Mechanics and Mechanical Engineering
Vol. 22, No. 1 (2018) 171–178

https://doi.org/10.2478/mme-2018-0015

Fracture Toughness Analysis of Delamineted Composites under Variable Temperatures

Meryem El Moufari
Larbi El Bakkali
Modelling and Simulation of Mechanical Systems Laboratory Faculty of Sciences
Abdelmalek Essaadi University, Tetouan, Morocco
elmoufari.meryem@gmail.com

Received (23 September 2017)
Revised (14 October 2017)
Accepted (18 November 2017)

The main goal of the present paper is to approach the modeling of one of the most important and critical failure modes for composite laminates which is known as interlaminar delamination in the aeronautical structures. The analytical model is based on a fracture mechanics approach; it's used to estimate the total mixed mode energy release rate for composite laminates. A finite element simulation has been achieved in combination with the virtual crack closure technique (VCCT) to analyze the effect of temperature on the interlaminar fracture toughness growth of a delaminated carbon/epoxy material, namely IM7/8552 subjected to mechanical loading at variable temperatures. The developed model may serve as the basis for treating different types of thermal and mechanical loading, different stacking sequences and thickness of lamina in order to build safe working conditions for composite laminates.

Keywords: interlaminar fracture, failure, composite laminate, fracture toughness, temperatures, delamination.

1. Introduction

The laminated composites are increasingly used for high-performance structural applications in many aerospace applications such as advanced aircraft fuselage, rocket motor cases, pressure vessels, containment structures, and other components with various shapes and sizes due to their number of advantages over conventional materials. They have exceptional characteristics such as: high specific strength and stiffness, low density, good fatigue performance, resistance to corrosion and high temperatures, ability to create complex shapes. For implementation of composite materials in aviation, the most important feature is their behaviour on dynamic loads and resistance to fatigue [1].
However, Fiber reinforced composites often exhibit complex failure mechanisms as an interaction among various damage modes on both microscopic and macroscopic

scales such as matrix cracking and interlaminar damage modes (Interfacial cracking between layers) or delamination.

This paper is an attempt to characterise delamination, the most frequent failure mode in laminated composite materials and it may cause catastrophic failure in critical aeronautic structures. In the composite materials literature, delamination or interlaminar cracking is generally assumed to take place at the interface between adjacent plies and rather treated as a fracture process between layers, than to consider it, more precisely, as a fracture between constituents or within one of the constituents. The growth of this phenomenon leads to a degradation of stiffness and an eventual failure of the composite structure.

In recent years, several studies have been carried out into the fracture of composites in their different stress modes under static loading, of which modes I and II have attracted more attention. A series of numerical investigations presented in literatures lead to excellent results. These methods are more suitable because of their low cost and time consuming. R. Krueger developed a finite element models using 3D shell elements which demonstrated good accordance with experimental results [2].

The calculation of delamination can be performed using cohesive elements [3, 4], which combine aspects of strength based analysis to predict the onset of damage at the interface and fracture mechanics to predict the propagation of a delamination. Initiation and propagation of delamination studied numerically with using cohesive elements and different constitutive laws lead to excellent results [5].

Over the past two decades, The criteria used to characterize the onset and growth of composite reinforced delamination under mixed-mode loading conditions are those usually established in terms of the components of the energy release rate and fracture toughness. It is assumed that the growth of delamination in composite structures starts when strain energy release rate G under service loads exceeds the fracture energy GC. Wang et al. evaluated strain energy release rates for the damage-tolerance analysis of skin-stiffener interfaces using Finite element analysis in conjunction with the virtual-crack-closure technique (VCCT) [6, 7]. They used a wall offset to move the nodes from the reference surfaces to a coincident location on the interface between the skin and the ?ange. The present study focuses on delamination testing under opening, shearing and mixed-mode loading conditions. A number of approaches have been proposed to develop test specimens with such combined normal and shear stresses on the delamination plane [8, 9]. The pure mode I values for delamination fracture toughness GIC were obtained using a split unidirectional laminate loaded as a double cantilever beam (DCB). Dattaguru et al. [10] calculated mode I and mode II energy release rates of Cracked-Lap Shear (CLS) specimen from a finite element analysis. They showed that the ratio of mode I to mode II energy release rate is strongly affected by the adhesive modulus and the adherent thickness. Mangalgiri and Johnson [11] described a technique for sizing the optimal thickness of adherents for CLS specimen to assure delamination instead of adherent failure.

In the present work, an attempt has been made to investigate the behaviour of the mixed-mode delamination mechanism of Carbone/Epoxy composite material by adopting one of the numerical intelligence concepts that have proved to be useful for various engineering applications. For this purpose, a numerical model has been developed by using a special shell finite element model that guarantees interlaminar shear stress continuity between different oriented layers, at a temperature range of operating conditions for composites in aeronautics. And then, generate mode I and mode II components of mixed-mode fracture toughness.

2. Finite Element methodology end data preparation

In theory, the failure criteria used, in general to predict delamination propagation under mixed-mode loading conditions are usually established in terms of the components of the energy release rate and fracture toughness. It is assumed that when the energy release rate, G, exceeds the critical value, the critical energy release rate G_c, delamination grows.
The expression proposed by Benzeggagh and Kenane [12] for the critical energy release rate G_c is:

$$G_c = G_{Ic} + (G_{IIc} - G_{Ic}) \left(\frac{G_{Shear}}{G_T}\right)^{\eta} \tag{1}$$

where η maintains the shape of the failure locus in the mixed-mode plane and the most accurate failure criterion is the one matching the material response when plotted on this mixed-mode diagram:

$$G_T = G_I + G_{Shear} \quad \text{with} \quad G_{Shear} = G_{II} + G_{III} \tag{2}$$

This criterion is recommended by ASTM standards [13] as the preferred interaction envelope for mixed-mode delamination.
Further, the Finite Element Method is a widely used technique for computing strain energy release rates for linear elastic fracture problems.
The mode I and mode II energy release rate were obtained using the virtual crack closure technique [14, 15]. That is, G I and G II were calculated as follows:

$$\begin{cases} G_I = \lim_{\delta a \to 0} \frac{F_y(\nu_c - \nu_d)}{2\delta a} \\ G_{II} = \lim_{\delta a \to 0} \frac{F_x(u_c - u_d)}{2\delta a} \end{cases} \tag{3}$$

The total energy release rate is:

$$G_T = G_I + G_{II} \tag{4}$$

where, Δa is a crack extension size, F_x and F_y are forces in x- and y-direction. The displacements, u_c (u_d) and v_c (v_d) are the sliding and opening displacements at node "c" (node "d") on the crack faces, respectively illustrated in Fig. 1.

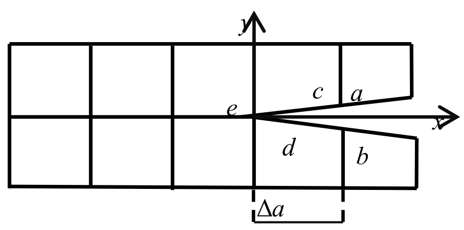

Figure 1 VCCT nodes near the crack tip

In this paper, a finite element modeling with a series of a virtual crack closure technique has been achieved.

The Cracked lap shear (CLS) specimen (Fig. 2) made of prepregs high-performance unidirectional carbon fiber reinforced epoxy Hexcel (IM7/8552) used for fracture tests was constructed using Abaqus Finite Element code [16] in order to determine the critical interlaminar fracture toughness components mode 1 and II of Carbon-Epoxy composite material:

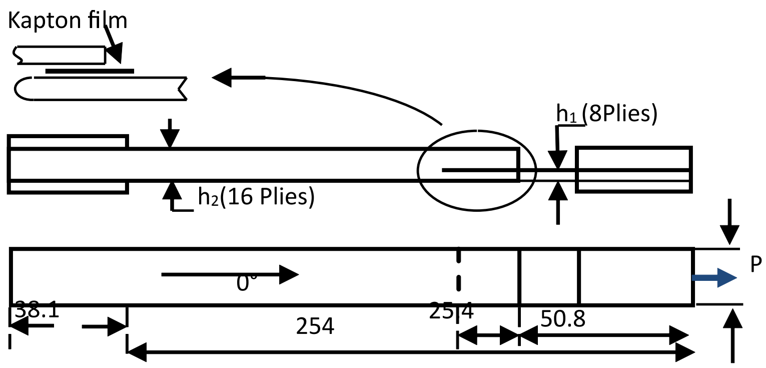

Figure 2 Shape and size of CLS specimen (All dimensions in mm)

The cracked-lap shear (CLS) specimen was originally developed for testing adhesive bonds between metals, and was first used to evaluate mixed-mode fracture toughness of composites.

The compliance of the CLS specimen is given by:

$$C(a) = \frac{1}{EBh_2}\left[2L + a\left(\frac{h_1}{h_2} - 1\right)\right] \tag{5}$$

The total energy release rate is given as:

$$G_T = G_I + G_{II} = \frac{P^2}{2EB^2}\left(\frac{h_1 - h_2}{h_1 h_2}\right) \tag{6}$$

The volume fraction of carbon fiber in the prepreg is 60%. The material properties used are shown in Tab. 1 and Tab. 2, measured in a previous investigation [17, 18, 19], With a nominal ply thickness of 0.0626 mm and the reference stacking sequence considered in the study is [90-0]8 s. The values for tensile and compressive fiber fracture can be obtained from compact tension (CT) and compact compression (CC) tests as proposed by Pinho et al. [20]. According to this formulation, the value depends on the laminate stacking configuration.

3. Results and discussions

The cracked-lap shear (CLS) specimen was modeled numerically in 3-D plane shell with the dimension parameters previously defined in Fig. 2.

Table 1 Elastic ply properties

Volume fraction Vf (%)	Elastic modulus (GPa)				
	E1	E2 = E3	G12 = G13	G23	ν12 = ν13
60	171.42	9.08	5.29	3.98	0.32

Table 2 Ply strength properties

Strengths (MPa)				
X^T	X^C	Y^T	Y^c	S^L
2326.2	1200.1	62.3	200.8	92.3

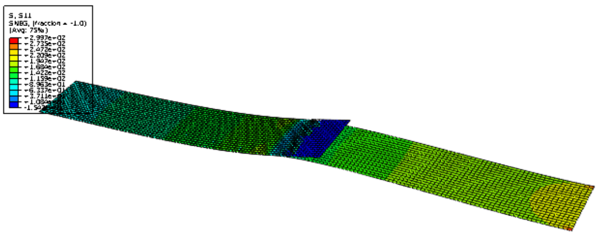

Figure 3 Maximum equivalent stresses distribution

For mechanical analysis first, the total energy release rate was obtained as:

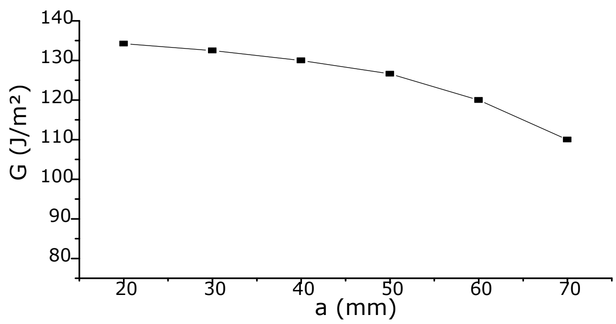

Figure 4 Total energy release rate as function of delamination length

The CLS specimen was loaded in tension by applying load to both the lap, and to the strap and lap part of the specimen at, -60°C, 20°C, 60°C and 80°C. This temperature range has been chosen in view of the fact that it is representative of

the operating temperatures experienced by the composite materials in aeronautical applications.
As a results, The mode I and mode II energy release rates, GI and GII, were calculated under a plane stress condition. The variation of GI and GII for the CLS specimen is shown in Fig. 5.

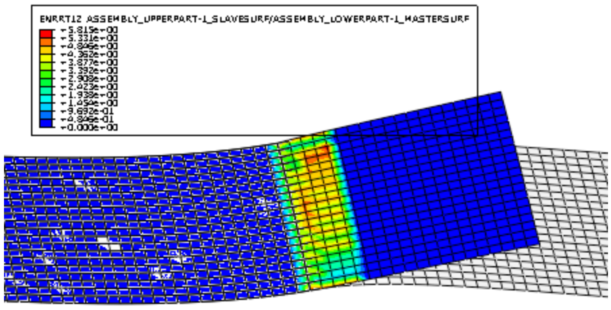

Figure 5 Mode I & Mode II Energy release rate distributions

The interlaminar fracture toughness under Mode I and Mode II, as well as mixed-mode loading, have been investigated for Carbone/Epoxy composite material under variable temperature values.

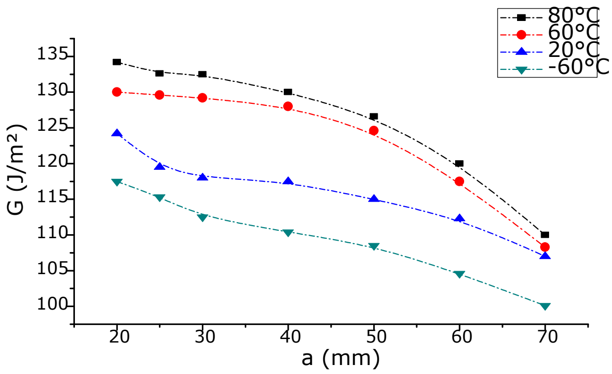

Figure 6 Total Energy release rate as function of delamination length at variable temperatures

It's concluded that the total energy release rate increases with the increasing in temperature values. Fig. 6 illustrates the variation of Total energy release rate as a function of delamination length under variable temperatures. It's well seeing that the major increasing of G is at 80°C and reaches its minimum values at -60° C.

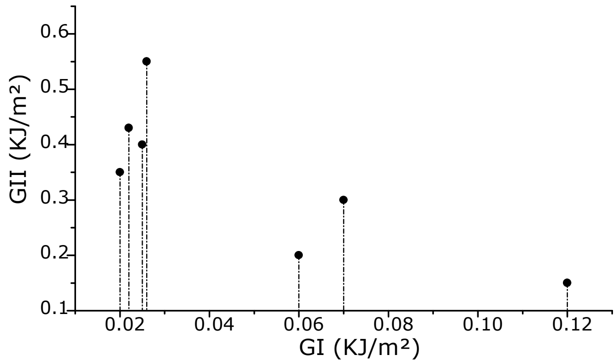

Figure 7 Mixed-Mode fracture Energies envelope

Fig. 7 illustrates the shape of the fracture envelope for Cracked Lap Shear specimen, it explores the interaction between the values of G II and G I in mixed-mode fracture toughness.

4. Conclusions

The objective is to attempt understanding more closely delamination mechanism under mode I, mode II and mixed-mode loading conditions.

Static fracture analysis were conducted for unidirectional CLS specimens to determine the total energy release rate of Carbone/Epoxy composite laminates and to characterize the mixed-mode delamination propagation at variable temperatures. The GI, GII, mode I and mode II energy release rates were calculated from finite element analysis in combination with the virtual closure technique. The total strain energy release rate G is evaluated, and it's concluded that the gradient of the temperature causes an effect of accelerating or retarding the growth of delamination, depending on the loading regime. It was found that a linear fracture envelope may be suitable for a CLS specimen.

References

[1] **Dragan, K., Rašuo, B.**: Review of impact damages modelling in laminated composite aircraft structures, *Tehnicki vjesnik/Technical Gazette*, 20, 3, **2013**.

[2] **Ronald Krueger**.: A shell/3D modelling technique for delamination in composite laminates, In proceedings of the American society for composites, 14th technical conference, *Technomic Publishing*, **1999**.

[3] **Camanho, P. P., Dávila, C. G., de Moura, M. F. S. F.**: Numerical Simulation of Mixed-Mode Progressive Delamination in Composite Materials, *Journal of Composite Materials*, 37, 16, 1415–1438, **2003**.

[4] **Turon, A., Camanho, P. P., Costa, J., Davila, C. G.**: A Damage Model for the Simulation of Delamination in Advanced Composites Under Variable-Mode Loading, *Mechanics of Materials*, 38, 11, 1072–1089, **2006**.

[5] **Krueger, R., Minguet, P. J., O'Brien, T. K.**: Implementation of interlaminar fracture mechanics in design: an overview, Presented at *14th international conference on composite materials (ICCM-14)*, San Diego, **2003**.

[6] **Wang, J. T., Raju, I. S.**: Strain Energy Release Rate Formulae for Skin-Stiffener Debond Modeled with Plate Elements, *Engineering Fracture Mechanics*, 54, 2, 211–228, **1996**.

[7] **Wang, J. T., Raju, I. S., Dávila, C. G., Sleight, D. W.**: Computation of Strain Energy Release Rates for Skin-Stiffener Debonds Modeled with Plate Elements, 34th AIAA/ASME/ASCE/AHS Structures, Structural Dynamics and Materials Conference, *AIAA, Washington, D.C.*, 1680–1692, **1993**.

[8] **Rao, B. N. and Acharya, A. R.**: Evaluation of fracture energy GIc using a double cantilever beam fiber composite specimen, *Engineering Fracture Mechanics*, 51, 317–322, **1995**.

[9] **Alif, N., Carlsson, L. A., Gillespie, Jr., J. W.**: Mode, I, mode II and mixed mode interlaminar fracture of woven fabric carbon/epoxy, *ASTM STP*, 1242, 82–106, **1997**.

[10] **Dattaguru, B., Everett, R. A., Whitcomb Jr., J. D., Johnson, W. S.**: Geometrically nonlinear analysis of adhesively bonded joints, *J. of Engineering Materials and Technology*, 106, 59–65, **1996**.

[11] **Mangalgiri, P. D., Johnson, W. S.**: Preliminary design of cracked lap shear specimen thickness for delamination of interlaminar fracture toughness," *J. of Composite Technology and Research*, 8, 58–60, **1986**.

[12] **Benzeggagh, M. L., Kenane, M.**: Measurement of mixed-mode delamination fracture toughness of unidirectional glass/epoxy composites with mixed-mode bending apparatus, *Compos. Sci. Technol.*, 49, 439–449, **1996**.

[13] Standard Test Method for Mixed Mode I – Mode II Interlaminar Fracture Toughness of Unidirectional Fiber Reinforced polymer Matrix Composites. ASTM Standard D 6671-01, *ASTM Annual Book of Standards*, 15.03, 392–403, **2001**.

[14] **Rybicki, E. F., Kanninen, M. F.**: A finite element calculation of stress intensity factors by modified crack closure integral, *Engineering Fracture Mechanics*, 9, 931–938, **1977**.

[15] **Krueger, R.**: Virtual crack closure technique: history, approach, and applications, *ASME Appl. Mech. Rev.*, 57, 2, 109–43, **1977**.

[16] ABAQUS v.6.10 documentation.

[17] **Camanho, P. P., Davila, C. G., Pinho, S. T., Iannucci, L., Robinson, P.**: Prediction of in situ strengths and matrix cracking in composites under transverse tension and in-plane shear, *Composites, Part A*, 37, 165–176, **2006**.

[18] ASTM D 5528-01.: Standard test method for mode I interlaminar fracture toughness of unidirectional fiber-reinforced polymer matrix composites, ASTM D 5528-01, American Society for Testing and Materials (ASTM), West Conshohocken, PA, USA.

[19] **Blatt, D., Nicholas, T., Grant Jr., A. F.**: Modelling the Crack Growth Rates of a Titanium Matrix Composite under Thermo mechanical Fatigue, *ASTM STP*, 1263, 352, **1996**.

[20] **Maimi, P., Camanho, P. P., Mayugo, J. A., Davila, C. G.**: A continuum damage model for composite laminates: Part II – Computational implementation and validation, *Mechanics of Materials*, 39, 909–919, **2007**.