figure 4.1 and Table 4.1 that the laminate with $[+45/-45/0]_s$ layup sequence, in which the 0° oriented layers are inside the laminate, has higher ultimate strength and failure strain than the laminate with $[+45/-45/0]_s$ layup sequence. This result accords with the phenomenon found by F. L. Matthews [2], who indicated that the strength of $[0/+45/-45]_s$ CFRP laminate is weaker than $[+45/-45/0]_s$ laminate. Comparing the $[+45/-45/0]_s$ and $[+45/-45/0]_{2s}$ laminates, the tensile strength and failure strain of the $[+45/-45/0]_{2s}$ laminate are slightly higher than that of $[+45/-45/0]_s$ laminate, which might be because thinner laminates are more sensitive to the damage initiation and propagation.

Table 4. 1 Mechanical properties and ratios comparing to $[+45/-45/0]_s$ laminate

Laminate	Ultimate strength		Young' s modulus		Failure strain	
	MPa	Ratio	GPa	Ratio	%	Ratio
[0/+45/-45]s	547(±18)	0.66	49.4(±0.80)	0.97	1.12(±0.04)	0.68
[+45/-45/0]s	828(±64)	1	51.1(±1.23)	1	1.64(±0.07)	1
[+45/-45/0]2s	920(±72)	1.11	52.5(±3.70)	1.03	1.76(±0.04)	1.07

However, it should also be noted that Young's modulus is very similar for the three laminate geometries, which can be explained using Classic Lamination Theory [3]. Specifically, during static tensile test, the specimens are just subjected an in-plane force N_x , x and y represents the longitudinal and transverse direction of the specimen, respectively. As the stiffness matrix $[B]=0$ due to the symmetric layup pattern, the curvatures k_x , k_y and k_{xy} are all zero. So the strains ε_x , ε_y and γ_{xy} in the global coordinate system are the uniform in the gauge region of the specimen, which can be shown in Equation 4.1.

$$\varepsilon_x = \varepsilon_x^0 = \alpha_{11} N_x \quad 4.1$$

Then the in-plane longitudinal modulus E_x is represented with Equation 4.2.

$$E_x = \frac{\sigma_x}{\varepsilon_x} = \frac{N_x/h}{\alpha_{11} N_x} = \frac{1}{\alpha_{11} h} \quad 4.2$$

Where h is the thickness of the specimen. And from Equation 2.19 and the layup sequences of the three laminate. The equation 4.3 can be obtained.

$$A_{ij(layup\ 1)} = A_{ij(layup\ 2)} = 0.5A_{ij(layup\ 3)} \quad (i, j = 1, 2, 6) \quad 4.3$$

And for symmetric laminate, the compliance matrix $[\alpha]$ is the inversion of stiffness matrix $[A]$. Then the Equation 4.4 can be obtained.

$$\alpha_{11(layup\ 1)} = \alpha_{11(layup\ 2)} = 2\alpha_{11(layup\ 3)} \quad 4.4$$

Theoretically, the Equation 4.5 can be assumed.

$$h_{(layup\ 1)} = h_{(layup\ 2)} = 0.5h_{(layup\ 3)} \quad 4.5$$

Where h is the thickness of the laminate. Combining Equation 4.2, 4.4 and 4.5, the Equation 4.6 can be obtained.

$$E_{x(layup\ 1)} = E_{x(layup\ 2)} = E_{x(layup\ 3)} \quad 4.6$$

4.4 Comparison of fatigue performance

4.4.1 Stress-Number of cycles relationship

For fatigue tests, specimens were subjected to different maximum stresses (S), which are percentages of ultimate strength (S_u) in static tensile test. In the

present study, the fatigue load where specimens failed after 10^6 cycles is regarded as an “endurance limit”.

The damage life curves for the three laminates are shown in Figure 4.2, Figure 4.3 and Figure 4.4, in which a Linear model and a Sigmoidal (Boltzman) model are used to represent the trend of fatigue life.

The equation of the Linear model is given by

$$y = P_1 * x + P_2 \quad 4.7$$

where

$$y = \frac{S}{S_u} \quad 4.8$$

and

$$x = \log_{10} N \quad 4.9$$

Sigmoidal curves (S-shaped) are extensively applied to represent the growth modes in biology science. Jitendra S. Tate [4] used Sigmoidal curve to describe the fatigue life pattern for 25° braided composite laminates. The equation for the sigmoidal (Boltzmann) curve is as follows:

$$y = \frac{A_1 - A_2}{1 + e^{(x - x_0)/dx}} + A_2 \quad 4.10$$

A_1 , A_2 , x_0 , and dx are the parameters in the equation. The value of y ranges from A_1 at the maximum to A_2 at the minimum. The higher value of A_2 means better fatigue life [5]. The dx represents the slope of the curve and larger value implies shallower curve. The x_0 indicates value of x when y is half between A_1 and A_2 . Table 4.2 shows the values of all the parameters, R^2 values and endurance limit loads for the three CFRP laminates. As the R^2 values for Sigmoidal model is more closer to 1, the Sigmoidal model can represent the fatigue life better than the Linear model for the three laminates. According to the Sigmoidal model, the predicted fatigue strength at 1 million cycles for the $[+45/-45/0]_s$ laminate is 668MPa ($S/S_u=0.81$), which is far higher than the $[0/+45/-45]_s$ laminate with 410MPa ($S/S_u=0.76$). In addition, the value of A_2 for the former is also higher than the latter. Thus, the fatigue performance of the $[+45/-45/0]_s$ laminate is more excellent than $[0/+45/-45]_s$ laminate. Comparing $[+45/-45/0]_s$ laminate and $[+45/-45/0]_{2s}$ laminate, the predicted fatigue strength

at 1 million cycles for the latter is 685MPa ($S/S_u=0.74$) is just a little higher than that for the former, which is 668MPa ($S/S_u=0.81$).

Figure 4. 2 S-N diagram representation for $[0/+45/-45]_s$ laminate

Figure 4. 3 S-N diagram representation for $[+45/-45/0]_s$ laminate

Figure 4. 4 S-N diagram representation for $[+45/-45/0]_{2s}$ laminate

Table 4. 2 Comparison of linear model and sigmoidal model for the two CFRP laminates

Laminate	Model	A1	A2	x0	dx	P1	P2	R2 value	Endurance limit	
									S/Su	S (MPa)
A	Linear	-	-	-	-	0.039	1.01	0.88	0.77	423
	Sigmoidal	1.00	0.69	4.28	1.369	-	-	0.90	0.76	414
B	Linear	-	-	-	-	0.033	0.94	0.56	0.74	610
	Sigmoidal	1.00	0.80	1.12	0.350	-	-	0.80	0.81	668
C	Linear	-	-	-	-	0.052	0.94	0.68	0.63	582
	Sigmoidal	1.00	0.74	1.10	0.393	-	-	0.80	0.74	685

*A:[0/+45/-45]_s; B:[+45/-45/0]_s; C:[+45/-45/0]_{2s}

4.4.2 Stiffness degradation

In fatigue studies, fatigue secant modulus E_n is used to define modulus at each cycle, which can be defined by Equation 4.11.

$$E_n = \frac{\sigma_{\max} - \sigma_{\min}}{\epsilon_{\max} - \epsilon_{\text{res}}} \quad 4.11$$

Where $\sigma_{\max}$ and $\epsilon_{\max}$ are the maximum stress and strain during the cycle, $\sigma_{\min}$ and ϵ_{res} are the minimum stress and the residual strain at the beginning of the cycle [6].

Figure 4.5, 4.6 and 4.7 show the stiffness degradation for all the specimens of the three laminate geometries during the fatigue test. The numbers of cycles (n)

are normalized with the number of fatigue failure cycles (N). And the fatigue secant modulus E_n is normalized with the fatigue secant modulus at the first cycle, E_1 . The fatigue secant modulus cannot be normalized with the initial tangent modulus E in tensile test, because the fatigue secant modulus E_1 usually differs from E due to the nonlinearity of the materials. In addition, the residual stiffness shown in the figures starts from the first cycle, which is actually the sixth cycle in the test. This is because the required forces are not reached in the first five cycles. For the three laminates, stiffness decreased with the number of cycles increasing until failure. The stiffness degradation can be split into three stages [7]. In the initial stage, stiffness fell quickly due to the matrix cracks or edge delamination. Then the stiffness decrease steadily, followed by a sharp drop due to the accumulation of large delamination, matrix cracks or fiber failure.

Figure 4. 5 Normalized stiffness vs. Normalized cycles for $[0/+45/-45]_s$ laminate

Figure 4. 6 Normalized stiffness vs. Normalized number of cycles for $[+45/-45/0]_s$ laminate

Figure 4. 7 Normalized stiffness vs. Normalized number of cycles for $[+45/-45/0]_{2s}$ laminate

To get more detailed information from the stiffness degradation curves, statistics were made to compare the stiffness degradation patterns of all the specimens with the three laminate geometries. Figure 4.8 displays the normalized residual stiffness at initial stage ($n/N=0.1$), mediate stage ($n/N=0.5$)

and final stage ($n/N=0.9$) of the three laminates. It is obvious that the residual stiffness at any of the three stages of the $[+45/-45/0]_s$ laminate is higher than that of $[0/+45/-45]_s$ laminate, and the residual stiffness of the $[+45/-45/0]_{2s}$ laminate is higher than the $[+45/-45/0]_s$ laminate. In short, the $[+45/-45/0]_{2s}$ laminate can resist the more stiffness degradation than the $[+45/-45/0]_s$ laminate, followed by the $[0/+45/-45]_s$ laminate subjected to tension-tension fatigue loading.

Figure 4. 8 Normalized residual stiffness at different normalized cycles for the three laminates

To investigate the damage patterns during fatigue test, the three kinds of specimens subjected to fatigue load were interrupted at intermediate stage for CT scanning. Figure 4.9(a), 4.9(b) and 4.9(c) shows the damage patterns in the $[0/+45/-45]_s$, $[+45/-45/0]_s$ and $[+45/-45/0]_{2s}$ specimen, respectively. It is obvious that initially transverse crack and fiber failure occurred in the two 0 plies from Figure 4.9(a). And delamination occurred at the 0/+45 interface. While in the $[+45/-45/0]_s$ specimen, transverse crack also occurred in the two 0 plies and delamination occurred at the 0/-45 interface. In $[+45/-45/0]_{2s}$ specimen, transverse crack and fiber failure also occurred initially in 0 plies, and delamination occurred at the interfaces adjacent to the 0 plies. In the case of

$[+45/-45/0]_s$ laminate, the failure patterns occurs initially in the inner plies interfaces.

In the case of $[+45/-45/0]_s$ laminate, the failure patterns occurs initially in the inner plies interfaces. This phenomenon accords with the results of Jiayu Xiao [8], who reported that the stacking sequence can affect the fatigue performance. And he indicated that the ply interfaces have resistance to damage development. Fu Kuo Chang [9] indicated that the increasing thickness of a group identical plies will bring about the decrease of strength, which is caused by the so-called 'adjacent ply constraint'. Generally speaking, interlaminar shear and interlaminar normal stresses are greatly influenced by both the ply thickness and the adjacent ply constraint effect. These effects influence stresses in the inner regions of a plate as well as the free edge. Also, the ply angle of that adjacent ply will determine its effectiveness in arresting such as a crack. Similarly, Ronald F. Gibson [10] also indicated that the adjacent ply constraint will make the laminate fail more gradually.

In composite laminates, it is obvious that the normal stress in the fiber direction in the 0 ply bear the highest load. So fiber failure occurred first in the 0 ply and induce transverse crack in this ply and delamination adjacent. So for the $[+45/-45/0]_s$ specimen, the damage occurred first in the 0 plies at the mid-plane, which may not easily overcome the obstacles constituted by the interfaces between plies with different fiber directions for developing outwards to the two surfaces of the specimen. And the outer plies as constraint may suppress the crack propagation from the damage existed in the inner plies. This may be the reason why the $[+45/-45/0]_s$ specimen shows better tensile properties than $[0/+45/-45]_s$ specimen. Comparing the $[+45/-45/0]_{2s}$ and $[+45/-45/0]_{2s}$ specimen, more adjacent ply constraints in the $[+45/-45/0]_{2s}$ specimen or the assumption that damage is more sensitive to thin laminates are may be the reasons for why the $[+45/-45/0]_{2s}$ specimens show better tensile properties. But theoretically the Young's modulus as well as tensile strength for the two kinds of laminates should be the same.

(a) Axial view of $[0/+45/-45]_s$ specimen

(b) Front view of $[+45/-45/0]_s$ specimen

(c) Axial view of $[+45/-45/0]_{2s}$ specimen

Figure 4. 9 Cross-sectional CT images of specimens during fatigue test

4.5 Summary

The tensile and fatigue behaviors of three kinds of CFRP laminates with stacking sequences of $[0/+45/-45]_s$, $[+45/-45/0]_s$ and $[+45/-45/0]_{2s}$ that have been extensively used by wind blade manufacturers were investigated. CT scanning technology was employed to observe the damage patterns during the fatigue tests. Two models were used to represent the trends of fatigue life for

the two kinds of laminates. And the differences of stiffness degradation were characterized for the three laminates.

- The $[+45/-45/0]_s$ CFRP laminate, in which the 0° oriented layers are inside the laminate, has higher ultimate strength and failure strain than the $[+45/-45/0]_s$ laminate. While the tensile strength and failure strain of the $[+45/-45/0]_{2s}$ laminate are slightly higher than that of $[+45/-45/0]_s$ laminate.
- Young's modulus of the three kinds of laminates is similar from the experiments, which can be explained using Classic Lamination Theory.
- Sigmoidal model can represent the fatigue life better than the Linear model for the three laminates. The fatigue performance of the $[+45/-45/0]_s$ laminate is more excellent than $[0/+45/-45]_s$ laminate. Comparing $[+45/-45/0]_s$ laminate and $[+45/-45/0]_{2s}$ laminate, the predicted fatigue strength at 1 million cycles for the latter is just a little higher than that for the former.
- The $[+45/-45/0]_{2s}$ laminate can resist the more stiffness degradation than the $[+45/-45/0]_s$ laminate, followed by the $[0/+45/-45]_s$ laminate subjected to tension-tension fatigue loading.

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Chapter 5 Static bending behaviors of CFRP composite laminates

In this chapter, the effects of delamination location on the flexural behaviors of CFRP laminates with symmetric layup sequence of $[+45/-45/0]_{2s}$ extensively used by wind blade manufacturers were investigated.

5.1 Introduction

The FRP composites developed for wind turbine blades are subjected to static or fatigue loading in service. And wind turbines are expected to sustain its service for around 20-30 years as required because the installation and replacement is time-consuming and inconvenient. So it is vital to investigate the static and fatigue behaviors of FRP composites associated with failure mechanisms. Many factors, such as manufacturing errors, operational, or maintenance activities, will bring about damage, which could decrease substantially the strength and stiffness of the FRP composites. But some wind turbine should continue to operate even there are slight damage in the blades due to manufacturing errors or in-service operation to save the cost of installation or replacement. On the other hand, wind turbine blades in service are subjected to highly variable loads such as aerodynamic, gravitational and inertial forces. The aerodynamic and gravitational forces will bring about the existence of bending stress in the blades [1]. So it is vital to study the flexural behaviors of FRP composites with existing damage. Delamination is a typical failure mechanism of FRP composites, many researchers investigated the effects of delamination on the mechanical behaviors of FRP composites. But most of the researchers focused on the effects of delamination on the buckling and compression behaviors of FRP composites. Seldom researchers investigated the effects of delamination on the flexural behaviors of FRP composites.

5.2 Experimental methods

Seven laminates with symmetric layup sequences of $[+45/-45/0]_{2s}$ were fabricated, which are extensively used by wind blade manufacturers. One of the seven laminates is normal without any embedded pre-delamination, which is named as NPD. For the other six laminates in which pre-delamination was created by embedding a Teflon layer with thickness of 20 μm at the first (+45/-45), second (-45/0), third (0/+45), ninth (+45/0), tenth (0/-45) and eleventh (-45/+45) interface are named as PD1, PD2, PD3, PD9, PD10 and PD11 respectively. Apparently, the positions of pre-delamination embedded in the PD1, PD2 and PD3 laminates are symmetric about the neutral plane with PD11, PD10 and PD9 laminates, respectively. The seven kinds of laminate configurations are list in Table 3.3 in Chapter 3, in which PD represents the embedded pre-delamination.

The prepreg laminates with dimensions of 190 mm $\times$ 250 mm were cured in an autoclave at 127°C for 60 min under the vacuum pressure of -0.069 MPa (i.e., 0.069 MPa lower than the atmospheric pressure) according to the manufacturer's recommendations. The pressure inside the chamber was maintained at 0.41 MPa during the whole curing process. After curing, the thickness of all the laminates is around 2.13 mm and the weight fraction of fibers is approximately 70%. Then specimens with dimensions of 150 mm $\times$ 12 mm $\times$ 2.13 mm were cut from all the laminates. For the specimens with Teflon layer at the interface, the pre-delamination is square-shaped in the central area of the specimen, which can be shown in Figure 3.3 in Chapter 3.

Three-point bending tests were conducted on eight specimens for each kind of laminate according to ASTM D790. MTS 810 Material Testing System was used in the present research. Based on specimen geometry, the crosshead velocity was set as 2 mm/min and the span length was 75 mm. The span direction is parallel with the fiber direction in the 0 ply. After failure, three of the eight specimens for each kind of laminate were scanned with computed tomography technique to analyze the damage in the specimens qualitatively and

quantitatively. Yxlon X-ray CT system was applied to conduct the computed tomography experiments in this study. The projection images were reconstructed into a three dimensional volume of the specimen by Power Recon, a software designed by Simtech. Subsequently, AVIZO/FIRE software packages were used to segment and quantify the damage (e.g. delamination, transverse crack and fiber failure) in each slice from the surrounding composite. After CT scanning, all the specimens were reloaded to determine the residual strength and modulus. Then JEOL JSM-5600LV Scanning Electron Microscope was used to investigate the fracture surfaces at the interfaces inside the specimens after failure. This will expect to obtain a further understanding of the failure mechanisms.

5.3 Mechanical properties

In the present study, three-point bend tests were performed on eight specimens of each laminate according to ASTM D790. The flexural strain in the outer surface of the specimen is calculated as Equation 5.1.

$$\varepsilon_f = 6Dd/L^2 \quad 5.1$$

Where D, d and L represents the width, thickness and span length of the specimen tested, respectively.

As the deflections of all the specimens are larger than 10% of the support span, so the flexural stress in the outer surface should be calculated with Equation 5.2 according to ASTM D790:

$$\sigma = \left(\frac{3PL}{2bd^2} \right) \left[1 + 6 \left(\frac{D}{L} \right)^2 - 4(d/L)(D/L) \right] \quad 5.2$$

P represents the bending load at a given point and b is the width of the specimen.

There are several ways to define the modulus of FRP composites [2]. In the present research, the modulus of elasticity E for three-point bending tests was calculated according to ASTM D790 standard as Equation 5.3.

$$E = \frac{L^3 P}{4bd^3 \Delta} \quad 5.3$$

It should be noted that the modulus of elasticity was calculated based on the plot data within the initial region that the deflection is less than 1% of support span. So the correction factor is not available for determination of modulus.

The representative stress-strain curves of the seven kinds of specimens are shown in Figure 5.1. After an initial elastic stage, the slopes of all the stress-strain curves are decreasing until failure when the flexural stress drops sharply. This non-linear feature is primarily caused by the viscoelastic nature of resin matrix. Comparing the seven curves, it is clearly that the flexural modulus of the seven laminates is almost the same, while the flexural strength as well as failure flexural strain at failure of the seven laminates is different. The NPD laminates without pre-delamination have the highest flexural strength and can resist the largest flexural strain, which displays that the embedded pre-delamination can reduce the flexural strength and failure flexural strain of the $[+45/-45/0]_{2s}$ CFRP laminates.

Figure 5. 1 Representative stress-strain curves of the seven kinds of specimens

After failure, all specimens were reloaded to obtain the values of residual strength. Figure 5.2 shows the average values of flexural strength of the seven laminates. Specifically, the flexural strengths of PD1, PD2, PD3, PD9, PD10 and PD11 laminates are 0.84, 0.76, 0.70, 0.86, 0.80 and 0.83 of that of NPD

laminates, respectively. This phenomenon demonstrates that the embedded delamination with the same size at different positions or interfaces has different extent of negative effects on the flexural strength of $[+45/-45/0]_{2s}$ CFRP laminates. Specifically, the pre-delamination embedded at the third interface (0/+45) can decrease the flexural strength furthest and the pre-delamination embedded at the ninth interface (+45/0) has the least negative effects. So it can be concluded that the flexural strength of the $[+45/-45/0]_{2s}$ CFRP laminates are the most sensitive to the delamination at the third interface (0/+45) and the least sensitive to that at the ninth interface (+45/0).

Figure 5. 2 Flexural strength of the seven kinds of specimens

In the present research, the modulus of elasticity E for three-point bending tests was measured as the slope of the curve at the initial elastic stage from origin point to the point when midspan deflection is 0.5 mm. Figure 5.3 compares the modulus of elasticity of the seven laminates at loading. The flexural strengths of PD1, PD2, PD3, PD9, PD10 and PD11 laminates are 0.99, 1.01, 1.04, 1.05, 1.05 and 1.04 of that of NPD laminates, respectively. Obviously the initial modulus of the seven laminates at loading is almost the same, which indicates that the embedded pre-delamination with the same size at any of the nine interfaces exerts no effects on the initial flexural modulus of elasticity of the

[+45/-45/0]_{2s} CFRP laminates. This is because delamination is interlaminar damage, the plane of which is perpendicular with the bending direction.

Figure 5. 3 Modulus of the seven kinds of specimens

The effective flexural longitudinal modulus of composite laminates subjected to moment M_x can be determined as Equation 5.4 [3].

$$E_x^f = \frac{12}{h^3 \beta_{11}} \quad 5.4$$

Where h is the thickness of the laminate. β_{11} is the element of compliance matrices in Equation 2.21. x represents longitudinal direction of the specimen in the global coordinate system. As the plies of NPD specimens and the other six kinds of specimens with imbedded relatively small delamination can almost be regarded as being perfectly bonded, so the classic lamination theory can be employed in this case to determine the effective flexural longitudinal modulus with Equation 5.4. The thicknesses for the seven kinds of specimens are almost the same and β_{11} depend on the ply thickness and layup sequences of the laminates. Thus the flexural modulus is almost the same for the seven kinds of specimens.

All the properties displayed above show a tiny standard deviation which represents that the experimental results are highly repeatable and reliable due to the soundly consistent material production and experimental setup.

5.4 Failure mechanisms in the specimens

In order to investigate the damage patterns of the CFRP laminates after failure, the edges of NPD specimens were observed with optical microscope. From the representative edge view shown in Figure 5.4, it is clear that the symmetric layup of the laminate was split into four ply groups. It is interesting that the fiber failure or transverse crack appeared in the 1st ply group and delamination only appeared at the first (+45/-45), second (-45/0) and third (0/+45) interface near the top surface of the specimen.

Previous literature [4-6] shows that the longitudinal compressive strength of unidirectional laminate is about 60%-70% of its tensile strength. One possible reason is that fiber misalignment causes fiber micro-buckling. The carbon fibers are allowed to buckle into the weaker resin in lower plies and finally break under in-plane compressive stress [7]. Because normally the compressive strength of the unidirectional laminates is lower than the tensile strength and the NPD laminate is symmetric configuration, so the plies in compression of the NPD laminate failed first, rather than tension. Figure 5.5 shows the SEM edge views of the NPD specimen after failure, in which Figure 5.5(a) presents the delamination, transverse crack and micro-buckling in the 1st ply group. Figure 5.5(b) gives a clear insight of the microbuckling and transverse crack in the 0 ply in the 1st ply group. It is obvious that the length of the broken fibers (λ) are around 7 fiber diameters long, which is similar to that shown in literature [5, 7]. β represents the inclination angle of the kinking band. According to Figures 5.5(b), (c) and (d), some fibers were buried or exposed in the two adjacent plies associated with ridges and valleys in the matrix, revealing the occurrence of fiber/matrix debonding at the locations of transverse crack [8]. Apart from the fiber/matrix debonding, matrix crack mainly characterized by shear cusps also occurred due to the extensive local yielding of matrix. Thus, the transverse

crack in the +45, -45 and 0 plies in the 1st ply group is characterized by fiber/matrix debonding and matrix crack.

Figure 5. 4 Optical edge view of the NPD specimen after failure

(a) SEM edge view of the 1st ply group (b) microbuckling and transverse crack in 0 ply

(c) transverse crack in -45 ply (d) transverse crack in +45 ply

Figure 5. 5 SEM edge views of the NPD specimen after failure

In order to investigate the phenomenon mentioned above, laminate analysis was conducted based on Classical Laminate Theory [9]. So the mechanical

properties of unidirectional lamina were determined. All the in-plane properties shown in Table 5.1 were obtained by experiments, which are described in Chapter 3. For the out-of-plane properties, the out-of-plane tensile (compressive) strength was assumed as the same as the in-plane transverse tensile (compressive) strength [10, 11]. The poison ratio ν_{23} is obtained referring to the literature [12]. The shear modulus in 2-3 plane G_{23} is calculated as $(E_2/2)*(1+\nu_{23})$ according to the reference book [13]. While the shear strength in 2-3 plane is approximated as the same as in-plane shear strength [11].

Table 5. 1 Mechanical properties of unidirectional lamina

Engineering constant	Value
Longitudinal Young's modulus (E_1)	133980 MPa
Transverse Young's modulus ($E_2=E_3$)	6560 MPa
Poisson ration in 1-2 and 1-3 planes ($\nu_{12}=\nu_{13}$)	0.232
Poisson ration in 2-3 planes (ν_{23})	0.3
Shear modulus in 1-2 and 1-3 planes ($G_{12}=G_{13}$)	5600 MPa
Shear modulus in 2-3 planes (G_{23})	2523 MPa
In-plane longitudinal tensile strength (X_t)	2173 MPa
In-plane longitudinal compressive strength (X_c)	952 MPa
In-plane transverse tensile strength (Y_t)	30 MPa
In-plane transverse compressive strength (Y_c)	124 MPa
Out-of-plane tensile strength (Z_t)	30 MPa
Out-of-plane compressive strength (Z_c)	124 MPa
Shear strength in 1-2 and 1-3 planes ($S_{12}=S_{13}$)	110 MPa
Shear strength in 2-3 plane (S_{23})	110 MPa

* '1' represents the fiber direction, '2' represents the direction perpendicular to the fiber direction. '3' represents the direction perpendicular to the ply. 't' represents tensile and 'c' represents compressive.

During three-point bending test, the specimens are just subjected a bending moment M_x , which can be calculated by Equation 5.5.

$$M_x = PL/4b \quad 5.5$$

Where P is the bending load. L and b represents the span length and width of the specimen, respectively. x and y represents the longitudinal and transverse

direction of the specimen, respectively. As the stiffness matrix $[B]=0$ due to the symmetric layup pattern, the Equation 5.6 can be obtained according to Equation 2.21.

$$\begin{aligned} k_x &= M_x \delta_{11} \\ k_y &= M_x \delta_{12} \\ k_{xy} &= M_x \delta_{16} \end{aligned} \quad 5.6$$

So the strains in the global coordinate system ε_x , ε_y and γ_{xy} can be calculated according to Equation 2.7, which is based on the fact that the strains in the reference plane ε_x^0 , ε_y^0 and γ_{xy}^0 are all zero because of the symmetric layup. Then the strains ε_1 , ε_2 , γ_{12} and stresses σ_1 , σ_2 , τ_{12} in the local coordinate system can be calculated according to strain transformation equation in reference book [13] and Equation 2.11.

Apart from the in-plane stresses σ_1 , σ_2 , τ_{12} , the interlaminar shear stresses τ_{xz} , τ_{yz} in global coordinate system and τ_{13} , τ_{23} in local coordinate system were calculated according to Chapter 2 (note that the transverse shear force $V_x=P/(2b)$). In this case, the interlaminar normal stress σ_z or σ_3 is approximated as zero.

After calculation of the stresses in the local coordinate system, 3D Tsai-Wu failure criteria [14] is employed to analyse the damage patterns in the NPD specimens, which is displayed as Equation.

$$\begin{aligned} F_1 \sigma_1 + F_2 \sigma_2 + F_3 \sigma_3 + F_{11} \sigma_1^2 + F_{22} \sigma_2^2 + F_{33} \sigma_3^2 + 2F_{12} \sigma_1 \sigma_2 + 2F_{23} \sigma_2 \sigma_3 + \\ 2F_{13} \sigma_1 \sigma_3 + F_{44} \tau_{23}^2 + F_{55} \tau_{13}^2 + F_{66} \tau_{12}^2 \geq 1 \end{aligned} \quad 5.7$$

Where the coefficients are determined with Equation 5.8.

$$\begin{aligned} F_1 &= \frac{1}{x_t} - \frac{1}{x_c}; \quad F_2 = \frac{1}{y_t} - \frac{1}{y_c}; \quad F_3 = \frac{1}{z_t} - \frac{1}{z_c}; \quad F_{11} = \frac{1}{x_t x_c}; \quad F_{22} = \frac{1}{y_t y_c} \\ F_{33} &= \frac{1}{z_t z_c}; \quad F_{44} = \frac{1}{s_{23}^2}; \quad F_{55} = \frac{1}{s_{13}^2}; \quad F_{66} = \frac{1}{s_{12}^2} \\ F_{12} &= \frac{-1}{2\sqrt{x_t x_c y_t y_c}}; \quad F_{23} = \frac{-1}{2\sqrt{y_t y_c z_t z_c}}; \quad F_{13} = \frac{-1}{2\sqrt{x_t x_c z_t z_c}} \end{aligned} \quad 5.8$$

Figure 5.6 shows the distribution of Tsai-Wu criterion factor through thickness of the specimen subjected to different extent of load. It is obvious that the highest failure factor exists in the 0 ply of the first ply group, which means

that the failure occurs first in the 0 ply. And it is obvious that the Tsai-Wu criterion exceed 1 when the specimen are subjected to bending load of 150 N. However, it should be noted that Tsai-Wu criterion normally underestimate the failure load of composites but still could provide useful information of the damage occurrence in the composites.

Figure 5. 6 Tsai-Wu criterion failure factor

5.5 3D analysis of damage patterns

However, more details of the damage inside the specimen cannot be obtained by optical observation. Then three out of eight specimens for each kind of laminate after failure were scanned with CT technology. To obtain the entirety of the damage in the specimen, the whole span length of the specimen is scanned. In manipulating the reconstructed and segmented volumes, the surrounding composite was removed with just damage remained to obtain a clear visualisation of the damage patterns. The CT scanning results show that the damage patterns are repeatable for the three specimens of each laminate. Figure 5.7 to 5.13 show the typical damage patterns in NPD, PD1, PD2, PD3, PD9, PD10 and PD11 specimens, respectively. It should be noted that in the figures the loading direction is perpendicular to the delamination plane. To distinguish different damage patterns, fiber failure, transverse crack, delamination and embedded pre-delamination were highlighted in red, yellow,

blue and black, respectively. Light, medium and deep blue colour represents the delamination at the +45/-45, -45/0 and 0/+45 interface of the first ply group, respectively. Light, medium and deep red colour represents the fiber failure in the +45, -45 and 0 ply in the first ply group, respectively. While light, medium and deep yellow colour represents the transverse crack in the +45, -45 and 0 ply in the first ply group, respectively.

Figure 5.7 shows the segmented damage in one of the NPD specimens after failure. Figure 5.7(a) represents the 3D view of all the damage, which displays that delamination occurred at the first (+45/-45), second (-45/0) and third (0/+45) interfaces in the 1st ply group. As the laminate consists of plies with different fiber directions, it can be assumed according to the beam theory and Classical Laminate Theory that the strain varies continuously across the depth of the specimen, while the stress, being a function of both the strain and elastic modulus, exhibits discontinuities through the specimen depth. So there is high interlaminar shear stress at the interface between the two adjacent plies, which bring about delamination in the specimens. According to bending theory, the 1st and 2nd ply group is in compressive stress state. So all the delamination at the three interfaces in the 1st ply group indicates how important the compressive stress state is [15]. Figure 5.7(b) is the magnified 3D view of the segmented damage, indicating that transverse crack and fiber failure appears in +45, -45 and 0 ply. Fiber failure occurred in the 0 ply and cross almost the width of the specimen continuously. In addition, the delamination is interconnected with the transverse crack or fiber failure in the adjacent plies.

Figure 5.8 shows the segmented damage in one of the PD1 specimens. It is obvious that fiber failure and transverse crack also occurs in +45, -45 and 0 plies in the first ply group like the NPD specimen. Delamination also occurs at the +45/-45, -45/0 and 0/+45 interfaces in the first ply group. It is interesting from Figure 5.8(b) that the fiber failure interconnects the two parts of the transverse crack in the +45 ply. And fiber in the 0 ply failed along the propagation direction of the transverse crack in the -45 ply. It can be concluded

that transverse crack provoked the fiber failure nearby because the transvers strength is weaker than the longitudinal strength of FRP composites. Figure 5.9 shows the damage patterns in one of the PD2 specimens, in which no delamination at the +45/-45 interface, transverse crack in +45 ply and fiber failure in +45 ply as well as -45 ply occurs, compared with NPD specimens. So the transvers crack in -45 ply is the only damage above the -45/0 interface, at which the pre-delamination were embedded. Figure 5.10 shows the damage patterns in one of the PD3 specimens, in which no fiber failure in +45 ply as well as -45 ply occurs, compared with NPD specimens.

As for the specimens with pre-delamination embedded at the interfaces in the tensile stress state, Figure 5.11 and Figure 5.12 displays the segmented damage in one of the PD9 and PD10 specimens, respectively. It is obvious that the damage patterns and locations are the same like NPD specimens. Moreover, there is even no delamination formed at the interface with embedded pre-delamination. While for the damage in one of the PD11 specimens shown in Figure 5.13, delamination formed from the embedded pre-delamination at the -45/+45 interface and transverse crack occurred in the +45 ply, apart from the other damage occurred in the 1st ply group like NPD specimens. This phenomenon indicates the effects of pre-delamination at this interface on the outer ply, which will be discussed in the next sections.

(a) 3D view of all the damage

(b) Magnified 3D view of the damage without delamination at -45/0 interface

Figure 5. 7 The segmented damage in NPD specimen

(a) 3D view of all the damage (b) Magnified 3D view of the damage without pre-delamination

Figure 5. 8 The segmented damage in PD1 specimen

(a) 3D view of all the damage (b) Magnified 3D view of the damage without pre-delamination

Figure 5. 9 The segmented damage in PD2 specimen

(a) 3D view of all the damage (b) Magnified 3D view of the damage without pre-delamination

Figure 5. 10 The segmented damage in PD3 specimen

(a) 3D view of all the damage (b) Magnified 3D view of the damage without pre-delamination

Figure 5. 11 The segmented damage in PD9 specimen

(a) 3D view of all the damage (b) Magnified 3D view of the damage without pre-delamination

Figure 5. 12 The segmented damage in PD10 specimen

(a) 3D view of all the damage (b) Magnified 3D view of the damage without pre-delamination)

Figure 5. 13 The segmented damage in PD11 specimen

In summary, most of the transverse crack, fiber failure and delamination occurred throughout thickness in the 1st ply group of the $[+45/-45/0]_{2s}$ CFRP laminates after flexural failure. But the damage locations can be affected by the embedded pre-delamination at the respective interface. The damage locations in PD2, PD3 and PD11 specimens are different with NPD specimens due to the effects of embedded pre-delamination. Although the damage locations in PD1, PD9 and PD10 specimens are the same like NPD specimens, the embedded pre-

delamination affected the damage areas, which will be discussed in the next section. Besides, multidirectional laminates in-ply and inter-laminar damage modes might interact with each other and this must always be taken into consideration, in order to achieve reliable predictions of the ongoing fracture mechanism. In addition, all the damage can be split into two types as damage whose plane is perpendicular and parallel to the loading direction. According to Figures 5.7 to 5.13, the plane of all the delamination and the transverse crack in 0 ply is perpendicular to the loading direction. While the plane of almost all the transverse crack and fiber failure in +45, -45 and 0 plies excluding the transverse crack in 0 ply is parallel to the loading direction.

5.6 Quantitative analysis of damage

The areas of delamination, transverse crack and fiber failure in the 1st ply group in the seven kinds of specimens were calculated from the CT scanning results using AVIZO/FIRE software packages. G. P. McCombe [16] used micro-X-ray computed tomography to quantify the damage volume in the quasi-isotropic CFRP laminates after impact test because all the damage measured is inside the laminates. But in the present research, the opening displacement of the delamination is arbitrary and random. So the investigation of damage areas is more reasonable than that of damage volumes.

Figure 5.14 shows the average values of the delamination areas without pre-delamination at the three interfaces in the 1st ply group of the seven kinds of specimens. It should be noted that the area of the pre-delamination was not accounted for calculation. The results show that, in all the seven kinds of specimens, the delamination at the third interface (0/+45) was the largest among all delamination at the three interfaces. This attributes to the higher interlaminar shear stress. The delamination area at the first (+45/-45) interface is the smallest among all the three interfaces for the NPD, PD2, PD3, PD9, PD10 and PD11 specimens, especially PD2 specimens in which no delamination occurred at the first (+45/-45) interface. For the PD1 specimens, the delamination area at the first interface (+45/-45) is slightly larger than that at the second (-45/0)

interface, which might be because the pre-delamination at the first interface (+45/-45) would lead to more delamination at this interface.

Figure 5. 14 Areas of formed delamination in the seven kinds of specimens

The columns with error bars in Figure 5.15 compare the average values of the damage whose plane is parallel to the loading direction, which includes all the transverse crack and fiber failure excluding the transverse crack in 0 ply. According to the values of initial flexural modulus at loading and reloading, the reduction ratios of flexural modulus for every kind of specimens were calculated, which are shown as the scatter data points. It is obvious that the reduction ratio of the flexural modulus is almost positively correlated with the areas of parallel damage. As section 3.1 mentioned, the pre-delamination exerts no effects on the initial flexural modulus of elasticity of the $[+45/-45/0]_{2s}$ CFRP laminates. It might be concluded that the more parallel damage produced in the specimens, the more the flexural modulus would reduce.

Figure 5. 15 Areas of parallel damage and reduction ratios of flexural modulus in the seven kinds of specimens

The columns with error bars in Figure 5.16 compare the average values of the all the formed damage. According to the values of flexural strength at loading and reloading, the reduction ratios of flexural strength for every kind of specimens were calculated, which are shown as the scatter data points. It is obvious that the reduction ratio of the flexural strength is almost positively correlated with the area of all the damage. But exceptionally, the reduction ratio of flexural strength of PD11 specimens is higher than that of PD2 specimens, even PD11 specimens have less formed damage. This exception is probably because PD11 specimens are the only kind of specimens in which damage occurred in the 4th ply group. So the damage formed in the specimens during loading process will exert negative effect on the flexural strength at reloading. The more damage produced in the specimens, the more the flexural strength would reduce.

Figure 5. 16 Areas of perpendicular damage and reduction ratios of flexural strength in the seven kinds of specimens

5.7 Stress analysis with FEA modelling

An FEA analysis on the bending behaviors of CFRP composite laminates was conducted. For this purpose, virtual crack closure technique (VCCT) was used to investigate the effects of embedded pre-delamination on the stress distributions in the specimens subjected to bending load. VCCT employs linear elastic fracture mechanics (LEFM) based on the strain energy release rate (SERR) of the crack tip deformations, and compares the SERR to interlaminar fracture toughness [17]. Figure 5.17 shows the configuration of VCCT. For pure mode I interlaminar fracture behavior, the nodes 2 and 5 will start to release when,

$$\frac{1}{2} \frac{v_{1,6} F_{v,2,5}}{bd} = G_I \geq G_{IC} \quad 5.9$$

Where G_I and G_{IC} are mode I energy release rate and the interlaminar toughness; b and d are the width and length of the element at the crack front; $F_{v,2,5}$ is the vertical force between nodes 2 and 5, and $v_{1,6}$ is the vertical displacement between nodes 1 and 6. Assuming that the crack closure is governed by linear behaviour, the energy to close the crack (and, thus, the

energy to open the crack) is calculated from the previous equation. Mode II and III can be treated similarly.

Figure 5. 17 Configuration of VCCT

For mixed mode crack propagation, the equivalent strain energy release rate G_{equiv} is to be calculated at a node, and G_{equivC} is the critical equivalent strain energy release rate calculated based on the user-specified mode-mix criterion. The crack-tip node will debond when the ratio of G_{equiv} to G_{equivC} reaches 1. In this section, the Benzeggagh-Kenane (BK) criterion was used to compute G_{equivC} , which is shown in Equation .

$$G_{equivC} = G_{IC} + (G_{IIC} - G_{IC}) \left(\frac{G_{II} + G_{III}}{G_I + G_{II} + G_{III}} \right)^\eta \quad 5.10$$

Where η is obtained by matching the failure criterion with material response when plotted on the mixed mode diagram, which is usually set as 1.75 in the thesis [17]. G_I , G_{II} and G_{III} represent the strain energy release rates (SERR) under the pure mode I, II and III, respectively. The subscript 'C' represents the critical strain energy release rate, namely the interlaminar fracture toughness under the corresponding pure modes.

The mode I interlaminar fracture toughness G_{IC} and mode II interlaminar fracture toughness G_{IIC} at the 0/0, 0/45 and 45/-45 interfaces was obtained by experiments shown in Section 3.2. While the value of mode III interlaminar fracture toughness G_{IIIC} was regarded as the same with mode II interlaminar fracture toughness like almost all the researchers does [18, 19], which are shown in Table 5.2.

Table 5. 2 Interlaminar fracture toughness of the CFRP laminates

Engineering constant	Value (N/mm)
G_{IC} (0/0 interface)	0.1651
G_{IC} (0/45 interface)	0.1679
G_{IC} (45/-45 interface)	0.1517
G_{IIC} (0/0 interface)	0.7724
G_{IIC} (0/45 interface)	0.6958
G_{IIC} (45/-45 interface)	0.7043
G_{IIIC} (0/0 interface)	0.7724
G_{IIIC} (0/45 interface)	0.6958
G_{IIIC} (45/-45 interface)	0.7043

The commercial software ABAQUS was employed for the simulation. Figure 5.18 illustrates the FEA model, in which the three rollers with diameter of 10 mm were considered as rigid body. The dimensions of the specimen accord with the experiments, which can be shown in Table 3.2. The 8-node quadrilateral continuum shell element SC8R was selected for the specimen. There are 12 elements along the width of the specimen. The critical central region of the model (64 mm by length) was meshed with much finer elements (mesh size of 0.2 mm along the longitudinal direction). While the each ply of the specimens was represented by one element along the thickness direction. The in-ply properties shown in Table 5.1 were input into the model. Based on the experimental results obtained from double cantilever beam (DCB) and end notched flexure (ENF) tests, the mode I, mode II and mode III interlaminar fracture toughness were set as 0.1651 N/mm, 0.7724 N/mm and 0.7724 N/mm, respectively.

Figure 5. 18 Schematic of the whole FEA model

Figure 5.19 shows the front view of the central region of the FEA model for PD1 specimen. Plies were bonded at the interfaces except the interface between

the +45 ply and -45 ply in the first ply group, at which just the central region (12 mm by length) was not bonded. All the red dots shown in Figure 5.19 represent the nodes that were bonded. For the other five kinds of specimens with embedded pre-delamination, the central regions at the according interfaces were also not bonded.

Figure 5. 19 Front view of the central region of the FEA model for PD1 specimen

To obtain the effects of the embedded pre-delamination on the stress distribution in the $[+45/-45/0]_{2s}$ specimens, all the seven kinds of specimens were loaded to 150 N (around the midpoint of the loading to failure) in the numerical model. Figure 5.20 shows the Tsai-Wu criterion failure factor distribution in the seven kinds of specimens. It is obvious that the Tsai-Wu failure criterion failure factor in the first ply group for all the seven kinds of specimens are relatively high comparing with the other plies. Also due to the contact pressure from the loading roller, most damage exists in the first ply group in all the specimens. It should also be noted that the embedded pre-delamination at the first interface increases the Tsai-Wu criterion failure factor in the +45 ply (first ply group) and the pre-delamination at the eleventh interface increases the Tsai-Wu criterion failure factor in the +45 ply (fourth ply group) to a large extent. This accords to the observation that the damage occurs in the +45 ply (fourth ply group) in the PD11 specimen, which is shown in Figure 5.13.

Figure 5. 20 Tsai-Wu criterion failure factor

The embedded pre-delamination exerts little effects on the stress distribution in the plies that near the interface at which the pre-delamination was embedded according to the numerical results. So the Figure 5.21 just illustrates the maximum absolute value of in-plane normal stress σ_1 in the plies adjacent to the interfaces with pre-delamination. Specifically, Figure 5.21(a) shows the effects of embedded pre-delamination at the first, second and third interface on the normal stress σ_1 (compression) in the adjacent plies. While Figure 5.21(b) shows the effects of embedded pre-delamination at the ninth, tenth and eleventh interface on the normal stress σ_1 (tension) in the adjacent plies. Comparing with the σ_1 in NPD specimens, it can be concluded that the embedded pre-delamination can increase the normal stress σ_1 in the adjacent plies that near the midplane and decrease the normal stress σ_1 in the adjacent plies that near the surface of the $[+45/-45/0]_{2s}$ laminates subjected to bending load. It is also interesting to see that the embedded pre-delamination at the second interface decrease the normal stress σ_1 in the +45 and -45 plies in the first ply group to a relatively low level, which may explains why no fiber failure occurs in these two plies in the PD2 specimens (Figure 5.9).

Figure 5. 21 Maximum absolute value of in-plane normal stress σ_1 in the plies adjacent to the interface with pre-delamination (a: compressive state, b: tensile state)

5.8 Fracture surfaces at different interfaces

To get a further understanding of the damage patterns, the fracture surfaces at the first (+45/-45), second (-45/0) and third (0/+45) interfaces in the 1st ply group were observed with SEM, as the delamination at the three interfaces is the predominant reason for the failure of the tested material. Figures 5.22 to 5.24 are the SEM images which display the typical fracture surfaces at the three significant interfaces obtained from the specimens after failure.

In Figure 5.22(a) and 5.22(b), it is obvious that most fibers on +45 ply are exposed, which lead to matrix valley on the -45 ply. This phenomenon indicates that fiber/matrix debonding is the dominant mechanism for crack initiation. Ridge and valley markings observed in the micrograph are recognised as a characteristic of combination of peel and shear failures during matrix fracture [8]. The crack initiates along fiber/matrix interface rather than inside the matrix because the interfacial debonding takes the least resistance along the fiber/matrix interface [8, 20]. This phenomenon is different with that observed by P. Naghipour [21], who found that resin adhered on the fibers in CF/PEEK laminates subjected to quasi-static and cyclic mix mode bending loads. As can be seen in Figure 5.22(b), the shear cusps oriented perpendicular to the matrix valley formed on the fracture surface due to the extensive local yielding of matrix. Morris [22] indicated that the shear cusps is caused by local bending of

the fracture surface just behind the crack tip. The bending degree of the upper section above this interface is different with the lower section below this interface, which will cause shear stress between the upper surface and the lower surface at the interface. It is interesting to note from Figure 5.22(a) that there are many strip-shaped epoxy adhered on the fibers in the +45 ply along -45 direction. And there are large amounts of fine epoxy debris scattered on the 0 ply, which is probably caused by sliding abrasion and large number of broken fibers during the bending process.

Figure 5. 22 Fracture surfaces at the +45/-45 interface in the 1st ply group

Figure 5.23 represents the fracture surfaces of the two adjacent plies at the -45/0 interface in the 1st ply group. Obviously the fibers were buried or exposed in the two adjacent plies associated with valleys in the matrix. Shear cusps formed on both plies, which are oriented perpendicular to the matrix valley and travel through the whole distance between two adjacent valleys.

(a) -45 ply

(b) 0 ply

Figure 5. 23 Fracture surfaces at the -45/0 interface in the 1st ply group

Figure 5.24 shows the fracture surfaces of the two adjacent plies at the 0/+45 interface. Similar to the phenomenon at the +45/-45 and -45/0 interfaces, exposed fiber, matrix valleys and shear cusps appear extensively on the two plies at the 0/+45 interface. In addition, it is interesting that fiber failure occurred at this interface, which is probably due to the presence of 0 ply and high interlaminar shear stress at the 0/+45 interface.

(a) 0 ply

(b) +45 ply

Figure 5. 24 Fracture surfaces at the 0/+45 interface in the 1st ply group

In summary, the fracture surfaces at the three interfaces are tough with exposed fiber, matrix valleys and shear cusps. Shear cusps are oriented to the matrix valleys and travel through the whole distance between the two matrix valleys. However, fracture surfaces of the three interfaces exhibit some other different features in addition to their similarities mentioned above. This is caused by the different oriented plies and stress state in the specimen subjected to bending loads.

5.9 Summary

Static three-point bending tests were conducted on CFRP laminates with layup sequence of [+45/-45/0]_{2s}, which has been extensively considered by wind turbine blade manufacturers. The CFRP laminates without artificial pre-delamination and with pre-delamination embedded at the first (+45/-45), second

(-45/0), third (0/+45), ninth (+45/0), tenth (0/-45) and eleventh (-45/+45) interface is named as NPD, PD1, PD2, PD3, PD9, PD10 and PD11, respectively.

The following conclusions can be drawn from the present research:

- The embedded delamination with the same size at different positions or interfaces has different extent of negative effects on the flexural strength of [+45/-45/0]_{2s} CFRP laminates. Specifically, the flexural strengths of PD1, PD2, PD3, PD9, PD10 and PD11 laminates are 0.84, 0.76, 0.70, 0.86, 0.80 and 0.83 of that of NPD laminates, respectively. And the embedded pre-delamination with the same size at any of the nine interfaces exerts no effects on the initial flexural modulus of elasticity of the [+45/-45/0]_{2s} CFRP laminates.
- Most of the transverse crack, fiber failure and delamination occurred throughout thickness in the 1st ply group of the [+45/-45/0]_{2s} CFRP laminates after flexural failure. Apart from fiber micro-buckling, transverse crack also occurs in the first ply group, which is characterized by fiber/matrix debonding and matrix crack. But the damage locations can be affected by the embedded pre-delamination at the respective interface. The occurrence of the damage in the specimen can be predicted by analysing the distribution of stresses in the [45/-45/0]_{2s} specimen. And 3D Tsai-Wu failure criterion is suitable to predict the damage occurrence of the specimen.
- The numerical model with VCCT shows that the embedded pre-delamination at the first interface increases the Tsai-Wu criterion failure factor in the +45 ply (first ply group) and the pre-delamination at the eleventh interface increases the Tsai-Wu criterion failure factor in the +45 ply (fourth ply group) to a large extent. And the embedded pre-delamination can increase the normal stress σ_1 in the adjacent plies that near the midplane and decrease the normal stress σ_1 in the adjacent plies that near the surface of the [+45/-45/0]_{2s} laminates subjected to bending load.
- In all the seven kinds of specimens, the delamination at the third interface (0/+45) was the largest and propagated the fastest among all delamination at the three interfaces. The delamination area at the first (+45/-45)

interface is the smallest among all the three interfaces for the NPD, PD2, PD3, PD9, PD10 and PD11 specimens.

- All the damage can be split into two types as damage whose plane is perpendicular and parallel to the loading direction. The plane of all the delamination and the transverse crack in 0 ply is perpendicular to the loading direction. While the plane of almost all the transverse crack and fiber failure in +45, -45 and 0 plies excluding the transverse crack in 0 ply is parallel to the loading direction. The reduction ratio of the initial flexural modulus is positively correlated with the areas of parallel damage and the reduction ratio of the flexural strength is almost positively correlated with the area of all the formed damage.
- The transverse crack in the +45, -45 and 0 plies in the 1st ply group is characterized by fiber/matrix crack and matrix crack. The fracture surfaces at the three interfaces are tough with exposed fiber, matrix valleys and shear cusps. Shear cusps are oriented to the matrix valleys and travel through the whole distance between the two matrix valleys.

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Chapter 6 Fatigue bending behaviors of CFRP composite laminates

In this chapter, the stiffness degradation and the fatigue life were investigated on the $[+45/-45/0]_{2s}$ CFRP laminate composites subjected to flexural fatigue loads. X-ray microtomography and fractography were then performed to characterise the 3D fatigue damage and the associated failure mechanism.

6.1 Introduction

Fatigue life/strength results of fiber reinforced polymer (FRP) composites are usually variable because of the inhomogeneous and anisotropic features of FRP composites, even under carefully controlled testing conditions. So the fatigue life could be predicted based on the analysis of the dispersed fatigue results with statistical methods. Global damage in CFRP composites subjected to fatigue loads is usually associated with stiffness degradation. Stiffness can reflect the laminate's condition and resistance capacity to fatigue loads, which can be helpful for life prediction or ranking residual strength data. While the stiffness degradation levels can change depending on composite components, loading type, loading level and layup sequences [1]. While for displacement-controlled fatigue tests, the most widely used fatigue failure criterion is when the stiffness falls to a specified level. As there is no constant standard of defining fatigue failure criterion regarding stiffness degradation, a greater understanding of stiffness degradation for a CFRP composite laminate subjected to different fatigue loads is worthwhile. In addition, the fatigue failure mechanisms of FRP composites are complex and generally in the forms of delamination, debonding, matrix crack, and fiber failure [2]. But it is an unresolved problem on which is the major mechanism responsible for fatigue failure of FRP composites [3-5]. So it is necessary to observe the damage progression patterns during fatigue tests and determine how the accumulated damage lead to properties degradation of the FRP composites. Among sorts of

non-destructive techniques, Computed Tomography (CT) technology has been proven to be a powerful tool for characterizing the damage of materials in recent years. Many researchers have used CT scanning technology to investigate the failure mechanisms of fiber FRP composites, which proved its strong practicability in this area. The use of X-ray CT technology associated with reconstruction and visualization software for analysing qualitatively and quantitatively the damage progression in CFRP laminates subjected to flexural fatigue loads has not yet been reported.

6.2 Experimental methods

The carbon fiber reinforced polymer laminates were fabricated using the L-930HT (Cytec Solvay Group, USA) flame retardant epoxy carbon preregs in the unidirectional form. The prepreg laminates with the symmetric layup sequence $[+45/-45/0]_{2s}$, which are extensively used in wind turbine blade structures, were cured in an autoclave at 127 °C for 60 min under the vacuum pressure of -0.069 MPa (i.e., 0.069 MPa lower than the atmospheric pressure). The pressure in the chamber was maintained at 0.41 MPa during the entire curing process. The laminates after curing were approximately 2.13 mm in depth (thickness). The rectangular specimens (150 mm×12 mm) for the flexural tests were machined from the cured CFRP laminates using the water jet cutter.

The flexural experiments were conducted in an in-house three-point bending fixture which was clamped to the MTS 810 (MTS Systems Corp., USA) servo-hydraulic universal testing machine by the wedge grips. Four specimens were subjected to the static bending tests (ASTM D790) with the crosshead velocity 2 mm min⁻¹ and the support span length 75 mm to determine the flexural properties. Displacement-controlled flexural fatigue tests with the sinusoidal waveform (ASTM D7774) were performed under five different cyclic deflection levels (CDL). The CDL normalised by the static failure deflection (SFD) was $\gamma_c = \text{CDL}/\text{SFD} = 0.58, 0.65, 0.71, 0.84$ and 0.91. Eight specimens were repeated in the flexural fatigue tests under each CDL. The ratio of the minimum and maximum displacement was constant at $R = 0.1$ for all the

fatigue experiments. The frequency was limited to 2 Hz given the large midspan deflection. The stiffness was measured and monitored in the fatigue tests.

To characterise the damage, the flexural fatigue test on a CFRP laminate specimen subjected to the normalised CDL $\gamma_c = 0.84$ was interrupted after the specific cycle. The midspan portion of the specimen was then scanned in the x-ray microtomography at 55 kV and 46 μA [14, 15, 21]. A set of 720 projections were recorded as the specimen was rotated through 360°. The 3D image was reconstructed from the projections using an in-house algorithm that enhances the reconstruction quality of planar objects [22, 23]. The internal damage including the fiber failure, matrix cracking and delamination was visualised in the AVIZO/FIRE software. The ImageJ software was used to quantify the projected area of the delamination in the specimen.

After fatigue testing under each CDL, the interlaminar fracture surfaces of the laminate specimens were examined in the SEM to explore the fatigue failure mechanism.

6.3 Static bending behavior

The static flexural response of the CFRP laminate was quantitatively characterised by the measured histories of both the bending load P and the midspan deflection D . As the deflection of the specimen exceeds 10% of the support span, the flexural stress at the midpoint of the outer surface was approximated with a correction factor (ASTM D790), which can be shown as Equation 5.2. The support span, the width and the depth (thickness) of the CFRP laminate are 75 mm, 12 mm and 2.13 mm, respectively. The flexural strain at the midpoint was calculated based on the Equation 5.1.

Figure 6.1 shows the representative flexural stress–strain curve of the CFRP laminate. After the initial linear elastic stage, the stress–strain response is nonlinear with the decreasing slope prior to the static failure deflection SFD at which the stress drops abruptly. The nonlinear feature is primarily due to the viscoelastic nature of the epoxy resin matrix. All the specimens were reloaded to measure the residual modulus and strength (Figure 6.1). The typical flexural

stress–strain curve for reloading also consists of the initial linear stage and the subsequent nonlinear portion before the flexural stress reaches the peak. However, compared to the response for the first loading, the flexural stress is smaller under reloading while the strain at the reloading failure is slightly larger, probably due to the contribution of residual strains. The flexural stress then gradually decreases with oscillation until the sharp drop of stress.

Figure 6. 1 The representative flexural stress–strain curves of CFRP laminates subjected static three-point bending and subsequent reloading

There are several methods to define the modulus of laminate composites subjected to the static three-point bending load [24]. The initial modulus E_0 is defined as the ratio of the flexural stress to the corresponding strain within the initial elastic limit (Fig. 1); thus the E_0 can be calculated:

$$E_0 = \frac{L^3}{4bd^3} m \quad 6.1$$

where m is the slope of the initial linear portion of the load P versus deflection D curve.

Table 1 summarises the flexural properties of CFRP laminates subjected to static three-point bending and subsequent reloading. The good repeatability indicates the consistent material fabrication, specimen preparation and experimental setup. Both the residual modulus and the residual strength are lower than the flexural properties. Table 6.1 also suggests that after failure the laminate can bear approximately 62% of the flexural strength with the residual modulus reduced by nearly 21% of the initial modulus.

Table 6. 1 The flexural properties of CFRP laminates subjected to static three-point bending and subsequent reloading

Initial modulus (GPa)	Flexural strength (MPa)	Flexural failure deflection (mm)	Residual modulus (GPa)	Residual strength (MPa)
28.1±0.8	872±24	14.4±0.5	22.2±2.9	542±41

6.4 Stiffness degradation under flexural fatigue

The stiffness degradation was monitored in the flexural fatigue tests to assess the fatigue resistance in the CFRP laminate. A convenient method to reflect the stiffness under fatigue is the fatigue secant modulus E_n which is defined by dividing the maximum stress with the corresponding strain in the n th cycle. In the present displacement-controlled flexural fatigue test, the maximum flexural deflection is constant under each normalised CDL γ_c , therefore the fatigue secant modulus is proportional to the measured bending load.

Figure 6.2 demonstrates the typical normalised stiffness as a function of the number of fatigue cycles under the five different CDL. Note that the stiffness degradation curves are similar with a very small deviation among the eight laminate specimens for each CDL. The fatigue secant modulus E_n was normalised with the modulus E_1 in the first cycle instead of the static initial modulus E_0 , because the secant modulus E_1 under each CDL usually differs from the E_0 thanks to the nonlinearity of the laminate (Fig. 1).

Figure 6. 2 The variation of the normalised stiffness with the number of cycles under different normalised CDL (γ_c)

The stiffness decreases with the increasing number of fatigue cycles prior to the final failure (Figure 6.2). The rate of stiffness degradation is also significantly affected by the applied CDL. In the laminate subjected to displacement-controlled flexural fatigue loads, the stiffness degradation process can be divided into three stages as follows.

Stage I: a rapid reduction of stiffness by 1–10% occurs within the first several (<10) cycles as a result of micro-cracking in the matrix which may be caused by the presence of voids or defects. Under higher fatigue CDL, the stiffness degrades with a faster rate in the laminate, and more reduction of stiffness is observed. However, this is different from the findings in the literature for force-controlled tension-compression fatigue tests of CFRP laminates in which the stiffness decreases faster under the lower stress level [25].

Stage II: the stiffness continues to degrade with a very low and steady rate due to the evolution of matrix cracks. The effect of the applied CDL on the degradation rate seems to be insignificant. This stage defines the major life of fatigue of the laminate prior to the rapid reduction of stiffness.

Stage III: the abrupt stiffness degradation represents the severe loss of resistance to flexural fatigue loads. This is caused by the formation of large delamination which is prone to interconnect with the fiber fracture and matrix cracks.

6.5 Flexural fatigue life

In the flexural fatigue test, the fatigue failure can be defined at the cycle when the stiffness degrades by a specific percentage. On the assumption that a relative reduction in cyclic stiffness during flexural fatigue tests leads to the same percentage reduction in the static flexural modulus, the fatigue life N is equal to the number of cycles in which the fatigue stiffness decreases by 21%.

The two-parameter Weibull analysis was performed to statistically characterise the flexural fatigue life N of CFRP laminates subjected to three-point bending at a specific normalised CDL γ_c . The cumulative distribution function $F(N)$ is defined:

$$F(N) = 1 - \exp\left(-\left(\frac{N}{\beta}\right)^\alpha\right) \quad 6.2$$

where α is the shape parameter (Weibull slope) and β is the scale parameter (characteristic life). The graphic method using the Weibull plot is applied to determine the parameters α and β . Equation 6.2 can be expressed in the following natural logarithm form:

$$\ln(-\ln(1 - F(N))) = \alpha \ln(N) - \alpha \ln(\beta) \quad 6.3$$

To plot a graph for Equation 6.3, the fatigue lives under a specific γ_c are rearranged in the ascending order. The empirical cumulative distribution function $F(N_i)$ for the i th fatigue life N_i is determined by the failure probability P_f for this series of lives using Bernard's approximation (or called Bernard's estimator) [6]:

$$F(N_i) = P_f = \frac{i - 0.3}{k + 0.4} \quad 6.4$$

where $k = 8$ is the number of specimens tested under the γ_c .

As shown in Figure 6.3, the expression $\ln(-\ln(1-F(N)))$ was plotted against the corresponding natural logarithm of fatigue life $\ln(N)$ for each γ_c as measured in the tests. The data was fitted using linear regression analysis to calculate the shape parameter α and the scale parameter β as listed in Table 6.2. A good correlation coefficient R^2 was achieved for each γ_c , thus indicating that the Weibull distribution model can reasonably characterise the statistical nature of the fatigue life of the CFRP laminate. The slope of the line represents the shape parameter, while the characteristic life corresponds to the life at the failure probability $P_f = F(N) = 0.632$ (i.e., $1-1/e$).

Figure 6. 3 The flexural fatigue life distribution of CFRP laminates under different normalised CDL (γ_c) fitted using the Weibull model

Table 6. 2 The Weibull parameters for the flexural fatigue life distribution of CFRP laminates under different normalised CDL (γ_c)

$\gamma_c = \text{CDL/SFD}$	Shape parameter, α	Scale parameter, β	Correlation coefficient
0.91	0.8051	55	0.9276
0.84	0.7811	382	0.9466
0.71	0.5086	3264	0.9372
0.65	0.7095	47437	0.9239
0.58	1.3184	210030	0.8626

To further validate the Weibull analysis, the Kolmogorov-Smirnov goodness-of-fit statistic D was calculated as follows:

$$D = \max_{i=1}^k \left(\left| \frac{i}{k} - F(N_i) \right| \right) \quad 6.5$$

where N_i is the i th fatigue failure life in the ascending ordered lives, and $F(N_i)$ denotes the theoretic cumulative distribution as in Equation 6.2. As listed in Table 6.3, the D value for each γ_c was compared to the critical value D_c as defined in the Kolmogorov-Smirnov table [26]. Since the D is less than the $D_c = 0.454$ at a 5% level of significance, the two-parameter Weibull analysis can be suitable to predict the life at flexural fatigue failure of the CFRP laminate.

Table 6.3 Kolmogorov-Smirnov goodness-of-fit test results for Weibull analysis of the flexural fatigue life distribution of CFRP laminates under different normalised CDL (γ_c)

$\gamma_c = \text{CDL/SFD}$	Goodness-of-fit statistic, D	Critical value, D_c	$D < D_c?$
0.91	0.189	0.454	Yes
0.84	0.151	0.454	Yes
0.71	0.178	0.454	Yes
0.65	0.212	0.454	Yes
0.58	0.139	0.454	Yes

The failure probability P_f is an important parameter to evaluate the fatigue behavior of laminates. Substituting $P_f = F(N)$ into Equation 6.4, the fatigue life can be predicted as a function of the failure probability using the shape and scale parameters (α and β) for each normalised CDL γ_c :

$$N = \exp \left(\frac{\ln(-\ln(1 - P_f)) + \alpha \ln(\beta)}{\alpha} \right) \quad 6.6$$

The median fatigue lives ($P_f = 0.5$), the characteristic lives ($P_f = 0.632$) and the fatigue lives for the failure probability $P_f = 0.95$ were thus calculated for all the five γ_c (refer to Figure 6.4).

Figure 6. 4 The predicted fatigue life at three failure probabilities under different normalised CDL (γ_c) and the normalised γ_c versus fatigue life curves fitted using the Sigmoidal model

For a specific failure probability P_f , the fatigue life N degrades with the increasing γ_c . A Sigmoidal model was used to quantify the relation between the normalised CDL γ_c and the common logarithm of life $\log(N)$:

$$\gamma_c = \frac{A_1 - A_2}{1 + \exp((\log(N) - N_0)/\Delta N)} + A_2 \quad 6.7$$

where A_1 , A_2 , N_0 and ΔN are the constants. The γ_c was regressed against the $\log(N)$ for the $P_f = 0.5, 0.632$, and 0.95 . The statistical correlation coefficient R^2 for each P_f is >0.98 , thus implying that the Sigmoidal model can well represent the variation of flexural fatigue life with the applied deflection level. The Sigmoidal model has been extensively applied to describe the growth mode in biological science, and has also been employed to predict the fatigue life of 25° braided composite laminates [24].

6.6 Characterization and quantification of flexural fatigue damage

Figure 6.5 illustrates the optical microscopic observation on the external edge surface of the CFRP laminate after 250 cycles in the flexural fatigue test under the normalised CDL $\gamma_c = 0.84$. The damage on the surface occurs near the

midspan portion of the specimen in the 1st ply group which is in the compressive stress state. The damaged portion was scanned in x-ray microtomography to reveal the details of the damage within the specimen (refer to Figure 6.6). In the μ XT image, the surrounding composite material was removed to better visualise the various damage features. Moreover, the depth of the reconstructed volume was stretched with the in-house algorithm such that the spatial resolution in the depth direction was approximately three times better than that in the length (and width) direction, with the voxel size 8 versus 28 (24) μm (Figure 6.6) [22, 23].

Figure 6. 5 Optical image of the CFRP laminate after 250 cycles in the flexural fatigue test under the normalised CDL $\gamma_c = 0.84$

As shown in Figure 6.6, the damage in the CFRP laminate after 250 flexural fatigue cycles under the $\gamma_c = 0.84$ comprises transverse cracks (yellow), delamination (blue) and fiber failure (red). The transverse cracks form in the -45 and 0 plies of the 1st ply group along the corresponding fiber direction. Note that the transverse crack includes matrix cracking and matrix/fiber debonding, both of which cannot be differentiated in the μ XT image due to the limited resolution and the small fiber diameter. The large delamination occurs at the -45/0 and 0/+45 interfaces while the small delamination at the +45/-45 interface (Figure 6.6(b)). The large delamination extends through the entire width; and the delamination tip is irregular. According to the beam theory and the classic laminate theory, in the laminate that consists of plies with different fiber directions, the strain varies continuously across the depth of the specimen; but

the stress, being a function of both the strain and the elastic modulus, exhibits discontinuities through the specimen depth. The high interlaminar shear stress thus arises at the interface between two adjacent plies, and results in the delamination in the specimen. The compressive stress in the 1st ply group gives rise to fiber failure (kinking) across the entire thickness and width (Figure 6.6(a) and (c)).

Figure 6. 6 X-ray microtomographic observations of the damages in the CFRP laminate after 250 cycles in the flexural fatigue test under the normalised CDL $\gamma_c = 0.84$

After CT scanning at the first stage, the specimen was flexural fatigue tested for another 19750 cycles. Figure 6.7 shows extracted damage in the specimen at this stage, indicating that the three types of damage interconnected to each other. It is evident that the areas of all the delamination expanded and the delamination thickness also increased. transverse crack propagated along the fiber direction in the corresponding ply but there was no new fiber failure occurred at other positions comparing with the previous stage. According to bending theory, the 1st and 2nd ply group is in compressive stress state. So all the delamination at the three interfaces of the 1st ply group indicates how important the compressive stress state is [7].

Figure 6. 7 Segmented damage volumes from the CFRP specimen after 20000 cycles of flexural fatigue

The damage pattern in the specimen after 50000 flexural fatigue cycles is shown in Figure 6.8. In contrast with the second stage, delamination at 0/-45 interface and -45/+45 interface associated with transverse crack in 0 ply of the 4th ply group occurred at this stage except the expansion of the existing damage. After 80000 cycles, transverse cracks formed in -45 ply of the 4th ply group and +45 ply of the 2nd ply group, which can be shown in Figure 6.9.

Figure 6. 8 Segmented damage volumes from the CFRP specimen after 50000 cycles of flexural fatigue

Figure 6. 9 Segmented damage volumes from the CFRP specimen after 10000 cycles of flexural fatigue

The areas of the delamination at +45/-45, -45/0 and 0/+45 interfaces in the 1st ply group were calculated from the plan view of the CT scanning results using Image-J software, which can be reflected in Figure 6.10. The results show that the delamination at 0/+45 interface was the larger and propagated faster comparing with the other delamination at the outer interfaces. The delamination at the -45/0 and 0/+45 interfaces is larger compared to that at the +45/-45 interface because the crack propagation is likely faster in the two interfaces adjacent to the 0 ply in which the fiber fracture leads to the reduced load carrying capacity. The interlaminar shear stress is higher at the 0/+45 interface, thus resulting in the largest delamination among the three interfaces. At the time point of around 50000 cycles, slight delamination occurred at the 0/-45 and -45/+45 interfaces in the 4th ply group, which is in tensile stress state. From 50000 cycles to 100000 cycles, the delamination at the 0/-45 interface in the 4th ply group extended at a considerable speed through the width comparing with all the other delamination that grew in a steady-state manner.

Figure 6. 10 Areas of delamination as a function of flexural fatigue cycles in the flexural fatigue test under the normalised CDL $\gamma_c = 0.84$

Thus, CT scanning associated with reconstruction and visualization software has proven to be an effective way for analysing the damage progression qualitatively and quantitatively in CFRP laminates subjected to flexural fatigue loads.

6.7 Fatigue failure mechanisms

Figure 6.11 shows the typical interlaminar fracture surfaces at the +45/-45, -45/0 and 0/+45 interfaces in the 1st ply group of the CFRP laminate as examined in the SEM after the flexural fatigue tests. Large quantities of fibres are exposed in these fracture surfaces, thus suggesting the matrix/fiber debonding. The fractured epoxy also dominates in the fracture surfaces as demonstrated by the different morphologies in Figure 6.11, such as ridges/valleys, strip-shaped epoxy, shear cusps and abraded epoxy. Therefore, the delamination propagates along the matrix/fiber interface as well as within the epoxy matrix [27]. However, the quantity of exposed fibers seems to be more than the fracture epoxy (Figure 6.11); this implies that the matrix/fiber interface is weaker than the interlaminar epoxy in the CFRP in the present study.

Figure 6. 11 SEM images of the fracture surfaces in the ply interfaces in the 1st and 2nd groups

The SEM examination also reveals the occurrence of fiber fracture in the 0 ply (refer to Figure 6.11(b) and (c)). Shear cups form on the fracture surface as a result of extensive local yielding of the matrix (Figure 6.11(b)). The frictional interaction of two adjacent plies under the cyclic load causes the fine epoxy debris especially in the 0 plies (Figure 6.11(b) and (c)) and the abrasion of the epoxy matrix, e.g., on the surface of the +45 ply as in Figure 6.11(c).

6.8 Summary

The stiffness degradation and the fatigue life were investigated on the $[+45/-45/0]_{2s}$ CFRP laminate composites subjected to flexural fatigue loads. X-ray microtomography and fractography were then performed to characterise the 3D fatigue damage and the associated failure mechanism. The conclusions were drawn as follows.

- The flexural fatigue life at failure can be quantitatively determined by the cycle when the tracked secant stiffness degrades by 21%. The two-parameter Weibull distribution well represents the statistical nature of the fatigue life data of the CFRP laminate. Combined with the Sigmoidal model, the Weibull analysis method reliably predicts the fatigue life as a function of the applied deflection level.
- The 3D damage in the CFRP laminate consists of matrix cracks, matrix/fiber debonding, fiber failure and delamination. The delamination at the interfaces in the 1st ply group is the predominant failure mode for the flexural fatigue. The delamination area at the $-45/0$ and $0/+45$ interfaces both adjacent to the 0 ply is larger than the area at the $+45/-45$ interface. Matrix/fiber debonding is the primary mechanism for delamination propagation, while matrix cracking the secondary mechanism.
- The fracture surfaces at $+45/-45$, $-45/0$ and $0/+45$ interfaces in the 1st ply group in the failed specimen are tough with fiber exposure and matrix fracture. But fibers exposed more than epoxy matrix, which indicates that the strength of epoxy-fiber interface is weaker than that of interlaminar epoxy in this case. However, fracture surfaces of the three interfaces exhibit some other different features due to the different oriented plies and stress state under the cyclic bending loads.

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Chapter 7 Mode I interlaminar fracture behaviors of CFRP composite laminates

In this chapter, DCB tests were conducted on three kinds of CFRP laminates $[0_{10}/0_{10}]$, $[0_8/-45/0//+45/0_9]$ and $[0_8/-45/45// -45/45/0_8]$ to investigate the effects of fiber orientation of adjacent plies on the mode I interlaminar fracture behaviors.

7.1 Introduction

As delamination is one of the most typical failure mechanisms in FRP composites, which will lead to the degradation in stiffness, strength and fatigue life [1]. And delamination is only kind of failure mechanism during interlaminar fracture tests. Numerous researchers have investigated the interlaminar fracture toughness of FRP composites, but mostly unidirectional FRP composites. However, the FRP laminates extensively employed in industrial applications are multidirectional (MD) and delamination propagation is more common between plies with different fiber directions in real composite structures [2]. Not much research has investigated the interlaminar toughness at the interface between plies with different fiber directions in multidirectional composites in comparison with unidirectional composites. The lack of research on this subject is mainly due to the difficulty that crack jumping may occur when delamination propagates, which might lead to inaccurate toughness values [3, 4]. But the selection on the layup of the FRP laminates for interlaminar fracture tests need to be considered to inhibit crack jumping. Due to the difficulty of determining the interlaminar fracture toughness at the interface between plies with different fiber directions in multidirectional composites, most of current engineering practices regard the interlaminar toughness at any interface of FRP composite structures as the same with that at 0//0 interface in unidirectional laminates. And most researchers use cohesive finite elements to model all the interfaces in multidirectional FRP laminates with interlaminar toughness at 0//0 interface in unidirectional FRP laminates [5-8]. However,

Zhao Libin indicated that the parameters of cohesive fracture model applicable for the unidirectional laminates may not be necessarily suitable for multidirectional composite laminates [9]. More importantly, it is a controversial issue on whether the interlaminar fracture toughness depends on the fiber directions at the interface. So it is very important to investigate the interlaminar toughness values for interfaces between plies with different directions for more accurate prediction of delamination development in real FRP composite structures [10, 11]. And more and more research on the interlaminar toughness of multidirectional FRP laminates should be carried out for optimized design of real FRP composite structures. In addition, the delamination simulation in multidirectional FRP laminates may be influenced by the curved delamination front [12]. Many researchers modelled the crack front shape of the FRP composites under DCB tests [4, 13-15]. But seldom researchers observed the crack front shape by experiments. In most literatures focused on interlaminar toughness of multidirectional specimens, the crack front is regarded as straight for the measurement of toughness. This assumption may not be reasonable, especially for multidirectional specimens [13, 14, 16].

7.2 Experimental methods

To satisfy the criteria presented in Section 3.1, three kinds of layup sequences: $[0_{10}/0_{10}]$, $[0_8/-45/0//+45/0_9]$ and $[0_8/-45/45// -45/45/0_8]$ were considered, where the symbol “//” represents the initial delamination introduced during the fabrication process. According to ASTM D5528, 170 mm×20 mm×3.7 mm specimens with 70 mm insert length were cut from the fabricated laminates for DCB tests to determine the Mode I interlaminar fracture behaviors of the three kinds of laminates. This insert length corresponds to an initial delamination length of 50 mm plus the length required for bonding the hinges. Five specimens were tested for each kind of layup sequence. Yxlon X-ray Computed Tomography system was applied to conduct the tomography experiments to observe the 3D crack front shapes in the specimens. Subsequently, AVIZO/FIRE software packages were used to segment the damage from the

surrounding composite to determine the crack front shape. Apart from X-ray CT scanning, JEOL JSM-5600LV Scanning Electron Microscope (SEM) was employed to observe the interlaminar fracture surfaces inside the specimen after DCB tests. All the details on the material fabrication, DCB test and microstructure is described in Chapter 3.

7.3 3D crack front shapes

In most literatures focused on interlaminar toughness of multidirectional specimens, the crack front is regarded as straight. This assumption may not be reasonable, especially for multidirectional specimens, as the crack front shape depends on the fiber orientations of the adjacent plies [13, 14, 16]. Figure 7.1 shows the different crack front shapes in the 0//0, 0//45 and 45//–45 specimens. In Figure 7.1, it is obvious that the real crack front in the specimen with 0//0 interface is thumbnail shaped. For the specimen with 0//45 and 45//–45 interface, the real crack fronts do not show thumb-nailed shape but display random shape from sample to sample. This is probably because the crack propagation is not that steady like the 0//0 specimen, which can be seen from the load-displacement curves shown in the subsequent sections.

When calculating the toughness, the difference between the real crack front and assumed crack front from the specimen edge was account for. Specifically, equivalent crack growth length was used for the calculation of initial fracture toughness. However, it shows not much difference (less than 4%) with the toughness calculated with the crack growth length obtained from the specimen edge.

Figure 7. 1 Crack front shapes of specimens (a: 0//0 sample 1; b: 0//0 sample 2; c:

0//45 sample 1; d: 0//45 sample 2; e: 45//45 sample 1; f: 45//45 sample 2)

7.4 Mode I interlaminar fracture toughness

The standard data reduction schemes to determine the mode I interlaminar fracture toughness G_{IC} , developed for unidirectional laminates, are usually applied for multidirectional ones [17-20]. One of the data reduction methods is modified beam theory (MBT) [21]. But the MBT method will overestimate G_I because the beam is not perfectly built-in (rotation may occur at the crack front). One way to correct this is introducing the correction factor F , which compensates for the reduction of the lever arm due to the eccentricity of the loading system and the rotation of the specimen [22]. The equation for calculation of G_I is as follow:

$$G_I = \frac{3FP\delta}{2ba} \quad 7.1$$

$$F = 1 - \theta_1 \left(\frac{\delta}{a}\right)^2 - \theta_2 \left(\frac{l\delta}{a^2}\right) \quad 7.2$$

$$l = 4 + \frac{h}{4} \quad 7.3$$

P and δ represent the applied load and the load point displacement. b represents the width of DCB specimen. a is the crack length. θ_1 and θ_2 was set as 0.3 and 1.5 for DCB test according to [23]. The distance between the rotation centre of hinge and the neutral axis of the arm is represented by l , which can be shown in Figure 3.9(a). h represents the thickness of DCB specimen.

Variable values of the interlaminar toughness G_{IC} at different crack growth length a^* for all the samples are shown in Figure 7.2. The crack growth length a^* is the distance between the insert and the crack front. The difference of the two crack lengths observed from both edges is less than 2 mm for all the tests, which coincides with the ASTM D5528. This can also be seen from the CT scanning results shown in Figure 7.1. The interlaminar toughness G_{IC} for the three interfaces increases as the crack grows at the initial stage until it arrives at a constant value, which characterizes the R-curve feature. The interlaminar toughness G_{IC} with R-curve feature was fitted by an approximate expression introduced by Foote et al. [24], which is shown as Equation 7.4:

$$G_{IC}(a^*) = G_{init} - 2(G_{prop} - G_{init})\left(\frac{\langle L_{bz} - a^* \rangle}{L_{bz}} - 1\right) - (G_{prop} - G_{init})\left(\frac{\langle L_{bz} - a^* \rangle}{L_{bz}} - 1\right)^2 \quad 7.4$$

Where $\langle \rangle$ represents the Macauley operator defined as $\langle x \rangle = 1/2(x + |x|)$. G_{init} is the interlaminar toughness when the crack starts to occur from the end of the insert. G_{prop} is the propagated fracture toughness, which exhibits as the constant value at the final stage. While L_{bz} represents the length of the fiber bridging zone. The gap between G_{prop} and G_{init} is the interlaminar toughness caused by fiber bridging. The fiber bridging zone length L_{bz} is defined as the distance between the start point of fiber bridging to the crack tip, which equals to the crack growth length when the toughness arrives at G_{prop} as shown in Figure 7.2.

There are three definitions for G_{init} according to ASTM D5528. For the present research, G_{IC} is determined using the load and displacement at the point when the load-displacement curve starts to be nonlinear (NL). Equation 7.4 is employed to fit a curve on the data of all the specimens corresponding to each kind of interface. Then G_{prop} and L_{bz} for each kind of interface are obtained from the fitting analysis. The statistical correlation coefficient R square value for the fitting of 0//0, 0//45 and 45//45 interface is 0.95, 0.97 and 0.86 respectively. This indicates that Equation 7.4 is reasonable to represent the fracture toughness G_{IC} with the crack growth length. This equation is different with the quadratic equation introduced by M.M. Shokrieh [25] and the power function introduced by BF Sørensen and TK Jacobsen [26] which only fit the features of the G_{IC} before the plateau region. As Figure 7.2 shows, the G_{IC} in specimens with 0//0 interface ramps with crack growth length at the lowest rate at the initial stage and reaches G_{prop} after the longest crack growth at the final stage. While the G_{IC} in specimens with 45//45 interface ramps at the highest rate at the initial stage and reaches G_{prop} after the shortest crack growth. As S.M. Spearing [27] and M.M. Shokrieh [25] reported, there is balance between fiber failure and pulling out so that the fiber bridging zone length is almost constant after the G_{IC} arrives at G_{prop} .

Figure 7. 2 Mode I interlaminar fracture toughness G_{IC} vs. crack growth length a^*

Figure 7.3 shows the front views of the three kinds of DCB specimens at the final stage of the crack propagation and the lengths of fiber bridging were measured. Based on DCB tests and fitting analysis, the G_{init} , G_{prop} and L_{bz} for each interface is listed in Table 7.1. The results show that the G_{init} at the three kinds of interfaces are almost the same, which displays its independence on layer sequences. While fiber bridging zone length is the longest in specimens with 0//0 interface because it is slower to reach to the balance between fiber failure and pulling out when the adjacent fiber orientation is smaller relative to the longitudinal direction [9]. While the G_{prop} in specimens with interface 45//45 is highest, followed by those with 0//45 interface and then 0//0 interface, which may be caused by the effect of bending-bending coupling and bending-twisting [9, 28].

Figure 7. 3 Front views of the DCB specimens during tests (a: 0//0 specimen, b: 0//45 specimen; c: 45//-45 specimen)

Table 7. 1 Results of average mode I interlaminar toughness and fiber bridging length values

Interface	G_{init} , N/mm($\pm$ SDV)	G_{prop} , N/mm	L_{bz} , mm	
			Fitted	Experiment
0//0	0.1651($\pm$ 0.011)	0.7579	66.65	$\approx$ 62
0//45	0.1679($\pm$ 0.004)	0.8674	46.03	$\approx$ 45
45//-45	0.1517($\pm$ 0.011)	0.9467	32.46	$\approx$ 40

7.5 Numerical mode for mode I interlaminar fracture behaviors

Delamination simulations have been conducted by many researchers, which proved that the cohesive zone model (CZM) is considered as the best numerical method. However, seldom researchers have investigated the interlaminar fracture behaviors in multidirectional composite laminates with variable fracture toughness. In this section, numerical model for the delamination propagation with variable fracture toughness is introduced.

Firstly, a set of suitable parameters to characterizing the CZM should be determined. In this thesis a quadratic stress failure criterion [29] and Benzeggagh-Kenane (BK) law [30] are employed to investigate mode I

interlaminar fracture behaviors, respectively. The quadratic stress failure criterion can be expressed as:

$$\left\{ \frac{\max(\sigma_I, 0)}{\sigma_I^0} \right\}^2 + \left\{ \frac{\sigma_{II}}{\sigma_{II}^0} \right\}^2 + \left\{ \frac{\sigma_{III}}{\sigma_{III}^0} \right\}^2 = 1 \quad 7.5$$

Where σ_i and σ_i^0 ($i=I, II, III$) are stress components and corresponding interfacial strength for pure mode I, II and III, respectively. The values of σ_I^0 , σ_{II}^0 and σ_{III}^0 are all set as 20 MPa according to the literatures [31]. All the stress components contribute proportionally to the initial crack, except for the interlaminar compressive normal stress, which does not react on the initial damage. The BK law for crack propagation evaluation is shown in Equation 5.10.

Initial interface stiffness is parameter with no physical meanings for undamaged cohesive elements to maintain a stiff connection between two neighboring layers. The range between 10^3 and 10^6 N/mm³ for initial interface stiffness is feasible [32, 33]. In this thesis the initial interface stiffness is set as 10^6 N/mm³.

The commercial software ABAQUS was employed for the simulation. The numerical model is shown in Figure 7.4, in which the dimensions of the DCB specimen accord with Table 3.2. There are 15, 15 and 300 solid elements (C3D8I) along the longitudinal direction of the Arm 1, Arm 2 and Arm 3, respectively. And a layer of cohesive elements (COH3D8) with thickness of 0.01 mm with the same element length with Arm 3 was embedded at the midplane of Arm 3 to simulate the crack initiation and propagation. There are 41 elements along the width for the solid elements and cohesive elements. The outer eight plies for the two halves of the DCB specimen are all 0 plies, which were simulated with just one element along the thickness direction to reduce the computation time.

Figure 7. 4 Schematic of the numerical model (a: the whole model; b: front view of the model; c: front view of the beginning point of the cohesive layer)

To implement the variable fracture toughness in the cohesive elements, a user-subroutine USDFLD in Abaqus was written and embedded into the numerical model. The flow chart of the simulation is shown in Figure 7.5. After every increment, the user-subroutine USDFLD is called at the integrated point of the cohesive element, with which the fracture toughness is defined on the cohesive element according to the fitted curve for G_{IC} vs a^* . Then the stress components of every element are determined based on the finite element analysis. Subsequently, damage onset is judged based on the quadratic stress failure criterion for the cohesive element. If the criterion is satisfied, the strain energy release rate G is calculated. Otherwise, the displacement load is increased. After that, the final failure is judged based on the BK law. If the final failure criterion is satisfied for the cohesive element, then the failed element will disappear and crack grows by one element size. Otherwise, the displacement load is increased. This loop will continue until the displacement load get to the setting value.

Figure 7. 5 Flow chart of the simulation with variable fracture toughness

Figure 7.6, 7.7 and 7.8 show all the experimental and numerical load-displacement curves for 0//0, 0//45 and 45//-45 specimens, respectively. Figure 7.6(a), 7.7(a) and 7.8(a) show the load-displacement curves in the initial loading process for the initial crack. And Figure 7.6(b), 7.7(b) and 7.8(c) show the load-displacement curves in the subsequent reloading process after unloading. From all the experimental curves, it is obvious that the curves in the initial loading process for the three laminate configurations indicate a linear pattern up to an abrupt drop (for 0//0 and 0//45) or nonlinearity point (for 45//-45) when the crack initiates from the insert. The critical values of the displacement and load at the nonlinearity point shown in Table 7.2 indicates that the critical load decreases from specimens with 0//0 interface to those with 0//45 interface and then to 45//-45 interface, but the critical displacements are nearly the same for the three laminate configurations. The values of initial stiffness in the linear part are also displayed in Table 7.2, which show that the specimens with 0//0

interface exhibit the highest structural stiffness and those with 45//45 interface have the lowest stiffness. This is because the increasing non-zero plies and their deviation degree from the longitudinal direction will decrease the structural stiffness of the specimen [9]. For the reloading process, all the curves exhibit gradual nonlinear rise pattern after the initial linear stage until the load reaches to the peak. As shown in Table 7.2, the specimens with 45//45 interface can resist the highest load in crack propagation, followed on the specimens with 0//45 interface and then those with 0//0 interface, which shows inverse relationship with the critical load at NL. The curves corresponding to 0//0 interface is relatively smooth after the peak, while the other two kinds of specimens exhibit a fluctuated behaviors due to variations in local material properties, such as the misalignment of the fibers [9, 34]. The load fluctuations for the specimens with 0//45 interface and 45//45 interface predicating unsteady crack propagation attributes to the occurrence of crack deviation and translaminar fracture, which will be shown in the subsequent sections.

The load-displacement responses from the numerical model with constant and variable fracture toughness are shown in the figures. Obviously, the numerical model with constant fracture toughness which has been widely used by other researchers cannot represent the load-displacements response of the three types of DCB specimens. While the numerical model with variable fracture toughness is in good agreements with the experimental results. It should be noted that no load drop occurs at the NL point on the numerical curves, which is slightly different with the experimental results. This is due to the fact that the numerical model represents relatively ideal phenomenon. In fact the crack initiation and propagation shows more steady pattern in the numerical model. However, the critical displacement and load are both in good agreements with the experimental results, which can be seen obviously from Table 7.2. In the reloading process, the numerical curves ramp smoothly to the peak and descend gradually until the end of reloading. This pattern is different with the experimental results, which show great fluctuations especially for the

45//45 specimens. However, the difference between the numerical and experimental values of the peak load during reloading process is within reasonable range.

Figure 7. 6 Load-displacement curves for 0//0 specimen (a: initial loading; b: reloading)

Figure 7. 7 Load-displacement curves for 0//45 specimen (a: initial loading; b: reloading)

Figure 7. 8 Load-displacement curves for 0//45 specimen (a: initial loading; b: reloading)

Table 7. 2 Results of average initial stiffness and peak load for DCB specimens
($\pm$ standard deviation)

Interface	Critical values at NL		Initial stiffness (N/mm)	Peak load (N)
	Displacement (mm)	Load (N)		
0//0 (Experiment)	3.15($\pm$ 0.32)	37.74($\pm$ 0.76)	12.26($\pm$ 1.07)	48.20($\pm$ 2.05)
0//0 (FEM, variable Gic)	3.06	37.79	12.89	47.79
0//45 (Experiment)	3.24($\pm$ 0.12)	30.06($\pm$ 1.74)	9.30($\pm$ 0.58)	51.67($\pm$ 1.83)
0//45 (FEM, variable Gic)	3.59	31.92	9.31	45.69
45//-45 (Experiment)	3.54($\pm$ 0.40)	24.73($\pm$ 2.58)	6.74($\pm$ 1.00)	55.13($\pm$ 3.65)
45//-45 (FEM, variable Gic)	3.92	26.28	6.97	46.60

The cohesive elements will stretch and disappear when they fail. Figure 7.9 shows the crack front shape for the three interfaces during crack propagation. It can be seen that all the crack front shapes of the three interfaces are thumb-nailed shape, but the shapes represents with different extent of nonuniformity, which can also be seen from the contour of the normal stress (S_{33}). It is obvious that the crack front shape for 45//-45 interface is the most nonuniform and 0//0 interface shows the least nonuniform. The thumb-nailed shape of the crack front accords with the CT scanning results. But the crack front shapes obtained from simulation and CT scanning are different for 0//45 as well as 45//-45 interface. T. A. Sebaey [4] concluded that the crack front shapes for different interfaces are all thumb-nailed based on his numerical investigation, which ignore the fiber bridging effect. While in this thesis the CT scanning results show that the crack front shape for 0//0 interface is thumb-nailed shape but the shapes for 0//45 and 45//-45 are not. This may be due to the fact that the fiber bridging in the 0//0 specimen is uniform along the width of the specimen but not uniform in the 0//45 and 45//-45 specimen, which can also be shown in M. M. Shokrieh's work [35]. And how the fiber bridging phenomenon affects the crack front shape could be investigated in the future.

Figure 7.9 Crack front shapes in simulation

7.6 Fracture surfaces at different interfaces

The optical images shown in Figure 7.10 display the crack initiation and propagation patterns observed from the edges of the specimens after DCB tests. Figures 7.10(a), 7.10(c) and 7.10(e) represent the crack initiation patterns near the insert in specimens with 0//0, 0//45 and 45//45 interface, respectively. Figure 7.10(b), 7.10(d) and 7.10(f) display the crack propagation patterns away from the insert in the three kinds of specimens. It is obvious that fiber bridging phenomenon can be observed in the edge photographs. For the specimens with 0//0 interface, the crack initiates and propagates with relatively smooth pattern in comparison with those with 0//45 and 45//45 interface. Comparatively speaking, the crack propagation patterns in specimens corresponding to 45//45 interface are more sensible to exhibit a stair-shape mode than those with 0//45 interface, which is because the 45//45 interface with higher ply mismatched angle exhibits higher interlaminar shear stresses as the arms bend. Although little crack jumping appears in specimen with 0//45 interface and double cracking appears in that with 45//45 interface, the light fiber tearing characterized by crack propagating inside one of the two adjacent layers close to the interface, is still the dominant mechanism, which can also be reasonable for the calculation of the fracture toughness [20].

(a) Edge view near the insert in 0//0 specimen (b) Edge view away from insert in 0//0 specimen

(c) Edge view near the insert in 0//45 specimen (d) Edge view away from insert in 0//45 specimen

(e) Edge view near the insert in 45//45 specimen (f) Edge view away from insert in 45//45 specimen

Figure 7. 10 Edge photographs of DCB specimens

In order to get a further understanding of the crack patterns, the interlaminar fracture surfaces at the interface of the three kinds of specimens after DCB tests were observed with SEM. The SEM samples were taken from the insert and the central area of the tested specimens to investigate crack initiation and propagation patterns, respectively.

Figures 7.11(a-d) show the fracture surfaces at the insert and the central area of the two adjacent layers at the interface in one of the specimens with 0//0 interface. Comparing the crack initiation patterns on the two 0 plies, it is obvious that most fibers on one 0 ply are exposed, which lead to matrix valley

on the opposite 0 ply. This phenomenon indicates that fiber/matrix debonding is the dominant mechanism for crack initiation. The crack initiates along fiber/matrix interface rather than inside the matrix because the interfacial debonding takes the least resistance along the fiber/matrix interface [36, 37]. It is interesting that scarps appear occasionally in Figure 7.11(c), which may give an indication of the direction of local crack growth [38]. In Figure 7.11(b) and 7.11(d), the exposed fibers and matrix valleys alternately appear in both 0 plies at the crack propagation stage. Apart from the fiber/matrix debonding, fiber bridging and matrix crack are also observed in the two plies, which indicates that the crack propagation patterns are diversified and the two adjacent plies interact more severe at the crack propagation stage than the initial stage.

Figures 7.12(a-d) show the fracture surfaces of the 0 and 45 plies at the interface in one of the 0//45 specimens. Obviously the 0 ply exhibits exposed fibers in most of the crack initiation area shown in Figure 7.12(a), which results in a vast number of matrix valleys in the 45 ply. This indicates that the crack would rather initiate from the insert along the 0 fiber/matrix interface than the 45 fiber/matrix interface. At the crack propagation stage, similarly as 0//0 specimen, the exposed fibers and matrix valleys alternately appear in 0 ply as well as 45 ply.

The crack initiation and propagation in the 45//45 specimen shown in Figures 7.13(a-d) exhibits similar patterns as those in 0//0 and 0//45 specimen. Fiber/matrix debonding is also the predominant failure mechanism for the 45//45 specimen under DCB test. However, unlike the 0//0 specimen and 0//45 specimen, a vast number of fine epoxy debris appear on the 45 and -45 plies.

It is also worth to note that, there is no cusps occurred on all these interlaminar fracture surfaces of carbon fiber epoxy composites mentioned above. In contrast, the mode I interlaminar fracture surfaces of carbon fiber PEEK composite have cusps due to the higher amount of matrix plastification [39]. And larger strain to failure will occur in the carbon fiber PEEK composites caused by the yielding of the thermoplastic matrix. So the mode I

interlaminar toughness of the here tested composite with epoxy matrix is smaller than that of the carbon fiber PEEK composites, which can be seen from Table 7.1 and [39, 40].

(a) Crack initiation patterns on 0 ply in 0//0 specimen (b) Crack propagation patterns on 0 ply in 0//0 specimen

(c) Crack initiation patterns on 0 ply in 0//0 specimen (d) Crack propagation patterns on 0 ply in 0//0 specimen

Figure 7. 11 Fracture photographs of 0//0 specimen

(a) Crack initiation patterns on 0 ply in 0//45 specimen (b) Crack propagation patterns on 0 ply in 0//45 specimen

(c) Crack initiation patterns on 45 ply in 0//45 specimen (d) Crack propagation patterns on 45 ply in 0//45 specimen

Figure 7. 12 Fracture photographs of 0//45 specimen

(a) Crack initiation patterns on 45 ply in 45//45 specimen (b) Crack propagation patterns on 45 ply in 45//45 specimen

(c) Crack initiation patterns on -45 ply in 45//45 specimen (d) Crack propagation patterns on -45 ply in 45//45 specimen

Figure 7. 13 Fracture photographs of 45//–45 specimen**7.7 Summary**

DCB tests were conducted on three kinds of CFRP laminates [0₁₀//0₁₀], [0₈/-45/0//+45/0₉] and [0₈/-45/45//–45/45/0₈] to investigate the effects of fiber orientation of adjacent plies on the mode I interlaminar fracture behaviors. The following conclusions can be drawn from the present research:

- CT scans enabled an insight into the real crack fronts of the three kinds of specimens after DCB tests. The real crack front in the specimen with 0//0 interface is thumbnail shaped and the real crack fronts of the specimen with 0//45 and 45//–45 interface do not show thumb-nailed shape but display random shape from sample to sample.
- The interlaminar toughness G_{IC} with R-curve features were fitted by an approximate expression introduced by Foote et al. [24]. The G_{IC} in specimens with 0//0 interface ramps with crack growth length at the lowest rate at the initial stage and reaches G_{prop} after the longest crack growth at the final stage. While the G_{IC} in specimens with 45//–45 interface ramps at the highest rate at the initial stage and reaches G_{prop} after the shortest crack growth. The G_{init} is almost the same for the three kinds of specimens. While the G_{prop} in specimens with interface 0//0, 0//45 and 45//–45 continuously increases.
- The specimens with 0//0 interface exhibit the highest structural stiffness and those with 45//–45 interface have the lowest stiffness. The specimens with 45//–45 interface can resist the highest load in crack propagation, followed on the specimens with 0//45 interface. The load-displacement curve corresponding to 0//0 interface is relatively smooth after the peak, while the other two kinds of specimens exhibit fluctuated behaviors.
- For the specimens with 0//0 interface, the crack initiates and propagates with relatively smooth patterns in comparison with 0//45 and 45//–45 specimens. For the latter two kinds of specimens, the crack near the insert is smoother than the propagated crack away from the insert.
- The numerical model with variable fracture toughness G_{ic} is applicable

to characterize the mode I interlaminar fracture behaviors at 0//0, 0//45 and 45//-45 interfaces in the FRP composites and in good agreements with experiments. In addition, all the crack front shapes of the three interfaces are thumb-nailed in the numerical model. The nonuniformity of the fiber bridging at the 0//45 and 45//-45 interfaces affected the crack front shapes during experiments.

- The morphologies of fracture surfaces by SEM technique indicate that fiber/matrix debonding, matrix crack and fiber bridging is the main crack patterns during DCB tests. At the crack initiation stage, the crack initiates along fiber/matrix interface rather than inside the matrix for the three laminate configurations. While the crack propagation patterns are diversified and the two adjacent plies interact more severe at the crack propagation stage than the initial stage.

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Chapter 8 Conclusions and recommendations

In this chapter, the conclusions from the results on the mechanical behaviors of static tensile, fatigue tensile, static bending, flexural fatigue and mode I interlaminar fracture were displayed.

8.1 Conclusions

This thesis focuses on the investigation of the mechanical behavior and the associated failure mechanisms of FRP composites subjected to different kinds of loads, which will contribute to improving the in-service properties, predicting the properties and response, optimization design of FRP composite structures. Specifically, the conclusions can be divided into four parts as follows.

8.1.1 Static and fatigue tensile behaviors of CFRP composite laminates

The tensile and fatigue behaviors of three kinds of CFRP laminates with stacking sequences of $[0/+45/-45]_s$, $[+45/-45/0]_s$ and $[+45/-45/0]_{2s}$ that have been extensively used by wind blade manufacturers were investigated. CT scanning technology was employed to observe the damage patterns during the fatigue tests. Two models were used to represent the trends of fatigue life for the two kinds of laminates. And the differences of stiffness degradation were characterized for the three laminates.

- The $[+45/-45/0]_s$ CFRP laminate, in which the 0° oriented layers are inside the laminate, has higher ultimate strength and failure strain than the $[+45/-45/0]_s$ laminate. While the tensile strength and failure strain of the $[+45/-45/0]_{2s}$ laminate are slightly higher than that of $[+45/-45/0]_s$ laminate.
- Young's modulus of the three kinds of laminates is similar from the experiments, which can be explained using Classic Lamination Theory.
- Sigmoidal model can represent the fatigue life better than the Linear model for the three laminates. The fatigue performance of the $[+45/-45/0]_s$ laminate is more excellent than $[0/+45/-45]_s$ laminate. Comparing $[+45/-45/0]_s$ laminate and $[+45/-45/0]_{2s}$ laminate, the predicted fatigue strength at 1 million

cycles for the latter is just a little higher than that for the former.

- The $[+45/-45/0]_{2s}$ laminate can resist the more stiffness degradation than the $[+45/-45/0]_s$ laminate, followed by the $[0/+45/-45]_s$ laminate subjected to tension-tension fatigue loading.

8.1.2 Static bending behaviors of CFRP composite laminates

Static three-point bending tests were conducted on CFRP laminates with layup sequence of $[+45/-45/0]_{2s}$, which has been extensively considered by wind turbine blade manufacturers. The CFRP laminates without artificial pre-delamination and with pre-delamination embedded at the first (+45/-45), second (-45/0), third (0/+45), ninth (+45/0), tenth (0/-45) and eleventh (-45/+45) interface is named as NPD, PD1, PD2, PD3, PD9, PD10 and PD11, respectively. The following conclusions can be drawn from the present research:

- The embedded delamination with the same size at different positions or interfaces has different extent of negative effects on the flexural strength of $[+45/-45/0]_{2s}$ CFRP laminates. Specifically, the flexural strengths of PD1, PD2, PD3, PD9, PD10 and PD11 laminates are 0.84, 0.76, 0.70, 0.86, 0.80 and 0.83 of that of NPD laminates, respectively. And the embedded pre-delamination with the same size at any of the nine interfaces exerts no effects on the initial flexural modulus of elasticity of the $[+45/-45/0]_{2s}$ CFRP laminates.
- Most of the transverse crack, fiber failure and delamination occurred throughout thickness in the 1st ply group of the $[+45/-45/0]_{2s}$ CFRP laminates after flexural failure. But the damage locations can be affected by the embedded pre-delamination at the respective interface.
- The numerical model with VCCT shows that the embedded pre-delamination at the first interface increases the Tsai-Wu criterion failure factor in the +45 ply (first ply group) and the pre-delamination at the eleventh interface increases the Tsai-Wu criterion failure factor in the +45 ply (fourth ply group) to a large extent. And the embedded pre-delamination can increase the normal stress σ_1 in the adjacent plies that near the midplane and decrease the normal stress σ_1 in the adjacent plies that near the surface of the $[+45/-45/0]_{2s}$

laminates subjected to bending load.

- In all the seven kinds of specimens, the delamination at the third interface (0/+45) was the largest and propagated the fastest among all delamination at the three interfaces. The delamination area at the first (+45/-45) interface is the smallest among all the three interfaces for the NPD, PD2, PD3, PD9, PD10 and PD11 specimens.
- All the damage can be split into two types as damage whose plane is perpendicular and parallel to the loading direction. The plane of all the delamination and the transverse crack in 0 ply is perpendicular to the loading direction. While the plane of almost all the transverse crack and fiber failure in +45, -45 and 0 plies excluding the transverse crack in 0 ply is parallel to the loading direction. The reduction ratio of the initial flexural modulus is positively correlated with the areas of parallel damage and the reduction ratio of the flexural strength is almost positively correlated with the area of all the formed damage.
- The transverse crack in the +45, -45 and 0 plies in the 1st ply group is characterized by fiber/matrix crack and matrix crack. The fracture surfaces at the three interfaces are tough with exposed fiber, matrix valleys and shear cusps. Shear cusps are oriented to the matrix valleys and travel through the whole distance between the two matrix valleys.

8.1.3 Fatigue bending behaviors of CFRP composite laminates

The stiffness degradation and the fatigue life were investigated on the [+45/-45/0]_{2s} CFRP laminate composites subjected to flexural fatigue loads. X-ray microtomography and fractography were then performed to characterise the 3D fatigue damage and the associated failure mechanism. The conclusions were drawn as follows.

- The flexural fatigue life at failure can be quantitatively determined by the cycle when the tracked secant stiffness degrades by 21%. The two-parameter Weibull distribution well represents the statistical nature of the fatigue life data of the CFRP laminate. Combined with the Sigmoidal model,

the Weibull analysis method reliably predicts the fatigue life as a function of the applied deflection level.

- The 3D damage in the CFRP laminate consists of matrix cracks, matrix/fiber debonding, fiber failure and delamination. The delamination at the interfaces in the 1st ply group is the predominant failure mode for the flexural fatigue. The delamination area at the -45/0 and 0/+45 interfaces both adjacent to the 0 ply is larger than the area at the +45/-45 interface. Matrix/fiber debonding is the primary mechanism for delamination propagation, while matrix cracking is the secondary mechanism.
- The fracture surfaces at +45/-45, -45/0 and 0/+45 interfaces in the 1st ply group in the failed specimen are tough with fiber exposure and matrix fracture. But fibers exposed more than epoxy matrix, which indicates that the strength of epoxy-fiber interface is weaker than that of interlaminar epoxy in this case. However, fracture surfaces of the three interfaces exhibit some other different features due to the different oriented plies and stress state under the cyclic bending loads.

8.1.4 Mode I interlaminar fracture behaviors of CFRP composite laminates

DCB tests were conducted on three kinds of CFRP laminates [0₁₀//0₁₀], [0₈/-45/0//+45/0₉] and [0₈/-45/45//+45/45/0₈] to investigate the effects of fiber orientation of adjacent plies on the mode I interlaminar fracture behaviors. The following conclusions can be drawn from the present research:

- CT scans enabled an insight into the real crack fronts of the three kinds of specimens after DCB tests. The real crack front in the specimen with 0//0 interface is thumbnail shaped and the real crack fronts of the specimen with 0//45 and 45//+45 interface do not show thumb-nailed shape but display random shape from sample to sample.
- The interlaminar toughness G_{IC} with R-curve features were fitted by an approximate expression introduced by Foote et al. [1]. The G_{IC} in specimens with 0//0 interface ramps with crack growth length at the lowest rate at the initial stage and reaches G_{prop} after the longest crack growth at the final stage.

While the G_{ic} in specimens with 45//–45 interface ramps at the highest rate at the initial stage and reaches G_{prop} after the shortest crack growth. The G_{init} is almost the same for the three kinds of specimens. While the G_{prop} in specimens with interface 0//0, 0//45 and 45//–45 continuously increases.

- The specimens with 0//0 interface exhibit the highest structural stiffness and those with 45//–45 interface have the lowest stiffness. The specimens with 45//–45 interface can resist the highest load in crack propagation, followed on the specimens with 0//45 interface. The load-displacement curve corresponding to 0//0 interface is relatively smooth after the peak, while the other two kinds of specimens exhibit fluctuated behaviors.
- For the specimens with 0//0 interface, the crack initiates and propagates with relatively smooth patterns in comparison with 0//45 and 45//–45 specimens. For the latter two kinds of specimens, the crack near the insert is smoother than the propagated crack away from the insert.
- The numerical model with variable fracture toughness G_{ic} is applicable to characterize the mode I interlaminar fracture behaviors at 0//0, 0//45 and 45//–45 interfaces in the FRP composites and in good agreements with experiments. In addition, all the crack front shapes of the three interfaces are thumb-nailed in the numerical model. The nonuniformity of the fiber bridging at the 0//45 and 45//–45 interfaces affected the crack front shapes during experiments.
- The morphologies of fracture surfaces by SEM technique indicate that fiber/matrix debonding, matrix crack and fiber bridging is the main crack patterns during DCB tests. At the crack initiation stage, the crack initiates along fiber/matrix interface rather than inside the matrix for the three laminate configurations. While the crack propagation patterns are diversified and the two adjacent plies interact more severe at the crack propagation stage than the initial stage.

8.2 Recommendations

Although the investigation of mechanical behaviors and failure mechanisms of FRP composites were conducted in this thesis, further research need to be conducted.

Firstly, more detailed insight into damage growth and propagation in the FRP composites subjected to mechanical loading is necessary. For this objective, in-situ loading fixture can be designed and amounted in CT machine to obtain instant three-dimensional insight into the damage propagation in the FRP composites during loading.

Secondly, the effects of embedded pre-delamination on the bending behaviors of FRP composites were illustrated by experiments. And the analysis of stress distribution based on numerical modelling and CLT were conducted to analyse the damage distribution in the FRP specimens subjected to bending load in this thesis. But further numerical modelling and theoretical analysis is necessary.

Thirdly, the numerical model with variable fracture toughness can be employed to analyse the mechanical behaviors of FRP composite structures subjected to other kinds of loads. And how the fiber bridging phenomenon affects the crack front shape in the micro-level can be investigated.

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