Order number:

PEOPLE'S DEMOCRATIC REPUBLIC OF ALGERIA
MINISTRY OF HIGHER EDUCATION AND SCIENTIFIC RESEARCH

DJILLALI LIABES UNIVERSITY
FACULTY OF TECHNOLOGY
SIDI BEL ABBES

DOCTORAL THESIS

Presented by

MALIKA KHODJA

Laboratory: Mechanics and Physics of Materials
Field of study: Mechanical Engineering

Specialisation: Mechanics of Materials

Entitled

The behaviour of cracks repaired with composite patch in Aluminium alloy 7075-T6

Signed by the final Examining Committee:

Chair's name:

Pr Belabbes Bachir Bouiadjra — University of Sidi Bel Abbes, Algeria

Examiner's name :

Dr Mehadjia Bezzerrouki — University of Sidi Bel Abbes, Algeria

Dr Leila Zouambi — Center University of Relizane, Algeria

Supervisor:

Pr Hamida Fekirini — University of Sidi Bel Abbes, Algeria

2020/2021

The behaviour of cracks repaired with composite patch in Aluminium 7075-T6

By

Malika Khodja

Submitted for the

Degree of Philosophy Doctor (Mechanics of Materials)

In the

Faculty of Technology, Mechanical Engineering Department

University of Djilalli Liabes

Sidi Bel Abbes, Algeria

Submission date: October 2020

Signed by the final Examining Committee: 27 January 2021

ملخص

لقد كان التصدع الناتج عن التعب تحت الضغط المتكرر مصدر قلق للهندسة لعقود .أصبحت حالات فشل التعب أكثر شيوعًا وأصبحت ذات أهمية متزايدة في التصميم الهندسي مع تزايد مصادر الاهتزاز والتحميل الديناميكي للمواد.

أثبت إصلاح الهياكل المعدنية التالفة باستخدام رقع من المواد المركبة الملتصقة أنها طريقة اقتصادية لزيادة مقاومة الإجهاد للهياكل التالفة. يتعلق هذا البحث بدراسة تجريبية عن عمر تعب ألواح صفائح الألمنيوم الألومنيوم AA7075-T 6 التي تم إصلاحها / تقويتها برقعة بورون / إيبوكسي ملتصقة من جانب أحادي. تركز الدراسة على سلوك الشقوق في AA7075 - T 6. أولاً ، تمت دراسة انتشار الشقوق من ثقب مسنن تحت شحن تعب أحادي المحور لصفيحة AA7075 – T 6 ، وهي غير مصححة .تم تحليل النتائج التي تم الحصول عليها لتأسيس فهم سلوك الكلال (التعب) لصفائح الألمنيوم6 AA7075 -T غير المشقوقة . يحدد عامل شدة الإجهاد (SIF) حدة أو خطورة الشق .لذلك، يتحكم SIF في معدل نمو شقوق التعب .المسار الذي حدثت فيه الصدوع كان من خلال الحبيبات (الكسر بين الحبيبات) في جميع مراحل نموها. لقد أظهرت نموًا بطيئًا مع وجود خط مستقيم بالقرب من الشق حيث لوحظ نمو مرتفع قبل الكسر النهائي، نظرًا للكسر السريع غير المستقر .ثانيًا، وقد تم تقديم معلومات أساسية عن مواد الإصلاح الملصقة وأيضا مناقشة عملية إصلاح الهياكل المعدنية / المركبة بالتفصيل .ثم تم فحص عمر الكلال لصفائح طائرة مصنعة من AA7075 -T 6متشققة بأطوال مختلفة التي تم إصلاحها باستخدام رقع من مركب بورون / إيبوكسي ومقارنتها بالصفائح الغير مصححة.

أظهرت النتائج المتحصل عليها أن ترميم الصفيحة المتصدعة باستخدام رقع من المواد المركبة الملتصقة كان فعالاً في تقليل معدل نمو الشقوق وبالتالي تحسين عمر الكلال للهيكل بنسبة 70٪. معدل نمو الشقوق في اللوحة غير المصقولة أعلى مقارنة بمعدلات التصحيح. يزيد معدل نمو الشق كلما زاد طوله بسبب معدل تركيز الضغط العالي عند طرفه. حيث أظهر التباين في معدل نمو الشق (da / dN، mm / cyc) كدالة لطول الشق للوحة المرقعة والغير مرقعة لإصلاح طول الشق المختلف أن وجود الرقعة يقلل بشكل كبير من معدل نمو الشق. لوحظ أنه من غير المناسب إصلاح الشقوق التي يزيد حجمها عن 30 مم، لذلك لا يوصى بإصلاح الشقوق الكبيرة التي يتم إصلاحها باستخدام رقع مركبة بطول الشقوق الكبيرة للصيانة طويلة المدى لأنها قد تؤدي إلى نتائج كارثية. ومع ذلك، فمن الفعال إصلاح الشقوق الكبيرة في حالات الطوارئ.

أخيرًا، ركز البحث على تأثير دورات المعالجة المختلفة على عمر الكلال للهياكل المعززة / المرممة. تم اختبار معاملات المعالجة المختلفة لتحديد الفروق من حيث قوة الترابط وتأثيرات التشويه (الاعوجاج) وعمر الكلال لصفائح AA7075 -T 6 التي تم إصلاحها. تمت دراسة السلوك المشوه للهياكل المعدنية التالفة والمصلحة باستخدام رقع من المواد المركبة الملتصقة. تم التحقيق في تأثير ظروف دورة المعالجة على عمر الكلال. أظهرت النتائج أن درجات حرارة المعالجة المرتفعة تعمل على تعزيز الالتصاق بين السطح المعدني والرقع من المواد المركبة والمتانة الكلية. ومع ذلك، قد تؤدي درجة حرارة المعالجة المرتفعة إلى تشويه مع الانحناء. يعتبر الجمع الأمثل بين درجة الحرارة ومدة دورة العلاج مهمًا لتقليل التشوه.

الكلمات الدالة:

شق؛ تعب أو كلال؛ كسر أو صدع؛ دوري؛ ألومنيوم 7075 -T6؛ مواد مركبة؛ الترابط اللاصق؛ درجة حرارة دورة المعالجة؛ ألياف البورون / ايبوكسي؛ تشوه؛ FM73K؛ عملية الإصلاح؛ نمو تشقق كلالي.

Abstract

Fatigue failure cracking under repeated stressing has been a concern of engineering for decades. Fatigue failures have become more common and increasingly significant in engineering design as sources of vibration and dynamic loading of materials have increased.

The repairs of damaged metallic structures using bonded composite patch have been proved as an economical method to increase the fatigue life of damaged structures. This research deals with an experimental study on the fatigue life of $AA7075-T6$ aluminium alloy plates repaired/reinforced with single-sided boron/epoxy patches. The study focuses on the behaviour of cracks in $AA7075-T6$. First, the crack propagation from a notched hole under uniaxial fatigue loading of $AA7075-T6$ sheet, namely unpatched was studied. The obtained results were analysed to establish the fatigue behaviour understanding for uncracked, notched $AA7075-T6$ aluminium alloy. The stress intensity factor (SIF) defines the severity of a crack. Therefore, SIF controls the rate of fatigue crack growth. The path along which fractures occurred was through the grains (transgranular fracture) in all crack growth stages. It showed slow growth with a straight line near the notch where,due to rapid unstable crack, growth just prior to final fracture a high crack growth rate was observed. Second, the background on bonded repair materials is introduced.Also, metal/composite bonded structures repair process is discussed in details. Then the fatigue life of $AA7075-T6$ aircraft sheet cracked with various crack length repaired with a boron/epoxy composite patch was investigated and compared to unpatched plate.

The obtained results showed that repairing of the cracked plate using bonded composite patches was effective in decreasing the crack growth rate and thereby enhancing the fatigue life of the structure by 70%. The crack growth rate in unpatched panel is higher compare to the patch ones. The crack growth rate increases as the crack length increases due to higher stress concentration rate at the crack tip. Where the variation of crack growth rate (da/dN, mm/cyc) as a function of the crack length for the patched and unpatched panel of different crack length repair showed that the patch presence highly decrease the crack growth rate. It was noted that it is inefficient to repair cracks of size more than 30 mm crack length.Therefore, in various crack length repaired with patches composite, it is not recommended to repair large cracks for long-term maintenance as it can lead to catastrophic results. However, it is effective to repair large crack for emergency conditions.

Finally, the research was focused on the effect of different curing cycles on the fatigue life of reinforced/repaired structure. Different curing parameters were tested to determine differences in terms of bond strength, distortion effects and fatigue life of repaired $AA7075-T6$ sheets. Distortion behaviour of bonded repair was studied. The influence of the curing cycle conditions on fatigue life was investigated. The results showed that the elevated curing temperatures enhance the adhesion between the metal surface and the composite patch and overall durability. However, elevated curing temperature may

result in distortion with bending curvature. The optimum combination of temperature and duration for the curing cycle is of importance to reduce the distortion.

Keywords:

Crack; fatigue; fracture; cyclic; Aluminium 7075-T6; composite materials; adhesive bonding; curing cycle temperatures; boron fibre/epoxy; distortion; FM73K; bonded repair processes; fatigue crack growth FCG.

Résumé

La fissuration par fatigue sous des contraintes répétées est une préoccupation de l'ingénierie depuis des décennies. Les défaillances dues à la fatigue sont devenues plus courantes et de plus en plus importantes dans la conception technique à mesure que les sources de vibration et de charge dynamique des matériaux ont augmenté.

Les réparations de structures métalliques endommagées à l'aide de patchs composites colléesse sont avérées être une méthode économique pour augmenter la résistance à la fatigue desstructures endommagées. Cette recherche porte sur une étude expérimentale sur la résistance à la fatigue de plaques d'alliage d'aluminium $AA7075-T6$ réparées / renforcées avec des simples patchs en bore / époxy. L'étude se concentre sur le comportement des fissures en $AA7075-T6$. Tout d'abord, la propagation des fissures à partir d'un trou entaillé sous charge de fatigue uniaxiale de la tôle $AA7075-T6$, à savoir le comportement d'une tôle sans patch, a été étudiée. Les résultats obtenus ont été analysés pour établir la compréhension du comportement en fatigue de l'alliage d'aluminium $AA7075-T6$ non fissuré et entaillé. Le facteur d'intensité de contrainte (FIC) définit la sévérité d'une fissure. Par conséquent, le FIC contrôle le taux de croissance des fissures de fatigue. Le chemin le long duquel les fracturesse sont produites était à travers les grains (rupture transgranulaire) à tous les stades decroissance des fissures. Il a montré une croissance lente avec une ligne droite près de l'entaille où, dû d'une fissure rapide et instable, juste avant la rupture finale, un taux de croissance de fissure élevé a été observé. Deuxièmement, le contexte sur les matériaux de réparation collés est présenté. De plus, le processus de réparation des structures collées métal/composite est discuté en détail. Ensuite, la résistance à la fatigue d'une tôle d'aéronef $AA7075-T6$ fissurée avec différentes longueurs de fissure réparée par patch en composite bore/époxy a été étudiée et comparée à une plaque non réparée.

Les résultats obtenus ont montré que la réparation de la plaque fissurée à l'aide de patchs composites collés était efficace pour réduire le taux de croissance des fissures et augmenter ainsi la résistance à la fatigue de la structure de 70%. Le taux de croissance des fissures dans les tôles non réparées est plus élevé que dans ceux réparées par patchs. Le taux de croissance des fissures augmente avec l'augmentation de la longueur de fissure en raison d'un taux de concentration de contraintes plus élevé à la pointe de fissure. La variation du taux de croissance des fissures (da/dN, mm/cyc) en fonction de la longueur de fissure pour une tôle réparée et non réparée de différentes longueurs de fissure a montré que la présence de patch diminue considérablement le taux de croissance des fissures. Il a été noté qu'il est inefficace de réparer des fissures de taille supérieure à 30 mm de longueur de fissure. Par conséquent, dans diverses longueurs de fissures réparées avec des patchs composites, il n'est pas recommandé de réparer de grandes fissures pour un entretien à long terme car cela peut conduire à des résultats catastrophiques. Cependant, il est efficace de réparer une grande fissure en cas d'urgence.

Enfin, la recherche s'est concentrée sur l'effet de différents cycles de traitement(c'est-à-dire l'historique de température en fonction du temps) sur la durée de vie en fatigue des structures renforcées/réparées. Différents paramètres de traitement ont été testés pour déterminer les différences en termes de résistance d'adhésion, d'effets de distorsion et de durée de vie en fatigue des plaques en $AA7075 - T6$. Le comportement de distorsion de la réparation collée a été étudié. L'influence des conditions du cycle de traitement sur la durée de vie en fatigue a été étudiée. Les résultats ont montré que les températures de traitement élevées améliorent l'adhérence entre la surface métallique et le patch composite et la durabilité globale. Cependant, une température de traitement élevée peut entraîner une distorsion avec courbure de flexion. La combinaison optimale de température et de durée du cycle de traitement est importante pour réduire la distorsion.

Mots Clés :

Fissure; fatigue; fracture; cyclique; Aluminium 7075-T6; matériaux composites; collage; températures du cycle de traitement; fibre de bore/époxy; distorsion; FM73K; processus de réparation collés; croissance des fissures par fatigue.

Acknowledgements

This thesis work is part of the research at the LMPM Laboratory , Faculty of Technology , Djilali Liabès University of Sidi Bel Abbés.

I want to thank Pr. BACHIR BOUIDJRA Bel Abbes, Professor at Djilali Liabès University of Sidi Bel Abbès for doing me the honour of chairing my PhD defence jury. My thanks also go to Dr BEZZERROUKI Mehadjia, senior lecturer at the University of Djilali Liabès University of Sidi Bel Abbès and Dr ZOUAMBI Leila, senior lecturer at the University Centre of Relizane by agreeing to be part of the jury and be examiners.

There is a long list of people I would like to thank for their help and understanding throughout this thesis project. Firstly, I would like to thank my:

- Advisor and supervisor, Pr. Fekirini Hamida at the University of Djillali Liabes for giving me this topic and guiding me to the successful completion of the research.
- I would like to thank Pr Laurent Dala from Northumbria University at Newcastle and John Morgan from CSIR for initiating the scientific collaboration between Algeria and South Africa to host my inter ship at CSIR.
- I wouldn't have been able to complete the research without the help of all of the colleagues at the CSIR. I express gratitude to Martin Williams, Marius and Andre Globler, Chris Mc Dulling, Ulyate Curle and Chris Kruger. Their technical expertise was invaluable.
- I would like to give a special thanks to Pierre Rossouw from CSIR for logistical management and technical and practical support. He allowed me to grow with knowledge in project management from planning to procurement to execution. Also thank you to Dr Heinrich Möller for his help and valuable discussions and guidance.

Lastly, I'd like to thank my family Khodja, for their love and continuous support through this whole journey. Keeping me in their prayers made this project possible.

Funding: The following contributions are gratefully acknowledged:

- A grant under the National Exceptional Program (PNE) from the Algerian Ministry of Higher Education and Scientific Research and Directorate General of Scientific Research and Technological Development is acknowledged.
- The Council for Scientific and Industrial Research (CSIR) supported this work through in kind funding by providing access to CSIR laboratories, expertise for testing and evaluation of samples.
- Dr Kevin Slattery and Selina Cobbs at Boeing and Dr Leonard Mac Adams at Cytec/Solvay Group-USA are acknowledged for assistance with acquiring materials.

- Soli Deo Gloria -

Contents

1 General Introduction **15**
1.1 Aircraft fatigue . . . 15
1.2 Repairs in aeronautical structures using bonded composite . . . 16
1.3 Problem Statement/Motivation . . . 19
1.4 Objective . . . 19
1.5 References . . . 22

2 Literature Survey **23**
2.1 Fatigue failure process in metals . . . 23
2.2 Background in fracture mechanics . . . 25
2.3 Theory of Linear Elastic Fracture Mechanics . . . 26
2.3.1 Failure modes . . . 27
2.3.2 Stress Intensity Factor Approach to Fracture . . . 28
2.3.3 The application of stress intensity factor for Fatigue crack propagation 31
2.4 Aluminium 7075 alloy in aircraft structures . . . 33
2.4.1 Aluminium Alloy AA7075 . . . 34
a) Composition of AA7075 . . . 34
b) AA7075 T6 Properties . . . 34
2.5 Repairability of aircraft structure . . . 35
2.6 Criteria of repairs in sheet metal in aircraft by patching . . . 36
2.7 Crack repair techniques used in aircraft crack repair . . . 36
2.8 Bonded crack patches . . . 37
2.9 Advantages and disadvantages of composite patches . . . 38
2.10 History and past efforts of aircraft crack repair using composite patches . . . 40
2.11 Current knowledge on composite crack repair of AA7075 T6 . . . 41
2.11.1 Experimental/Numerical and Theoretical efforts . . . 42
2.11.2 Numerical Analysis using Finite Element Analysis FEA . . . 45
2.12 References . . . 48

3 Experimental Methods **53**
3.1 Introduction . . . 53
3.2 Materials Selection & Description . . . 54
3.2.1 Aluminium AA7075-T6 . . . 54
3.2.2 The Patch Composite: Boron Fibre Epoxy . . . 55
3.2.3 Adhesive Layer: Cytec FM73-K . . . 56
3.3 Manufacturing of the Specimens . . . 57

3.3.1 AA7075-T6 Aluminium Specimen Design and Fabrication 57
3.3.2 The Composite Patch Repair Design Process 60
3.3.2.1 Design of the Boron/Epoxy Patch 61
3.3.2.2 Stiffness Ratio Calculation 61
3.4 The Repair Methodology & Manufacturing of the Repaired Samples 61
3.4.1 Surface Preparation of the Aluminium Panels 61
3.4.2 Bonding Boron/Epoxy Patch onto the Aluminium Panel 62
3.4.3 Curing process 63
3.4.4 Curing parameters 64
3.5 Testing methodology 66
3.5.1 Mechanical Properties Characterisation of AA7075-T6 66
3.5.1.1 Tensile Test Procedure of AA7075-T6 66
3.5.1.2 Microstructural Analysis of AA7075-T6 67
3.5.2 Distortion Measurements 70
3.5.3 Fatigue testing procedure 70
3.5.3.1 Pre-cracking the Aluminium Panels with Fatigue testing . . 70
3.5.3.2 Fatigue crack growth (FCG) Test Procedure 71
3.5.3.3 Fatigue crack growth measurement 73
3.6 References 74

4 Results & Discussion **75**
4.1 Part I: UNPATCHED PLATE 75
CRACK PROPAGATION FROM A NOTCHED HOLE UNDER UNI-AXIAL FATIGUE LOADING OF AA7075-T6 PLATE 75
Summary 75
4.1.1 Introduction 75
4.1.2 AA7075-T6 material characterisation 78
4.1.2.1 Chemical composition 78
4.1.2.2 Mechanical properties 78
4.1.3 Optical microscopy of AA 7075-T6 plates 82
4.1.4 Fatigue crack growth (FCG) 84
4.1.5 Fatigue crack growth & Failure relationship 89
4.1.6 Conclusion on fatigue behaviour of unpatched, notched AA7075-T6 plate 97
4.1.7 References 98
4.2 PART II: PATCHED PLATE 99
EXPERIMENTAL STUDY ON THE BEHAVIOUR OF CRACKS REPAIRED WITH BORON FIBRE/EPOXY COMPOSITE PATCH IN ALUMINIUM 7075-T6 99
Summary 99
4.2.1 Introduction 99
4.2.2 Materials and methods 100
4.2.2.1 Materials 100
4.2.2.2 Surface treatment 101
4.2.2.3 Composite plate manufacturing 102
4.2.2.4 Curing parameters 103
4.2.2.5 Fatigue testing 104

4.2.3 Fatigue life of a notched hole in AA7075-T6 aircraft plate reinforced with a boron/epoxy and carbon/ epoxy composite patches (A) . . . 105
4.2.3.1 Fatigue life enhancement with bonded composite repair . . . 106
4.2.3.2 Fatigue crack growth behaviour (FCG) . . . 110
4.2.3.3 Tension test after the fatigue testing . . . 112
4.2.4 Fatigue life of AA7075-T6 aircraft plate pre-cracked with various crack length repaired with a boron/epoxy composite patch (B) . . . 117
4.2.4.1 Effect of the patch presence on FCG & SIF . . . 117
4.2.4.2 Effect of the boron/epoxy patch repair on the fatigue life with various crack length . . . 122
4.2.5 Effect of curing cycle on fatigue life of cracked AA7075-T6 aircraft plate reinforced with a boron/epoxy composite patch (C) . . . 127
Summary . . . 127
4.2.5.1 Introduction . . . 127
4.2.5.2 General observations . . . 129
4.2.5.3 Plate distortion . . . 131
4.2.5.4 Fatigue life . . . 133
4.2.5.5 Fatigue-distortion relationship . . . 134
4.2.5.6 Failure mechanism . . . 135
4.2.6 References . . . 138

5 Conclusions & Recommendations **140**
Future work . . . 144

List of Figures

1.1 Examples of aircraft fatigue failures [1]. 16

2.1 Crack initiation at the interface between the slip band and the matrix [3]. . 24
2.2 Basic fracture modes [10]. 25
2.3 Co-ordinate system under Mode I loading and basic modes of loading [16]. 29
2.4 Dependence of stress intensity parameter (factor) K on thickness of specimen or cracked component [16]. 30
2.5 Schematic elastic/plastic stress distribution near a notch. (a) in plane stress ($\sigma_{33} = 0$); (b) in plan strain ($\sigma_{33} = 0$) [17]. 30
2.6 Variation of fatigue crack propagation rates da/dN with stress intensity factor range $\triangle K$ [19]. 32
2.7 Advanced materials/processes for major structural areas of the Airbus A380 [23]. 33
2.8 Composite bonded repair patch being applied to an aircraft structure [36]. 37
2.9 Bolted vs. Riveted Repair Strength [46]. 39
2.10 Reduction in the Stress Intensity Factor with Repair [47]. 39
2.11 Effect of Patching on the Lifetime of a Cracked Structure [47]. 40

3.1 Geometry of the plate and composite repair patch. 54
3.2 SEM image of boron fibre and boron fibre/epoxy patch composite prepreg tape. 55
3.3 A roll of epoxy FM73 Cytec film structural adhesive with supporting fibres. 57
3.4 Tensile specimens with dimensions in mm machined from plate. 58
3.5 The Trumpf Lasercell 1005. 58
3.6 Rivet hole milling process using a Leaderway V650 machine. 59
3.7 The designed clamping plates for the panels. 59
3.8 (a) The panel sketch dimensions (b) Final samples machined (d) Panel fitted with the clamping plate. 60
3.9 Surface preparation with sand blasting method. 62
3.10 Sequence of bonding boron/epoxy patch onto the Aluminium panel. 63
3.11 Detail of vacuum bag lay-up and manufactured specimens. 64
3.12 The steps to the final patched test panel. 64
3.13 Computer Curing cycle log. 65
3.14 Tensile test procedure set-up of AA7075-T6. 66
3.15 Calculation of elongation after fracture. 67
3.16 Equipment used for sample preparation process at the metallographic laboratory at LM/MSM/CSIR. 68

3.17 Selected samples for microstructural analysis in the rolling direction. 68
3.18 Equipment used for sample preparation process at the metallographic laboratory at LM/MSM/CSIR. 69
3.19 The CSIR mechanical testing laboratory. 70
3.20 Aluminium Panel in Instron Test Machine Fitted with COD "Clip-Gauge". 71
3.21 Completed panel with Boron/Epoxy loaded in tension. 72

4.1 Illustration of the different fatigue - fracture modes in an AlZnMg alloy according to Forsyth [2]. 76
4.2 Examples of aircraft fatigue failures investigation [7] [8]. 77
4.3 Stress-strain curve of aluminium alloy 7075-T6 in the rolling direction with five tested samples for reproducibility test. 79
4.4 Stress-strain curve of aluminium alloy 7075-T6 in the rolling direction (0Deg). 79
4.5 Stress-strain curve of aluminium alloy 7075-T6 in perpendicular to the rolling direction (90Deg). 80
4.6 Load-extension curve for 2mm thick dog bone AA7075-T6 sample (a) in the rolling direction (0), (b) is perpendicular to the rolling direction (90). . 81
4.7 The AA7075 alloy plate and direction of analysis. 82
4.8 Schematic showing the three principal orthogonal directions in a rolled plate[12]. 82
4.9 Micrograph revealing a microstructure characteristic of AA 7075-T6 (a) Longitudinal (rolling) direction (RD), (b) Short Transverse directions (ST), (c) long transverse (LT). 83
4.10 Geometric model of the aluminium plate. 84
4.11 The sample sketch of the plate in crack opening mode. 85
4.12 Unpatched aluminium Plate inInstron Test Machine Fitted with COD "Clip-Gauge". 85
4.13 Fatigue life of AA7075-T6 sample without a patch. 86
4.14 Stress intensity factor vs crack length of AA7075-T6sample without a patch. 86
4.15 Variation of fatigue crack propagation rates da/dN with SIF range $\triangle K$ [15]. 88
4.16 Plot of log fatigue crack growth rate da/dN versus log $\triangle K$ of AA7075-T6 sample. 89
4.17 Crack propagation path of AA7075-T6 uncracked sample during the fatigue test. 90
4.18 Fractured sample of AA7075-T6 fatigue test. 91
4.19 Transgranular failure mechanism in AA7075-T6 fractured sample after fatigue test. 92
4.20 Optical microscope analysis of crack propagation path after fracture. . . . 93
4.21 Geometry of the plate and composite repair patch. 101
4.22 Surface preparation by grit blasting. 102
4.23 Detail of vacuum bag lay-up. 103
4.24 The steps to the final patched test plate. 103
4.25 Curing cycle log. 104
4.26 Completed plate with composite patch loaded in tension. 105
4.27 Fatigue life of notched/ unrepaired plate 106
4.28 Fatigue life of notched/ repaired plate with boron fibre epoxy composite patch. 107

4.29 Fatigue life of notched/ repaired plate with carbon fibre epoxy composite patch. . . . 108
4.30 Fatigue life of notched/ repaired plate with boron fibre epoxy and carbon fibre epoxy composite patch. . . . 109
4.31 Comparison of the repair material on Fatigue life of notched/ repaired plate. 110
4.32 Comparison of the crack propagation rate between unpatched and patched plates. . . . 111
4.33 Crack propagation path behaviour of patched plates. . . . 112
4.34 Effect of patch strength after plate failure. . . . 113
4.35 Tensile test of patched samples after the fatigue testing (a)Unpatched sample, (b) Sample patched with carbon fibre/epoxy composite and (c) Sample patched with boron fibre/epoxy composite. . . . 115
4.36 Fractured samples after tension test. . . . 116
4.37 Effect of the patch presence on Fatigue Life. . . . 118
4.38 Reduction in Stress Intensity Factor with Repair. . . . 119
4.39 Fatigue crack growth FCG of AA7075-T6sample with and without patch. . 120
4.40 Fatigue crack growth FCG of AA7075-T6 sample with and without patch. 121
4.41 Variation of the crack growth rate of unpatched and patched plates. . . . 121
4.42 Different crack length pre-cracked to be repaired. . . . 122
4.43 Crack growth behaviour in the repaired plate. . . . 123
4.44 Fatigue life of 1 mm pre-cracked plate repaired with composite patch. . . . 123
4.45 Fatigue life of AA7075-T6 repaired with composite patch of different pre-crack length size(a) Crack size length repaired and (b) Crack length size shifted to zero as the origin for comparison. . . . 125
4.46 Fatigue life cycles variations of different crack length damage repaired with patch composite. . . . 125
4.47 Fatigue life cycle behaviour of small and large cracks repaired with patch composite. . . . 126
4.48 Curvature of plate s results in bending due to curing process. . . . 130
4.49 Summary of the measured thicknesses. . . . 130
4.50 Bending profile from the data for curing condition D. . . . 131
4.51 Distortion comparison of bonded repaired samples in different curing conditions. . . . 132
4.52 Fatigue life of cracked/ unrepaired plate. . . . 133
4.53 Effect of curing condition on crack propagation in the repaired plates. . . . 134
4.54 The relationships between fatigue life and plate distortion. . . . 135
4.55 Fractured samples in shear mode after tension test. . . . 136

List of Tables

2.1 Description of the fracture modes [10]. 26
2.2 Major modes of mechanical failures in engineering structure [7]. 27
2.3 AA 7075 Typical Chemical Composition Limits (in %) [24]. 34
2.4 Typical physical, mechanical thermal properties of AA 7075 T6 [21]. 35

3.1 Thickness of the material used for the bonded structural repair. 53
3.2 Chemical Composition Limits (in wt%) [ALUMINIUM DATA SHEET ALLOY GRADE 7075]. 55
3.3 Material Properties values of the boron fibre and boron prepreg tape [3]. . 56
3.4 Bond curing cycles. 65
3.5 Kellers etchant used for sample preparation. 69

4.1 Chemical Composition Limits (in wt. %) [ALUMINIUM DATA SHEET ALLOY GRADE 7075 T6]. 78
4.2 Tensile properties of AA7075T6 sample according (A) the rolling direction, (B) perpendicular to the rolling direction (90). 81
4.3 Average grain size of 7075-T6 aluminium sheet used in this study. 84
4.4 Mechanical properties of different repair materials. 101
4.5 The FCG rate & $\triangle K$ variation of unpatched and patched plate. 111
4.6 Tensile test data summary of unpatched and patched plate after the fatigue testing. 114
4.7 Distortion variation of bonded repaired samples in different curing conditions. 132

Chapter 1

General Introduction

1.1 Aircraft fatigue

Many materials contain physical discontinuities, which are given the generic term of notches. When a notched material is loaded, local stress and strain concentrations occur. These phenomena may result in micro-cracks that develop and coalesce with one another until they form a macroscopic crack, which propagates until the total breakdown of the material occurs. Aeronautical components may be subjected to multiaxial stress states of varying amplitude leading to fatigue damage.This can lead to catastrophic failure as shown in Figure 1 [1].

Therefore, fatigue and reliability analysis and prediction of structural performance are critical to investigate, especially with the development of composite materials. Studies on this have been done on various cases and using different techniques [2-4]. Structural fatigue cracking is a result of usage and will eventually occur in all ageing aircraft. These cracks are common in aluminium alloys such as 7075 and 7079, especially in the T6 temper condition. These alloys are tailored for high strength, but compromise crack initiation/growth performance and corrosion resistance [5]. Structural shapes also significantly affect the fatigue life; square holes or sharp corners lead to elevated local stresses where fatigue cracks can initiate [5].



Figure 1.1: Examples of aircraft fatigue failures [1].

1.2 Repairs in aeronautical structures using bonded composite

One efficient means of extending the life of aircraft components is to retard or arrest the crack growth. Traditional repair methods involve the use of riveted metallic reinforcements or patches. The most common repair/reinforcement technique is the mechanical fastening of a plate over the damaged area to redistribute the load and reduce the stress concentration in the damaged zone. Although these methods give good results

in most cases, they require extra fastener holes, which create more stress concentrations in an already weakened structure. Furthermore, poorly drilled holes may create additional crack nucleation sites. Other operational damage, such as fretting and corrosion, can also produce failure of the components. Therefore the mechanical fastening technique is less desirable, because of the non-uniformity of the stress distribution and the creation of stress concentration zones around the fasteners [8].

Composite materials are becoming more important in the construction of aerospace structures. Bonded joints fulfil a significant role in the development of new technologies, particularly in the aerospace industry where they give new possibilities for connecting structural elements. The composite patch bonded repair (CPBR) method can be used for repairing metallic and composite structures [6].ThereforeBonded fibre composite repair on damaged metal structures can be an effective method to restore load carrying capacity or increase damage resistance. The purpose of this method is to slow or arrest crack growth, and thus, prolong the life of the structure. The primary advantages of composite materials for repair are their high strength, relatively low weight, and corrosion resistance [7].The technology has been successfully used in repair applications on both military and civil aircraft for fatigue enhancement and crack patching. Depending on the patch design and the quality of the installation, adhesively bonded repairs provide an efficient way to reduce stress levels in the damaged structure by introducing minimum stress concentrations into the structures.

Composite bonded repairs to metallic aircraft structures aregenerally used for fatigue enhancement, crack patching, as well as different kinds of damage repairs, such as corrosion damage or machining defects. In addition to the advantages mentioned above, there are many other advantages of using bonded composite repairs instead of using conventional metallic riveted repairs. Thoseare discussed in chapter 2.

Bonded composite repairs of locally damaged metallic structures have gained considerable interest in aircraft structural maintenance and life extension in the recent decades, where the US Air Force and the aerospace industry have shown considerable interest in the use of bonded composites as a repair technique for metallic structures [8]. The Aeronautical Research Laboratory in Australia has pioneered composite repair doublers bonded to metal substructures, which have been used in development programs and partially in fleet repair programs, as an alternative to bolted repairs. They showed that the bonded composite repair improved the durability of aircraft structures under practical conditions [9].

Whenrepaired aeronautical componentsare subjected to cyclic loading, the presence of defects in structural components reduces the operating life. This can cause barely visible damage to the interior structure of laminated composites. Fatigue behaviour may also create debonds between the patch and metal substrate[10]. Bond integrity plays a big role in the repaired structure fatigue life. The bond line is the most important part of the patch repair structure since it is the key to transferring stress from the cracked plate to the patch [11].

Significant efforts have been devoted to the repair or reinforcement repairs by CPBR

for various applications of the damaged or weakened part of a structure to restore the structural effectiveness and thereby ensure the safety of the structures. Analysis of metallic aircraft structures development and qualification indicated secondary bending structures curvature of single side in the repair area. This secondary bending structures curvature influenced the mechanical behaviour and further detailed investigations were recommended [12].

The Safe Life approach/design philosophy expects no cracking during pre-determined service life [13]. Therefore, if cracking is found, engineers can decide to replace the damaged components and alter the design if necessary. The safelife aircraft requires only minimal inspection of the structures during the intended service life. This approach minimizes aircraft maintenance costs, but means that major components, or even the whole aircraft, must be retired at the end of the intended service life. Replacement of a damaged component can be very costly, in both man-hours and loss of aircraft serviceability, especially if disassembly of complex structures is required. Alternatively, to the Safe Life philosophy, the Damage Tolerance Approach (DTA) can be used after the Safe Life service life to allow the cracked structure to withstand a high proportion of its design loads after suffering corrosion, fatigue, accidental or impact damage by monitoring and slowing crack growth [14,15]. The DTA allows the operator to set safe inspection intervals to detect and monitor any significant structural cracking before it becomes a flight safety concern. DTA requires a fundamental understanding of fracture mechanics, including the critical crack length as well as the crack propagation rate to determine the inspection interval.

Recently, in March 2020, the US aviation regulator has launched an investigation after a 12-inch (30.5cm) crack ruptured the skin of a Boeing 737 aircraft operated by Southwest Airlines, causing the plane to gradually lose cabin pressure.Flight records revealed that the pilots began a rapid, six-minute descent from 39,000 feet (990 meters) to 22,000 feet. No injuries were reported from the incident. Fortunately, the cabin pressure was safe at the lower altitude. The FAA investigation reported that the crack in the skin of the plane's crown was discovered in prior inspection but that it was too early to say whether the frequency of inspections should be increased. This incident that might have caused a catastrophic consequence is due to not conducting mandatory inspections or followup inspections on some of its planes regularly. Maintenance experts say cracks are usually caused by fatigue or wear and tear and are not uncommon in older planes.Inspections must put in place to ensure they are repaired before they pose a threat to safety [16].

The CPBR method includes the following steps: surface preparation before bonding, the application of Composite Patch, a cure cycle of the composite and the adhesive film, and the final cooling [17]. The main disadvantages of using this technology are that very careful preparation of bonded surfaces is required. Heat and pressure are also required to cure the adhesive. Due to the low relative thermal expansion coefficient compared to metals such as aluminium, composite bonded repairs made of carbon or boron fibres induce thermal residual stresses during the curing process. Finally, inspection and crack monitoring are more complex as the patch is installed over the damaged structure.

1.3 Problem Statement/Motivation

As will be shown in the next chapter, the review of current studies in the area of bonded composite patches applied to cracked aluminium panels can be summarized as follows:

- Few studies have addressed fatigue characteristics of repaired thin components.

- Those which have investigated this problem have not addressed secondary bending effects or debonding.There is a need to perform experiments to characterize fatigue crack growth after thin components are repaired with bonded composite patches and to provide data to validate analytical or numerical solutions currently available.

- A simple, inexpensive approach to crack growth prediction in thin components with repairs is needed to reduce inspection intervals and to predict the service life of aircraft repairs with more accuracy.

- Development of new analytical tools is an important next step to expand the usefulness of bonded composite repair technology.

1.4 Objective

The objectives of this research project were as follows:

This study, therefore, was proposed to conduct experiments to investigate the fatigue crack growth behaviour of repaired aluminium panels with centre crack emanating a rivet hole.

a. To improve the understanding of fatigue life estimation techniques of metallic structures repaired using boron/fibre composite in crack emanating a rivet hole.

b. To develop a systematic methodology for the evaluation of the fatigue life of thin structures repaired using bonded composite.

c. The effect of the curing conditions were investigated for single-sided repairs to define the influence of the curing time and temperature on the fatigue life of bond repaired structures of $7075-T6$ with boron fibre epoxy composite patches.The novelty of our research is that we have tested different curing parameters to determine differences in terms of bond strength, distortion effects and fatigue life of repaired $AA7075-T6$ sheets. There are limited investigations available on the influence of curing parameters of structural epoxies on fatigue life of repaired metallic panels considering different curing regimes in a single study. The study of fatigue life, various cycle conditions on the formation of distortion levels and how that is influenced by cure cycles where

the time and temperature cycle combination in various conditions are discussed. Initial curvatures (due to residual thermal stresses from bonding repairs), fatigue crack growth rates, were obtained in the experiments. Also, experiments with cracked yet unrepaired panels were conducted to establish baseline data for comparison.

d. To quantify the impact and behaviour of repair in various crack size cases on the fatigue life predictions.

e. To provide a detailed method of boron/epoxy composite patch repair procedures of damaged metallic structure in order to provide results data for future finite element analysis (FEA) to investigate the failure modes and more in-depth the mechanics of materials in the repair field. Examples include FEA analysis on the effect of different curing conditions in order to find a model that gives the optimum combination of temperature and duration for suitable cure cycles to eliminate secondary bending and formation of distortion of repaired structures after curing. Also quantifying residual stresses that may be formed and its relation on the issue of the resulting distortion with single-sided reinforcement.

This thesis has been separated into five distinct chapters, each summarizing a different area of the study. This first chapter outlines the motivation behind the research. The second chapter discusses the background information and past research efforts in bonded repair technology with the focus on **Mechanical Physical Materials Laboratory** (LMPM) efforts and contributions. The third chapter discusses the experimental setup and testing procedures. Chapter four details the results and discussion of the study.This part consists of two main experimental studies on:

Part 1: Crack propagation from a notched hole under uniaxial fatigue loading of $AA7075-T6$ **sheet**.

Part 2: The behaviour of cracks repaired with boron fibre/epoxy composite patch in $AA7075-T6$. The latter is divided into

I. Fatigue life a notched hole in $AA7075{-}T6$ aircraft sheet repaired with a boron/epoxy and carbon/epoxy composite patches.

II. Fatigue life of $AA7075{-}T6$ aircraft sheet cracked with various crack length repaired with a boron/epoxy composite patch.

III. Effect of curing cycle on fatigue life of cracked $AA7075-T6$ aircraft sheet repaired with a boron/epoxy composite patch.

The fifth chapter provides a summary of the research and closing remarks, including recommendations of future areas of research that should be performed.

Publications

Effect of curing cycle on fatigue life of cracked $AA7075-T6$ aircraft sheet repaired with a boron/epoxy composite patch.

1.5 References

[1] Bureau of Accident Investigation, Aircraft Accident Report: Aloha Airlines Flight 243, Boeing 737-200, N73711, Near Maui, Hawaii, April 28, 1988. Report No. NTSB/AAR-89/o3, National Transportation and Safety Office, Washington, 14 June 1989.
[2] Khashaba UA, Aljinaidi AA, Hamed MA. Fatigue and reliability analysis of nanomodified scarf adhesive joints in carbon fibre composites. Composites Part B: Engineering 2017; 20: 103-117.
[3] Hugo C. Biscaia, Carlos Chastre, David Cruz, André Viegas. Prediction of the interfacial performance of CFRP laminates and old timber bonded joints with different strengthening techniques. Composites Part B: Engineering 2017; 108: 1-17.
[4] Mohsen Heshmati, Reza Haghani, Mohammad Al-Emrani. Durability of bonded FRP-to-steel joints: Effects of moisture, de-icing salt solution, temperature and FRP type. Composites Part B: Engineering 2017; 119: 153-167.
[5] Chakherlou TN, Vogwell J. The effect of cold expansion on improving the fatigue life of fastener holes. Engineering Failure Analysis 2003; 10(1): 13-24.
[6] Heatcon Abaris Training; Advanced Composite Structures: Fabrication and Damage Repair Phase I, II, III; Training Materials 2008; Cwmbram; UK.
[7] Niedernhuber M, Holtmannspötter J, Ehrlich I. Fibre-oriented repair geometries for composite materials. Composites Part B: Engineering 2016; 94: 327-337.
[8] Ting T, Jones R, Chiu W, Marshall I, Greer J. Composites repairs to rib stiffened panels. Journal of Composite Structure 1999; 47: 737-743.
[9] Baker A.A, Jones R. Bonded Repair of Aircraft Structures. Composite part B 1988 ; Chapter 6 ; Section 7.
[10] Bureau of Accident Investigation, Aircraft Accident Report: Aloha, Airlines Flight 243; Boeing 737-200; N73711; Near Maui; Hawaii; April 28; 1988. Report No. NTSB/AAR-89/o3; National Transportation and Safety Office 14 June 1989; Washington.
[11] Ouinas D, Bachir Bouiadjra B, Himouri S and Benderdouche N. Progressive edge cracked aluminium plate repaired with adhesively bonded composite patch under full width disband. Composites Part B:Engineering 2012; 43(2): 805-811.
[12] Günther G, Maier A. Composite repair for metallic aircraft structures. Proceedings of 27th International congress of the aeronautical sciences ICAS2010; 19-24 September 2010; France.
[13] Baker A.A. (1997). Fatigue studies related to certification of composite crack patching for primary metallic aircraft structure. In: **FAA-NASA Symp. on Continued Airworthiness** of **Aircraft Structures**, Dept. of Transportation Report No. DOT/FAA/AR-97-2, I, July.
[14] Fredell, R.S. (1994). Damage Tolerant Repair Techniques for Pressurized Aircraft Fuselages [Ph.D. dissertation]. Delft University.
[15] Jones, K.M. and Shah, S. (1996) Composite Repair - Upper Forward Corner of P-3 Door - Model [14] L-1011 Aircraft, Strength and Damage Tolerance Analysis. Report No. LG95ER0157, Part of Documentation Package for FAA Atlanta Aircraft Certification Office Project No. SPl798AT-Q, analysis plan December, (1995),final report October.
[16] US probes large crack in Boeing 737 jet, https://www.dw.com/en/us-probes-large-crack-in-boeing-737-jet/a-52771903. Article accessed March 2020.
[17] Heatcon Abaris Training; Advanced Composite Structures: Fabrication and Damage Repair Phase I, II, III; Training Materials 2008; Cwmbram; UK.

Chapter 2

Literature Survey

2.1 Fatigue failure process in metals

Morris et al.[1], described fatigue as the process during which progressive and permanent localized deformation or structural change occurs in a material when subjected to cyclic loading. This structural change occurs during the forward and backward movement of loading causing continuous fluctuation strains at nominal maximum stresses below the static elastic limit of the material. When a material is subjected to permanent structural changes over time during cyclic loading, micro-cracks develop which eventually fail.

These permanent structural changes also referred to as fatigue damage is dependent on the simultaneous action of cyclic stress, tensile stress, and plastic strain. If anyone of these three is not present, a fatigue crack will not initiate and propagate, according to Morris et al.[1].

Studies by Hertzberg [2] show that plastic strain resulting from cyclic stress, initiate cracks while tensile stress promotes crack growth. Crack initiation develop from plastic strain during cyclic stress application and crack growth depends on tensile stress [2].

When the alternating stress amplitude is high enough, slip steps are created on the surface of the material because of plastic deformation in the form of dislocation motions. Subsequently, these dislocations may converge against an obstacle, such as an inclusion, a cracked particle, or grain boundary, and form a slip band, decohesion between particle and matrix, or decohesion along the grain boundary [3]. As shown in Figure 1, slip bands form during cyclic loading resulting in crack initiation between the slip band and the matrix [3].

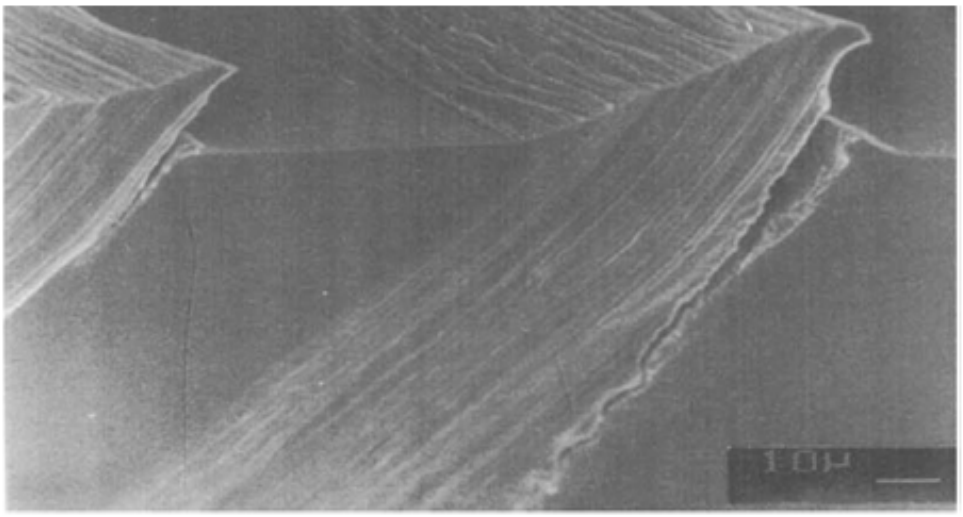


Figure 2.1: Crack initiation at the interface between the slip band and the matrix [3].

During the normal process of fatigue failure in a metal-free of imperfections, microcracks continue to merge or grow into macrocracks that propagate until the fracture toughness of the material is exceeded and final fracture occurs [2]. During controlled cyclic loading, microcracks initiate near or at individual imperfections like surface scratches, corrosion pits, inclusions, or embrittled grain boundaries that lie just below or on the surface[2], [4].

These initial cracks are usually very small and their size is not defined clearly, because it is difficult to determine when a slip band or other deformation feature develops into a crack [5]. However, modern metallographic tools such as the scanning electron microscope or transmission electron microscope are very effective to observe cracks as small as a fraction of a micron. The resolution of modern inspection instruments has increased which resulted in the portion of the fatigue lifetime attributed to fatigue crack initiation has decreased. For practical purposes, it is useful to define the "initiation stage" as that portion of the fatigue lifetime before a crack is detectable by usual non-destructive evaluation techniques [6].

The basic fatigue failure process is divided into five stages [6]:

1. Cyclic plastic deformation before fatigue crack initiation.
2. Initiation of one or more micro-cracks.
3. Propagation or coalescence of micro-cracks to form one or more micro-cracks.
4. Propagation of one or more macro cracks.
5. Final failure.

To predict the shape of fatigue cracks propagating in thin aluminium structures, it was necessary to first understand the fundamental concepts of fracture mechanics. Several structural failures can be associated with the fracture of one or more of the materials

making that structure. When such events occur, they are mostly unexpected, sudden, and unfortunate, and it is natural for us to focus attention on minimizing the undesired consequences when designing and analysing modern-day structures. The study of crack behaviour, prevention and analysis of fracture of materials is known as fracture mechanics.

2.2 Background in fracture mechanics

Introduced in 1920 by Griffith [7], fracture mechanics has matured to become a science that provides the theoretical knowledge required to quantify the effects of cracks in structures. Fracture mechanics provides tools to assess the structural integrity of structural components and thus, is the key to damage tolerance programs for aircraft. Although fracture mechanics is a complex science, the analysis of a structural component is often based on the computation of the Stress Intensity Factors (SIF), which quantify the severity of the structural damage. Based on earlier work performed by Westergaard, Irwin [8,9] published solutions for crack-tip stress distribution associated with three independent failure modes of fracture. The three fracture modes are illustrated in Figure 2 and described in Table 1. In fracture mechanics, a distinction is made between different modes of fracture propagation, depending on the loading situation.

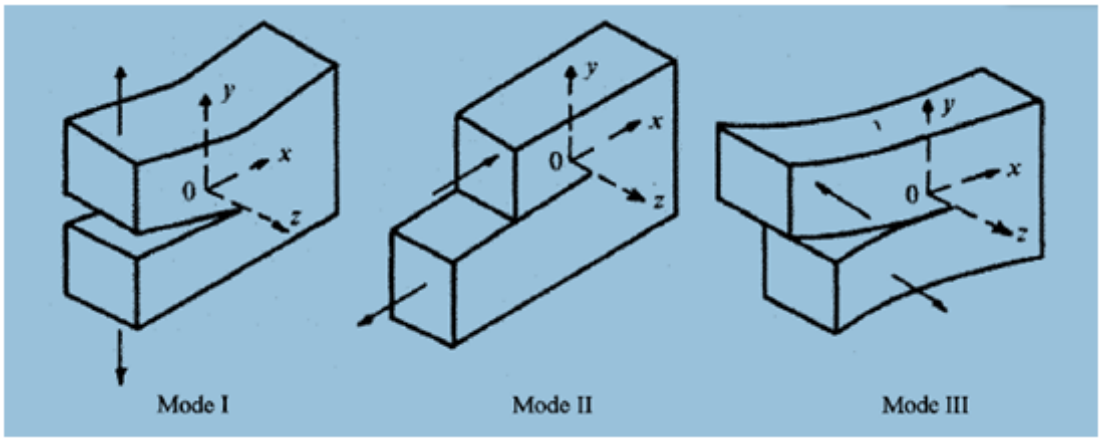


Figure 2.2: Basic fracture modes [10].

Fracture modes	Descriptions
Mode l	Mode I, known as the opening or tensile mode exists when the crack surfaces move directly apart. This mode is the most encountered in engineering problems
Mode II	Mode II, known as the sliding or in-plane shear mode, exists when the crack surfaces slide over one another in a direction perpendicular to the leading edge of the crack. This mode is found less frequently on its own in engineer ing problems but more often in combination with Mode I, referred to the mixed Mode I-II
Mode III	Mode III, known as the tearing or anti-plane shear mode, exists when the crack surfaces move relative to one another parallel to the leading edge of the crack

Table 2.1: Description of the fracture modes [10].

2.3 Theory of Linear Elastic Fracture Mechanics

To better understand how an aircraft repair works and the different advantages of mechanically fastened or bonded repairs, it is helpful to understand some of the theory involving cracks in thin metallic structure. This section focuses on some of the elementary theory of Linear Elastic Fracture Mechanics, since all structures have inherent pre-existing cracks or flaws that cause the failure. The stress intensity factor is the driving force for crack growth. The stress intensity factor is the linear elastic fracture mechanics parameter relating remote load, crack size, and structural geometry.Under elastic conditions, linear-elastic fracture mechanics characterizes the stress/strain field ahead of the crack tip using a single field parameter known as the stress intensity factor K. The magnitude of the stress field surrounding the crack is related to K as follows:

$$K = \sigma\sqrt{\pi a}.Y$$

where

σ = remotely applied stress(MPa),
a = crack size (mm),
Y = calibration factor that depends on the geometry of the cracked body and the size and location of the crack.

When K exceeds a critical value, the component fails. This critical value is referred to as the fracture toughness K_{Ic}. Cracks in structural materials do not show the behaviour of "Griffith cracks" because crack instability is preceded by the development of a plastic zone ahead of the crack tip as the loading increases. Valid K_{Ic} values may be obtained under small scale yielding conditions if the plastic zone remains small relative to the crack length and the dimensions of the component/test piece. This restriction is manifested in

the requirements for minimum specimen dimensions given in the recommended testing procedures [11, 12].

Before we discuss the stress intensity factor any further, it is at this point important to present the different fracture modes defined in fracture mechanics.

2.3.1 Failure modes

The degradation and fracture in engineering materials and structures have been and still are of continuing concern for both material scientists and mechanical engineers. These failures may occur under quasi-static and/or alternating (dynamic) loads. The path along which fracturesoccur can go through the grains (transgranular fracture) or along the grain boundaries (intergranular fracture). Regardless of the fracture path, there are essentially only four principal failure modes namely, dimple rupture, cleavage, fatigue a nd decohesive rupture [13]. Each of the principal failure modes is controlled by specific mechanisms that result in typical fracture surface morphologies. Consequently, the fractographic analysis in a scanning electron microscope (SEM) has emerged as a valuable tool towards the understanding of fracture processes [14]. The mechanisms of the various fracture modes have been explained in terms of dislocation interaction involving complex slip and crystallographic relationships [7]. Major modes of mechanical failures in engineering structure may be summarized as follows in Table 2:

Failure mode	Main controlling parameter
Buckling or jamming	Young's modulus
Necking	Ultimate tensile strength
Slow cracking - Low cycle fatigue (LCF) - High cycle fatigue (HCF) - Fatigue crack propagation (FCP)	 - Ductility - Strength - Fracture toughness, Young's modulus
Fast cracking	Fracture toughness

Table 2.2: Major modes of mechanical failures in engineering structure [7].

It should be noted that variations in temperature (i.e. high-temperature creep) and environment (i.e. corrosion fatigue), which have a significant effect on the degradation and failure of engineering materials, have not been considered for the above failure modes.

2.3.2 Stress Intensity Factor Approach to Fracture

When a remote uniaxial tensile load is applied to a cracked body, a multi-axial state of stress develops ahead of the crack tip, which is dependent on the thickness of the cracked body. In the case of a thin plate, the stress distribution can be given as follows [15]:

$$\sigma_x = \sigma \left(\frac{a}{2r}\right)^{0.5} .f_1(\theta)$$

$$\sigma_y = \sigma \left(\frac{a}{2r}\right)^{0.5} .f_2(\theta)$$

$$\sigma_{xy} = \sigma \left(\frac{a}{2r}\right)^{0.5} .f_3(\theta)$$

(5)

Where σ is the nominal stress, θ is the angle between the x-axis and the radius vector r and a is the half-crack length. The above equations show that the local stress near a crack is proportional to the product of nominal stress σ and the square root of the half-crack length. This has led to the definition of the stress intensity factor K. Hence, the local stresses near the crack tip can be written as follows:

$$\sigma_x = \frac{K}{\sqrt{2\pi}} f_1(\theta)$$

$$\sigma_y = \frac{K}{\sqrt{2\pi}} f_2(\theta)$$

$$\sigma_{xy} = \frac{K}{\sqrt{2\pi}} f_3(\theta)$$

There are various modes of loading that can be applied to a crack as shown in Figure3 [16]. Mode I or crack opening mode, where the tensile stress is applied normal to the faces of the crack, is regarded as the most important type of loading. If not otherwise stated, the following discussions refer to the crack opening mode or Mode I only.

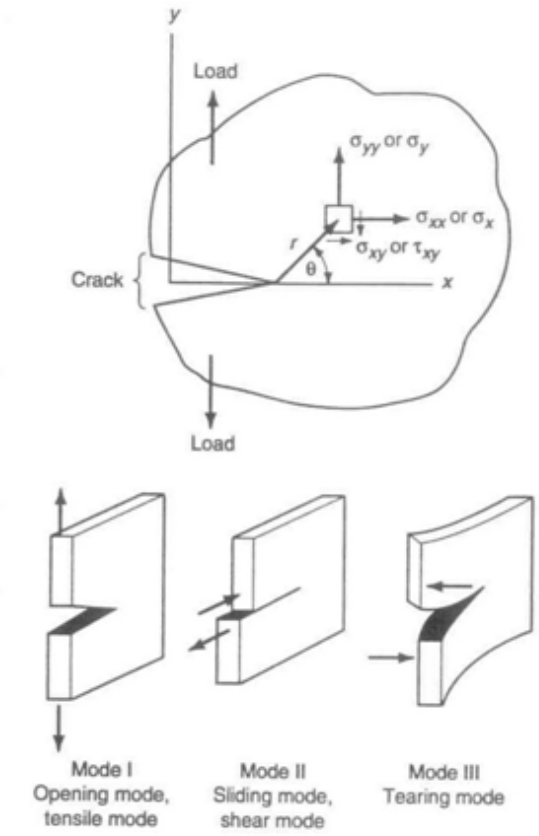


Figure 2.3: Co-ordinate system under Mode I loading and basic modes of loading [16].

For linear elastic conditions, the stress intensity factor K (crack opening mode) characterizes uniquely the stress field around a crack and brittle fracture occurs when the stress intensity factor reaches a critical value of K_c, termed fracture toughness. Figure 4 shows a plot of critical stress intensity factor K_c vs the specimen thickness. For thin specimens, the stress in the thickness direction approaches zero and a biaxial stress state prevails ahead of the crack. This situation is referred to as plane stress and Kc is strongly dependent on the specimen thickness. In the case of thick specimens, the strain in the thickness direction approaches zero and the crack tip is characterized by a triaxial state of stress. In this situation, referred to as plane strain state, a material has its lowest resistance to crack instability (rapid fracture). The critical stress intensity factor becomes now independent of the thickness of the specimens and is a true material property referred to as plane strain factor toughness K_{1c}. The difference in the stress distribution between plane stress and plane strain is shown in Figure 5.

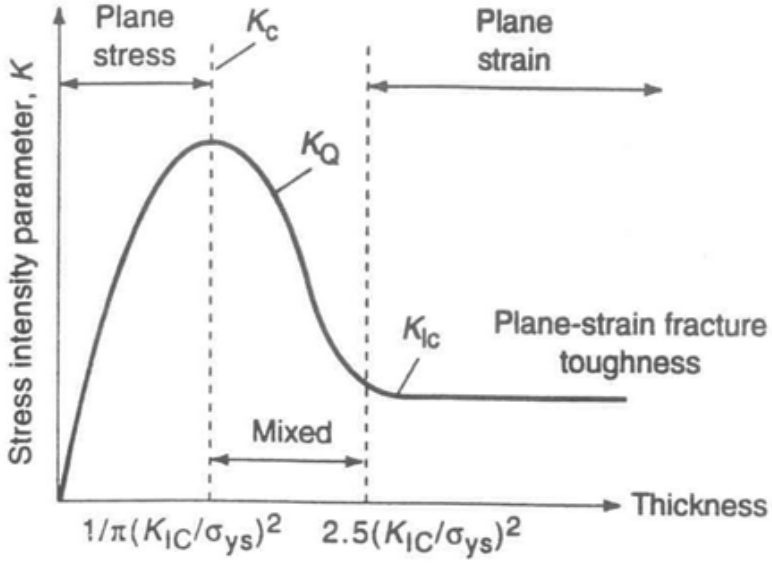


Figure 2.4: Dependence of stress intensity parameter (factor) K on thickness of specimen or cracked component [16].

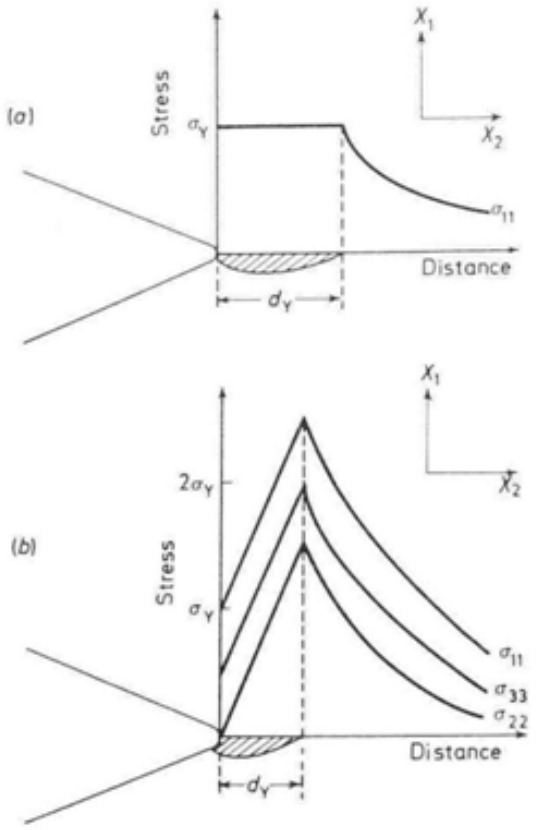


Figure 2.5: Schematic elastic/plastic stress distribution near a notch. (a) in plane stress ($\sigma_{33} = 0$); (b) in plan strain ($\sigma_{33} = 0$) [17].

2.3.3 The application of stress intensity factor for Fatigue crack propagation

As mentioned earlier, the presence of cracks can significantly reduce the strength of structures. Even small flaws can by time develop and become a real threat to a structure. Under cyclic loading, a small crack of an initially negligible size can grow until it reaches the critical size (a_c) for brittle fracture, a behaviour called fatigue crack growth. Consider a growing crack that increases by an amount $\triangle a$ due to the application of a number of cycles $\triangle N$. The rate of growth with cycles can be characterized by the ratio $\triangle a/\triangle N$ or, for small intervals, by the derivative da/dN. A value of the fatigue crack growth rate da/dN is the slope at a point on a crack length versus cycles curve [18]. Now that the Mode I stress intensity factor, K_I has been defined, it must be correlated with crack growth due to fatigue. One of the simplest and most useful equations to accomplish this is the Paris Law.

Paris Law

The stress intensity factor concept provides a very useful basis for the quantitative description of fatigue crack propagation (FCP) rates, i.e. the rate at which an existing crack advances during cyclic loading. It has been recognized that the FCP rates depend on the stress field close to the crack tip. Given this, Paris et al. [19] related the FCP, da/dN (mm/cycle), to the stress intensity factor range, $\triangle K(MPa\sqrt{m})$, using the following equation, also known as Paris law:

$$da/dN = C(\triangle K)^m$$

where

$$\triangle K = K_{max} - K_{min},$$

K_{max} = Stress intensity factor corresponding to the upper load limit,
K_{min} = Stress intensity factor corresponding to the lower load limit,
C_1m = Constants to be determined experimentally for a particular material and set of conditions.

The sigmoidal variation of the FCP rates with the alternating stress intensity factor range is shown schematically in Figure 6.

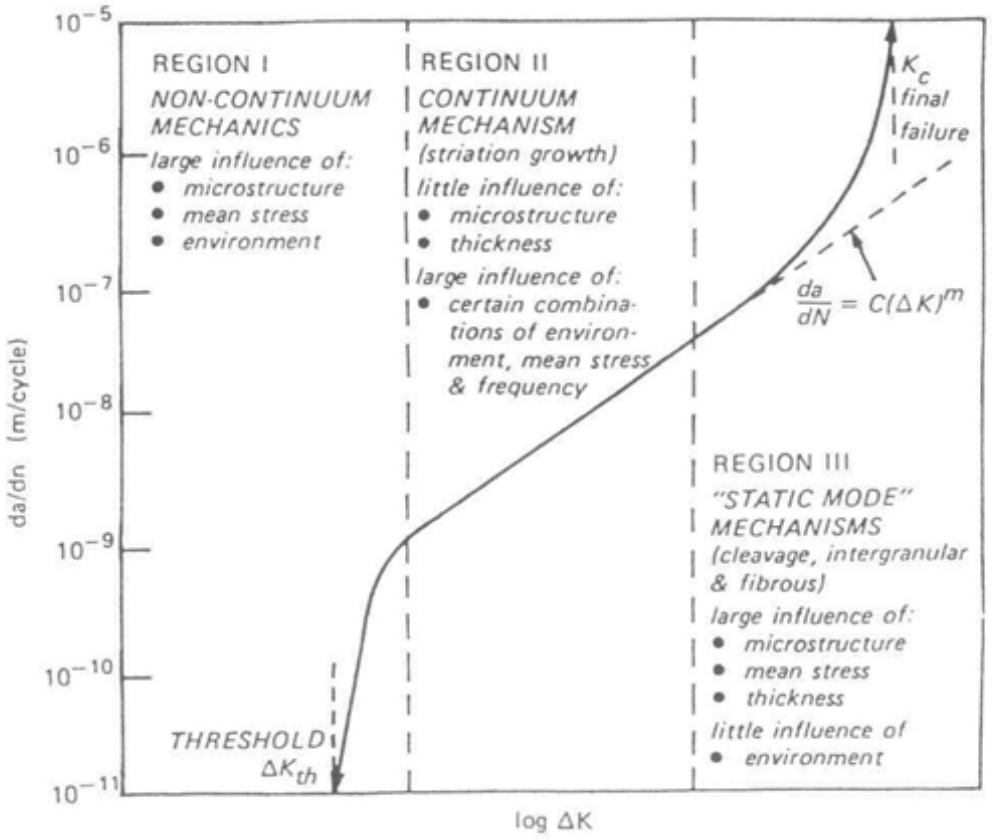


Figure 2.6: Variation of fatigue crack propagation rates da/dN with stress intensity factor range $\triangle K$ [19].

Three regimes, governed by different failure mechanisms, are discernible:

i. The near-threshold propagation rates (Region I in Fig. 6) are strongly influenced by the mean stress. The growth rates increase and the threshold alternating stress intensity ranges decrease with rising load ration R. Microstructure and environment also have a significant effect on crack propagation.

ii. In the mid growth rate regime (Region II in Fig. 6) where the Paris law is valid, failure is governed by a transgranular ductile striation mechanism and is slightly dependent on microstructure and mean stress. The environment may additionally affect growth rates.

iii. At high crack growth rates (Region III in Fig.6) where the maximum stress intensity approaches the critical stress intensity factor K_{1c} or K_c. The microstructure and mean stress has a large effect on the growth rates. Cleavage, intergranular cracking and microvoid coalescence are now the failure mechanisms in addition to striation growth.

At intermediate values of $\triangle K$, the curve is linear (stage II). At low growth rates (stage I), the curve becomes steep and da/dN approaches zero at a threshold $\triangle K$, the fatigue crack growth threshold, $\triangle K_{th}$. Crack growth does not ordinarily occur below this limiting

value of $\triangle K$. At high values of $\triangle K$, the growth rate may increase rapidly (stage III). This is due to rapid unstable crack growth just before the final fracture.

2.4 Aluminium 7075 alloy in aircraft structures

Aluminium alloys which are used in transport aircraft are alloys primarily such as 2024-T3, and the alloys having still higher strength (2014-T6, 7075-T6, 7079-T6 and 7178-T6) [26]. Where sheet is used, the Alclad form is preferred. The upper skins and spar caps of wings often are of 7075-T6 and 7178-T6, because the critical requirement is high compressive strength [20]. Aluminium alloy plate is used in a large number of aerospace applications, ranging in complexity and performance requirements from simple components through to primary load-bearing structures in aircraft such as the Airbus A340 and Boeing 777. Alloy producers develop similar alloys for different product forms and applications. The most important contribution to this flexibility is the development of a range of alloy tempers that allow optimizations and trade-offs of properties, and hence the ability to match the alloys to particular applications [21]. More specifically, the alloy property requirements vary depending on the application. A generic example is given in figure 7, which illustrates the engineering property requirements for several of the main structural areas in a transport aircraft, namely (i) fuselage and pressure cabins, (ii) wings and (iii) empennage (horizontal and vertical stabilizers). The engineering properties required for these structures are tensile strength (TS), compressive yield strength (CYS), stiffness (E), damage tolerance (DT) properties: (fatigue, fatigue crack growth, fracture toughness), and corrosion (general and stress corrosion). The rankings of the requirements differ for different areas, but there is much commonality [22].

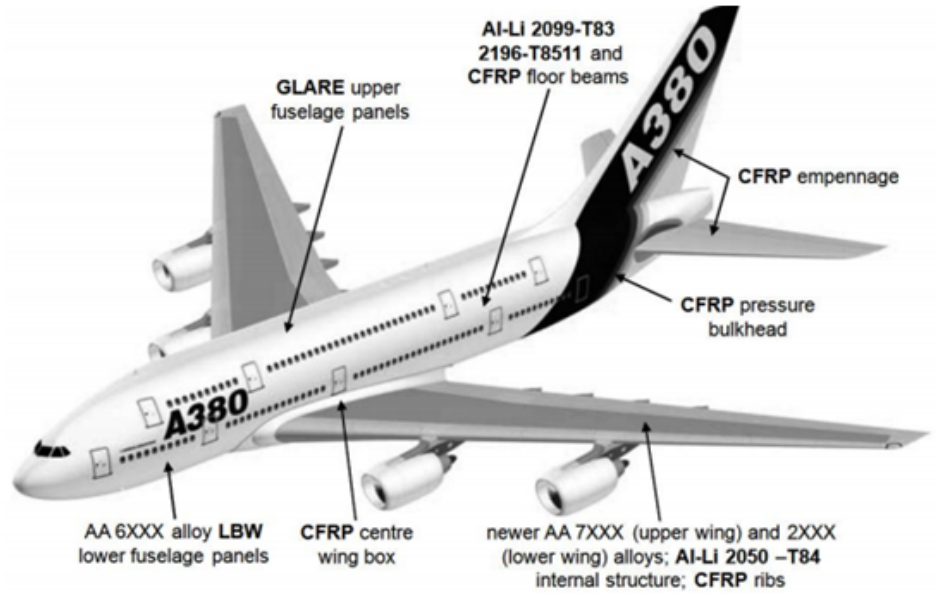


Figure 2.7: Advanced materials/processes for major structural areas of the Airbus A380 [23].

2.4.1 Aluminium Alloy AA7075

General 7075 grade has outstanding characteristics and is used as high strength material for highly stressed structural parts and mainly used in manufacturing aircraft and other aerospace applications.

a) Composition of AA7075

With zinc as the dominant alloying element, the 7075 aluminium alloy was the first alloy of high strength composed by Al-Zn-Mg-Cu that was able to successfully combine the benefits of the inclusion of chromium to develop high stress-corrosion cracking resistance in sheet products. The composition of the AA 7075 aluminium alloy consists of 87.1 - 91.4% aluminium, while the zinc is present in a concentration of 5 - 6%. It also contains 2.1 - 2.9% magnesium, 1.2 - 2% copper, and no more than 0.5% of silicon, titanium, chromium, iron, manganese, and other constituting metals as summarized in Table 3[24].

Cu	Mg	Si	Fe	Mn	Zn	Ti	Cr	Other elements Each	Total
1.2 2.0	2.1 2.9	0.4	0.5	0.3	5.1 6.1	0.2	0.18 0.28	0.05	0.15

Table 2.3: AA 7075 Typical Chemical Composition Limits (in %) [24].

The composition of the 7075 aluminium alloy series makes it the one that offers the most strength, to the extent that it is comparable to that of many steels. Although this is one of its most appealing properties, it is the combination with the rest of its properties that makes it the perfect candidate for highly stressed structural parts; its low density, great machinability, and its great ability to resist corrosion. It is worth noting that compared to other aluminium alloys; its corrosion resistance is relatively low, although it is significantly better than many of the 2000 alloys. The use of the 7075 aluminium usually parallels with the use of AA2024, but the 7075 will remain the first choice when the job requires higher mechanical properties.

In our study, the material of focus is AA7075 -T6. The AA 7075-T6 is a 7075 aluminium alloy with a T6 temper, which is achieved bya solution heat treatment, at 480 $^{\circ}C$, quenching, and then artificially ageing it at 120 $^{\circ}C$ for 24 hours. This process guarantees to achieve the peak strength of this alloy [25].

b) AA7075 T6 Properties

Table 4 below gives more details about the typical physical, mechanical thermal properties of AA7075 T6 [25].

Properties	Metric
Density	3.0 g/cm^3
Melting point	483$^{\circ}C$
Tensile strength: Ultimate (UTS)	560 MPa
Yield strength: Yield (Proof)	480 MPa
Shear strength	330 MPa
Shear Modulus	26 GPa
Fatigue strength	160 MPa
Elastic modulus	70 GPa
Poisson's ratio	0.33
Elongation at break	11%
Brinell Hardness	150
Thermal Properties	
Thermal expansion	23.2 ($10^{-6}/^{o}C$)
Thermal conductivity	130 W/mK

Table 2.4: Typical physical, mechanical thermal properties of AA 7075 T6 [21].

2.5 Repairability of aircraft structure

Aircraft structural members are designed to perform a specific function or to serve a definite purpose. The primary objective of aircraft repair is to restore damaged parts to their original condition. Very often, replacement is the only way this can be done effectively. When repair of a damaged part is possible, first careful study of the part is required to fully understand its purpose or function. Strength may be the principal requirement in the repair of certain structures, while others may need entirely different qualities. For example, fuel tanks and floats must be protected against leakage; cowlings, fairings, and similar parts must have such properties as neat appearance, streamlined shape, and accessibility. The function of any damaged part must be carefully determined to ensure the repair meets the requirements [26].

In general, failures occur when a component or structure is no longer able to withstand the stresses imposed on it during operation. Commonly, failures are associated with stress concentrations, which can occur for several reasons including . Design errors, e.g. the presence of holes, notches, and tight fillet radii; . The microstructure of the material may contain voids, inclusions etc.; . Corrosive attack of the material, e.g. pitting, can also generate a local stress concentration [27].Structural damage of an aircraft includes dents, scratches, cracks, and buckles. The damage to an aircraft is typically discovered by a visual inspection but can also be discovered because of impaired flight performance. Damages may be grouped into four general classes as follow:

1. **Negligible damage**.

2. **Damage Repairable by Patching**.

3. **Damage Repairable by Insertion**.

4. **Damage Necessitating Replacement of Parts**.

In many cases, the availabilities of repair materials and time are the most important factors in determining if a part should be repaired or replaced [28].

2.6 Criteria of repairs in sheet metal in aircraft by patching

In aircraft construction, stressed skin is a form of construction in which the external covering (skin) of an aircraft carries part or all of the main loads. Stressed skin is made from high strength rolled aluminium sheets. Stressed skin carries a large portion of the load imposed upon an aircraft structure. Various specific skin areas are classified as highly critical, semi-critical, or noncritical. Minor damage to the outside skin of the aircraft can be repaired by applying a patch to the damaged zone. In any aircraft sheet metal repair, it is critical to maintain original strength, maintain original contour, and minimize weight [29-31].

The Basic Principles of Metal Repair in the Aircraft structures is firstly to perform an inspection of the damage and getting an accurate estimate of the type of repair required. The inspection includes an estimate of the best type and shape of repair patch to use; the type, size, and the number of rivets if needed; and the strength, thickness, and kind of material required to make the repaired member no heavier (or only slightly heavier) and just as strong as the original. When any component or group of components has been damaged, both the damaged members and the attaching structure must be investigated, since the damaging force may have been transmitted over a large area, sometimes quite remote from the point of the original damage [32].

2.7 Crack repair techniques used in aircraft crack repair

Cracks are surface or subsurface fissures that develop in a material. Propagation energy derived from mechanical, thermal, chemical, and metallurgical effects, or a combination of these may influence crack initiation and growth. Various types of cracks exist in metals and can be categorised as cooling, solidification, centreline, crater, grinding, pickling, heat treatment, machining tears, plating, fatigue, creep, stress corrosion and hydrogen cracks. Cracks can grow and lead to a complete fracture of the component posing significant threats to component life and may lead to serious injuries or loss of life. Brittle fracture in metals occurs with little or no visible warning [33,34].

Discovery of any cracks warrants immediate interventions to stop the cracks before they propagate to the point of fracture. Dexter et al [33] defined fracture as rupture in tension or rapid extension of a crack which leads to gross deformation, loss of function or serviceability, or complete separation of the component. When fracture critical cracks are detected in mechanical components and structures, several methods are employed to

arrest or stop them from further growth or propagation. The choice of the repair technique or a combination to be used depends on many factors which include the nature of the crack, crack position, crack orientation, crack size, crack accessibility, component application, expected repair precision, availability of tools, metal type, component thickness and the required expertise [35]. The commonly employed techniques Common crack repair techniques in metals are as follow:

- Metal crack stitching.
- Vee-and-weld.
- Stop-hole technique.
- Laser additive crack cladding.
- Pulsed electron beam irradiation.
- Grinding.
- Hammer peening.
- **Bonded crack patches**.

In our study, the focus will be on cracks as the type of damage and only the bonded patches crack repair techniques will be discussed in the following section. An example of the bonded patches crack repair techniques is shown in figure 8.

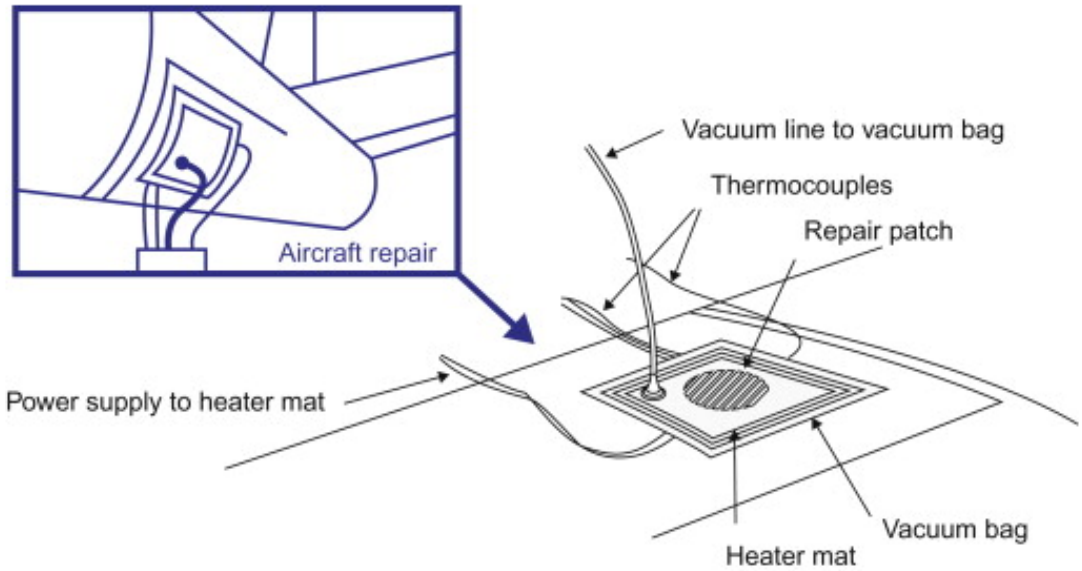


Figure 2.8: Composite bonded repair patch being applied to an aircraft structure [36].

2.8 Bonded crack patches

These can be in the form of bonded composite patches, carbon, boron, glass fibres reinforced polymer patches or doublers or splice plates [35-40]. The repair of cracked

components by an adhesively bonded composite patch has gained acceptance in aerospace structures [41]. The technique was found to be effective when optimally used. Emdad, and Al-Mahaidi [37] studied the effect of prestressed carbon fibre reinforced polymers (CFRP) patches on crack growth of centre-notched steel plates. The authors found that multi-layers of CFRP increased the fatigue life of the plate up to 30% and showed an increase in the cracked life of between 6.5 times and 10 times as also reported by Srilakshmi et al [40]. Increases in pre-stressed patches lowered the crack growth rates thus increasing the crack life of the specimen. The pre-stressing technique proved to be effective and the authors recommended that the correct use of pre-stressing, in general,can lead to increased fatigue life of cracked members by applying a compressive force to the crack edges. The authors further indicated that the compressive force will impose artificial crack closure on the flawed structure arresting its further growth. It was postulated that the method cannot be used for double-sided repairs due to limited access and the use of the pre-stress force on one side can cause significant bending moments to the member leading to unforeseen failure, as also cautioned by Ahn and Basu [42]. Doublers or splice plates can be used to repair through-thickness cracks by adding them to either increase a cross-section or provide continuity at a cracked cross-section which in turn reduces stress ranges and can be added after the repair is made [35].

2.9 Advantages and disadvantages of composite patches

The most common repair/reinforcement technique is the mechanical fastening of a plate over the damaged area to redistribute the load and reduce the stress concentration in the damaged zone. The mechanical fastening technique is less desirable, because of the non-uniformity of the stress distribution and the creation of stress concentration zones around the fasteners. The primary advantages of composite materials for repair are their high strength, relatively low weight, and good corrosion resistance [43]. Compared to mechanical methods, such as riveting or bolting, for attaching the patch, adhesive bonding provides very efficient load transfer into the patch from the cracked component and introduces minimum stress concentrations into the structure [44]. While mechanically fastened patches transfer stress with concentrated loads (bolts or rivets), bonded composite repairs transfer stress along the entire surface area of the patch, providing much more efficient load transfer.

Examples of the use of adhesive bonding can also be found in many aircraft components including fuselage longerons, fuselage skin panel splice areas, and wing stiffeners. In recent years, the advances in adhesive bonding techniques and advanced composites have sparked new interest in repairing cracked aircraft components with bonded composite patches in addition to the fabrication of new fatigue-resistant parts [45]. Figure 9 from Baker [46] shows a comparison of bolted vs. bonded repair strengths as simulated by symmetric lap shear joint. Baker conducted a comparison test of a bolted composite/aluminium joint and the same joint bonded with adhesive. Results showed more than a 300 percent increase in strength of the bonded joint over the bolted joint.

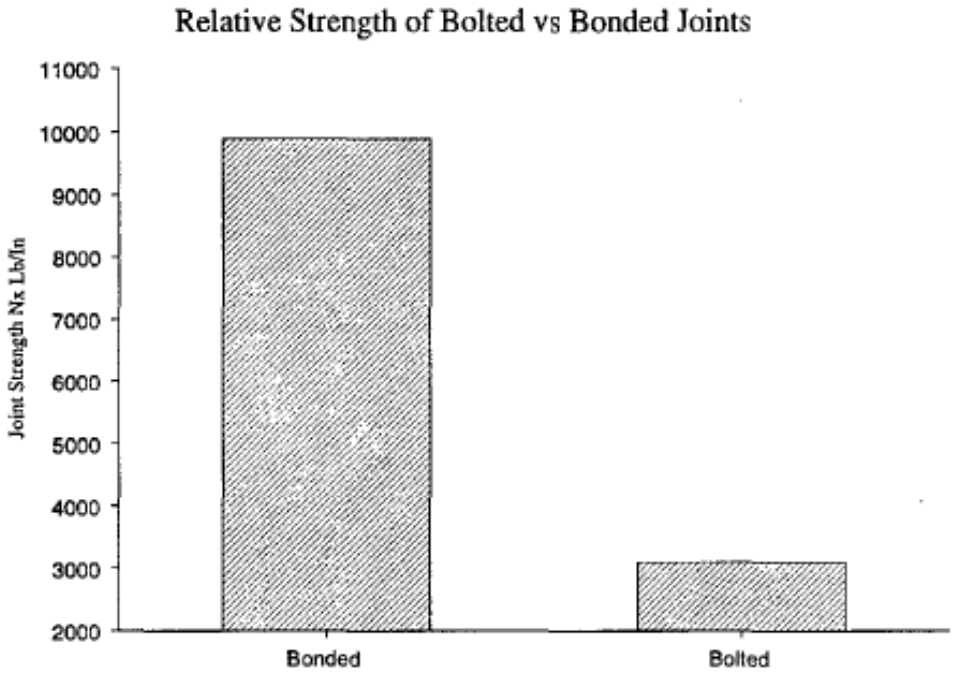


Figure 2.9: Bolted vs. Riveted Repair Strength [46].

A well-designed and bonded patch causes the stress intensity at the crack tip to reach a limiting factor, no matter how long the crack length becomes. This is shown schematically in figure 10 [47,48]. Further, a well-designed patch, because it efficiently carries the stress of the aircraft part, significantly reduces the need for short inspection intervals. This can be seen in figure 11 [47,49]. Also, adhesive bonding provides a sealed interface, protecting the damaged area from corrosion and reducing the threat of stress corrosion cracking.

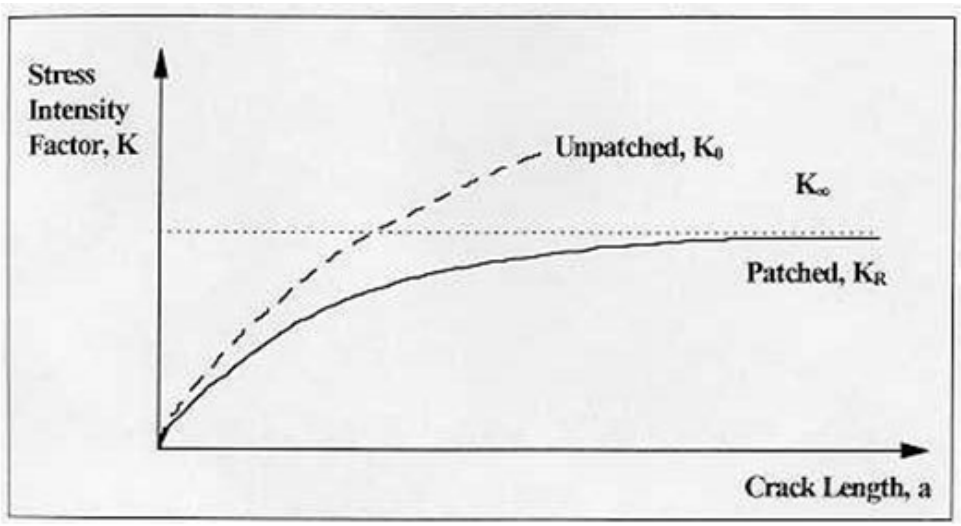


Figure 2.10: Reduction in the Stress Intensity Factor with Repair [47].

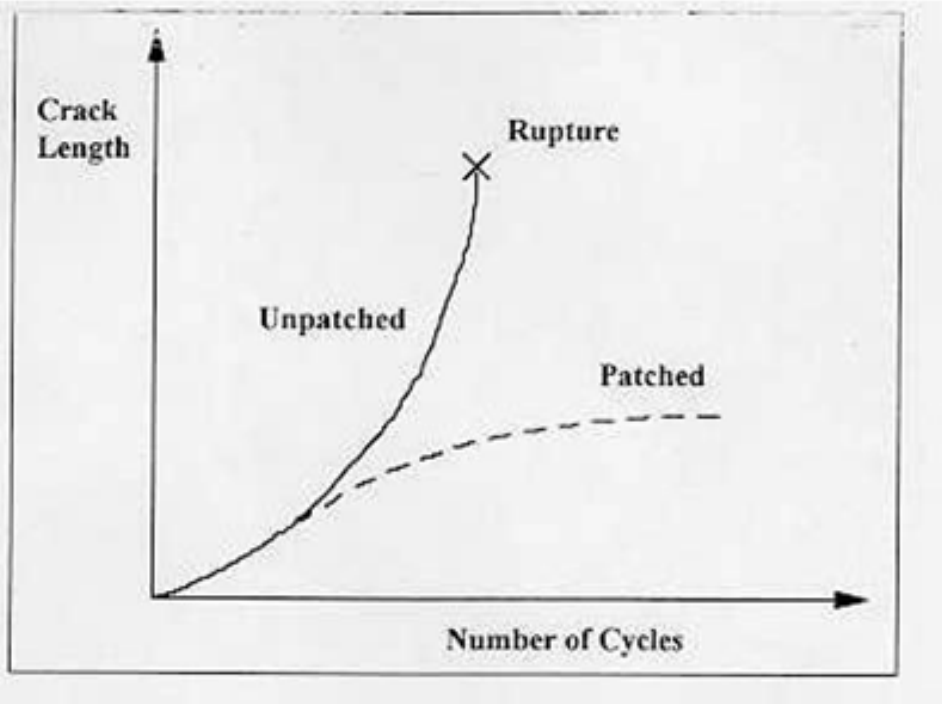


Figure 2.11: Effect of Patching on the Lifetime of a Cracked Structure [47].

Because composite materials are much stronger and stiffer than metallic materials, patches can be made approximately three times thinner. This provides not only a weight savings, but the air stream is not affected nearly as much with composite patches. Lastly, the biggest advantage gained by performing bonded repairs is the lack of additional damage in the repair area. No bolts or rivets are needed, allowing maintainers to simply fix the area without further weakening of the structure. While crack patching is a technology with a lot of advantages, it has its share of disadvantages [44]. For one, performing a bonded repair is more complicated than performing a bolted or riveted repair. Some technical equipment is involved which requires specialized training. Also, field repairs are more difficult to perform because a lot of the materials, such as the adhesives and composites, need to be refrigerated and have short shelf lives, usually about six months. Besides the repair process, there are also some concerns with material compatibility. Composite materials, such as boron/epoxy and graphite/epoxy, have lower coefficients of thermal expansion (CTE) than aluminium. This creates residual stresses in the repair that could lead to stress corrosion cracking. As a last note, some composites, particularly graphite, can create galvanic corrosion problems when in direct contact with aluminium.

2.10 History and past efforts of aircraft crack repair using composite patches

This section details past efforts to better understand bonded composite repairs. Because of the importance of addressing ageing aircraft concerns and the potential advantages of bonded composite repairs, much research has gone into characterizing bonded repairs, both experimental and theoretical, Some of the first research performed in this

area was done by Baker at the Australian Aeronautical and Maritime research laboratory [44],[50-52]. Baker characterized crack growth in aluminium panels utilizing edge cracked panels bonded together over a honeycomb core. He obtained constant crack growth rates while the crack was underneath the patch. Every crack tip has an associated stress intensity factor, K. When a patch is applied to a crack, the stress in the panel is lowered, resulting in a new repaired stress intensity factor. Baker showed that the latter remains approximately constant and stable within the composite patch bondline. Denney [47] later confirmed this same mechanism for the repair of thin aluminium panels ($t = 1mm$). Baker also researches the effect of debonds and elevated temperature on bond strength and durability. His research showed that as debond size increased, bondline life decreased. His elevated temperature testing showed that as temperature increased, life decreased. This effect is due to an increase in stress intensity factor with elevated temperature as a result of increased adhesive plasticity and decreased adhesive shear stiffness. This initial work was used to validate bonded repairs as a viable alternative to mechanically fastened repairs.

One of the United State's first efforts in bonded repair research was the Primary Adhesively Bonded Structure Technology (PABST) program of 1978. One of the primary objectives of the Air Force multidiscipline Primary Adhesively Bonded Structure Technology (PABST) Development Program was to achieve significant improvement in the durability of primary fuselage structure through the development and validation ultimately by a full-scale test of adhesively bonded structures technology [53]. This program, which involved testing a fuselage manufactured completely by adhesive bonding, showed that adhesive bonds were damage tolerant and durable. Adhesive bonds were found to have higher shear strength than the aluminium adherends. The PABST program also monitored three debonds located in various fuselage splice joints. The debonds did not decrease joint strength over the life of the program. This testing concluded that most debonds, except those at the edges of the patch, could go unrepaired. As a result of this program, confidence in bonded repairs was substantially increased. Fredell [30] performed tests using both boron/epoxy and Glare (aluminium/fibreglass laminate) to characterize the effects of coefficient of thermal expansion (CTE) mismatch on the life of bonded repairs. His research showed that residual stresses in the metal/patch system can be significantly reduced by using patch materials with CTE's close to that of the damaged part.

2.11 Current knowledge on composite crack repair of AA7075 T6

Repair of thin-walled structures with cracks by composite patching has been established as a reliable technology by the aircraft and aerospace industry. Application of this method requires a knowledge of the design stress and knowledge in adhesive bonding.Because of the importance of the benefits of bonded patch repairs, a great deal of research has been done to characterize bonded repairs from an experimental and theoretical perspective [27]. The efforts made in previous years provide a better understanding of the repairs cracked structures (plate, panel or sheet) with bonded composite patchmostly in AA 2024 aluminium alloy. In the following section, we will discuss the significant efforts

undertaken in the area of the plate, panel or sheet repair at the Mechanical Physical Materials Laboratory (LMPM)- University of Djilalli Liabes, Sidi Bel Abbes.

2.11.1 Experimental/Numerical and Theoretical efforts

The behaviour of repaired cracks, located in aluminium alloy sheets AA2024-T3, with bonded composite patch is analyzed experimentally and numerically by **F. Benyahia, et al. [54]**. The experimental study was conducted through fatigue tests on aluminium cracked plate repaired with Carbon/epoxy patch. In the numerical analysis, the stress intensity factor at the crack front is computed using a three-dimensional finite element method. The obtained results show that the stress intensity factor at the crack front is highly reduced by the presence of the patch repair. Therefore, the fatigue life of the damaged structure can be significantly improved especially if the patch repair is applied at small crack lengths.

In the paper of **M.A. Sohail, et al. [55]**, the fatigue behaviour of a crack emanating from v-notched aluminium (AA7075-T6) plate is studied. To analyze the fatigue life of notched specimens, variable cyclic block-loading sequences is applied, including increasing (L-H) and decreasing (H-L) blocks. The authors also analyzed the effect of the stress ratio(R) on the fatigue life of aluminium specimen subjected to stepped variable cyclic loading. The obtained results show that the effect of stress ratio on the fatigue life is significant. The patch repair efficiency is found to be limited for increasing (L-H) loading blocks but is significantly higher for decreasing (H-L) cyclic loading blocks.

The performances of aged bonded composite repair of aircraft structures have been evaluated by conducting experimental and numerical investigations in the study of **F. Benyahia, et al. [56]** The accelerating ageing of the patch was realized by immersion in water for 120 days. The performance of aged and non-aged bonded composite patches in the repair of cracked structures was evaluated. The experimental results showed that the fatigue life of the aircraft structures can be significantly improved with the patch repair. The patch ageing reduces the repair performances because of the reduction of the composite rigidity by humidity absorption. Numerical results show that the stress intensity factor at the crack tip is significantly reduced by the presence of the patch. The humidity absorption increases the stress intensity factor at the crack tip and decreases the adhesive stresses. Nevertheless, the repair durability cannot be improved by the humidity absorption because it leads to a reduction of the mechanical resistance of the adhesive.

A. Albedah, et al. [57] analyzed the fatigue behaviour of a V-notch crack in an aluminium alloy AA7075-T6 plate repaired with bonded composite patch under constant and stepped variable amplitude loading. The effect of adhesive debond on the repair performance was also analyzed. The obtained results confirm the need of repairing cracks at their early creation for a maximum repair performance and showed that the increases in the debond width and the stress ratio have a large negative effect on the repair efficiency. The latter is low for increasing loading blocks but significantly high for decreasing blocks.

The fatigue behaviour of aluminium alloy 2024-T3 V-notched specimens repaired with

composite patch under block loading was analyzed experimentally by **A. Albedah, et al. [58]** Two loading blocks were applied: increasing and decreasing at two stress ratios: $R = 0$ and $R = 0.1$. Failed samples were examinedusing scanning electron microscopy at different magnifications to analyze their fractured surfaces. The obtained results show that under increasing blocks, the crack growth is accelerated for both repaired and unrepaired specimens. This is attributed to the increase of the loading amplitude in the second block. A retardation effect was observed for decreasing blocks loading in unrepaired specimens. However, this retardation effect is attenuated by the presence of the patch which lead to lower fatigue life for repaired specimens.

The objective of the study of **L. Rezgani, et al. [59]** is to analyze by experimental tests, the effect of hygrothermal ageing of the patch and the adhesive on the fatigue behaviour of a damaged AA2024-T3 aluminium plate repaired by a carbon/epoxy patch composite. The effect of the water absorption by the composite patch and the adhesive has a significant effect on the load transfer from the damaged plate and therefore on the delay of the crack propagation. The more water immersion time is important, the less is the fatigue strength of the repaired structure. The experimental results showed that the number of cycles to fracture is directly related to the ageing time.

A Albedah, et al. [60] analyzed experimentally and numerically the behaviour of fatigue crack in aluminium plates repaired with bonded composite patch. They studied the behaviour of repaired crack in AA2024-T3 and AA7075-T6 under two levels of applied fatigue stresses: maximal stresses of 70 and 120 MPa at a load ratio of 0.1. In the experimental part, the fatigue life of unrepaired and repaired notched specimens were determined. In the numerical part, the J integral around repaired and unrepaired crack tips was calculated. The numerical and the experimental results were used to plot the crack velocity (da/dN) as a function of the J integral. The analysis was completed with scanning electron microscopic observations on fracture surfaces of repaired and unrepaired specimens. It was found that the patches improve the fatigue life, but this improvement is considerably reduced with the increase in the applied fatigue load. The AA2024-T3 presented better resistance to fatigue crack propagation in both repaired and unrepaired cases.

A.Sahli, et al. [61] examined patch debonding by calculating the influence of notch tip stress on debond area and verified computational results with acoustic emission (AE) measurements obtained from specimens subjected to uniaxial tension. The FE results showed that debonding first occurs between the patch and the substrate close to the free edge of the patch followed by failure around the tip of the notch, both highest stress regions. Experimental results revealed that cement adhesion at the aluminium interface was the limiting factor in patch performance. The patch did not appear to strengthen the aluminium substrate when measured by stress-strain due to early stage debonding. Analysis of the AE signals provided insight to the debond locations and progression at the metal-adhesive interface. Crack growth from the notch in the aluminium was not observed until the stress reached a critical level, an instant before final fracture, which was unaffected by the patch due to early-stage debonding. The FE model was further utilized to study the effects of patch fibre orientation and increased adhesive strength. The model revealed that the effectiveness of patch repairs is strongly dependent upon the

combined interactions of adhesive bond strength and fibre orientation.

Under real working conditions, structures are subjected to variable and peak loads. It is well known that cracks growth rate is load history-dependent. Overloads generate crack growth retardation while underloads produce acceleration in crack growth. The effect of load history on patch repair has not been covered so far. **M.A. Sohail, et al. [62]** investigated experimentally, how a single overload peak affects the performance of a bonded composite patch repair in cracked AA2024-T3 and AA7075-T6 samples fatigued under cyclic loading. They studied the effect of three overload peaks; 9, 12 and 14 kN, applied just before and soon after the bonded composite repair. The fatigue lives and retardation cycle numbers exhibited that subjecting the test sample to overload before the repair will cause more retardation of the crack growth, while the reverse will result in less retardation, yet still better than that subjected to an overload alone. They carried out SEM observations to analyse the fractography of failed specimens subjected to overloads and observed mixed-mode fracture type.in other research **M.A. Sohail, et al. [63]** studied the effect of a single overload on the fatigue life of cracked aluminium AA2024-T3 alloy repaired with a bonded composite patch. Special consideration was given to the load history, sequence of the bonded patch repair and the overload application (repair-overload and overload-repair). The two cases were compared to a reference case involving a single overload on unrepaired specimen. Fatigue tests were performed on V-notched repaired and unrepaired specimens naturally pre-cracked under constant amplitude loading. Fractographic analysis of the failed specimens was carried out to analyze the obtained fatigue results. It was found that when the crack is repaired before the overload application, the retardation effect due to the overload is attenuated by the patch. Nonetheless, when the patch precedes the overload, very high fatigue life extension of about 38.9 times is observed due to a cumulative retardation effect of the two events (patching and overload).

The technology of bonded composite repair has displayed its effectiveness in increasing the residual fatigue life of cracked aeronautical structures. However, two major phenomena reduce the performance of this technique; the adhesive debond by the fatigue load and the thermal residual stresses due to the adhesive curing. **A. Albedah, et al. [64]** analyzed experimentally and numerically the effects of the adhesive debond and the thermal residual stresses on the fatigue life of repaired cracked aluminium plate. Fatigue tests were performed on cracked 2024-T3 aluminium alloy specimens repaired using carbon/epoxy patches that have artificial disbonds, to highlight the adhesive debond effect. Fatigue tests were performed on repaired specimen bonded with elevated temperature cure and ambient temperature cure adhesives to evaluate the effect of the thermal residual stresses. To analyze the effects of the two phenomena, the finite element method was used to compute the stress intensity factor at the front of repaired cracks. The experimental results indicate that fatigue life is significantly reduced by the presence of an initial adhesive debond and thermal residual stresses. The numerical results are consistent with the experimental observations, and reveal that the stress intensity factor increases with an increase in the debond width. Moreover, it also increases in the presence of thermal residual stresses.

The objective of the study of **A. Benkheira, et al. [65]** is an experimental analysis of the repair of a laminate composite damaged by central circular notch. The effects of the repair technique used, single or double patch, the nature of the patch and its stacking

sequence, were highlighted. The main results show that the performance of the repair depends on several parameters, namely the nature of the patch and its stacking sequence, the technique repair used.

The effect of single overload on the fatigue crack growth in AA2024-T3 and AA7075-T6 alloys was analyzed by **A. Albedah, et al. [66]**. Fatigue tests under constant amplitude loading with overload peak were carried out on V-notched specimens. Fractographic analysis was used as a principal approach to explain the crack growth retardation due to the overload. Scanning electron microscopy (SEM) analyses were conducted on the fractured surface of failed specimens to study the retardation effect. The obtained results show that the overload application generates a plastic zone in both aluminium alloys. The generated plastic zone is three times larger in the case of 2024-T3 compared to 7075-T6, and thus, a significant crack retardation was induced for 2024-T3. The retardation effect due to the overload for 2024-T3 and 7075-T6 lasted for about 10 mm and 1 mm, respectively, from the point of overload application.

2.11.2 Numerical Analysis using Finite Element Analysis FEA

Numerical analysis is a powerful tool, which can provide very useful information regarding the mechanical performance of composite bonded repairs. The Finite element method (FEM) provides the means to conduct virtual experiments and analyse the behaviour of the cracked plate under various configurations. The determination of the stress intensity factor K or the integral J at the crack tip is one of the most studied parameters, where it can be a very effective means to study the performance of the composite patches used for repairing cracks and defects. The following section will discuss research that has applied numerical methods to analyze the behaviour of cracks structures repaired by composite patches.

In the study of **F. Benyahia, et al. [67]** the adhesive damage in bonded composite repair of aircraft structures is analyzed using a three-dimensional finite element method. Four different patch shapes were chosen to analyze the adhesive damage: rectangular, trapezoidal, circular and elliptical. The modified damage zone theory was implemented in the FE code to evaluate the adhesive damage. The obtained results show that the rectangular shape is beneficial for the repair durability but it presents a disadvantage for the repair efficiency. The elliptical patch is found the optimal whereas the trapezoidal patch represents the worst shape among the four types analyzed.

Other research of **F. Benyahia, et al. [68]** in this study, the mechanical properties of the aged adhesive Adekit A140 epoxy were used in a finite element model to evaluate the effect of water absorption on the adhesive damage in bonded composite repair of aircraft structures. The damage zone theory was implemented in the finite element code to achieve this objective. In addition, the effect of the water absorption, by the adhesive, on the repair efficiency was analyzed by computing the stress intensity factor at the crack tip. The obtained results show that, when the water absorption increases, the adhesive loses its rigidity, which reduces the repair durability. Besides, it leads to the increase of the stress intensity factor at the crack tip indicating a reduction in the repair efficiency.

In the paper of **M. Salem, et al. [69]**, a three-dimensional finite element method is used to analyze the J integral for repaired cracks in plates with bonded composite patch and stiffeners. For elastic behaviour, the effect of cracks, the thickness of the patch and properties of the patch are presented for calculating the J integral. For elastic-plastic,the obtained results show that the presence of the composite patch and stiffener reduces considerably the size of the plastic zone ahead of the crack.

In bonded composite repair of aircraft structures, the damage of the adhesive can thus reduce significantly the efficiency and the durability of the bonded composite repair. The adhesive damage models using critical zone have proven their effectiveness due to the simplicity and applicability of the damage criteria in these models. This is the scope of the research of **Ibrahim Nour Chafak, et al. [70]** they analyzed the effects of the patch thickness and the adhesive thickness on the damage in bonded composite repair of aircraft structures by using modified damage zone theory. The obtained results show that, when the thickness of adhesive increases, the damage zone increases and the adhesive loses its rigidity, inversely when the patch is reduced the adhesive damage becomes more significant.

Many military and commercial ageing aircrafts flying beyond their design life may experience severe crack and corrosion damage, and thus lead to catastrophic failures. **M. Berrahou, et al. [71]** used a finite element model to evaluate the effect of corrosion on the adhesive damage in bonded composite repair of aircraft structures. The damage zone theory was implemented in the finite element code to achieve this objective. Also, the effect of the corrosion, on the repair efficiency was studied. Four different patch shapes were chosen to analyze the adhesive damage: rectangular, trapezoidal, circular and elliptical. The modified damage zone theory was also implemented in the FE code to evaluate the adhesive damage. The obtained results show that the adhesive damage localized on the level of corrosion and in the sides of the patch, and the rectangular patch offers high safety because it reduces considerably the risk of the adhesive failure.

In another study **R. Mhamdia, et al. [72]** used the three-dimensional finite element method to analyze the effects of the thermal residual stresses due to the adhesive curing on the efficiency of bonded composite repair of cracked aluminium panels. The variation of the stress intensity factor at the crack tip is used for evaluating the repair performances. The effects of the geometrical and the mechanical properties of the composite patch and the adhesive on the repair efficiency were highlighted. The obtained results show that the thermal residual stresses significantly increase the stress intensity factor at the crack tip which reduces the repair efficiency.

The objective of the part I study of **M. Elhannani, et al. [73]** is to analyze by three-dimensional finite element method, the effect of the position, number and shape of the bonding defect on the shear stress distribution in the adhesive layer used to joint two aluminium AA2024-T3 plates. Numerical analysis showed that the bonding defect presence affected negatively the value of the shear stresses in the adhesive layer and therefore the durability of the assembly. Whatever the size of the defect, the maximum stresses are always localized at the adhesive edge. If the overlap length is important, the effect of the defect on joint strength is lower.

M. Salem, et al. [74] presented a three-dimensional finite element method analysis of repairing plate with bonded composite patch subjected to tensile load. The effect of the corrosion on the damage of the adhesive (FM73) in the length of the inclined crack is presented. The obtained results show that the increase of ratio of the damaged zone is due to the increase of the inclination of the crack against it leads to the decrease of the values of the stress intensity factor in mode I (K_I).

The plasticity in the crack region has a great influence on the fatigue life of aircraft structures, and the adhesive damage has negative effects on the durability of bonded joints. In the research of **N. Demmouche, et al. [75]** the interaction effect between the crack tip's plasticity and adhesive damage in a bonded composite repair of cracked metallic panels was analyzed using the three-dimensional finite element (FE) method. A damage criterion was implemented in the FE model to evaluate the damaged zone in the adhesive layer, and the plastic strain around the crack was extracted from the FE calculations. In this research, they demonstrated that the applied load has very significant effects both on crack plasticity and adhesive damage, but the ductility of the repaired material has no significant effect on adhesive failure. Fatigue tests and SEM observations were performed to validate the numerical results.

Bonded repairs using composite patches over metallic structures have been evaluated as a cost-effective method to increase the life of damaged structures. The J-integral is a widely applied fracture mechanics parameter relating to the energy release associated with crack growth and is a measure of the deformation intensity at the crack tip. In practice, the calculated J-integral can be compared with a critical value for the material under consideration to predict fracture. The study of **L. Zouaambi, et al. [76]** aimed at providing an overview of the behaviour of a cracked plate of AA7075-T6 alloy repaired with a boron-epoxy patch bonded with FM73 Adhesive layer. The Finite Element Method (FEM) using Abaqus Software 6.14 predicted the performance. The results show a considerable decrease in the value of the J-integral. This is due to the beneficial effect of the patch on the stress state at the crack tip. The best results were obtained from a unidirectional composite fibres orientation of 0°, where the fibres oriented parallel to the direction of load. A parametric analysis has been carried out to evaluate the effect of lay-up, load variation and crack mouth opening on the J-integral. It was found that the crack mouth opening displacement (CMOD) was reduced by 90-97%.

2.12 References

[1] Morris et al, "Fatigue failure in metals," in ASM Handbook, Volume 19, Fatigue and Fracture Anonymous 1996.
[2] R. W. Hertzberg, Deformation and Fracture Mechanics of Engineering Materials. (4th ed.) United States of America: Wiley, 1996.
[3] D. Michael Sangid, "The physics of fatigue crack initiation," International Journal of Fatigue, vol. 57, pp. 58, 2013, 2012.
[4] B. S. Basinski ZS, "Fundamental aspects of low amplitude cyclic deformation in face-centred cubic crystals. "Prog Mater Sci, vol. 36, pp. 89, 1992.
[5] H. Mughrabi, "Microstructural fatigue mechanisms: Cyclic slip irreversibility, crack initiation, non-linear elastic damage analysis " International Journal of Fatigue, vol. 57, pp. 5-7, 2013.
[6] Anonymous "Fatigue," in Elements of Metallurgy and Engineering Alloys Anonymous United States of America: ASM International, 2008, pp. 243-264.
[7] GRIFFITH A.A. (1920) The phenomena of rupture and flow in solids, Philosophical Transactions of the Royal Society of London, London, vol. A-221, p. 163 198.
[8] IRWIN, G.R. (1957) Analysis of stresses and strains near the end of a crack traversing a plate, Trans. ASME, Journal of Applied Mechanics, vol. 24, p. 361-364.
[9] G R Westergaard, J A Kies and H L Smith (1958): Am. Soc. Test. Mater., Proc., Vol. 58, 640-660.
[10] HERTZBERG, R.W. (1996) Deformation and fracture mechanics of engineering materials, New York, 4th edition, John Wiley & Sons, Inc., 786 p.
[11] ASTM E399-83: Standard Test Method for Plane Strain Fracture Toughness of Metallic Materials, American Society for Testing and Materials, Philadelphia, PA 19103, USA.
[12] BS 5447: Methods of test for: Plane Strain Fracture Toughness (K_{1c}) of Metallic Materials. British Standard Institution.
[13] V Kerlins (1987). Modes of Fracture. Metals Handbook, Ninth Edition, Vol. 12, 12-71.
[14] B L Gabriel (1987). Scanning Electron Microscopy. In ref. 1, 166-178.
[15] G E Dieter (1988): Mechanical Metallurgy, McGraw-Hill Book Company, London, 351.
[16] ASTM E561: Standard Practice for R-Curve Determination. American Society for Testing and Materials, Philadelphia, PA 19103, USA.
[17] J F Knott (1973): Fundamentals of Fracture Mechanics, Butterworths, London, 94-113.
[18] Dowling, N.E., 2012. Mechanical behavior of materials: Engineering methods for deformation, fracture and fatigue. 4th edition, Prentice Hall.
[19] P C Paris, M P Gomet and W E Anderson (1961). The Trend in Engineering, Vol. 13, 9-14.
[20] https://www.totalmateria.com/Article96.htm ,2018.
[21] M.R. Mansoret al, Natural fiberpolymer composites,Biomass, Biopolymer-Based Materials, and Bioenergy, 2019.
[22] A. Baker, R. Jones, Bonded Repair of Aircraft Structures, (1988).
[23] Russell James Hugh Wanhill, Fatigue Requirements for Aircraft Aerospace Materials and Material Technologies, Structures November 2017.
[24] International Alloy Designations and Chemical Composition Limits for Wrought Alu-

minum and Wrought Aluminum Alloys, ALUMINIUM DATA SHEET ALLOY GRADE 7075, January 2015.
[25] https://www.engineeringclicks.com/7075-t6-aluminium/ , Accessed Dec 2019.
[26] Alan Baker, Francis Rose, Rhys Jones. Advances in the bonded composite repair of metallic aircraft structures. ISBN: 0-08-042699-9. Vol 1.Elsevier 2002.
[27] T.H.G.Megson. Aircraft structures for engineering students, third edition, Elsevier 1999.
[28] Basic-principles-of-sheet-metal-repair, Aviation Maintenance Technician Handbook - Airframe Volume 1 - Part 1, 2018.
[29] Baker, A.A. "Bonded Composite Repair of Metallic Aircraft Components - Overview of Australian Activities," Proceedings of the 1994 AGARD Specialists' Meeting on Composite Repair of Military Aircraft Structures. Seville, Spain: October 1994.
[30] Fredell, R.S. Damage Tolerant Repair Techniques for Pressurized Aircraft Fuselages. Ph.D. dissertation. Delft University of Technology, Delft, The Netherlands, June 1994 (WL-TR-93- 3134).
[31] Bruhn, E.F. and others. Analysis and Design of Flight Vehicle Structures. Carmel IN: Jacobs Publishing, Inc. 1973.
[32] Guidelines for Composite Repair of Metallic Structure. AFRL-ML-WP-TR-1998-4113.
[33] R. Dexter, K. Anami, P. Albrecht, B. Brakke, O. Bucak, R. Connor, G. Decorges and J. Fisher, "Manual for Repair and Retrofit of Fatigue Cracks in Steel Bridges," U.S Department of Transportation Federal Highway Administration (FHWA), New York, 2013. Processing Technology, vol. 212, pp. 2642-2651, 2012.
[34] Hashemite-University, "The Hashemite University NDT Centre: Defectology," [Online]. Available: https://eis.hu.edu.jo/ACUploads/10526/Defectology.pdf. [Accessed 26 March 2015].
[35] Tawanda Marazani et al, Repair of cracks in metals: A review, 14th Global Conference on Sustainable Manufacturing, GCSM 3-5 October 2016, Stellenbosch, South Africa.
[36] Niedernhuber M, Holtmannspötter J, Ehrlich I. Fibre-oriented repair geometries for composite materials. Composites Part B: Engineering 2016; 94: 327-337.
[37] R. Emdad and R. Al-Mahaidi, "Effect of prestressed CFRP patches on crack growth of centre-notched steel plates," Journal of Composite Structures, vol. 123, pp. 109-122, 2015.
[38] Sohail M. A. Khan, F. Benyahia and B. Bachir Bouiadjra, "Analysis and Repair of Crack Growth Emanating from V-Notch under Stepped Variable Fatigue Loading," XVII International Colloquium on Mechanical Fatigue of Metals (ICMFM17), vol. 74, pp. 151-156, 2014.
[39] D. Seo and J. Lee, "Fatigue crack growth behavior of cracked aluminum plate repaired with composite patch," Journal of Composite Structures, vol. 57, pp. 323-330, 2002.
[40] R. Srilakshmi, M. Ramji and V. Chinthapenta, "Fatigue crack growth study of CFRP patch repaired Al 2014-T6 panel having an inclined center crack using FEA and DIC," Journal of Engineering Fracture Mechanics, vol. 134, pp. 182-201, 2015.
[41] V. Sabelkin, V. S. Mall and J. B. Avram, "Fatigue crack growth analysis of stiffened cracked panel repaired with bonded composite patch," Engineering Fracture Mechanics, vol. 73, pp. 1553-1567, 2006.
[42] S. J. Ahn and P. K. Basu, "Locally refined p-FEM modeling of patch repaired plates," Journal of Composite Structures, vol. 93, pp. 1704- 1716, 2011.However, alignment prob-

lems have been reported to affect this technique.
[43] Niedernhuber M, Holtmannspötter J, Ehrlich I. Fibre-oriented repair geometries for composite materials. Composites Part B: Engineering 2016; 94: 327-337.
[44] Baker, A. A. "Repair of Cracked or Defective Metallic Aircraft Components with Advanced Fibre Composites-an Overview of Australian Work." Composite Structures. Elsevier Applied Science Publishers Ltd. Printed in Great Britain. 1984. Pgs. 153-181.
[45] Belason, E. B. "Bonded Doublers for Aircraft Structure Repair," Aerospace Engineering, July 13-18 1995.
[46] Baker, A. A. Bonded Repair of Aircraft Structures. Martinus Nijhoff, 1988.
[47] Denney, J.J. Fatigue Response of Cracked Aluminum Panel with Partially Bonded Composite Patch. MS thesis, AFIT/GAE/ENY/95D-7. School of Engineering, Air Force Institute of Technology (AU), Wright-Patterson AFB OH, 1995.
[48] Baker, A.A. "Crack Patching; Experimental Studies, Practical Applications," in Bonded Repair of Aircraft Structures. 107-172. A.A. Baker and R. Jones, editors. Dordrecht: Martinus Nijhoff Publishers, 1988.
[49] United States Air Force Academy. Calcurep for Windows: Beta Version. Colorado Springs: USAFA Department of Engineering Mechanics, May 1995.
[50] Grabovac, I, R. A. Bartholomeusz, and A. A. Baker. "Composite Reinforcement of a Ship Superstructure-Project Overview." Composites, 24: 501-509, (Number 6, 1993).
[51] Reinhart, T.J. "Surface Treatments for Bonded Repairs of Metallic Components," in Bonded Repair of Aircraft Structures. 19-29. A.A. Baker and R. Jones, editiors. Dordrecht: Martinus Nijhoff, Publishers, 1988.
[52] Baker, A.A, R.J. Chester, M. J. Davis, M.A. Retchford, and J.D. Roberts. "The Development of a Boron/Epoxy Doubler System for the F-111 Wing Pivot Fitting-Materials Engineering Aspects," Proceedings of the International Conference on Aircraft Damage Assessment and Repair. August 1991.
[53] Primary Adhesively Bonded Structure Technology (PABST) Full Scale Test Report. Douglas Aircraft Company, McDonnell Douglas Corporation. November 1980. AFWAL-TR-80-3112.
[54] F. Benyahia, A. Albedah, B. Bachir Bouiadjra, Bonded Composite Repairs of Aluminum Alloy 2024T3 Cracked Plates , ASME 2014 International Mechanical Engineering Congress and Exposition, November 14-20, 2014, Montreal, Quebec, Canada, Volume 1: Advances in Aerospace Technology.
[55] Sohail M. A. Khan, F. Benyahia, B. Bachir Bouiadjra, A.Albedah, Analysis and Repair of Crack Growth Emanating from V-notch under Stepped Variable Fatigue Loading, Procedia Engineering, Volume 74, 2014, Pages 151-156.
[56] F. Benyahia, L. Aminallah, A. Albedah, B. Bachir Bouiadjra, T. Achour, Experimental and numerical analysis of bonded composite patch repair in aluminum alloy 7075 T6, Materials & Design, Volume 73, 2015, Pages 67-73.
[57] A. Albedah, Sohail M. A. Khan, F. Benyahia, B. Bachir Bouiadjra, Experimental analysis of the fatigue life of repaired cracked plate in aluminum alloy 7075 with bonded composite patch, Engineering Fracture Mechanics, Volume 145, August 2015, Pages 210-220.
[58] A. Albedah, Sohail M. A. Khan, F. Benyahia, B. Bachir Bouiadjra, Effect of load amplitude change on the fatigue life of cracked Al plate repaired with composite patch, International Journal of Fatigue, Volume 88, July 2016, Pages 1-9.
[59] L. Rezgani, K. Madani, X. Feaugas, S. Touzain, S. Cohendozc, J. Valette, Influence

of water ingress onto the crack propagation rate in a AA2024-T3 plate repaired by a carbon/epoxy patch, Aerospace Science and Technology, Volume 55, August 2016, Pages 359-365.
[60] A. Albedah, Sohail M. A. Khan, B. Bachir Bouiadjra, F. Benyahia, Fatigue crack propagation in aluminum plates with composite patch including plasticity effect,Proceedings of the Institution of Mechanical Engineers, Part G: Journal of Aerospace Engineering, Volume: 232 issue: 11, page(s): 2122-2131, 2017.
[61] Abderahmane Sahli, Mokhtar Bouziane M, Smail Benbarek, W. Steven F, Zhang Liang, Bachir Bouadjra Belabbes, Boualem Serier, Experimental and numerical disbond localization analyses of a notched plate repaired with a CFRP patch, Structural Engineering & Mechanics 63(3):361-370, 2017.
[62] Sohail M. A. Khan Mohammed, A. Albedah, F. Benyahia, B. Bachir Bouiadjra, Effect of single tensile peak overload on the performance of bonded composite repair of cracked Al 2024-T3 and Al 7075-T6 plates, Composite Structures, Volume 193, 1 June 2018, Pages 260-267.
[63] Sohail M. A. Khan Mohammed, B. Bachir Bouiadjra, F. Benyahia, A. Albedah, Analysis of the single overload effect on fatigue crack growth in AA 2024-T3 plates repaired with composite patch, Engineering Fracture Mechanics, Volume 202, 15 October 2018, Pages 147-161.
[64] Abdulmohsen Albedah, Belabbes Bachir Bouiadjra, Faycal Benyahia, Sohail M.A. Khan Mohammed, Effects of adhesive disbond and thermal residual stresses on the fatigue life of cracked 2024-T3 aluminum panels repaired with a composite patch, International Journal of Adhesion and Adhesives, Volume 87, December 2018, Pages 22-30.
[65] A. Benkheira, M. Belhouari & S. Benbarek, Comparison of Double-and Single-Bonded Repairs to Symmetrical Composite Structures, Journal of Failure Analysis and Prevention volume 18, pages 1601-1606. 2018.
[66] A. Albedah, B. Bachir Bouiadjra, S.M.A.K. Mohammed, and F. Benyahia, Fractographic analysis of the overload effect on fatigue crack growth in 2024-T3 and 7075-T6 Al alloys, International Journal of Minerals, Metallurgy and Materials, Volume 27, Number 1, January 2020, Page 83.
[67] F. Benyahia, A. Albedah, B. Bachir Bouiadjra, Analysis of the adhesive damage for different patch shapes in bonded composite repair of aircraft structures, Materials & Design, Volume 57, May 2014, Pages 435-441.
[68] F. Benyahia, M. Fari Bouanani, A. Albedaha, B. Bachir Bouiadjrab, T. Achour, Effect of water absorption on the adhesive damage in bonded composite repair of aircraft structures, Advances in materials Research, Volume 4 Issue 2, Pages.87-96.
[69] Salem Mokadem, Bachir Bouadjra Belabbes, Mechab Belaid, Kaddouri Khacem, Elastic-plastic analysis of the J integral for repaired cracks in plates, Advances in materials Research, Volume 5 Issue 1, Pages 11-20, 2016.
[70] Ibrahim Nour Chafak, Bouanani Morad Fari, Bachir Bouadjra Belabbes, Serier Boualem, Analysis of the adhesive damage between composite and metallic adherends: Application to the repair of aircraft structures, 2016.
[71] Berrahou Mohamed, Bel Abbes Bachir Bouiadjra, Analysis of the adhesive damage for different patch shapes in bonded composite repair of corroded aluminum plate, Structural Engineering and Mechanics, Vol. 64, No. 3 (2017) 323-328.
[72] R. Mhamdia, B. Serier, A. Albedah, B. Bachir Bouiadjra, K. Kaddouri, Numerical analysis of the influences of thermal stresses on the efficiency of bonded composite repair

of cracked metallic panels Journal of Composite Materials, vol. 51, 26: pp. 3701-3709, Aerospace Science and Technology, Volume 62, March 2017, Pages 122-135.
[73] M. Elhannani, K. Madani, E. Legrand, S. Touzain, X. Feaugas, Numerical analysis of the effect of the presence, number and shape of bonding defect on the shear stresses distribution in an adhesive layer for the single-lap bonded joint; Part I, Aerospace Science and Technology, Volume 63, April 2017, Pages 372-386.
[74] Salem Mokadem, Berrahou Mohamed, Mechab Belaïd, Bachir Bouadjra Belabbes, Effect of the angles of the cracks of a corroded plate in bonded composite repair, Frattura ed Integrità Strutturale, 46 (2018) 113-123; DOI: 10.3221/IGF-ESIS.46.12.
[75] N. Demmouche, A. Albedah, Sohail M. A. Khan Mohammed, B. Bachir Bouiadjra & F. Benyahia, Interaction between adherend plasticity and adhesive damage in metal/composite joints: application to bonded composite repair of metallic structures. Welding in the World volume 63, pages211-219(2019), https://doi.org/10.1007/s40194-018-0659-6.
[76] Zouambi L, Khodja M, Oudad W, Fekirini H, Moller H, Bachir Bouiadjra B. J-integral evaluation of repaired cracks in AA7075-T6 structures subjected to uniaxial tensile stresses. Polymer Testing 2019.

Chapter 3

Experimental Methods

3.1 Introduction

This Research was performed at CSIR (Council for Scientific and Industrial Research) at Material Sciences & Manufacturing (MSM) Operating Unit / Light Metals (LM) Competency Area. Tests were conducted at LM/MSM Laboratories. Machining was done at the LM workshop and at the National Laser Centre (NLC) at CSIR. Test materials utilized in this research for the bonded repair of damaged structures, as shown in Table 1, include an AA7075-T6 aluminium panel bonded to six-ply unidirectional boron fibre epoxy with the use of FM 73 adhesive layer as bonding agent. The design and fabrication of these test specimens involved:

1. Aluminium Specimen Design and Fabrication.
2. Machining of the aluminium panels to test configuration dimensions.
3. Pre-cracking of the aluminium panels.
4. Surface preparation of the aluminium panel bonding surfaces.
5. Designing of the six-ply boron/epoxy patch.
6. Bonding of the boron/epoxy patch with aid of FM 73-K adhesive layer.
7. Curing the repaired structures in a vacuum furnace.

Materials	**Thickness (mm)**
AA7075-T6 plate	2
Boron Prepreg 5521 tape 4 mil boron/epoxy Laminate with 5521 resin	0.13 per layer 0.78 (6 Layers)
FM73K Structural film adhesive	0.25

Table 3.1: Thickness of the material used for the bonded structural repair.

3.2 Materials Selection & Description

Aluminium alloy AA7075-T6 was bonded to boron/epoxy prepreg composite patch, with the use of FM73-Knit Cytec/Solvay Structural Adhesive that is widely used in the aircraft industry [1]. The dimensions of the AA7075-T6 aluminium plate were 400 mm x 160 mm x 2 mm and contained a 6 mm size rivet hole with a wire-cut notch of 1 mm length perpendicular to the length of the plate emanating from the hole's edge. The geometry of the plate and composite repair patch is represented in Figure 1. Cytec/Solvay FM73 knitted epoxy film of 0.25 mm thickness was used as an adhesive between the aluminium and the boron/epoxy patch. The structural composite layer was composed of plies, 5.2-mil thick, of Type B4/5521 unidirectional 4-mil Boron/epoxy prepreg tape from Specialty Materials, Inc.

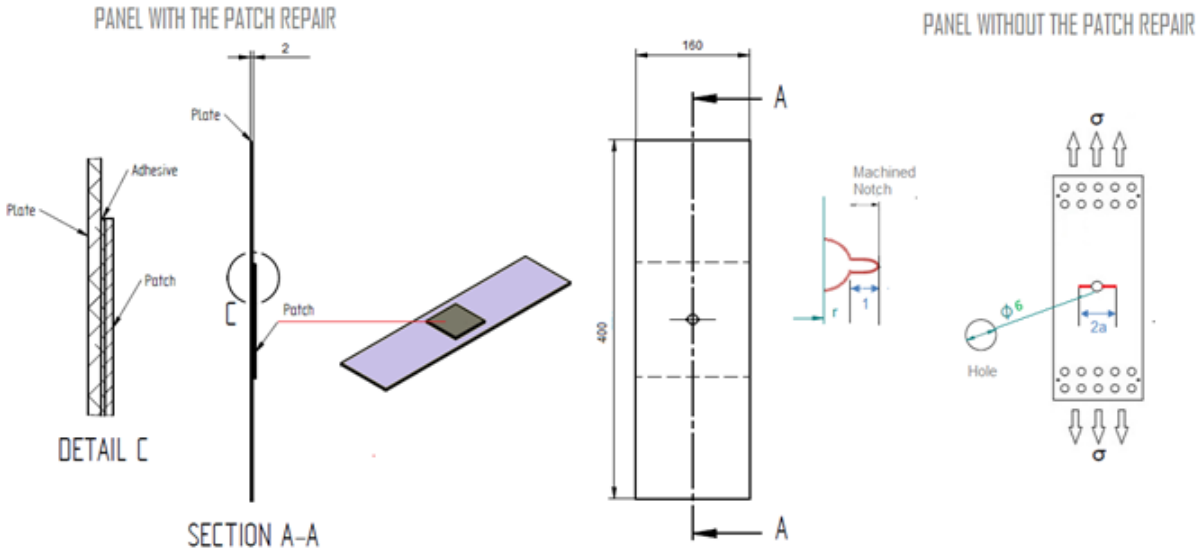


Figure 3.1: Geometry of the plate and composite repair patch.

3.2.1 Aluminium AA7075-T6

Aluminium alloys have good corrosion resistance. Their strength decreases when they are subjected to high temperatures. The 7xxx aluminium-zinc-magnesium alloys are notable for their combinations of good finishing characteristics, good general corrosion resistance, and the capability of developing high strength through ageing. AA7075 grade has high strength and is used for highly stressed structural parts such as in the manufacturing of aircraft and other aerospace applications [2]. The aluminium sheet used in this study is an aerospace-grade material that was supplied by Denel Aerostructures SOC Ltd, South Africa, purchased from Castle Metals France (Nantes Castle Metals Aerospace). The bare sheet 7075-T6 Aluminium with protective vinyl coating (specification: QQ. A-250/12) had dimensions of 2.03mm x 1219.2mm x 3657.6mm. Table 2 shows the chemical composition information provided by the manufacturer of the aluminium 7075-T6 alloy.

Material	Elements									
AA 7075	**Si**	**Fe**	**Cu**	**Mn**	**Mg**	**Cr**	**Zn**	**Ti**	**Ti+Zr**	**Other EA**
Spec. min.	**-**	**-**	**1.2**	**-**	**2.1**	**0.18**	**5.1**	**-**	**-**	**-**
Spec. max	**0.40**	**0.50**	**2.0**	**0.30**	**2.9**	**0.28**	**6.1**	**0.20**	**0.25**	**0.05**
Actual	**0.07**	**0.13**	**1.4**	**0.01**	**2.6**	**0.19**	**5.6**	**0.03**	**0.04**	**0.02**

Table 3.2: Chemical Composition Limits (in wt%) [ALUMINIUM DATA SHEET ALLOY GRADE 7075].

3.2.2 The Patch Composite: Boron Fibre Epoxy

Boron fibres are used primarily in military aviation applications to repair cracked aluminium aircraft skins. Advantages include the fact that the thermal expansion of boron is close to aluminium and there is also no galvanic corrosion potential. It is used because it has high stiffness, high tensile and compressive strength. An epoxy matrix is often used with the boron fibre. The boron fibre is difficult to use if the parent material surface has a contoured shape. The fibres have a relatively large diameter and do not flex well; therefore, they are available only as a prepreg tape product. The boron 5521 Prepreg Tape as composite patch repair was used in this investigation and is shown in Figure 2 with a SEM image of boron fibre. Their properties are shown in Table 3 [3].

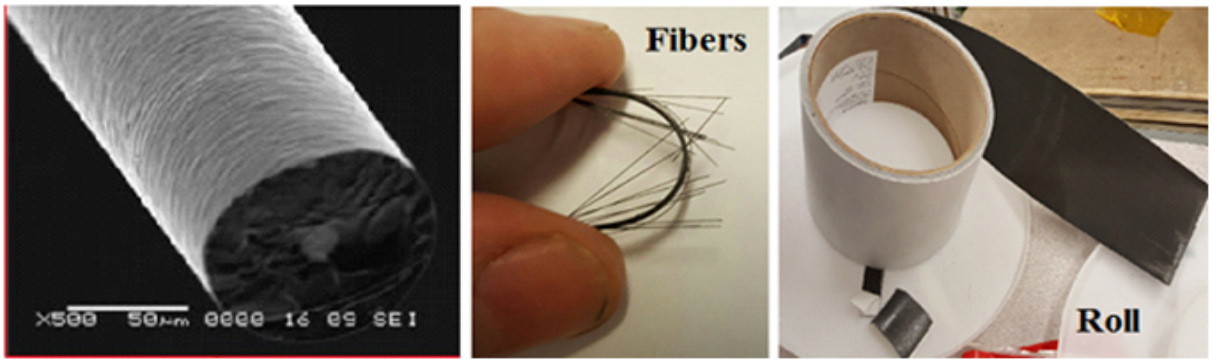


Figure 3.2: SEM image of boron fibre and boron fibre/epoxy patch composite prepreg tape.

Specialty Materials, Inc.

BORON FIBER PROPERTIES

Property	Units	4-mil Boron (Typical)
Diameter	um	102
	in.x10⁻³	4
Cross-Section	----------	
Aspect Ratio	----------	
Density	g/cc	2.61
	lbs/in³	0.094
Thermal Expansion	PPM/°C	4.5
	PPM/°F	2.5
Tensile Strength	MPa	3,600
	ksi	520
Tensile Modulus	GPa	400
	msi	58
Compression Strength (est.*)	MPa	>6000
	ksi	>900
Hardness	Knoop	3,200

BORON PREPREG TAPE PROPERTIES

Property	Units	Type B4/5521F
Cure Temperature	Deg-F	250
Test Temperature	Deg-C	R.T.
	Deg-F	
Tensile Strength	MPa	1,484
	ksi	215
Tensile Modulus	GPa	192
	msi	28
Flexural Strength	MPa	1,790
	ksi	260
Flexural Modulus	GPa	190
	msi	28
Compression Stregth	MPa	2,930
	ksi	425
Compression Modulus	GPa	210
	msi	30
Interlaminar Shear Strength	MPa	97
	ksi	14.1
Cured Ply Thickness	mm	0.13
	in.x10⁻³	5.2
Areal Weight	g/m²	290
	lbs/ft²	0.06
Density	g/cc	2.0
	lbs/in³	0.072

Table 3.3: Material Properties values of the boron fibre and boron prepreg tape [3].

3.2.3 Adhesive Layer: Cytec FM73-K

Structural adhesives are generally supplied as thin films supported on a release paper and stored under refrigerated conditions ($-18^{\circ}C$) for aerospace applications. The upper-temperature limit of $121 - 177^{\circ}C$ is usually determined by the degree of toughening required and by the overall choice of resins and curing agents. To improve the handling of the films prior to cure, film materials are frequently supported by fibres incorporated as short-fibre mats with random orientation or as woven cloth that serves to control adhesive flow during bonding and assist in bond line thickness control. This is shown in the structural film adhesive used in this investigation in Figure 3.

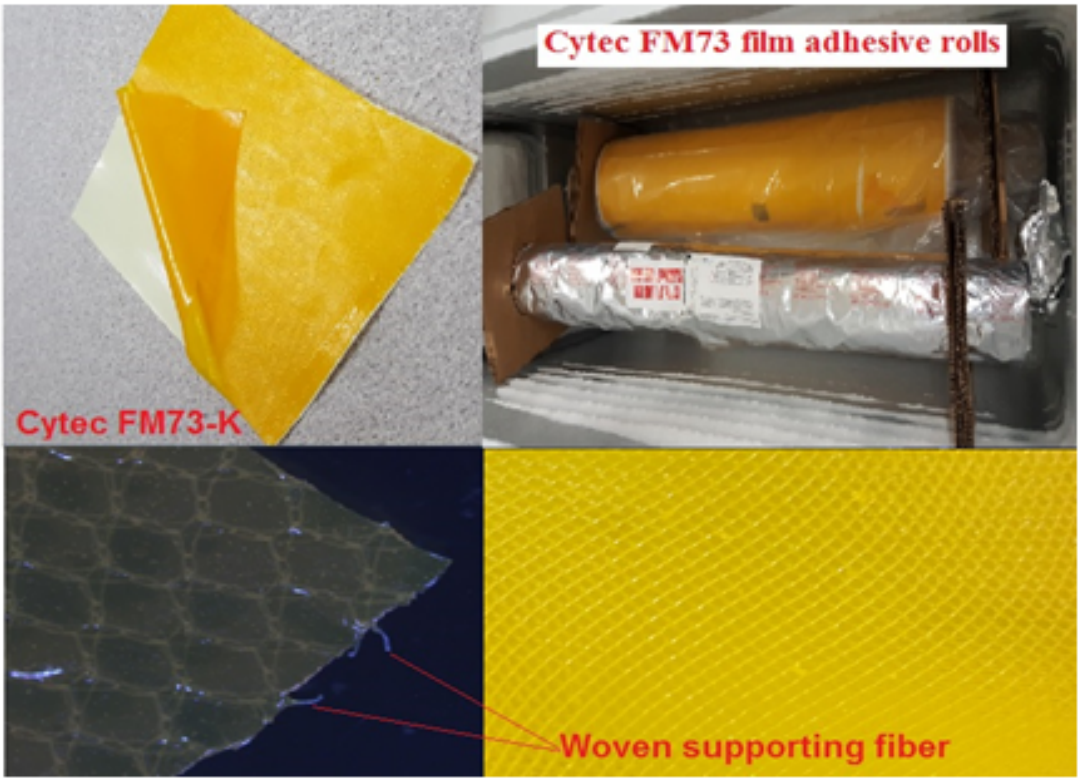


Figure 3.3: A roll of epoxy FM73 Cytec film structural adhesive with supporting fibres.

3.3 Manufacturing of the Specimens

3.3.1 AA7075-T6 Aluminium Specimen Design and Fabrication

The dimensions of the fatigue sample panels for this research investigation for repaired and unrepaired structures were chosen where the width of the panels was wider to comply with the spacing in stiffened structures of aeroplane skin.

For material characterisation of unrepaired aluminium 7075-T6, specimens were fabricated in parallel and perpendicular to the rolling directions.

For low cycle fatigue testing, unrepaired and repaired sheets were fabricated in the rolling direction as per fabrication standard of aircraft structures. The dimensions of dog-boned ASTM E8 Standard [4] test specimens machined for material characterisation are shown in Figure 4.

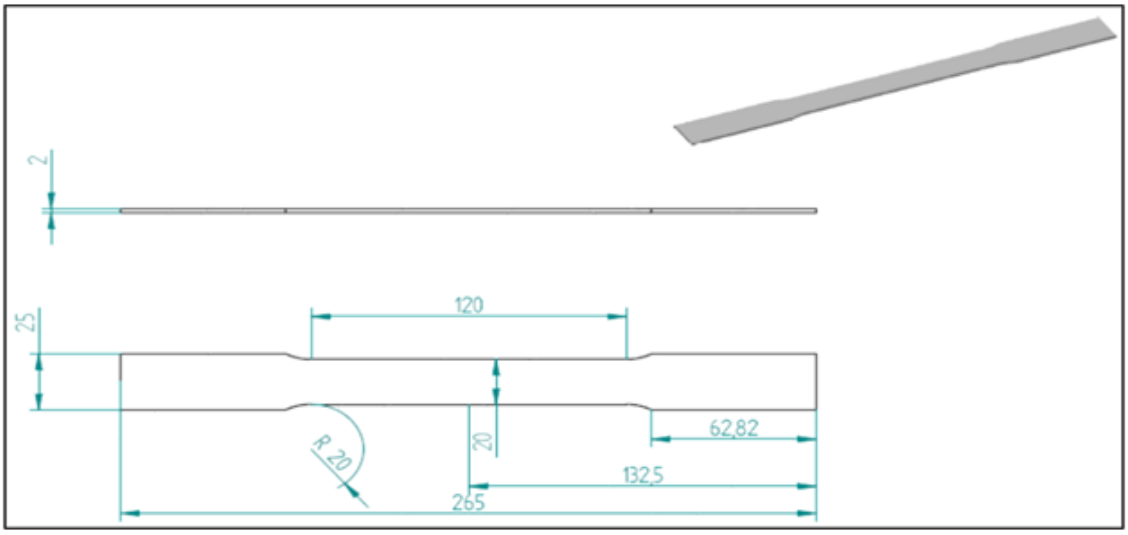


Figure 3.4: Tensile specimens with dimensions in mm machined from plate.

The aluminium sheet was first cut to smaller size sheet of 1.8m x 1.25m dimensions with a waterjet cutting machine. The sheets were sent to the National Laser Centre (NLC-CSIR) at the Laser Enable Manufacturing Unit (LEM) within the Council for Scientific and Industrial Research (CSIR) to be cut to final size. A Trumpf Lasercell 1005 was used to performed the laser cutting on the sheet and is shown in Figure 5.



Figure 3.5: The Trumpf Lasercell 1005.

Five dog bone tensile samples for material characterisations were machined, as well as four fatigue samples from the sheet. After determining the final test rolling direction, a final test panels dimension of 400 mm x 160 mm for required fatigue test panels were also machined. A notch in the centre of the fatigue test panels was created at 3 mm diameter by laser machining for machining accuracy. Then, the hole diameter was milled to the exact final dimension of 6 mm diameter rivet hole size to eliminate the possible effects from laser cutting. All edges of the final finished samples were slightly polished

for the same reason. The milling procedure was performed at the Material Science and Manufacturing unit workshop at CSIR. Figure 6 shows the plate mounted and levelled on a Leaderway milling machine to do accurate milling on the rivet hole.

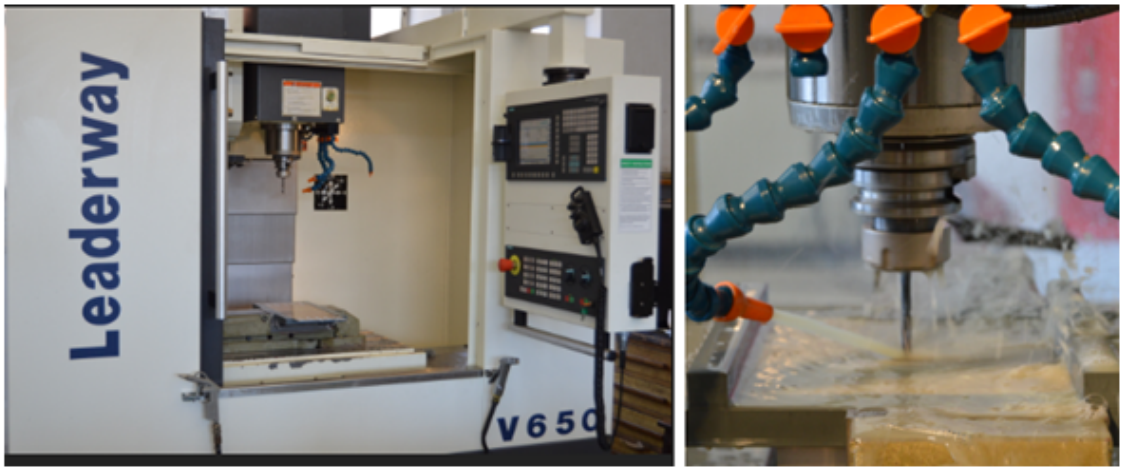


Figure 3.6: Rivet hole milling process using a Leaderway V650 machine.

Four clamping plates were designed in order to clamp the samples to withstand the fatigue testing and to ensure uniform load distribution and an aligned setup. A schematic of the fatigue testing clamping plate is shown in Figure 7. Figure 8 shows the panel sketch dimensions, final machined samples and fitted with the clamping plate respectively.

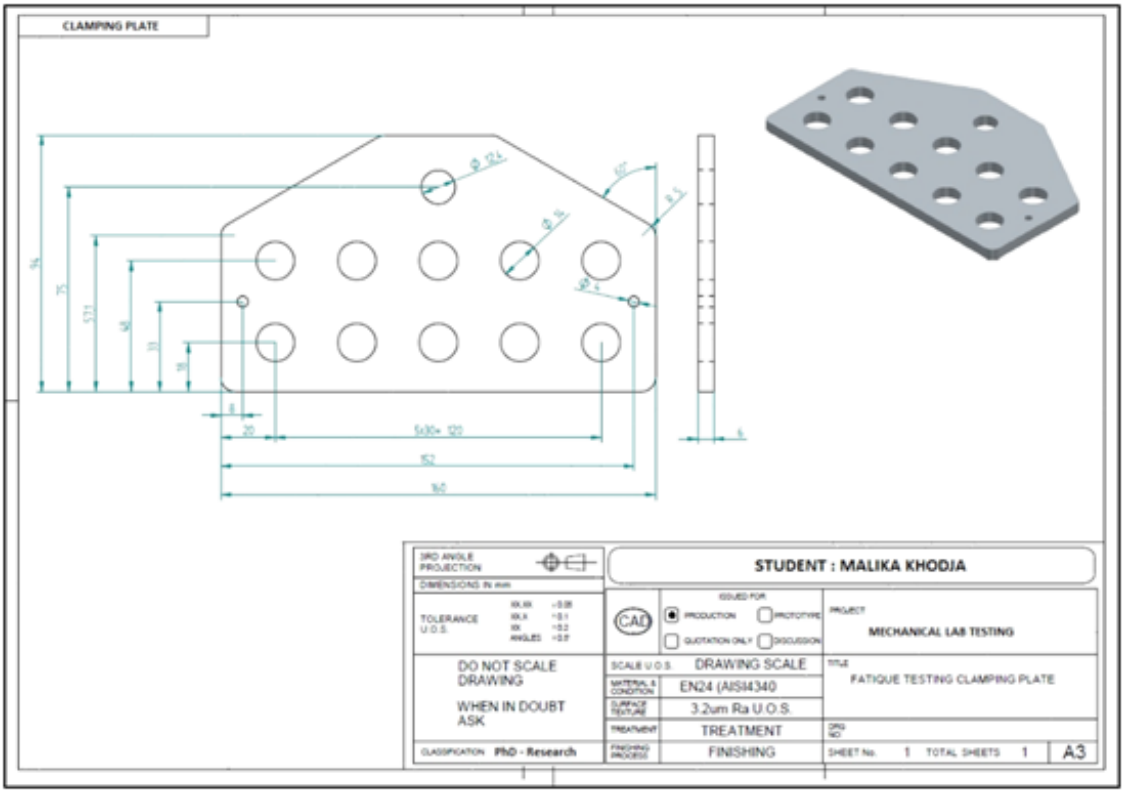


Figure 3.7: The designed clamping plates for the panels.

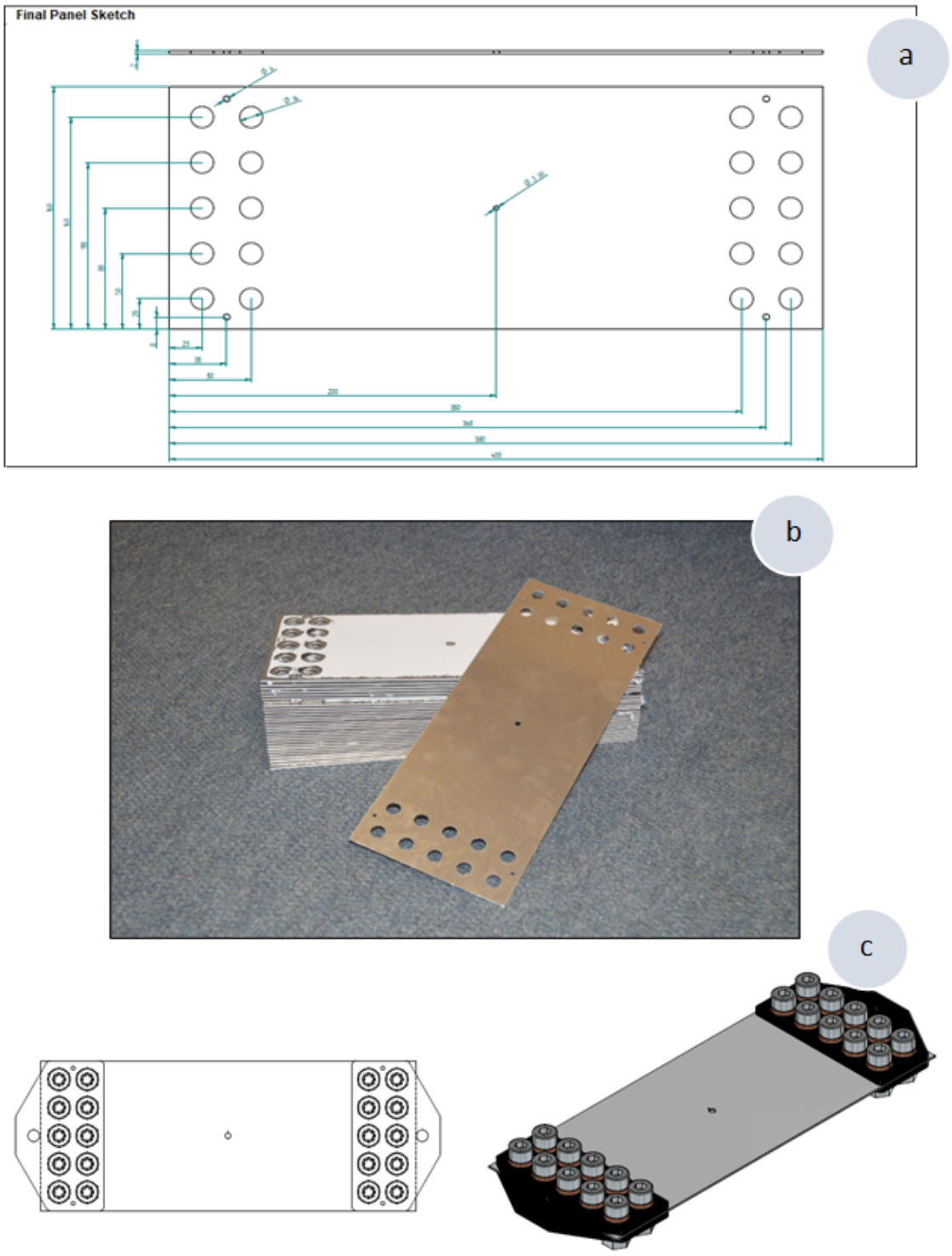


Figure 3.8: (a) The panel sketch dimensions (b) Final samples machined (d) Panel fitted with the clamping plate.

3.3.2 The Composite Patch Repair Design Process

The goal of a properly designed bonded repair is to restore the damaged structure's ultimate load-carrying capability. Damage (crack) growth should either be arrested or significantly retarded. The repair must be carried out without causing further damage or

creating a weak link in the structure. In short, the repair allows the structure to fulfil its original intended function [5]. That means choosing repair materials (composites, adhesives) that have load-carrying capability greater than or equal to that of the parent material. Boron/epoxy was chosen as the material of repair from the composite patches because it is the most common composite used for patching damaged aircraft structure. This is due to several advantages gained by using boron fibres. Firstly, it is stiffer and stronger than other possible materials, such as aluminium, fibreglass, or graphite. This results in thinner, more aerodynamic repairs. While aluminium or aluminium/fibreglass laminate patches would solve CTE mismatch problems in the repair area, which induce bending and residual stresses, these materials are approximately one third as stiff and result in much bulkier repairs. Secondly, boron does not cause galvanic corrosion problems when in contact with aluminium, while graphite does [6].

3.3.2.1 Design of the Boron/Epoxy Patch

When possible, it is always better to perform a double-sided repair. This removes the neutral axis shift, hence removing out-of-plane bending stresses and reducing peel stresses at the patch ends. This study utilized a single-sided repair, since most aircraft repairs are single-sided, therefore this section will focus on a single-sided composite repair. To reduce patch-tip stresses, there are several guidelines to follow with regards to patch ends. Patch termination is defined as the edge of the patch with the taper [7]. Therefore, the thickness of the repair patch tips was tapered to relieve adhesive peel stresses.

3.3.2.2 Stiffness Ratio Calculation

The stiffness ratio is calculated to a value of 1 in order to find the desired patch thickness [6]. When dealing with cracks and damages, to maintain a stiffness ratio (S) of 1 between the patch and the repaired structure, can be shown by equation 1 below

$$S = \frac{E_p t_p}{E_s t_s} \tag{3.1}$$

Where

E_p = Young's modulus of the patch,
t_p = Thickness of the patch,
E_s = Young's modulus of the structure,
t_s = Thickness of the structure.

3.4 The Repair Methodology & Manufacturing of the Repaired Samples

3.4.1 Surface Preparation of the Aluminium Panels

An important factor influencing the strength of the bonded joint is the surface preparation of the metal. To prepare the bonded area, the metal surface was subjected to mechanical abrasion (sandblasting) with uniform grit blasting of the repair surface area as shown in Figure 9. The process employed 210 μm alumina grit at a pressure of 300 kPa

and a working distance of 10 to 15 cm to obtain the surface roughness without inducing compressive stresses. Trichloroethylene was used for cleaning to remove impurities and to avoid any oxidation.

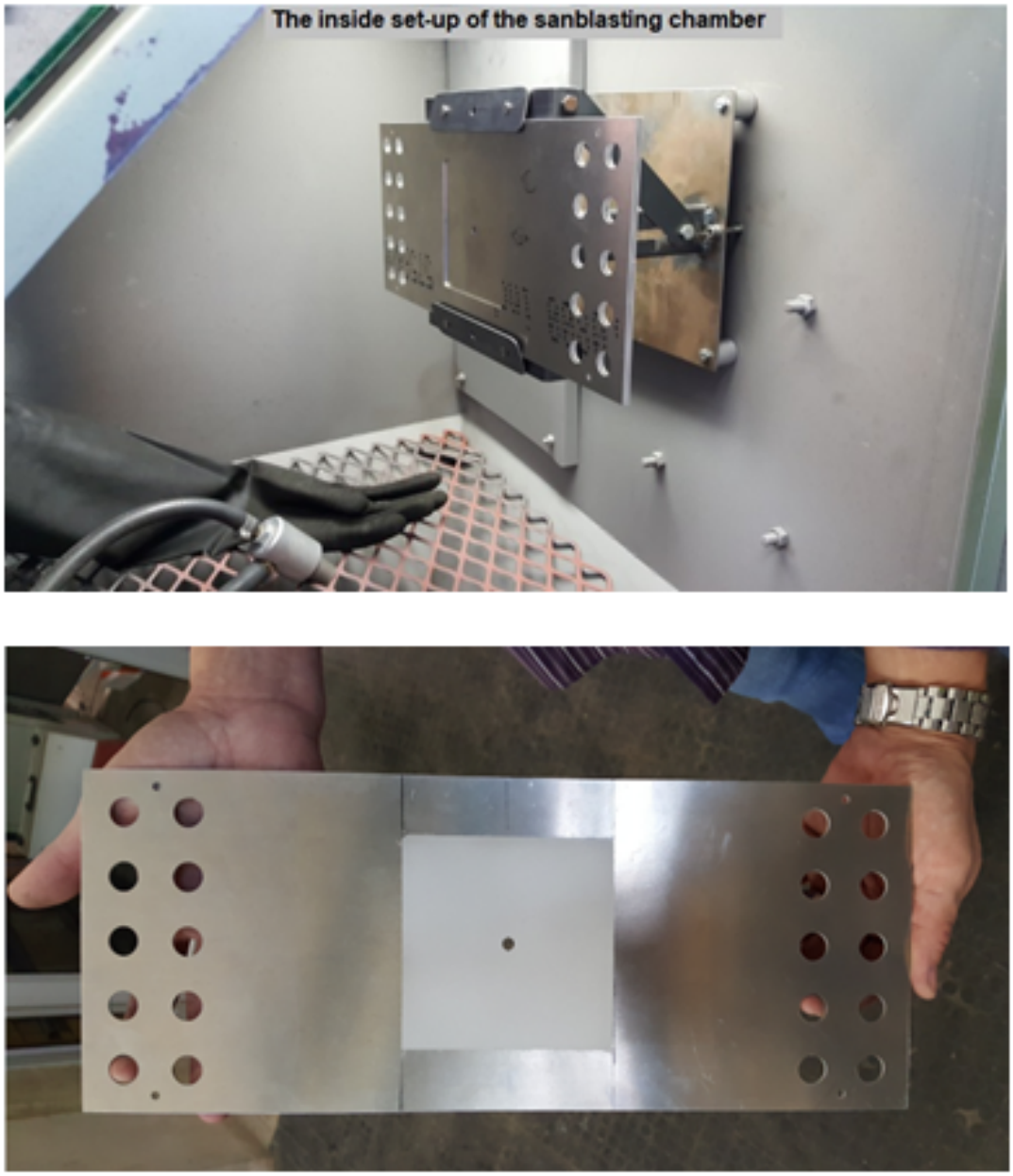


Figure 3.9: Surface preparation with sand blasting method.

3.4.2 Bonding Boron/Epoxy Patch onto the Aluminium Panel

Once both the aluminium panel was prepared [machined, pre-cracked (description of pre-cracking in Section 5.4.1.), and surface preparation applied] and the boron/epoxy patch had been sectioned to the desired size, then the patch had to be bonded on to the panel. The bond areas were marked on the panel while sandblasting (outside of the bonding surface). This made for much more accurate placement of the patch in the middle

of the panel. The bonding surface of the patch was lightly grit blasted so a physical bond could be formed with the adhesive as explained in Section 4.1. The adhesive FM73-K was then cut out slightly larger than the perimeter of the patch. This was done to create a fillet around the perimeter of the patch and reduce stress concentrations at the edges. Finally, the patch was added layer by layer. The patch should have a dimension of 100 mm x 100 mm x 1.042 mm (thickness including 6 prepreg tape layers and the adhesive layer). Figure 10 shows the practical sequence.

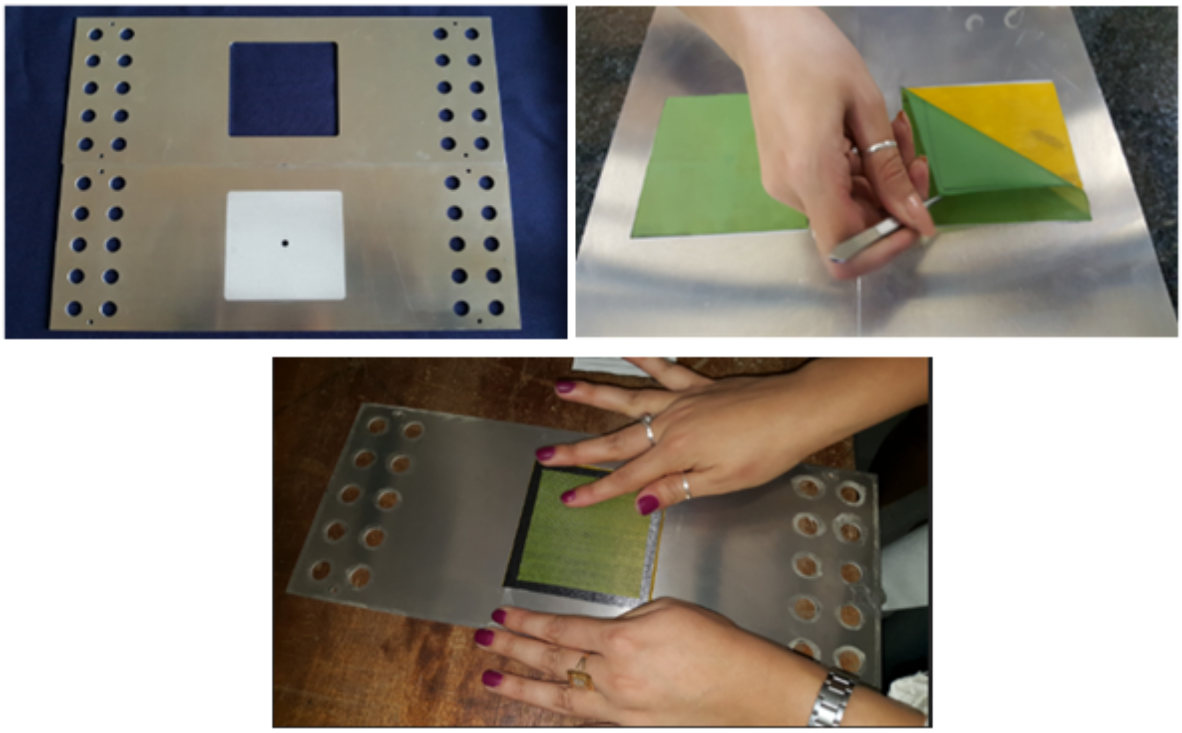


Figure 3.10: Sequence of bonding boron/epoxy patch onto the Aluminium panel.

3.4.3 Curing process

The lay-up is shown schematically in Figure 11 and shows the sequence. The AA7075-T6 aluminium was the base plate; it was bonded with the FM73 adhesive to six layers Type B4/5521 unidirectional prepreg tape. Plies were all stacked with the unidirectional orientation running perpendicular to the crack direction, the rivet hole being at the centre of the patch. The following conventional layers were applied over the prepreg tape: peel ply, release film, breather fabric cloth, and then the vacuum bag. The vacuum was sealed off at the edge of the vacuum bag with Cytec sealant tape. The full panel with the layup was put in an air convection furnace for curing while maintaining a vacuum of 80 mbar drawn over the layup. An overview of the final samples is shown in Figure 12 and the steps to the final patched test panel.

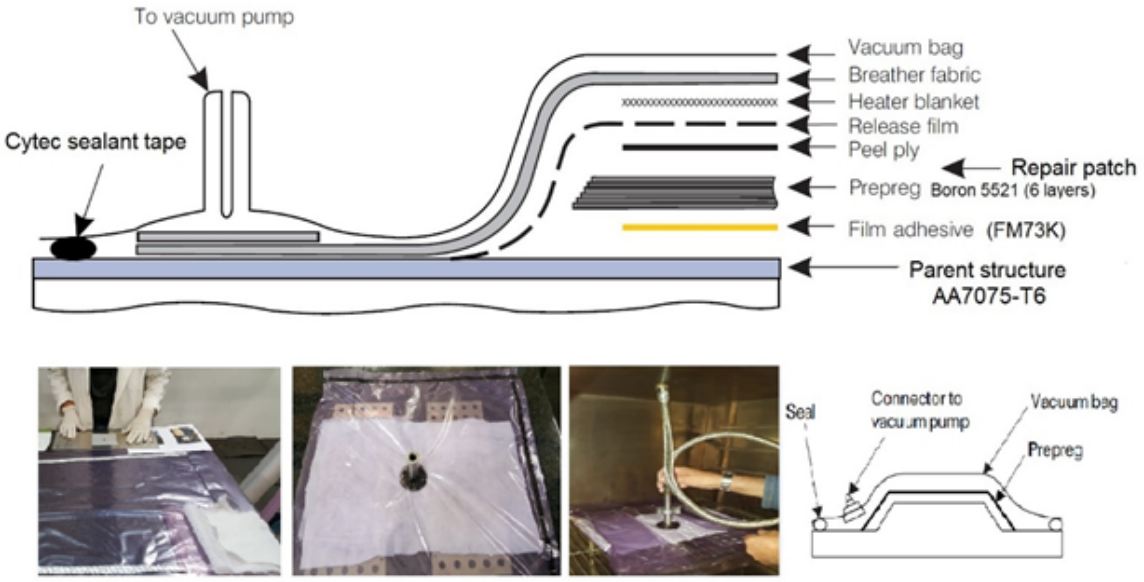


Figure 3.11: Detail of vacuum bag lay-up and manufactured specimens.

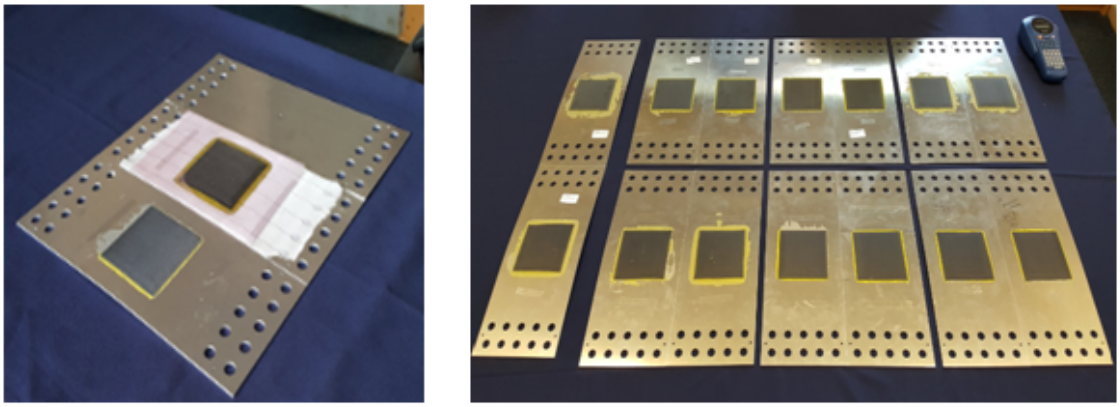


Figure 3.12: The steps to the final patched test panel.

3.4.4 Curing parameters

Three panels were made for each curing condition to ensure repeatability and an extra panel for traceability. The full plate with patch was heated in a convection furnace. Four curing conditions were used:

- (A) 120 $^{\circ}C$ for 60 minutes (as per manufacturer specification).
- (B) 80 $^{\circ}C$ for 480 minutes with post curing ramped to reach 120 $^{\circ}C$ in 60 minutes.
- (C) 80 $^{\circ}C$ for 480 minutes.
- (D) 120 $^{\circ}C$ for 60 + 45 minutes.

The curing cycle was selected as per the data sheet - heating up in 30 minutes and held at 120$^{\circ}C$ for 60 min. To evaluate the prolonged curing time, cure cycle as per data sheet was cured for a longer time. Also, to evaluate prolonged time effects during curing at lower temperature; a temperature of 80oC was selected and held for a longer time of 480 min. A reduction in the cure temperature may affect important properties of the bonded repair integrity, and therefore a cure cycle of a combination of low temperature over a long cycle and post cured at the recommended cycle as per the data sheet was done to ensure adhesion at elevated temperatures. These cure cycles are also summarized in Table 4.

	Cure parameter	
Cure Conditions	**Cure temperature °C**	**Time (min)**
A	120	60
B	80 post cure at 120	480 post cure for 60
C	80	480
D	120	105

Table 3.4: Bond curing cycles.

The curing process was controlled by computer to ensure repeatability, while the temperature of the workpiece (composite panel) was continuously monitored by a thermocouple fixed to the aluminium base plate. The ramp rate of all the curing temperatures was 2 $^{\circ}C/min$. Plates were taken out of the furnace after curing and left to cool at ambient conditions. An example of a curing sample log on a computer is shown in Figure 13.

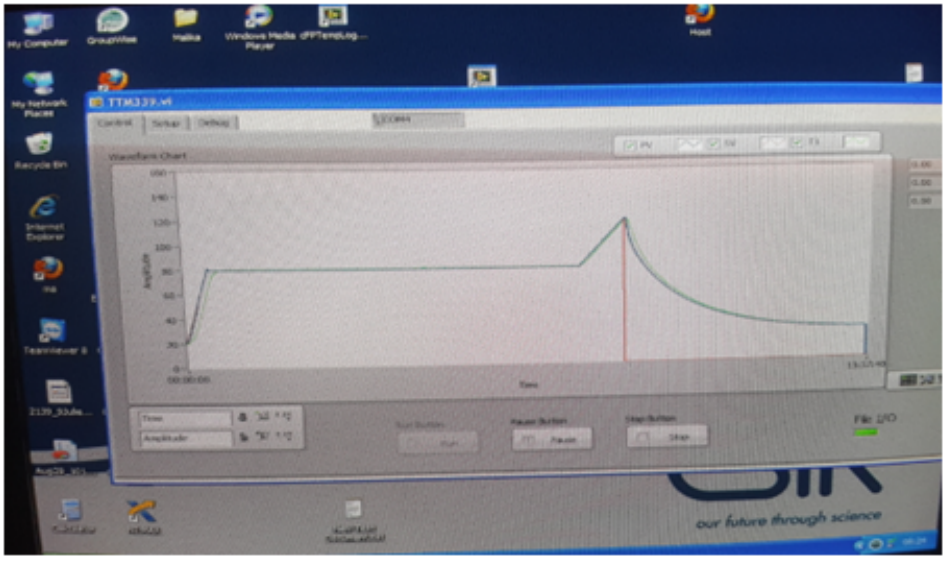


Figure 3.13: Computer Curing cycle log.

3.5 Testing methodology

3.5.1 Mechanical Properties Characterisation of AA7075-T6

3.5.1.1 Tensile Test Procedure of AA7075-T6

The tensile properties of the AA7075-T6 alloy were determined using an INSTRON 1342/H1314 with 25 kN load cell capacity and an INSTRON Model 2620-602 extensometer with a gauge length of 12.5mm. The tensile test procedure set-up of AA7075-T6 is shown in Figure 14.

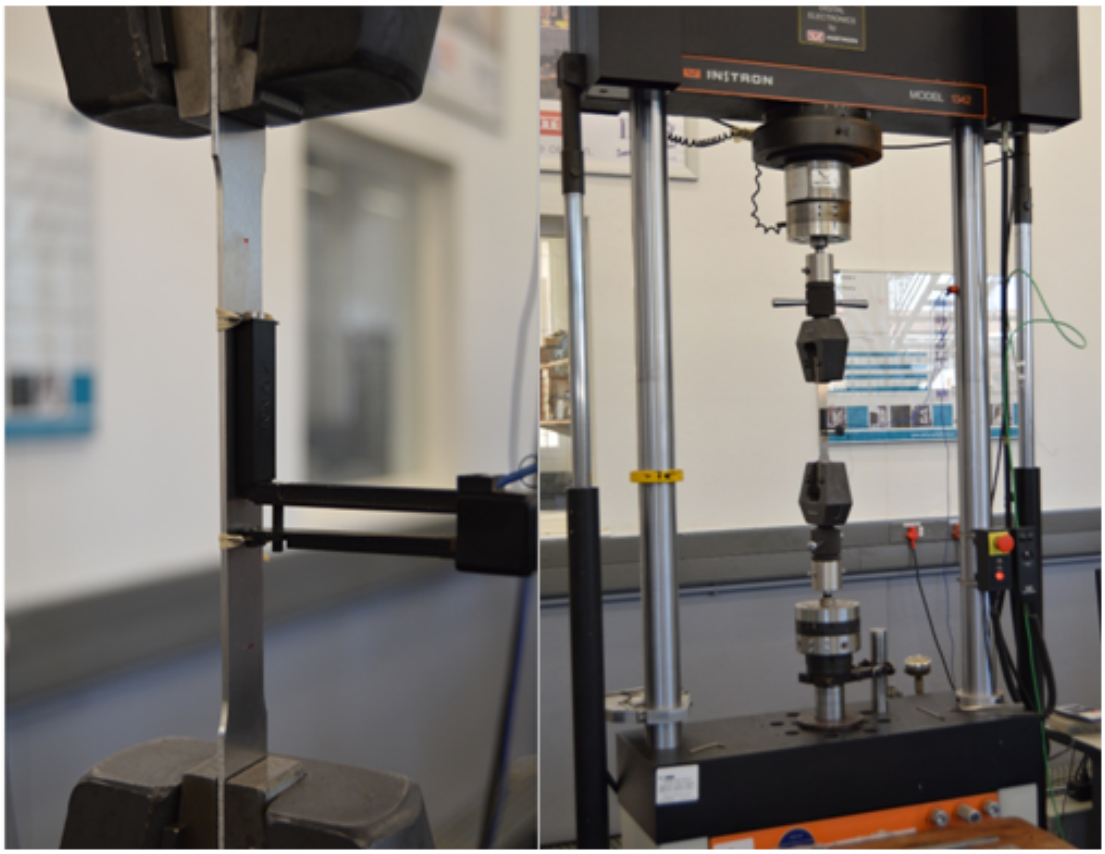


Figure 3.14: Tensile test procedure set-up of AA7075-T6.

To determine the 0.2% proof stress, a stress rate of 10 MPa/s was used and for the ultimate tensile stress (UTS) determination a displacement rate of 10mm/min. These parameters were selected based on the American Society for Testing and Materials (ASTM) standard E8M-04[4]. The extensometer was removed during tensile testing after reaching the 0.2% plastic strain value (typically after 1-2% strain) to prevent damage to the extensometer in the case of premature fracture. The % elongation after fracture was calculated in accordance with ASTM standard E8M-04, where gauge marks were drawn with ink on the tensile specimens. After fracture, the ends of the fractured specimen were fitted together carefully and the distance between the gauge marks measured as shown in Figure 15.

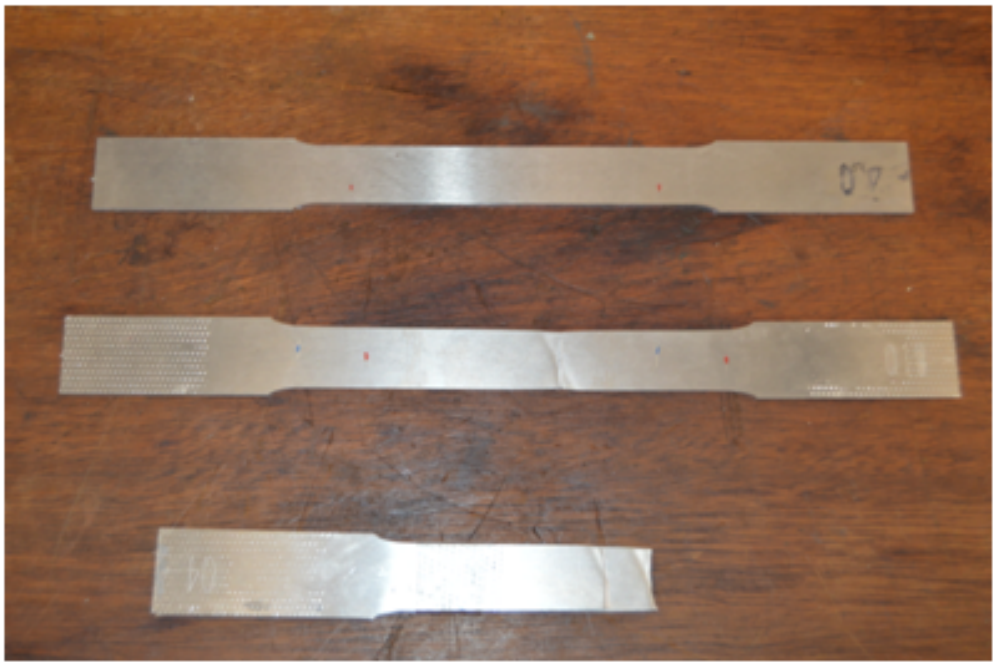


Figure 3.15: Calculation of elongation after fracture.

3.5.1.2 Microstructural Analysis of AA7075-T6

Metallography is the art of embedding a specimen in resin and preparing it to examine and reveal the microstructure of the materials. Specimens are prepared in this manner for analyses using optical microscopes to magnify features of the material under analysis. In order to identify and evaluate the microstructure of the material, it is very important to prepare the test sample carefully and properly. The various steps in sample preparation for microstructural examination include:

- Selecting a representative sample of the materials.
- Sectioning the sample to avoid altering or destroying the structure of interest.
- Mounting the section without damage to the test sample.
- Grinding to achieve a flat sample with a minimum amount of damage to the sample surface.
- Polishing the mounted and ground sample.
- Etching in the proper etchant to reveal the microstructural details.

The microstructural examination was performed using fully automated polishing equipment (for the final stages of sample preparation) as well as customized sample preparation equipment for the initial stages (grinding). All of these types of equipment were used to prepare the samples for the evaluation process without causing excessive damage to the component's microstructure are shown in Figure 16. Those analyses were performed at the MSM/ CSIR metallographic laboratory at the Light Metals Competency area.



Figure 3.16: Equipment used for sample preparation process at the metallographic laboratory at LM/MSM/CSIR.

In order to have more understanding of crack behaviour, specimens were selected from the same machined samples that were tested. Figure 17 shows the selected samples before preparation.

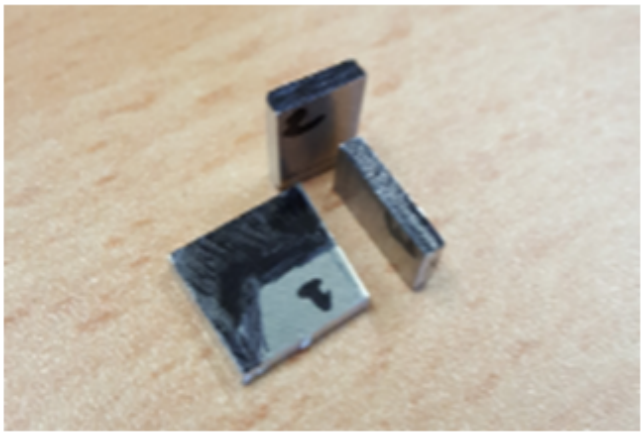


Figure 3.17: Selected samples for microstructural analysis in the rolling direction.

The same procedure was done for samples after fatigue crack growth tests. Figure 18 shows the fractured samples that were evaluated for the crack path direction according to the microstructural analysis.

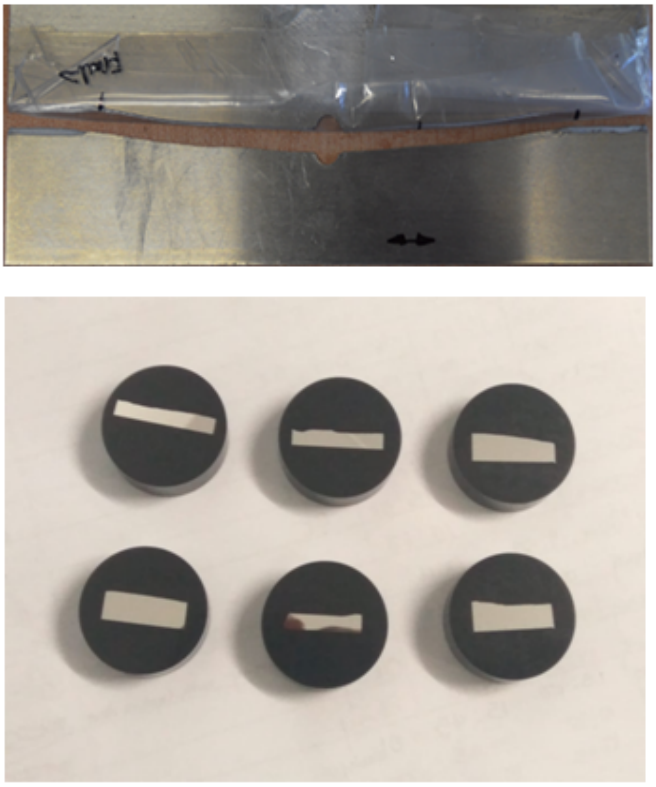


Figure 3.18: Equipment used for sample preparation process at the metallographic laboratory at LM/MSM/CSIR.

The samples used for optical microscopy were mounted in an electro-conductive resin before grinding on SiC papers from P120 to P2400 grit. This was followed by polishing using 9 to 1 um diamond suspension, finishing with 50 nm colloidal silica. An optical microscope (LEICA DMI5000 M) equipped with a camera (LEICA DFC480) and imaging software (Image-Pro MC v6.0) was used to study the microstructures of the alloys.

It is important to select the best etchants to reveal the microstructure. Etching was performed using a Kellers etch for 20 seconds. Table 5 shows the volume and the chemical composition of the etchant.

Etchant	Concentraion	Condition
Kellers Etch		
Distilled water	190 ml	10-30 Second
Nitric acid	5ml	immerssion
Hydrochloric acid	3ml	Used fresh
Hydrofluric acid	2ml	

Table 3.5: Kellers etchant used for sample preparation.

Examination of grain structure measurement of features is done after the grain structure is revealed by polishing and etching as appropriate for the material of interest. Grain sizes of the structure were determined by the mean linear intercept for each line on each image taken along with overall average values. Microstructural analyses were performed before and after the failure of the material. Observations of the complex interaction between mechanical load and the metallurgical structure are discussed in Chapter 4.

3.5.2 Distortion Measurements

The thickness and distortion due to bending of the samples before and after curing was measured. Distortion measurements were done accurately with metrology equipment controlled by computer at NMISA (National Metrology Institute of South Africa, www.nmisa.org). Distortion was measured by a laser metrology scanning technique using a laser scanner. Composite plates were placed convex-side down with the bare aluminium surface facing up for measurement. Scanning the length of the plate took place over 83 intervals.

3.5.3 Fatigue testing procedure

Figure 19 shows the testing facility of Mechanical Testing Laboratory at CSIR were all tensile and fatigue testing were performed.



Figure 3.19: The CSIR mechanical testing laboratory.

3.5.3.1 Pre-cracking the Aluminium Panels with Fatigue testing

Once machining of the aluminium test panels was completed, pre-cracking of the panels was accomplished. Various crack lengths were grown from a 6mm size rivet hole with a

wire-cut notch of 1mm length perpendicular to the length of the plate emanating from the hole's edge. The cracks were grown from 1mm length until the crack size length of 30mm, and further until 45 mm for a parametric study in case of field damage. The cracks were grown in 1.5mm step interval until 5mm length, then in 5mm step after that. All of the panels were pre-cracked at 10 Hertz at a load of 22 kN with an Instron 1342-H1314 machine. The aluminium panels were then fitted with crack-opening displacement (COD) clip gauges to measure the crack growth per cycle. These gauges stopped the testing when the crack length reached a certain point as desired per each panel. The pre-cracking set-up is shown in Figure 20.

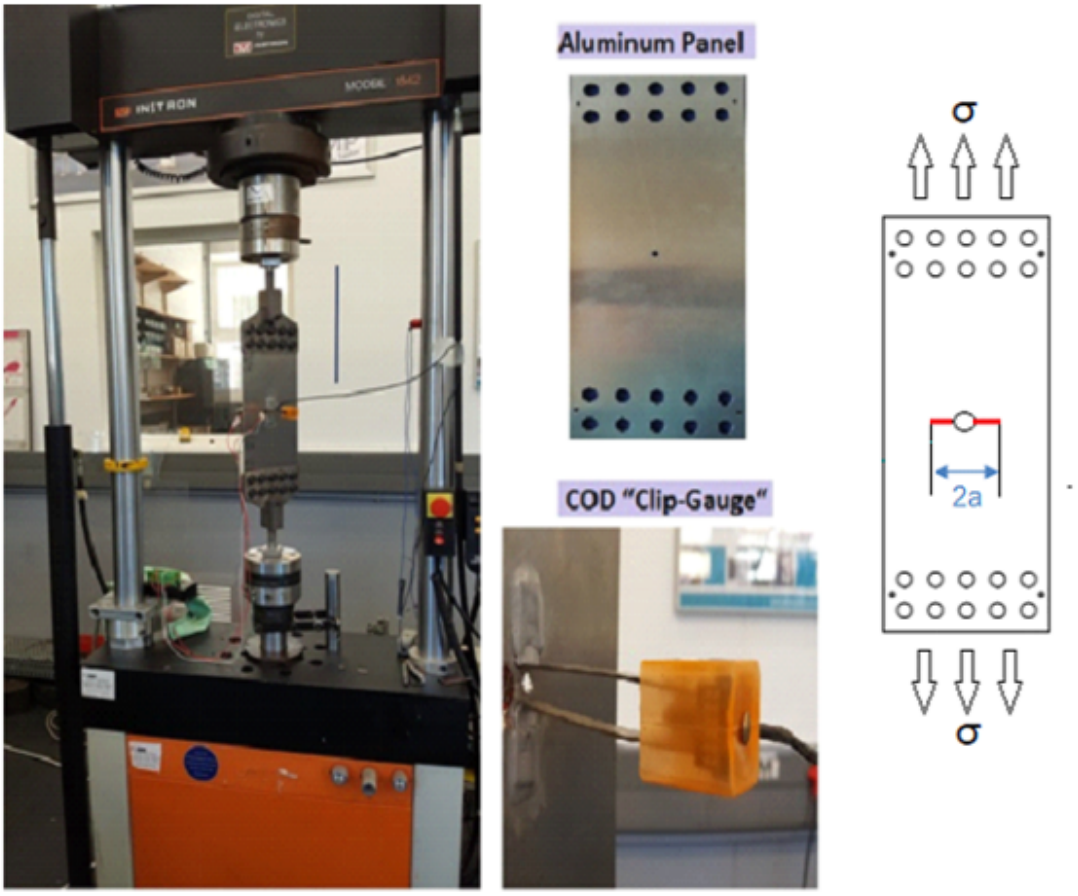


Figure 3.20: Aluminium Panel in Instron Test Machine Fitted with COD "Clip-Gauge".

3.5.3.2 Fatigue crack growth (FCG) Test Procedure

Fatigue crack growth tests were performed on an Instron 1342-H1314 servo-hydraulic system on 25 kN dynamic capacity. The fatigue sample panels for this research investigation were on the fatigue behaviour response of repaired and unrepaired structures.

- The repaired samples with four different cure cycles, to evaluate the effect of curing condition on the fatigue life of the repair. In this case, the test started only from the notch emanating from the rivet hole (no pre-cracking was done).

- The repaired panels with various crack lengths, to investigate the effect of the repair with various crack lengths on the crack growth.

The test panel set-up with the completed repaired structure with boron fibre composite patch is shown in Figures 21. All fatigue tests were conducted at cyclic constant-load amplitude of 22 kN, frequency of 10 Hz and stress ratio of 0.1. Crack length data and number of cycles were logged on a computer using the visual methodology. Panels were fatigued until:

1. The crack lengths approached the edge of the patch.
2. The panels were cycled until catastrophic failure of the specimen.
3. The panels were cycled until the crack tips reached the panel edge and crack lengths could no longer be measured.



Figure 3.21: Completed panel with Boron/Epoxy loaded in tension.

3.5.3.3 Fatigue crack growth measurement

From the knowledge of the material's fracture toughness and the constants in the Paris equation - these properties have to be determined by testing laboratory specimens - the number of cycles to failure can be calculated for specific loads. The standard Test Method E647 issued by the American Society for Testing and Materials (ASTM) describes the determination of fatigue crack growth rates using specimen configurations. The testing and data reduction procedure are briefly summarized in the following:

- Fit COD gauge over the notch mouth of CT specimen, for example.
- Cycle the specimen between two fixed load limits.
- Determine crack length from compliance measurements as a function of the number of loading.
- Construct plot of crack length a vs number of cycles N.
- Compute the crack growth rate da/dN at each crack length a_i.
- Compute the stress intensity factor range $\triangle K_i$ at each crack length a_i

$$\triangle K_i = \frac{\triangle P}{B\sqrt{W}} \int (a/W).$$

- Construct a plot of da/dN vs $\triangle K$ on a double-logarithmic scale to obtain the fatigue crack propagation (FCP) curve.
- Perform a linear regression analysis over the truncated linear range of the FCP curve to obtain the Paris equation

$$da/dN = C(\triangle K)^m.$$

3.6 References

[1] Cytec Engineered Materials; FM73 Epoxy Film Adhesive; Technical Data Sheet.
[2] Park, J. K., and A. J. Ardell. "Microstructures of the Commercial 7075 AI Alloy in the T651 and T7 Tempers". Metall. Trans. A. 14A (1983): 1957. Print.
[3] Boron 5521 Prepreg Tape, SPECIALTY MATERIALS, INC, Technical Data Sheet.
[4] ASTM E8 standard , Guide to Metal Tensile Testing, www.instron.us/en-us/testing-solutions/by-standard/astm/multiple-testing-solutions/astm-e8.
[5] Fredell, R.S. Damage Tolerant Repair Techniques for Pressurized Aircraft Fuselages. PhD dissertation. Delft University of Technology, Delft, The Netherlands, June 1994 (WL-TR-93-3134).
[6] Baker A.A. (1988). Advanced in bonded composite repair in metallic aircraft structures, Chapter eight, **Recent Expansions in the Capabilities of Rose's Closed-form Analyses for Bonded Crack-patching L.J. Hart-Smith**, pp. 186.
[7] Guidelines for Composite Repair of Metallic Structure. AFRL-ML-WP-TR-1998-4113.
[8] ASTM International, ASTM E647 - 13, Standard Test Method for Measurement of Fatigue Crack Growth Rates, 2013.

Chapter 4

Results & Discussion

4.1 Part I: UNPATCHED PLATE

CRACK PROPAGATION FROM A NOTCHED HOLE UNDER UNI-AXIAL FATIGUE LOADING OF AA7075-T6 PLATE

Summary

Aluminium is the predominant material used in the manufacture of commercial aircraft. Fatigue cracks in aircraft skin commonly initiate from stress concentrations, such as fastener holes and failure will occur in the mode that corresponds with the lowest load-carrying capability of the material. Crack propagation from a notched hole under cyclic tension-tension mode was studied. Crack growth rates were obtained from load-controlled low cycle fatigue testing of aluminium 7075-T6 loaded in Uniaxial fatigue. In addition, material characterization and microstructural analyses were performed before and after the failure of the material. Observations of the complex interaction between mechanical load and the metallurgical structure are discussed.

4.1.1 Introduction

The phenomenon of decreased strength of a material under repeated loading is commonly called fatigue. Fatigue life is defined as the number of cycles to failure at a specified stress level, whereas fatigue strength, also known as the endurance limit, is the stress level below which failure does not occur due to fatigue [1]. The fatigue phenomenon in metallic materials is characterised by four stages: cyclic hardening or softening, the nucleation of microcracks, the propagation of the microcracks and then the final failure, see Figure 1[2]. Cracks initiate, either due to persistent slip bands at surface inclusions, or the boundaries of grains. There are a number of factors influencing nucleation of microcracks such as cycling amplitude and asymmetry, specimen shape and environment during the fatigue process [3].

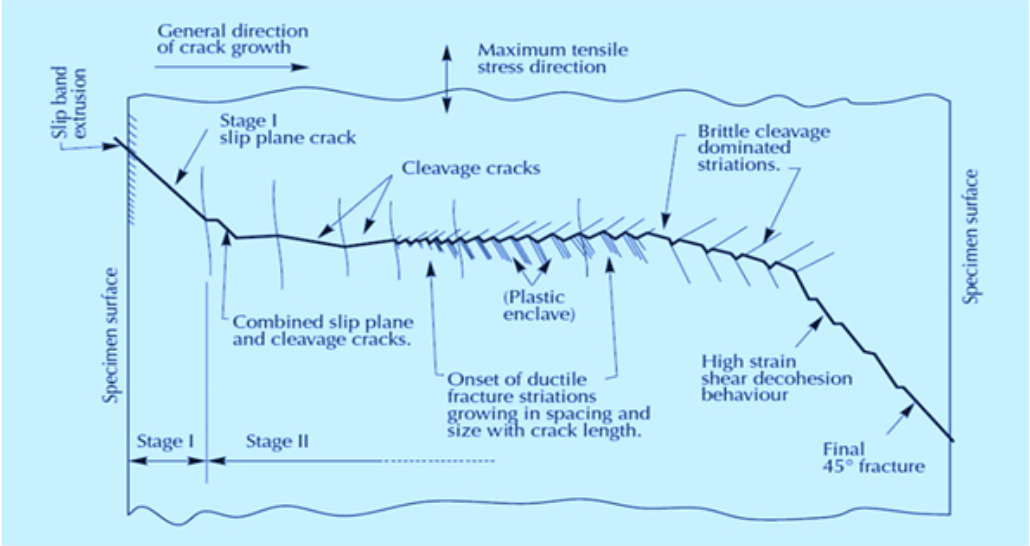


Figure 4.1: Illustration of the different fatigue - fracture modes in an AlZnMg alloy according to Forsyth [2].

Fatigue life prediction has been one of the most important issues in failure analysis since the first half of the nineteenth century [4]. The investigation of fatigue life is essential for reliable life prediction of critical engineering components, such as aircraft parts. Over the past three decades, various models have been developed to correlate fatigue life and crack growth under variable loading conditions [5]. Many materials contain physical discontinuities, which are given the generic term of notches. When a notched material is loaded, local stress and strain concentrations result in the notched area. These phenomena may result in micro-cracks that develop and coalesce with one another until they form a macroscopic crack, which propagates until the total breakdown of the material occurs. Structural fatigue cracking is a result of usage and will eventually occur in all ageing aircraft. These cracks are common in aluminium alloys such as 7075 and 7079, especially in the T6 temper condition. These alloys are tailored for high strength, but compromise crack initiation/growth performance and corrosion resistance. Structural shapes also significantly affect the fatigue life; square holes or sharp corners lead to elevated local stresses where fatigue cracks can initiate [6]. Aeronautical components may be subjected to multiaxial stress states of varying amplitude leading to fatigue damage that could lead to catastrophic failure [7].

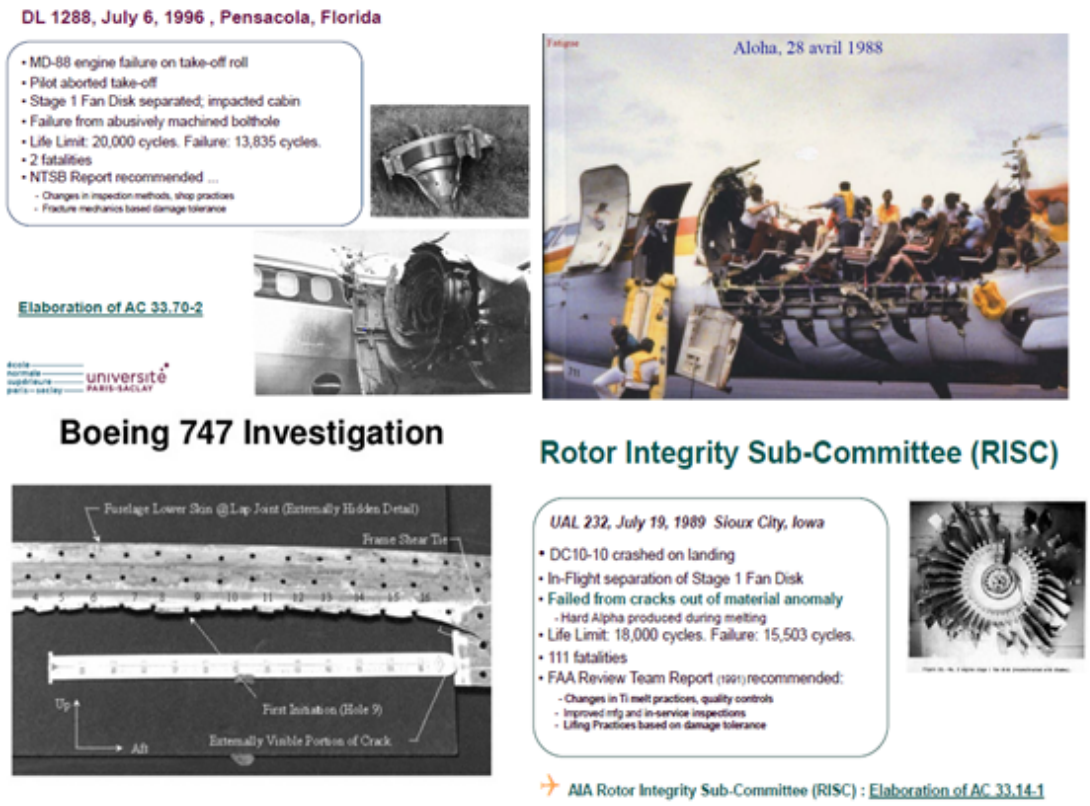


Figure 4.2: Examples of aircraft fatigue failures investigation [7] [8].

A fundamental issue in fracture mechanics is the determination of criteria for the growth of pre-existing cracks in materials. Past research has been aimed at understanding the phenomenon of initiation and propagation of cracks to predict the life of aeronautical structures [9-10]. There are limited investigations available on crack propagation behaviour in an existing damage or pre-existing crack on metallic plates evaluating the influence of different cracking parameters on fatigue life of AA7075-T6 plate. Additional understanding of material properties should not be depended only on manufacturer data sheet, as it is not enough for full behaviour analysis.The purpose of **Part I** from this research is to investigate the behaviour of cracks in AA7075-T6 plates emanating a rivet hole using notched plate aircraft structure subjected to tensile cyclic fatigue loading. The fatigue process consists of crack propagation to failure.Cracked specimens are used to investigate the crack growth behaviour and to characterize crack propagation(The conditions until the crack has grown to a certain length). This crack length is the maximum acceptable crack length (considering the material's service conditions) and is not necessarily the ultimate fracture of the specimen. There are three reasons for performing this type of test:

a. To gather data for the calculation of crack growth used in determining the life of a component or structure that already has a crack in it, that is subjected to cyclic loading.

b. To determine a threshold value of stress intensity below which crack growth will not occur.

c. The fatigue life of the specimen is easily determined by cycling the specimen under specified loading.

The understanding gained from this work will be the basic principle on fatigue crack growth behaviour for repair method where it is to adhesively bond a composite patch over the damaged area in order to restore the load-carrying capacity of a damaged structure discussed in the following partII.Those experiments will form as guidelines to predict the rate of continued damage growth after repair. This will be required in some cases for validation of these experiments numerically and further development of them to ensure that they can accurately predict the effect of various parameters such as the nature of the loading (spectrum and R ratio effects), the influence of residual stresses, process and environmental effects.

A detailed method of material characterisation and fatigue crack growth procedures of damaged metallic structure is presented to provide necessary material data. This can provide input guidelines also for future Finite Element Analysis (FEA) to investigate more into failure modes and internal stresses. This will save time and investigation costs,to investigate in details the AA7075-T6 fracture mechanics mechanism.

4.1.2 AA7075-T6 material characterisation

Material characterisation is presented with chemical composition in Table 1. Stress-strain curve of aluminium alloy AA7075-T6 plate is presented with a reproducibility test in Figure 3. The Stress-strain curves with load vs extension in the rolling direction (0 Deg) and transverse to the rolling directions (90 Deg) are presented respectively in Figures 4, 5 and 6 followed by a summary of tensile properties of our material are shown in Table 2, which confirms the typical Mechanical Properties of AA7075 [11].

4.1.2.1 Chemical composition

Material	**Elements**									
AA7075-T6	Si	Fe	Cu	Mn	Mg	Cr	Zn	Ti	Ti+Zr	Other EA
Spec.min.	-	-	1.2	-	2.1	0.18	5.1	-	-	-
Spec.max	0.40	0.50	2.0	0.30	2.9	0.28	6.1	0.20	0.25	0.05
Actual	0.07	0.13	1.4	0.01	2.6	0.19	5.6	0.03	0.04	0.02

Table 4.1: Chemical Composition Limits (in wt. %) [ALUMINIUM DATA SHEET ALLOY GRADE 7075 T6].

4.1.2.2 Mechanical properties

The specimens with 0 Deg and 90 Deg orientations to the rolling direction gave similar results that confirm the material isotropy in those directions.

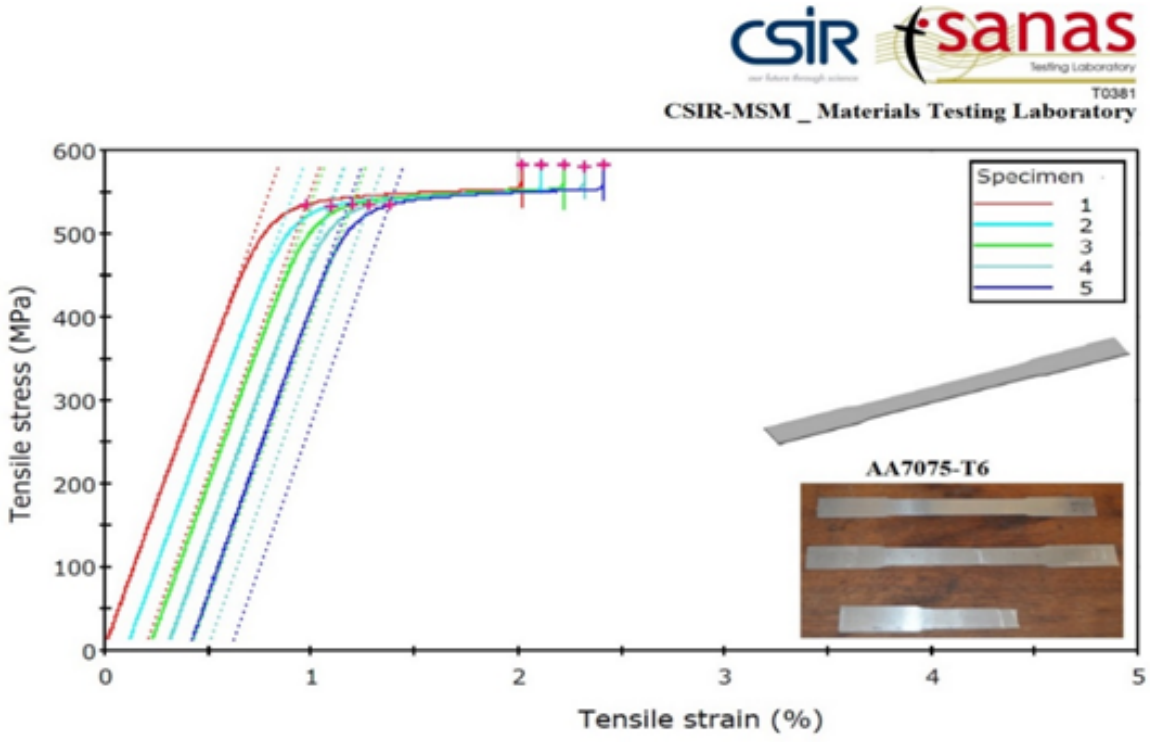


Figure 4.3: Stress-strain curve of aluminium alloy 7075-T6 in the rolling direction with five tested samples for reproducibility test.

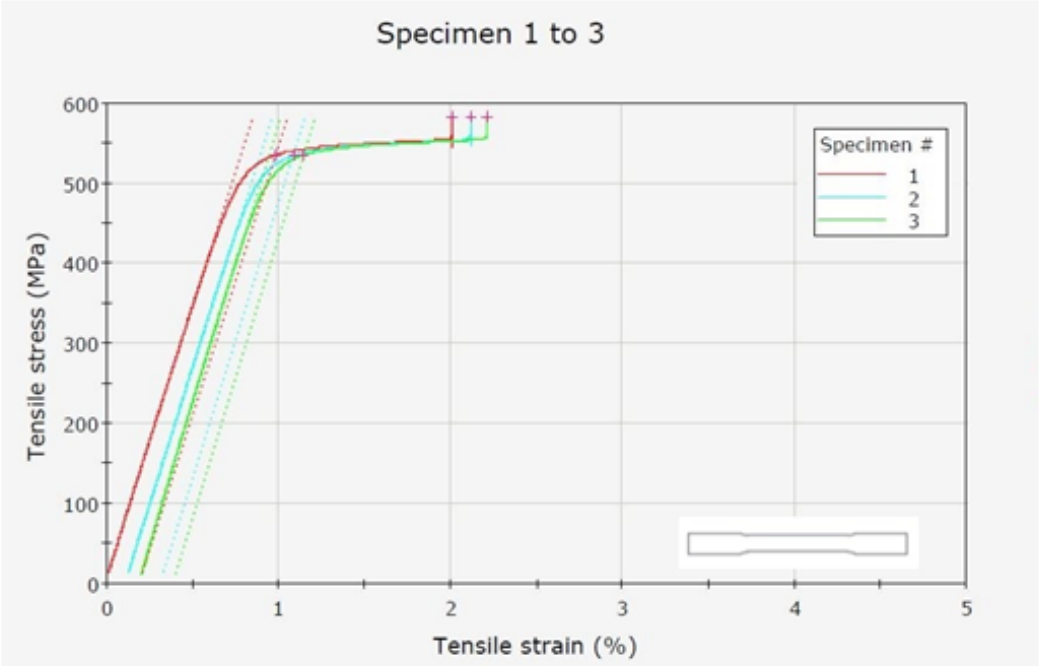


Figure 4.4: Stress-strain curve of aluminium alloy 7075-T6 in the rolling direction (0Deg).

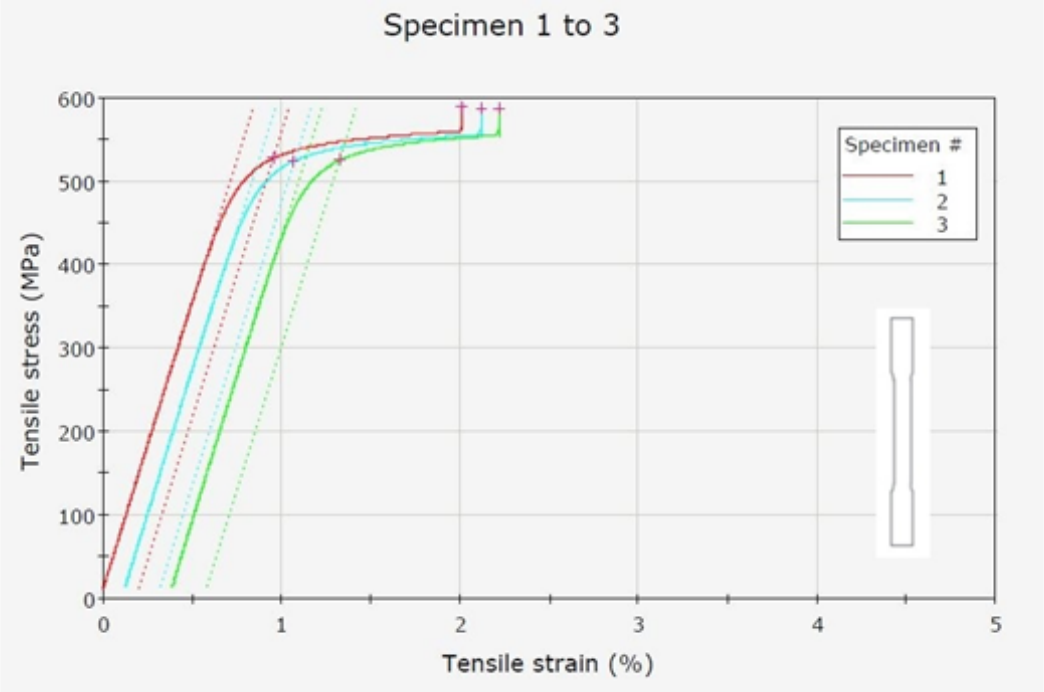
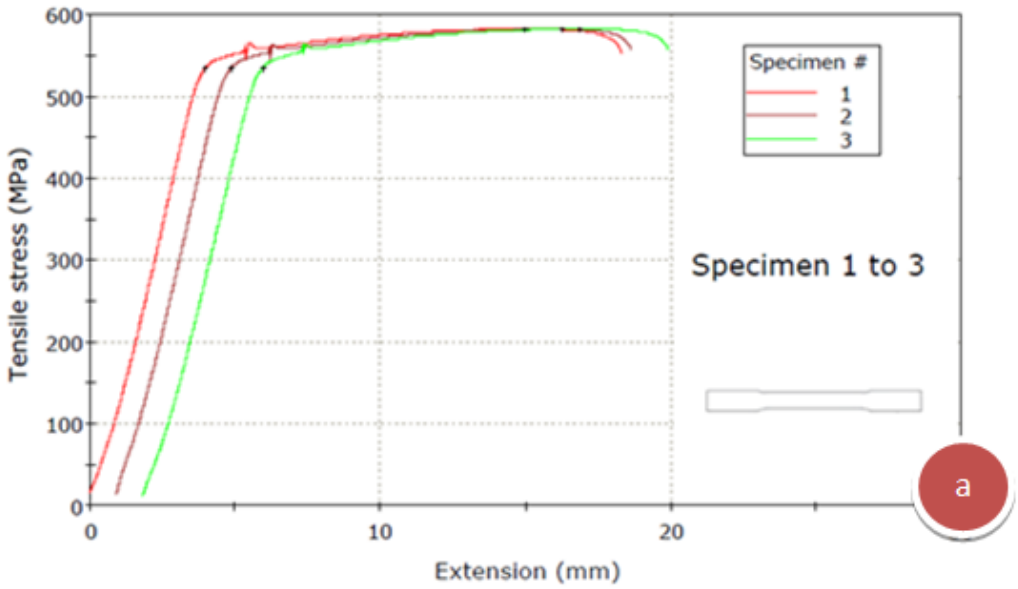


Figure 4.5: Stress-strain curve of aluminium alloy 7075-T6 in perpendicular to the rolling direction (90Deg).



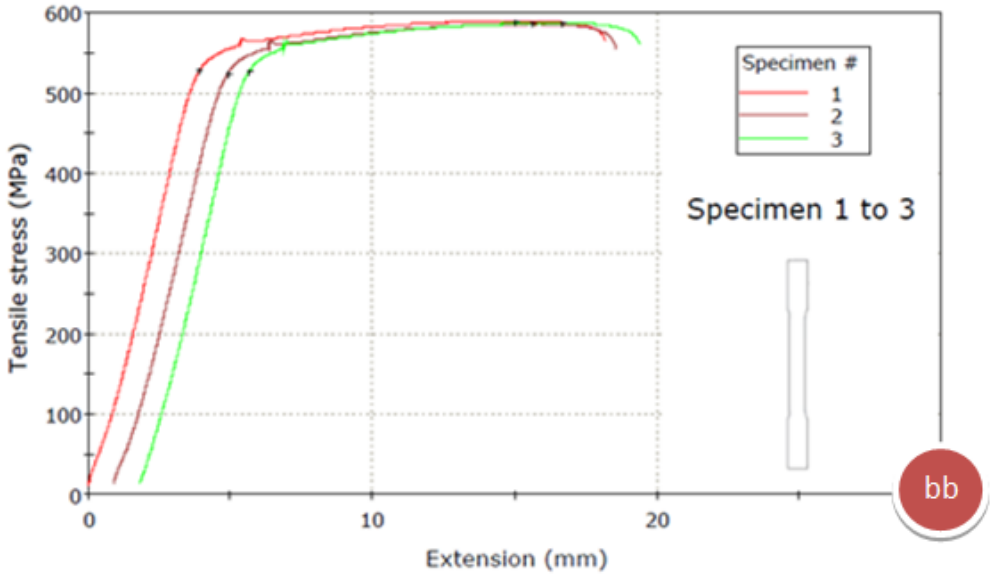


Figure 4.6: Load-extension curve for 2mm thick dog bone AA7075-T6 sample (a) in the rolling direction (0), (b) is perpendicular to the rolling direction (90).

a

	Specimen label	Area (mm^2)	Tensile stress at Yield (Offset 0.2 %) (MPa)	Modulus (Chord 100 MPa - 400 MPa) (GPa)	UTS (MPa)	Elongation % (5*W)
1	T-L-Q-1	40.54	534.93	67.1	582.49	12.63
2	T-L-Q-3	40.64	534.48	68.3	582.16	11.50
3	T-L-Q-5	40.62	534.41	70.0	582.59	12.28
Mean		40.60	534.61	68.5	582.42	12.14
Standard Deviation		0.05298	0.28627	1.45808	0.22196	0.57848
Coefficient of Variation		0.13048	0.05355	2.12962	0.03811	4.76634
Range		0.10	0.53	2.9	0.42	1.13
Maximum		40.64	534.93	70.0	582.59	12.63
Minimum		40.54	534.41	67.1	582.16	11.50

b

	Specimen label	Area (mm^2)	Tensile stress at Yield (Offset 0.2 %) (MPa)	Modulus (Chord 100 MPa - 400 MPa) (GPa)	UTS (MPa)	Elongation % (5*W)
1	T-T-Q-6	40.78	527.32	68.2	589.73	11.93
2	T-T-Q-8	40.94	523.87	68.2	585.94	12.14
3	T-T-Q-10	40.84	525.30	68.6	587.21	12.28
Mean		40.85	525.50	68.3	587.63	12.12
Standard Deviation		0.08060	1.73522	0.21305	1.92875	0.17616
Coefficient of Variation		0.19729	0.33020	0.31172	0.32823	1.45389
Range		0.16	3.45	0.4	3.79	0.35
Maximum		40.94	527.32	68.6	589.73	12.28
Minimum		40.78	523.87	68.2	585.94	11.93

Table 4.2: Tensile properties of AA7075T6 sample according (A) the rolling direction, (B) perpendicular to the rolling direction (90).

4.1.3 Optical microscopy of AA 7075-T6 plates

Examination of grain structure measurement of features on the rolled sheet is done after the grain structure is revealed by polishing and etching as appropriate for the material of interest as presented in Figure 7. (Microstructural analysis is explained in details in the methodology section - chapter 3, section 5.1.2).

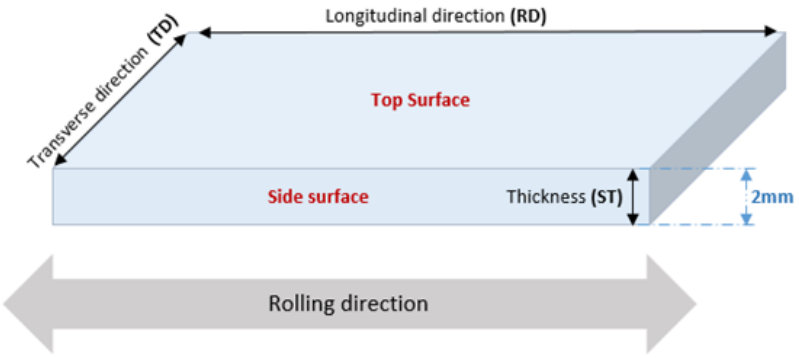


Figure 4.7: The AA7075 alloy plate and direction of analysis.

The typical schematic showing the three principal orthogonal directions in a rolled plate is shown in Figure 8. The grain sizes of the structure were determined by the mean linear intercept for each line on each image taken along with overall average values as presented in Table 3. The grain structure of rolled AA7075-T6 aluminium alloy sheet consists of elongated grains along the rolling direction with typical "pancake" microstructure in the short transverse direction. Figure 9(a) to (c) shows the optical micrographs of the sheet that was used in this study. It can be seen that the typical microstructure as shown in Figure 8 is also observed. The average grain thickness, length and "pancake" diameters are shown in Table 3.

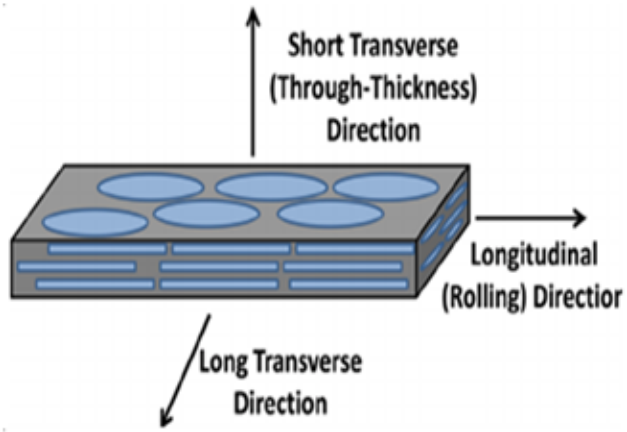


Figure 4.8: Schematic showing the three principal orthogonal directions in a rolled plate[12].

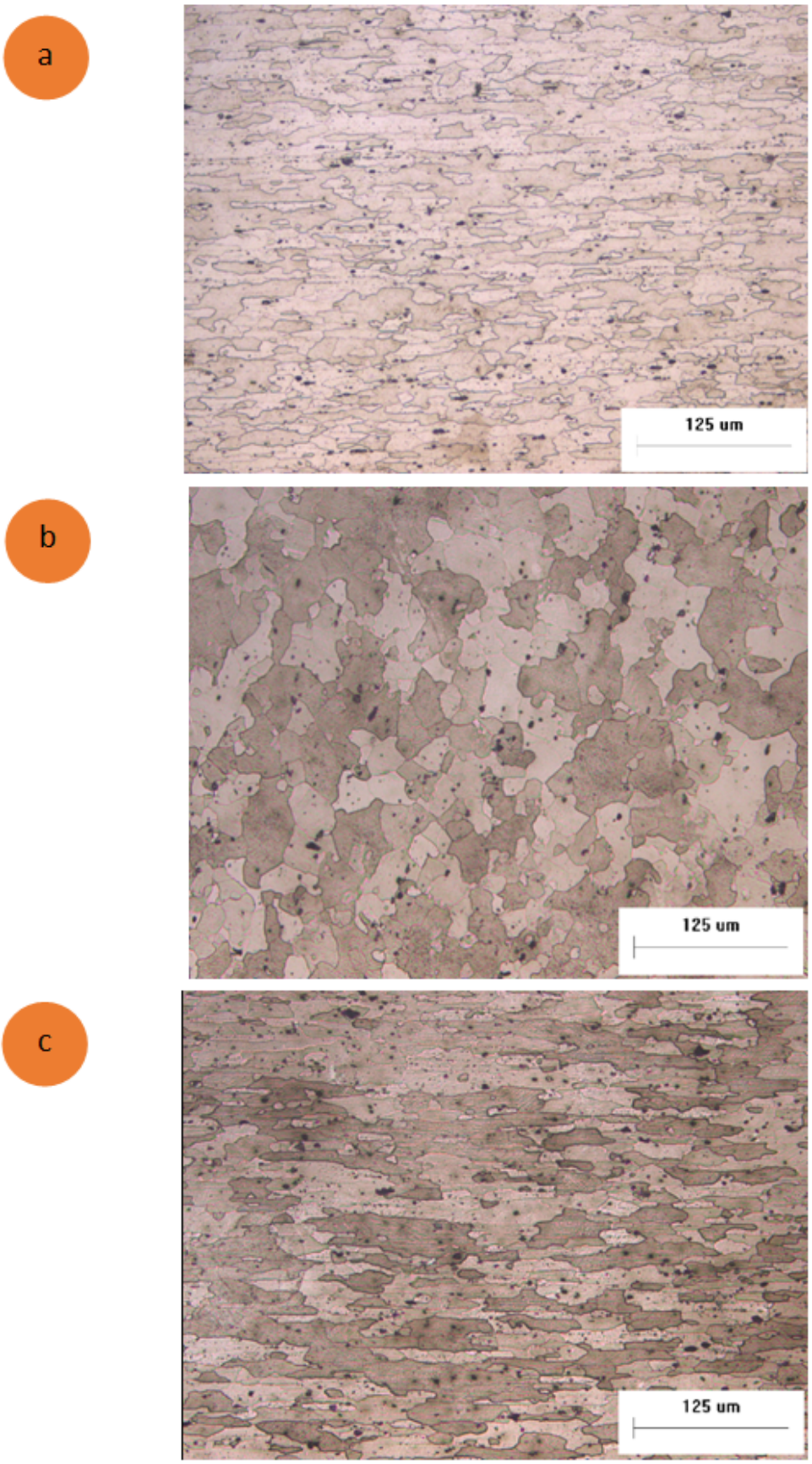


Figure 4.9: Micrograph revealing a microstructure characteristic of AA 7075-T6 (a) Longitudinal (rolling) direction (RD), (b) Short Transverse directions (ST), (c) long transverse (LT).

Description	Sizes
Thickness	10.20 μm
Length	30 μm
Pancake diameter	28 μm

Table 4.3: Average grain size of 7075-T6 aluminium sheet used in this study.

4.1.4 Fatigue crack growth (FCG)

The final dimensions of the AA7075-T6 aluminium plate were 400 mm x 160 mm x 2 mm and contained a 6mm size rivet hole. Specimens are prepared with a sharp-tipped notch, which provides a stress concentration to stimulate crack growth. The wire-cut notch was 1mm length perpendicular to the length of the plate emanating from the hole's edge. The geometry of the plate dimensions is represented in Figure 10 (More details on the machining and dimensions can be found in the methodology section 3.1 - Chapter 3).

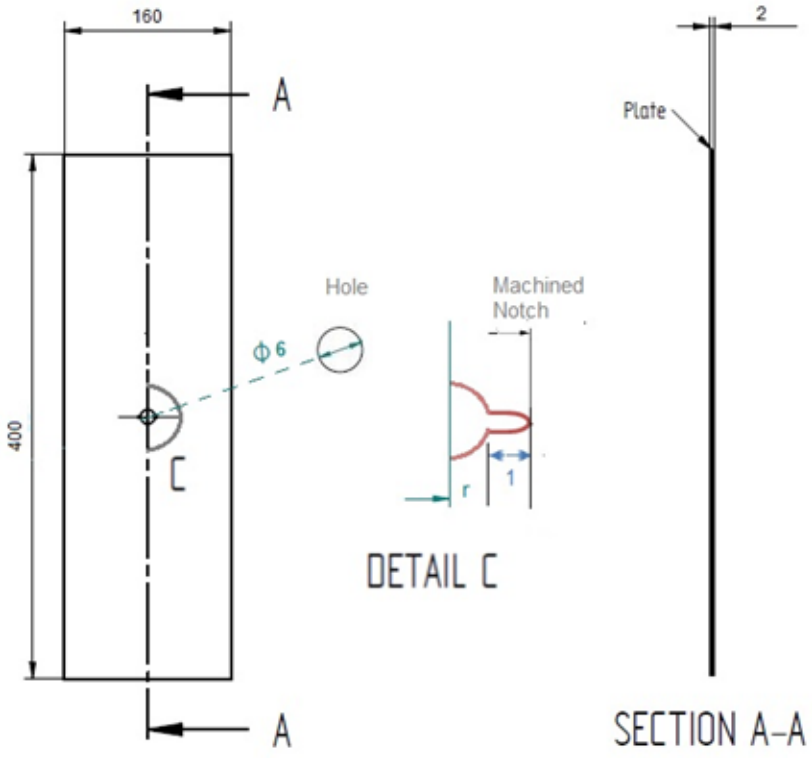


Figure 4.10: Geometric model of the aluminium plate.

In fracture mechanics, a distinction is made between different modes of fracture propagation, depending on the loading situation. There are various modes of loading that can be applied to a crack. Mode I or crack opening mode, where the tensile stress is applied normal to the faces of the crack, is regarded as the most important type of loading [13]. The following discussions refer to the crack opening mode or Mode I only as shown in Figure 11. The testing set-up is shown in Figure 12. Fatigue crack growth tests were performed on an Instron 1342-H1314 servo-hydraulic system on 25 kN dynamic capacity. The fatigue sample plates for this research investigation were on the fatigue behaviour

response to study the crack behaviour of AA7075-T6 plates in the rolling direction. All fatigue tests were conducted at cyclic constant-load amplitude of 22 kN, frequency of 10 Hz and stress ratio of 0.1. Crack length data and number of cycles were logged automatically on a computer.

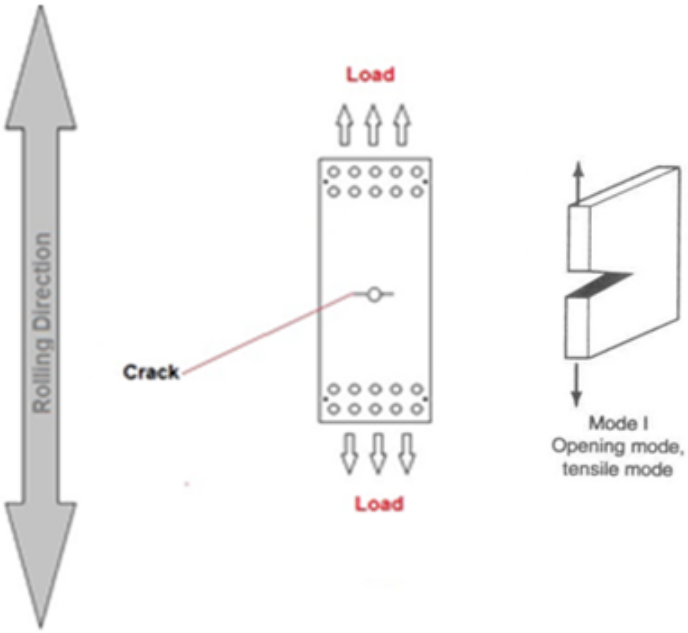


Figure 4.11: The sample sketch of the plate in crack opening mode.

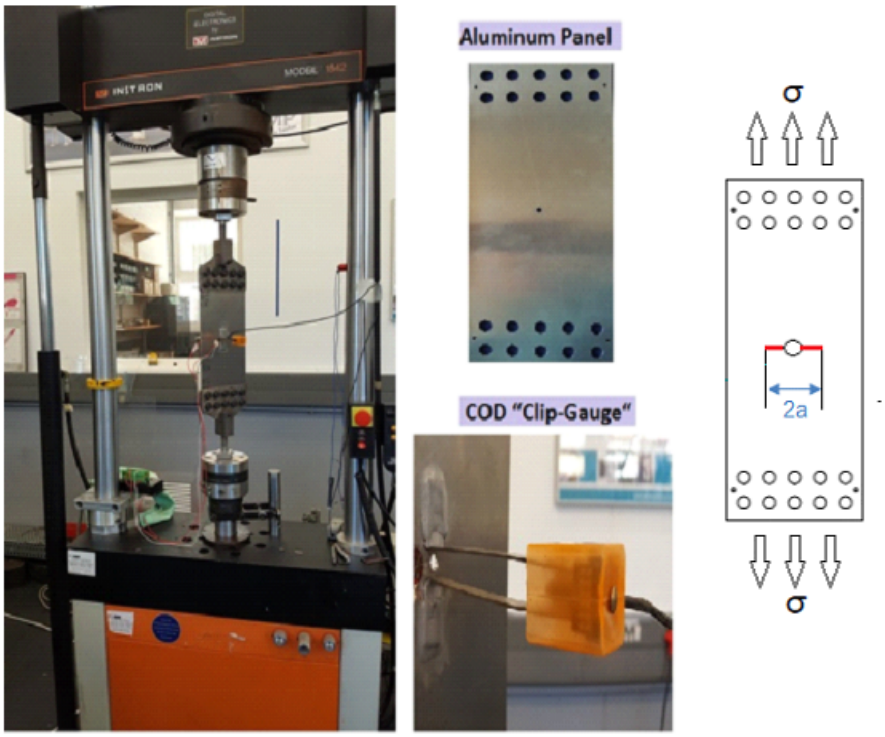


Figure 4.12: Unpatched aluminium Plate inInstron Test Machine Fitted with COD "Clip-Gauge".

Results of Fatigue life and Stress intensity factor vs crack length of uncracked, unpatched AA7075-T6 sample are presented in Figures 13 and 14. The number of cycles increases with the increase of the crack length during propagation. That can be seen in Figure 14, where the stress intensity factor K values increase with the increase in crack length. That can be explained that rigidity of a sample in the presence of a crack from a notch is weakened. Under repeated strain, the crack propagates rapidly towards the free edge of the plate until fracture.

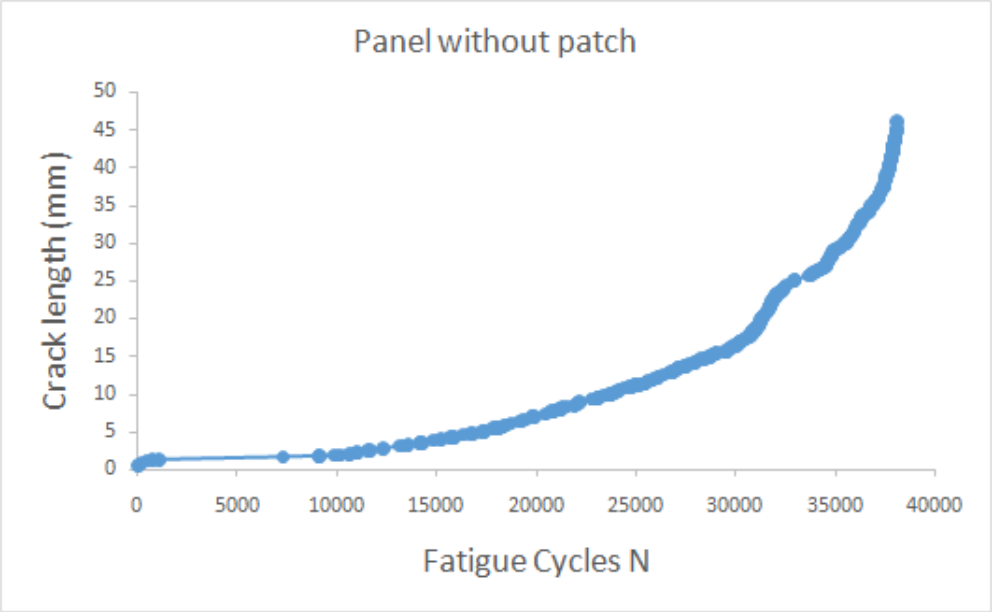


Figure 4.13: Fatigue life of AA7075-T6 sample without a patch.

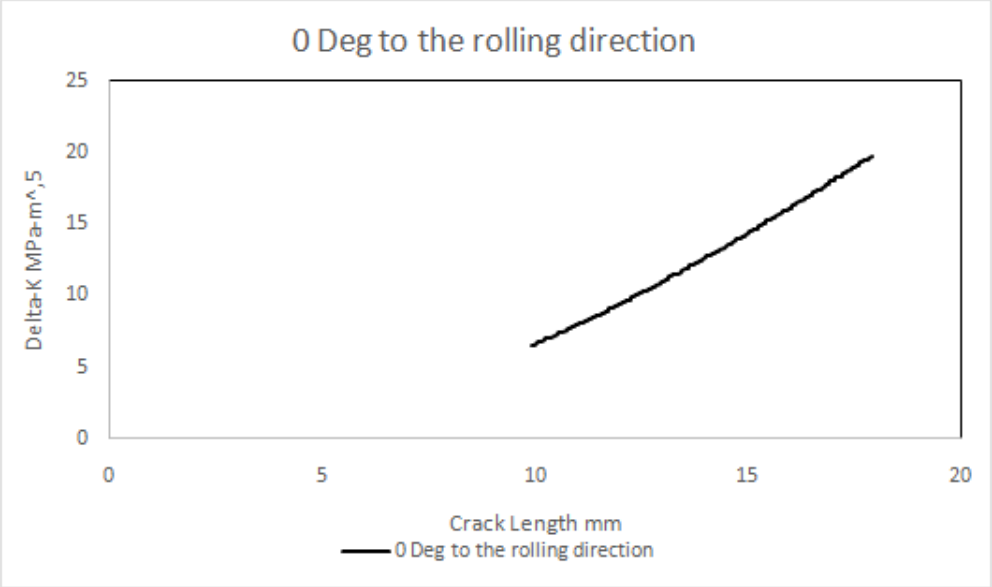


Figure 4.14: Stress intensity factor vs crack length of AA7075-T6sample without a patch.

Even small flaws can by time develop and become a real threat to a structure. Under

cyclic loading, a small crack of an initially negligible size can grow until it reaches the critical size (a_c) for brittle fracture, which is a behaviour called fatigue crack growth.Consider a growing crack that increases by an amount $\triangle a$ due to the application of a number of cycles $\triangle N$. The rate of growth with cycles can be characterized by the ratio $\triangle a/\triangle N$ or, for small intervals, by the derivative da/dN. A value of the fatigue crack growth rate da/dN is the slope at a point on a crack length versus cycle's curve [14]. The stress intensity factor K, can be used as a basis for fatigue crack growth analysis. The stress intensity factor defines the severity of a crack, and K. therefore controls the rate of fatigue crack growth. K depends on the combination of crack length, geometry and loading, for cyclic loading the stress intensity varies with the fluctuating nominal stresses. It is therefore necessary to look at the range of the stress intensity factor, $\triangle K$. The stress intensity range is given by the difference between the maximum and minimum stress intensity of the load cycle, which is determined by the stress range $\triangle S$. It is also convenient to introduce the stress ratio R.

The stress intensity factor concept provides also a very useful basis for the quantitative description of fatigue crack propagation (FCP) rates, i.e. the rate at which an existing crack advances during cyclic loading. It has been recognized that the FCP rates depend on the stress field close to the crack tip. In view of this, Paris et al. [15] related the FCP, da/dN (mm/cycle), to the stress intensity factor range, using the equation. The linear region of the plot in Figure 15 can be described by the Paris region known in the following equation as the Paris Law equation after Paul Paris who first presented it in 1960.
Where:

K_{max} = Stress intensity factor corresponding to the upper load limit.
K_{min} = intensity factor corresponding to the lower load limit.
C and m = Constants to be determined experimentally for a particular material and set of conditions.

The sigmoidal variation of the FCP rates with the alternating stress intensity factor (SIF) range is shown schematically in Figure 15. Three regimes, governed by different failure mechanisms, are discernible: which illustrates typical fatigue crack growth behaviour [15].

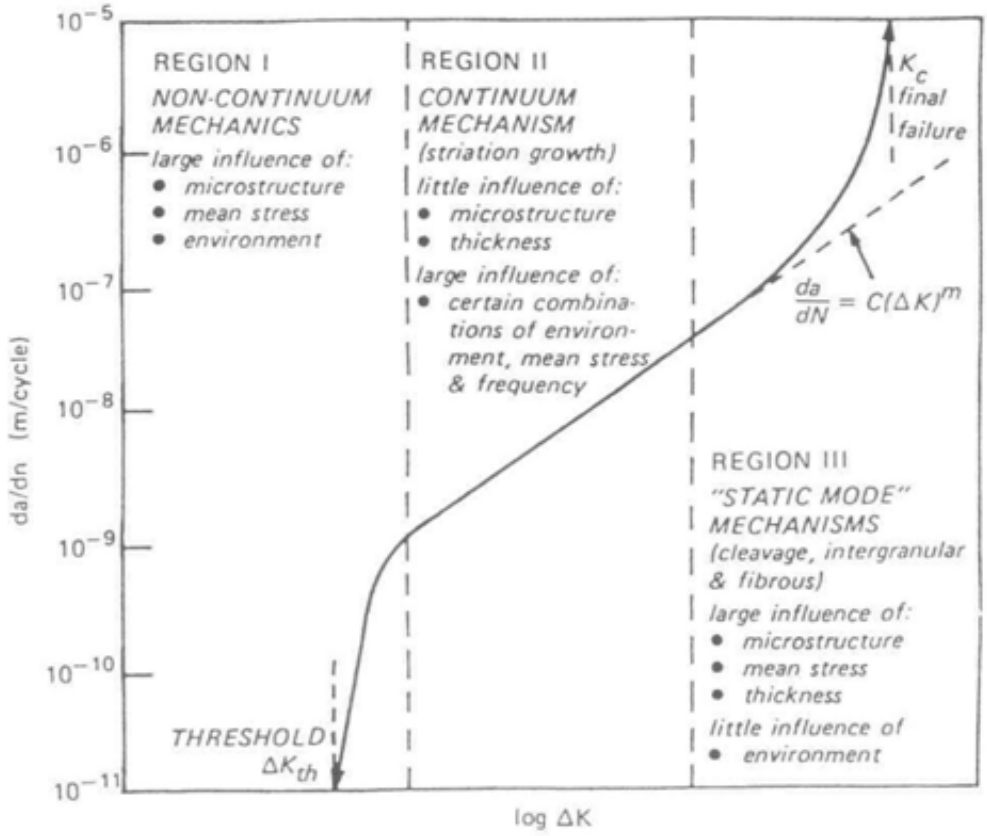


Figure 4.15: Variation of fatigue crack propagation rates da/dN with SIF range $\triangle K$ [15].

The standard Test Method E647 issued by the American Society for Testing and Materials describes the determination of fatigue crack growth rates. The data reduction procedure are briefly summarized in the following[16]:

- Construct a plot of da/dN vs $\triangle K$ on a double-logarithmic scale to obtain the fatigue crack propagation (FCP) curve.
- Perform a linear regression analysis over the truncated linear range of the FCP curve to obtain the Paris equation $da/dN = C(\triangle K)^m$.

An Instron software written following the specifications of ASTM E 647 is used at the mechanical testing laboratory of CSIR for machine control and data reduction. From the knowledge of the material's fracture toughness and the constants in the Paris equation,these properties have to be determined by testing laboratory specimens - the number of cycles to failure can be calculated for specific loads. Figure 16 represents the mid growth rate regime (Region II in Fig. 15); where the Paris law is valid, the procedure discussed in the form of fatigue crack growth FCG of AA7075-T6 sample results.

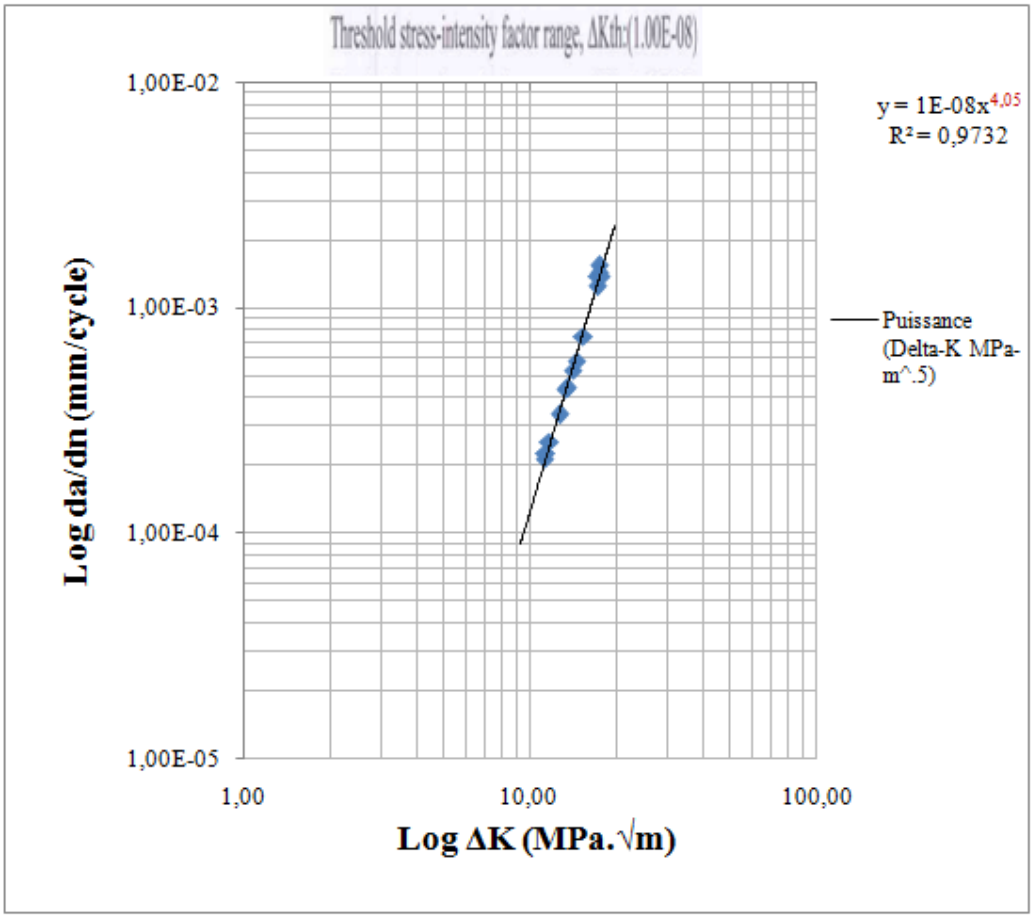


Figure 4.16: Plot of log fatigue crack growth rate da/dN versus log $\triangle K$ of AA7075-T6 sample.

This test method involves cyclic loading of notched specimens which have been acceptably cracked in fatigue. Crack size is measured, either visually or by an equivalent method, as a function of elapsed fatigue cycles and these data are subjected to numerical analysis to establish the rate of crack growth. Crack growth rates are expressed as a function of the stress-intensity factor range, $\triangle K$, which is calculated from expressions based on linear elastic stress analysis. A crack growth rate of 0.036 μm /cycle was reached at a $\triangle K$ threshold of 1e-8 where C and m are two parameters of the material defining, respectively, the position and the slope of the line of Paris where in Fig.16 the Paris law parameters $C = 1e - 8$ at $\triangle k = 1$ and m= 4.05 of depend strongly on the material properties. The parameters C and m found are in a good agreement with Paris law parameters and confirm our results,where m is typically around 4, and C around 10^{-8} to 10^{-9} [15].

4.1.5 Fatigue crack growth & Failure relationship

Fracture mechanics is a powerful science for predicting the integrity, i.e. the load-carrying capability of structures and components containing sharp flaws and cracks of known size and location. The degradation and fracture in engineering materials and structures have been and still are of continuing concern for both material scientists and mechanical engineers. These failures may occur under quasi-static and/or alternating

(dynamic) loads. Consequently, this concept has evolved, as the basis for fracture control, i.e. the combination of nominal stress and existing defect or crack size must be kept below a critical level for a structure or component under service conditions.

The path along which fractures occur can go through the grains (transgranular fracture) or along the grain boundaries (intergranular fracture). Regardless of the fracture path, there are essentially only four principal failure modes namely, dimple rupture, cleavage, fatigue and decohesive rupture [17]. Each of the principal failure modes is controlled by specific mechanisms that result in typical fracture surface morphologies. Consequently, the fractographic analysis in the scanning electron microscope (SEM) has emerged as a valuable tool towards the understanding of fracture processes [18]. The mechanisms of the various fracture modes have been explained in terms of dislocation interaction involving complex slip and crystallographic relationships [19]. The dislocation-based models which have been proposed to explain fracture mechanisms are mathematically complex and are beyond the scope of this research. The modes of mechanical failures investigated here is fatigue crack propagation in low cycle fatigue (LCF).This research deals only with slow cracking, fatigue crack growth which is controlled by the fracture toughness (resistance to unstable crack extension) of the material. In the following section,firstly, a crack propagation path emanating a hole with a notch is shownbefore failure and after failure in Figure 17 and Figure 18 respectively. Secondly, crack propagation on the microstructural scale is discussed which are used to determine the fracture mechanism on the basic concept of fracture mechanics and shown in Figure 19. Thirdly, a microstructural analysis on the crack propagation path is shown in Figure 20 with the cracking stages until failure.



Figure 4.17: Crack propagation path of AA7075-T6 uncracked sample during the fatigue test.

In a visual observation and computer monitoring, all tested samples exhibited the same crack growth behaviour. The crack started in a very slow steady straight path state emanating from the notch until the crack reaches between 20-25 mm length; after that,the crack started changing orientation in 45 angle with zigzag path and it started growing rapidly after exceeding a crack length of 35 mm. The crack propagation path can be seen in Figure 17 where the case of the crack length approached the edge of pre-failure of the plate.

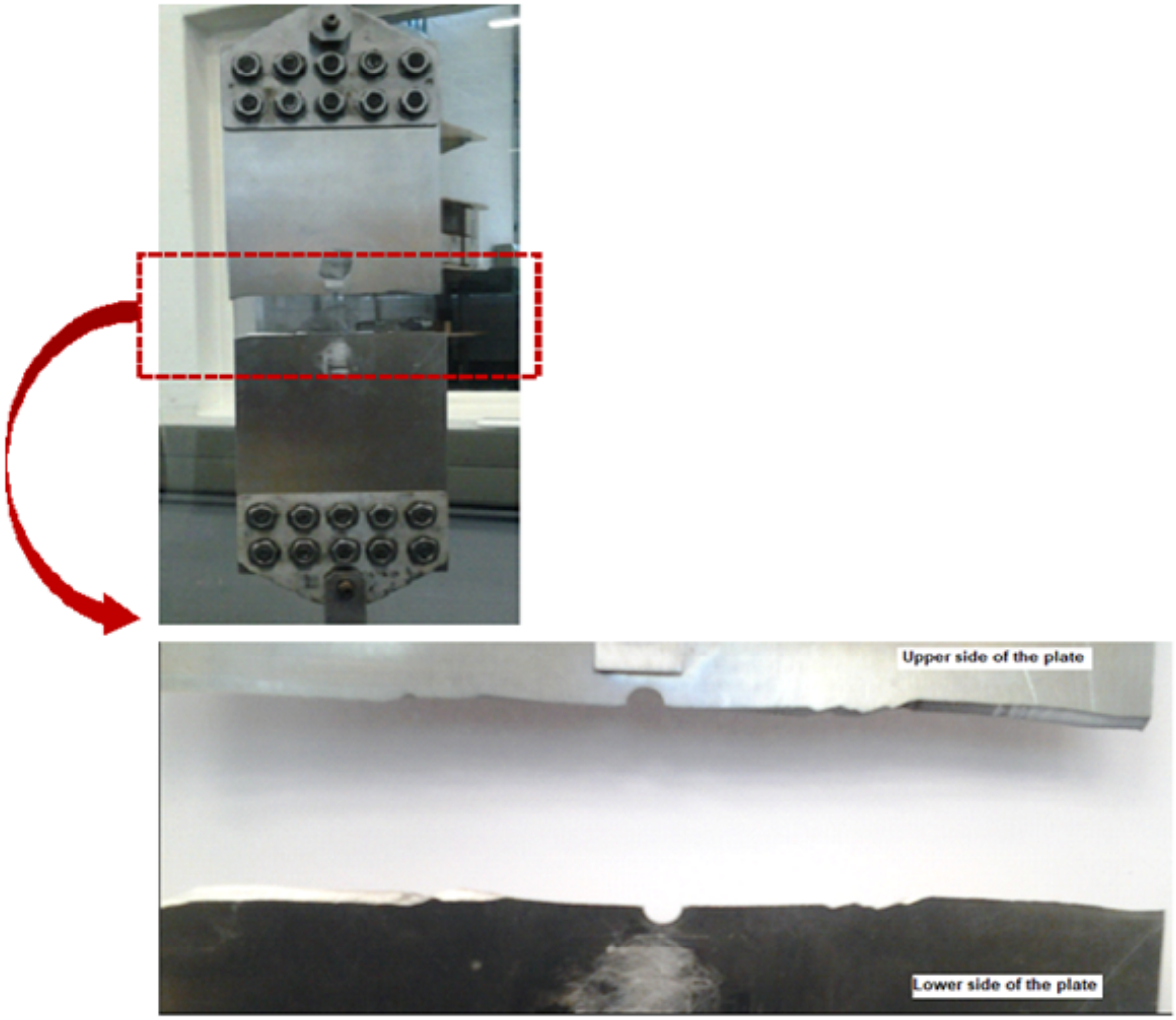


Figure 4.18: Fractured sample of AA7075-T6 fatigue test.

From the lower side of the plate in the fractured sample shown in Figure18. Figure 20 shows the overall type of failure mechanism along the crack propagation path on microstructural view.

i) Region 1: Near the notch and slow growth rate with a straight line.

ii) Region 2: In the midsection where transition is between the straight line region and the change of crack growth direction.

iii) Region 3: At high crack growth rates. This is due to rapid unstable crack growth just prior to final fracture.

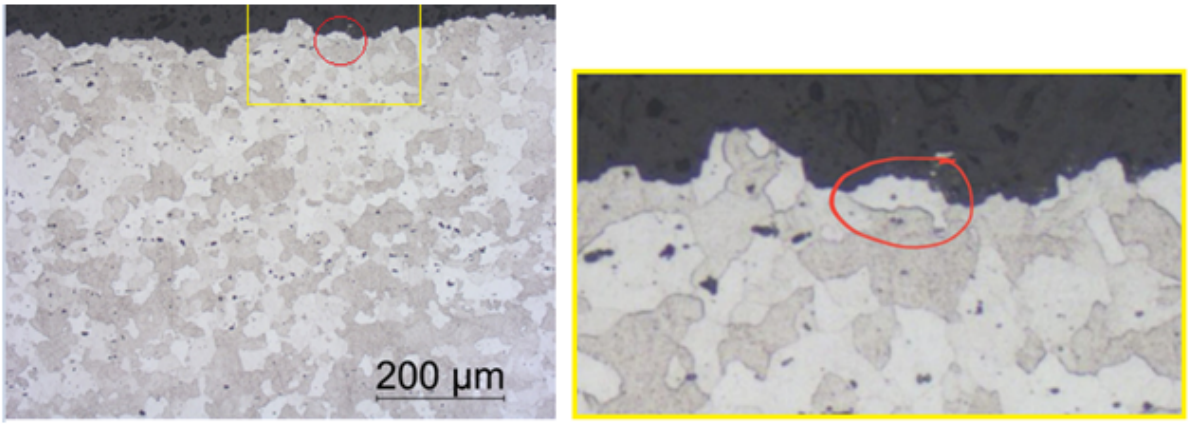


Figure 4.19: Transgranular failure mechanism in AA7075-T6 fractured sample after fatigue test.

Our results shows transgranular fracture as presented in Figure 19. The crack propagation path was perpendicular to the loading. The crack propagation was analysed with an optical microscope. It was noted that the crack propagation was transgranular in all crack growth stages. The crack propagation path through the grain can be seen in detail in Figure 20.

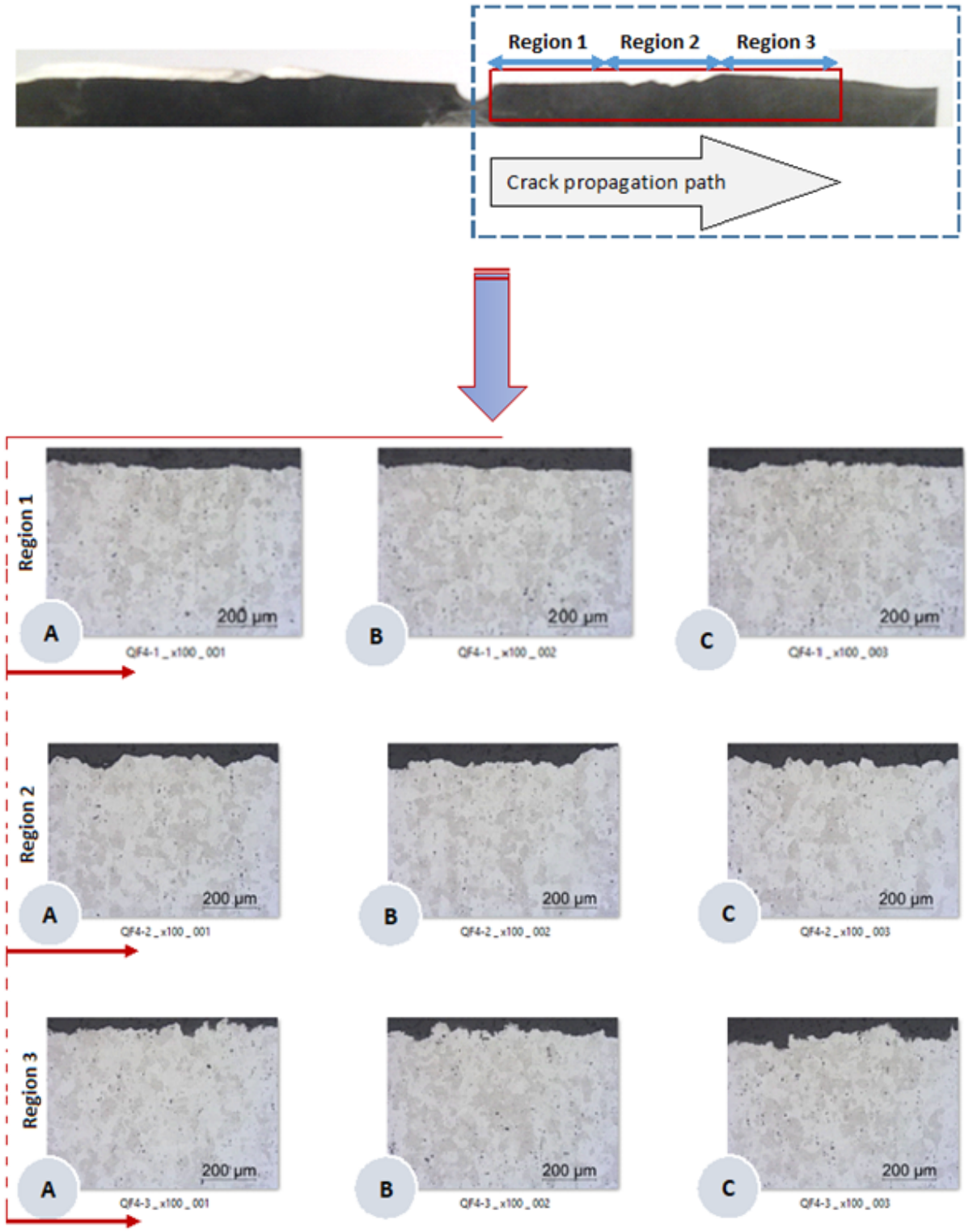
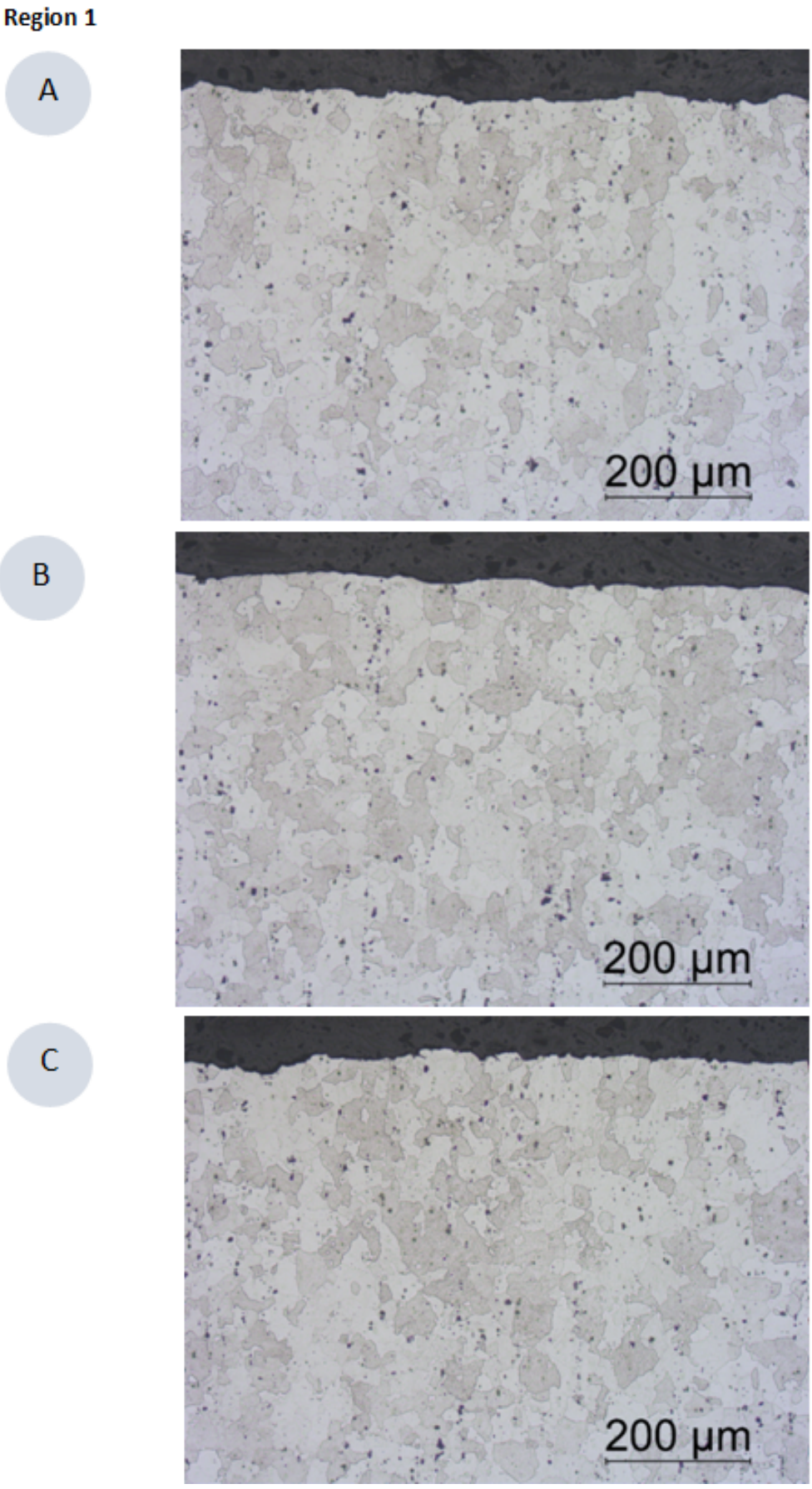
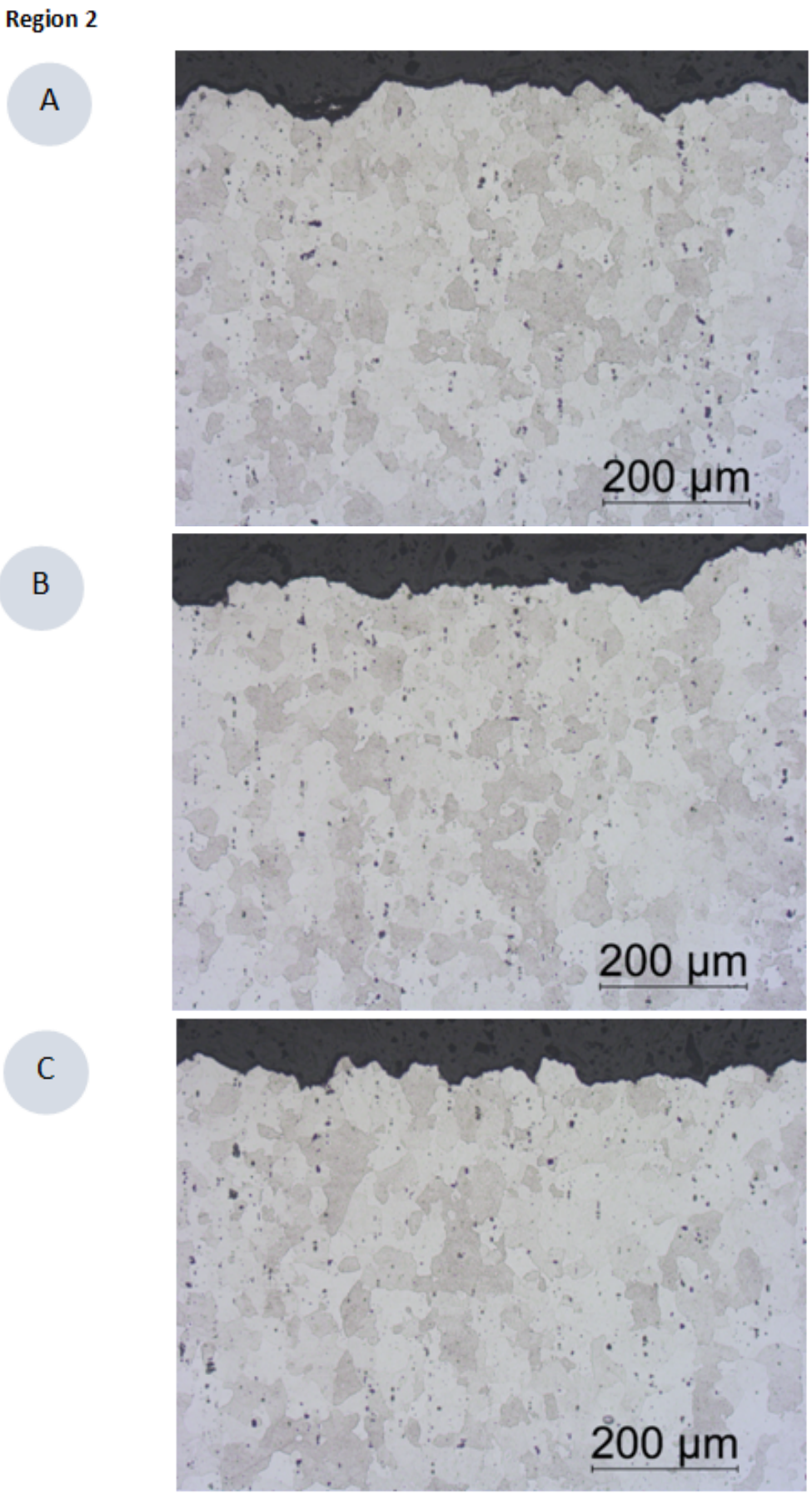
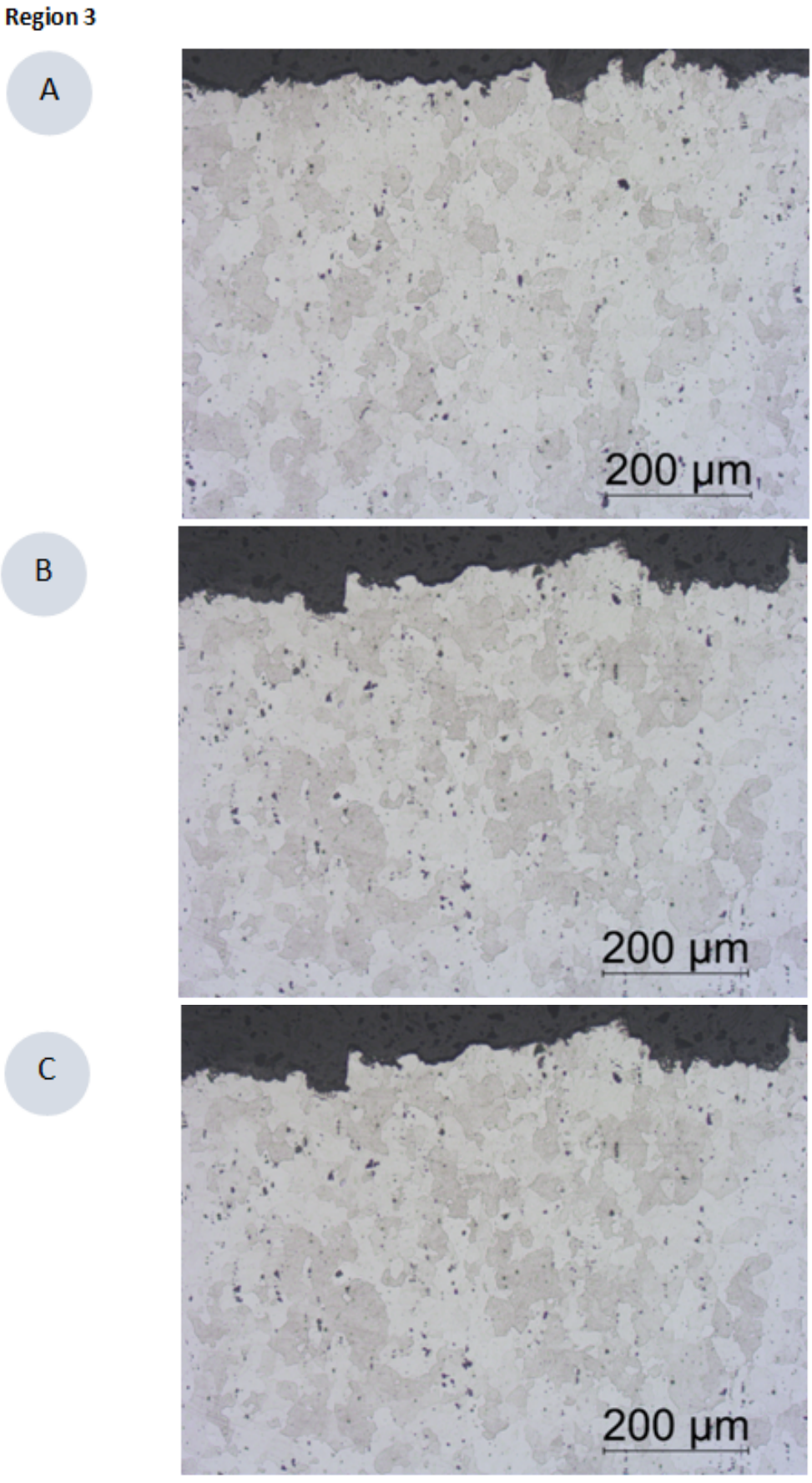


Figure 4.20: Optical microscope analysis of crack propagation path after fracture.


Region 1
A
200 µm
B
200 µm
C
200 µm


Region 2
A
200 µm
B
200 µm
C
200 µm


Region 3
A
200 µm
B
200 µm
C
200 µm

4.1.6 Conclusion on fatigue behaviour of unpatched, notched AA7075-T6 plate

The crack growth behaviour was monitored in correlation with fracture mechanics, the obtained results were analysed to establish the fatigue behaviour understanding for uncracked, notched AA7075-T6 aluminium alloy. The stress intensity factor (SIF) defines the severity of a crack. Therefore, SIF controls the rate of fatigue crack growth. The path along which fractures occurred was through the grains (transgranular fracture) in all crack growth stages. It showed slow growth with a straight line near the notch while,due to rapid unstable crack, growth just prior to final fracture a high crack growth rate was observed.For unpatched plate, our study results have shown a good agreement with data in the literature. The understanding gained from this work will be the basic principle on fatigue crack growth behaviour for comparison with the specimens repaired using adhesively bonded composite patch (boron/epoxy). The repair method where it is to adhesively bond a composite patch over the damaged area to restore the load-carrying capacity of a damaged structure is discussed in the following **part II** in details with the findings.

4.1.7 References

[1] Anonymous "Metallic materials and elements for aerospace vehicle structures," in MIL-HDBK-5H, Military Handbook Anonymous United States of America: Department of Defence, 1998.
[2] P. J. E. Forsyth, "Fatigue damage and crack growth in aluminium alloys." in ActaMetallurgica, 11th ed.Anonymous 1963, pp. 703-715.
[3] R. W. Hertzberg, Deformation and Fracture Mechanics of Engineering Materials. (4th ed.) United States of America: Wiley, 1996.
[4] Suresh, S. (2006). Fatigue of Materials 2nd Edition. Cambridge University Press, New.
[5] Beden, S.M. Abdullah, S. Ariffin, A.K. & Al-Asady, N.A. (2010). Fatigue crack growth simulation of aluminium alloy under spectrum loadings. Mater. Design, 31: 3449-3456.
[6] Chakherlou TN, Vogwell J. The effect of cold expansion on improving the fatigue life of fastener holes. Engineering Failure Analysis 2003; 10(1): 13-24.
[7] Bureau of Accident Investigation, Aircraft Accident Report: Aloha, Airlines Flight 243; Boeing 737-200; N73711; Near Maui; Hawaii; April 28; 1988. Report No. NTSB/AAR-89/o3; National Transportation and Safety Office 14 June 1989; Washington.
[8] Sylvie Pommier, Basics elements on linear elastic fracture mechanics and crack growth modelling Sylvie Pommier, 2017; 11-12.
[9] Hoffman ME, Hoffman PC. Current and future fatigue life prediction methods for aircraft structures. Navel Res Rev 1998; 50(4):55-68.
[10] Jiang Y, Feng M. Modeling of fatigue crack propagation. ASME Journal of Engineering Material Tech 2004; 126:77-86.
[11] http://asm.matweb.com/search/SpecificMaterial.asp?bassnum=MA7075T6 , 2018.
[12] R. L. Whelchel, Spall behaviour of rolled aluminum 5083-H116 plate Journal of Applied Physics 113(23), June 2013.
[13] ASTM E561: Standard Practice for R-Curve Determination. American Society for Testing and Materials, Philadelphia, PA 19103, USA.
[14] Dowling, N.E. 2012. Mechanical behaviour of materials: Engineering methods for deformation, fracture and fatigue. 4th edition, Prentice Hall.
[15] P C Paris, M P Gomet and W E Anderson (1961). The Trend in Engineering, Vol. 13, 9-14.
[16] da/dN of precracked metallic materials Fatigue Crack Propagation" Standard Test Method for Measurement of Fatigue Crack Growth Rates1," in ASTME647: ASTM International, 2013.
[17] V Kerlins (1987). Modes of Fracture. Metals Handbook, Ninth Edition, Vol. 12, 12-71.
[18] B L Gabriel (1987). Scanning Electron Microscopy. In ref. 1, 166-178.
[19] AA Griffith (1920): Phil. Trans. Royal Soc, A221, 163-198.

4.2 PART II: PATCHED PLATE

EXPERIMENTAL STUDY ON THE BEHAVIOUR OF CRACKS REPAIRED WITH BORON FIBRE/EPOXY COMPOSITE PATCH IN ALUMINIUM 7075-T6

Summary

The composite bonded repair technology has been proven an efficient and reliable method for use on metallic aircraft structures. The purpose of this method is to slow or arrest crack growth, and thus, prolong the life of the structure. The technology has been successfully used in repair applications on both military and civil aircraft for fatigue enhancement and crack patching. Aircraft part repair is the easiest and cheapest way to address the problem. By focusing on fixing individual part damage, as opposed to replacing the entire part or airframe, the time and money needed to get the airframe up and running again is reduced. Bonded Repair technology is becoming more and more advanced and accepted, but additional research is always needed to help predict how well bonded composite patches can perform their job. In this study, the performances of the composite patch were compared to those of unpatched one. Experimental investigations were carried out in order to evaluate and characterize the effect of patch repair on the fatigue life of thin AA7075-T6 aluminium plates as follow:

A. **Fatigue life of a notched hole AA7075-T6 aircraft plate reinforced with a boron/epoxy and carbon/ epoxy composite patches**.

B. **Fatigue life of AA7075-T6 aircraft plate cracked with various crack length repaired with a boron/epoxy composite patch**.

C. **Effect of curing cycle on fatigue life of AA7075-T6 aircraft plate reinforced with a boron/epoxy composite patch**.

4.2.1 Introduction

Advanced composite technology has been widely used for several years to reinforce metallic structures in aerospace applications. One of the resulting technologies is the bonded patch, which is now in some sense finding applications in other areas such as civil engineering. Design, analysis and technology for repair integrity continue to be an engineering challenge. Bonded composite repairs of locally damaged metallic structures have gained considerable interest in aircraft structural maintenance and life extension in the recent decades, where the US Air Force and the aerospace industry have shown considerable interest in the use of bonded composites as a repair technique for metallic structures [20].A badly implemented repair can be more dangerous than an unrepaired configuration. The materials evaluation shows that the most suitable applications are boron/epoxy and carbon/epoxy composites. The thermal expansion coefficient of boron/epoxy makes it a better choice [21]. The primary advantages of composite materials for repair are

their high strength, relatively low weight, and corrosion resistance [22]. The Aeronautical Research Laboratory in Australia has pioneered composite repair doublers bonded to metal substructures, which have been used in development programs and partially in fleet repair programs, as an alternative to bolted repairs. They showed that the bonded composite repair improved the durability of aircraft structures under practical conditions [23]. Therefore, fatigue and reliability analysis and prediction of structural performance are critical to investigate, especially with the development of composite materials. Studies on this have been done on various cases and using different techniques [24-26].

A detailed method of boron/epoxy composite patch repair procedures of damaged metallic structure was presented in details in the methodology section 3 and section 4 in Chapter 3 and briefly presented in the following section. In this effort to provide necessary material data. Thin aluminium plates were machined to the desired configuration, precracked, repaired with an adhesively bonded composite patch, and then fatigued under constant-amplitude stress fatigue cycling until the crack tips reached the plate edge and crack lengths could no longer be measured or failure occurred. This part details the results of the study and discusses exactly what impact the results have on the technology.

4.2.2 Materials and methods

4.2.2.1 Materials

The layers per plate were: an aluminium 7075-T6 plate, an adhesive layer and a structural composite layer. The dimension of the AA7075-T6 aluminium plate was 400 mm x 160 mm x 2 mm, and contained a 6 mm size rivet hole with a wire-cut notch of 1 mm length perpendicular to the length of the plate emanating from the hole's edge. The geometry of the plate and composite repair patch is represented in Figure 21. Cytec/Solvay FM73 knitted epoxy film of 0.25 mm thick was used as an adhesive between the aluminium and the boron/epoxy patch. The structural composite layer was composed of plies, 5.2-mil thick, of type B4/5521 unidirectional 4-mil Boron/epoxy prepreg tape from Specialty Materials, Inc. Test materials utilized in this study for the bonded repair of structures are presented in details in the methodology section.

The chemical composition (in weight %) given for the 7075-T6 plate by the manufacturer was 5.6 %Zn, 2.6 %Mg, 1.4 %Cu, 0.19 %Cr, 0.13 %Fe, 0.07 %Si, and aluminium the balance. The whole study was carried out based on the repair with boron fibre epoxy composite patches. However, in the following section, a comparison between repair materials was investigated, a carbon fibre epoxy (SigratexKDU1092/Epolam2022) composite material was utilised in a unidirectional direction with woven structure [27]. Material properties of the used in this study are summarised in Table 4.

Properties	Materials			
	AA7075-T6	Boron/epoxy	Carbon epoxy[26]	Adhesive (FM73K)
Longitudinal Young Modulus (GPa)	68.5	200	93	2.778
Transverse Young Modulus (GPa)	68.3	18	7.37	2.778
Poison Ratio	0.33	0.23	0.3060	0.38
Hardness (HV20)	186	-	-	-
Elongation at break (%)	12	-	-	-
Shear Modulus (GPa)	-	5.59	3.94	1.006
Thickness (mm)	2	0.13/ply	0.33/ply	0.25

Table 4.4: Mechanical properties of different repair materials.

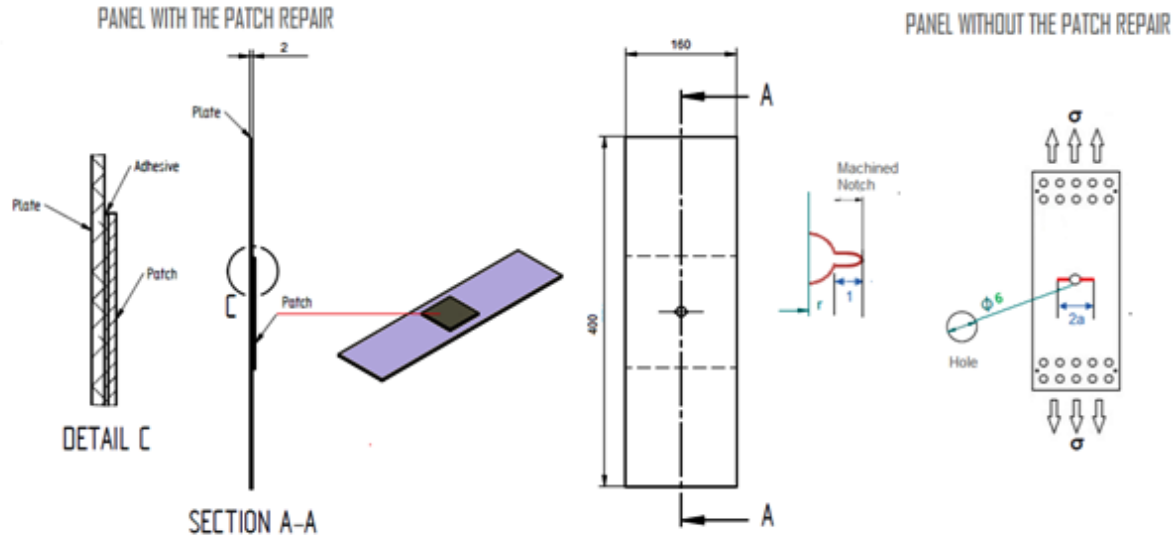


Figure 4.21: Geometry of the plate and composite repair patch.

4.2.2.2 Surface treatment

An important factor influencing the strength of the bonded joint is the surface preparation of the metal. Many studies have been done on the influence of surface treatment on the mechanical behaviour of bonded assemblies [28-30]. To prepare the bonded area, the metal surface was subjected to mechanical abrasion (grit blasting) with uniform alumina grit. The process employed 210μm alumina grit at a pressure of 300 kPa and a working distance of 10 to 15 cm to obtain the required surface roughness without induc-

ing compressive stresses. The surface was then cleaned with trichloroethylene to remove impurities. Figure 22 shows the bare prepared aluminium plate.

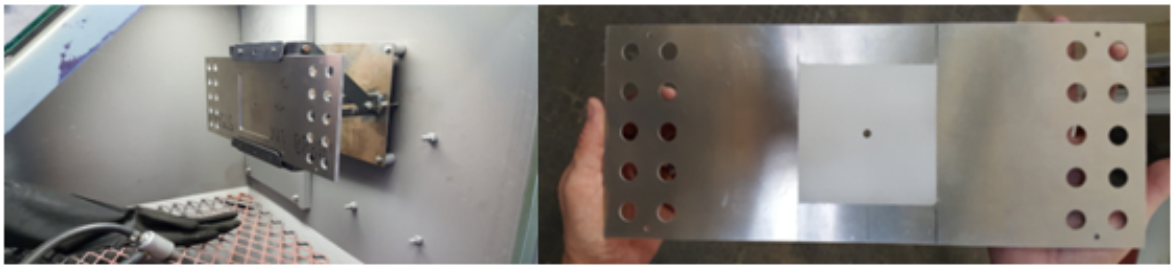


Figure 4.22: Surface preparation by grit blasting.

4.2.2.3 Composite plate manufacturing

The lay-up is shown schematically in Figure 23 and shows the sequence. The 7075-T6 aluminium was the base plate; it was bonded with the FM73 adhesive to six layers, type B4/5521 unidirectional prepreg tape. In the case when the crack position is perpendicular to the loading, the best stacking layup is with the fibres in the unidirectional orientation;as the fibres will inhibit the opening of the notch in fatigue [31]. It is recommended to utilise the orientation of the fibres for the composites layups in the perpendicular direction to the damage direction to obtain the optimum results, as also noted by other authors [32,33]. Plies were all stacked with the unidirectional orientation running perpendicular to the crack and parallel to the fatigue loading direction, the rivet hole being at the centre of the patch. The following conventional layers were applied over the prepreg tape: peel ply, release film, breather fabric cloth, and then the vacuum bag. The vacuum was sealed off at the edge of the vacuum bag with Cytec sealant tape.

The full plate with the layup was put in an air convection furnace for curing while maintaining a vacuum of 80 mbar drawn over the layup. An overview of the final samples is shown in Figure 24 and the steps to the final patched test plate. The patch and adhesive had dimensions of 100 mm x 100 mm x 1.03 mm (thickness including six prepreg tape layers and the adhesive layer).

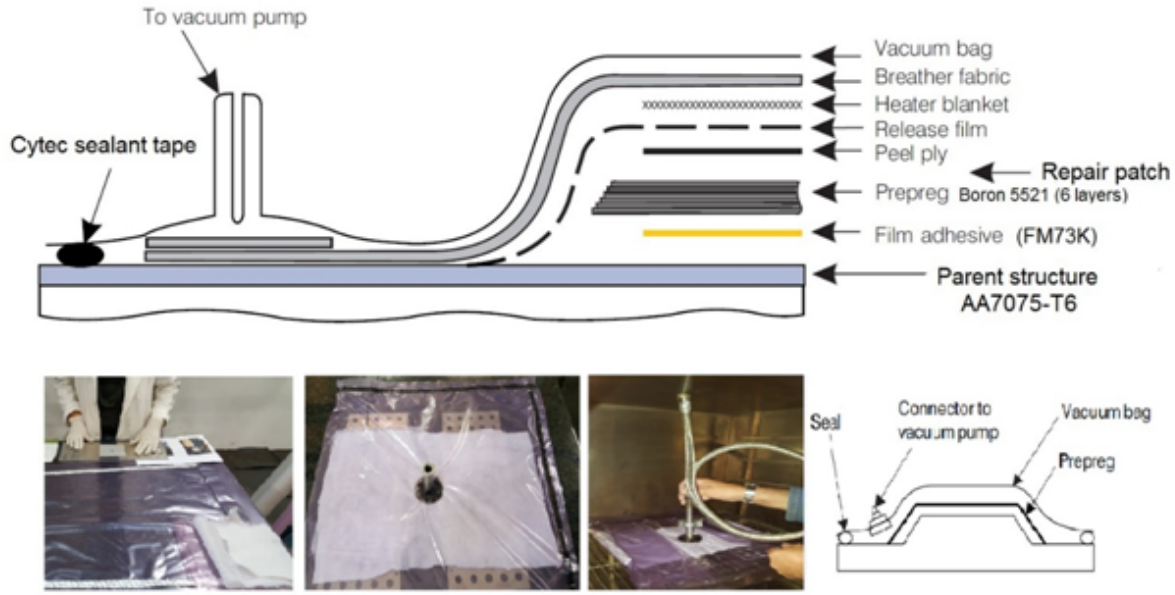


Figure 4.23: Detail of vacuum bag lay-up.

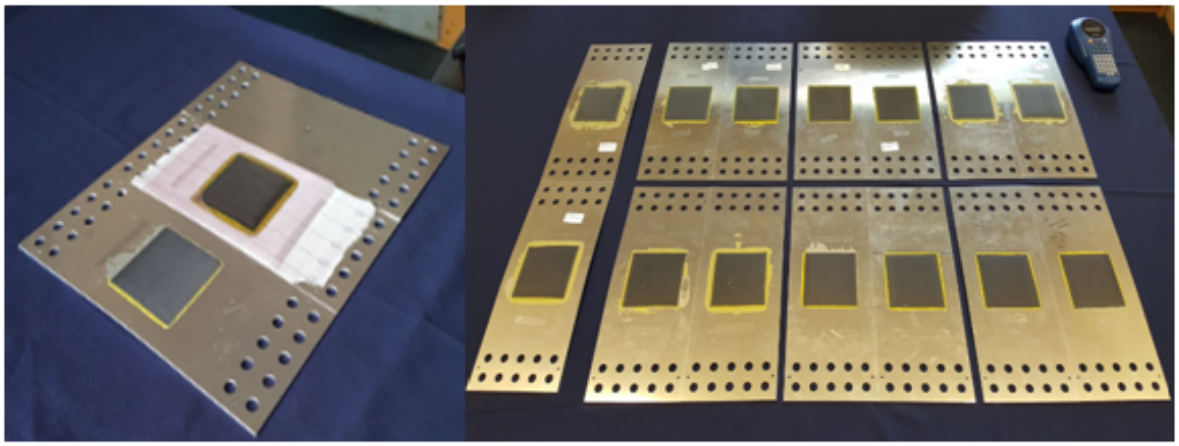


Figure 4.24: The steps to the final patched test plate.

4.2.2.4 Curing parameters

Three plates were made for each cure condition to ensure repeatability and traceability. The full plate with patch was heated in a convection furnace. The ramp rate to all the curing temperatures was 2 $^{\circ}C/min$. Plates were taken out of the furnace and left to cool at ambient conditions after curing. To ensure repeatability, the curing process was controlled by the computer while the temperature of the workpiece (composite plate) was continuously monitored by a thermocouple fixed to the aluminium base plate. The curing condition was used as per manufacturer specification (120 $^{\circ}C$ for 60 minutes). An example of a curing sample log on a computer is shown in Figure 25.

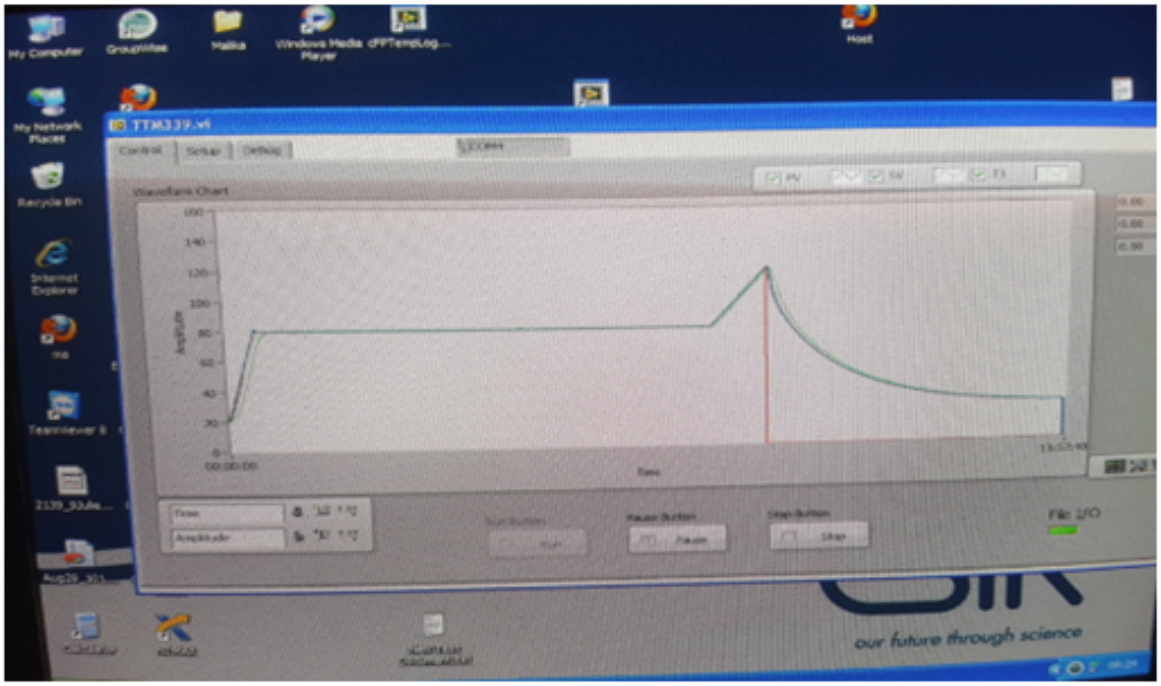


Figure 4.25: Curing cycle log.

4.2.2.5 Fatigue testing

Fatigue crack growth tests of repaired and unrepaired structures were performed on an Instron 1342-H1314 servo-hydraulic system with a dynamic capacity of 25 kN. Samples dimensions were chosen to closely match dimensions where the width of the plate is wider to comply with the spacing in stiffened structures of aeroplane skin. The test plate setup is illustrated and the completed repaired structure with single-sided composite patch is shown in Figure26. Fatigue tests were conducted at a frequency of 10 Hz under constant amplitude loading $\triangle F$ equal to 22 kN and a load ratio $R = F_{min}/F_{max} = 0.1$. This ratio is a decisive factor in the fatigue life of structures. Non-negative stress ratio was chosen to avoid the closure of the crack under the minimum load Fmin, and thereby ensure its full opening during the test. Recall that the amplitude of solicitation $\triangle F$ does not vary whatever the state of the specimen is: repaired or unrepaired.Crack length data and number of cycles were logged on a computer. Plates were fatigued until crack lengths approached the edge of the patch or failure occurred.



Figure 4.26: Completed plate with composite patch loaded in tension.

4.2.3 Fatigue life of a notched hole in AA7075-T6 aircraft plate reinforced with a boron/epoxy and carbon/ epoxy composite patches (A)

This part deals with the fatigue life a notched holein AA7075-T6 aircraft plate repaired with a boron/epoxy and carbon/ epoxy composite patches. First, boron and carbon/ epoxy patches repair results datais introduced. Second, the effect of patch repair with a boron/epoxy and carbon/ epoxy composite patches in AA7075-T6 aircraft plate repaired is discussed. Then, the behaviour of bonded repair is studied. The comparison between using different repair material is concluded. Also, the repair patch effectiveness on the stress state at the crack tip in increasing the fatigue life of the unrepaired structure is shown.

4.2.3.1 Fatigue life enhancement with bonded composite repair

The plate is symmetrical along its length with respect to the hole and notch. The crack length starts at 4 mm, which is the 3 mm radius of the hole plus 1 mm for the machined notch.The crack length measurement is calibrated at the start of each test and measurement data was automatically logged. The fatigue propagation curves terminate at 45 mm which was chosen and defines fatigue life, in the case of the repaired plates to prevent the crack from growing past the edge of the half-patch at 50 mm.

The fatigue life (crack length vs number of cycles) for unrepaired is shown in Figure27. The fatigue life (crack length vs number of cycles) for repaired plate with boron fibre epoxy and carbon fibre epoxy composite patchis presented respectively in Figures28 and Figure29. The effect of the patch presence on fatigue life of the repaired structure and compared to unrepaired plate is presented in Figure30.From this, we can note that the fatigue life is highly improved by the presence of the composite patch. Indeed, the number of cycles to failure is increased up to 3 times with the presence of the composite patch.The crack delay for carbon/epoxy is less compared to that of plate with boron/epoxy patch. The number of cycles to crack length 45 mm prior to patch edge with carbon patch is about 78000 cycles, and this number is about 96000 cycles with boron patch; there is, therefore, a 23 % decrease in fatigue life with the use of carbon patch compare to the use of boron patch.We can thus conclude that the repair material selection plays a significant role in the crack growth retardation due to high-stress concentration absorption around the crack tip.

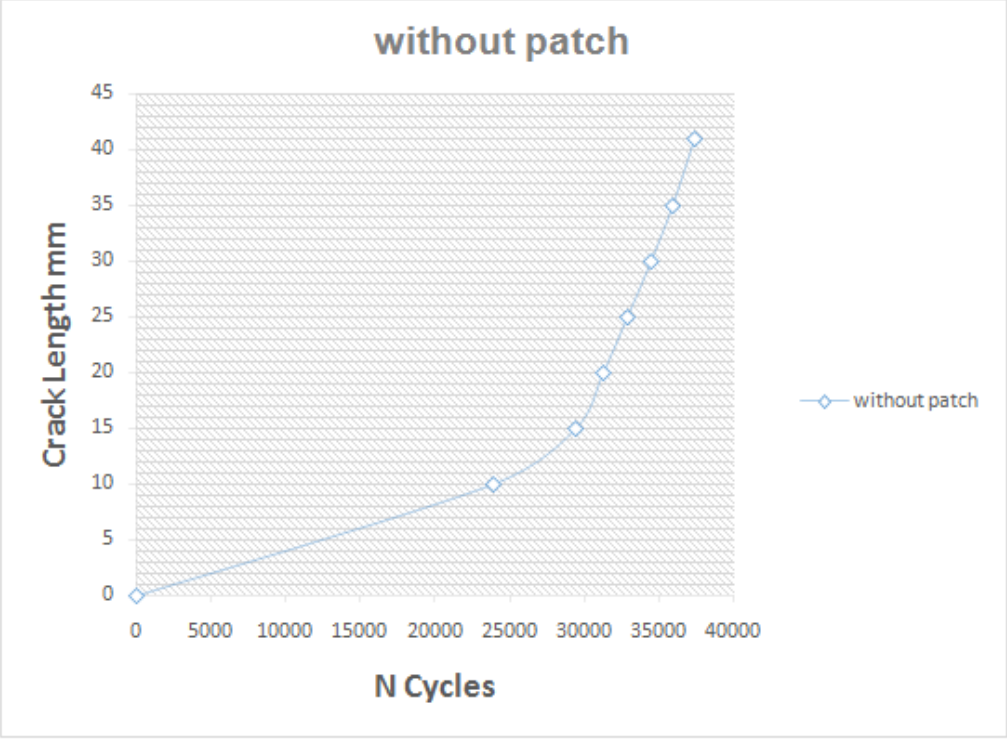


Figure 4.27: Fatigue life of notched/ unrepaired plate .

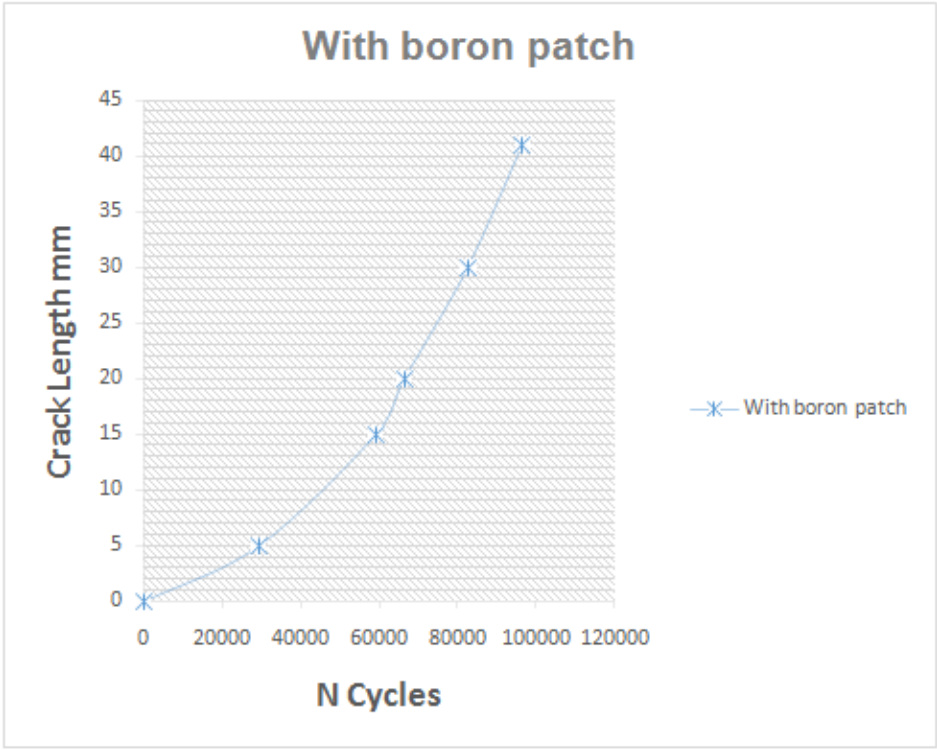


Figure 4.28: Fatigue life of notched/ repaired plate with boron fibre epoxy composite patch.

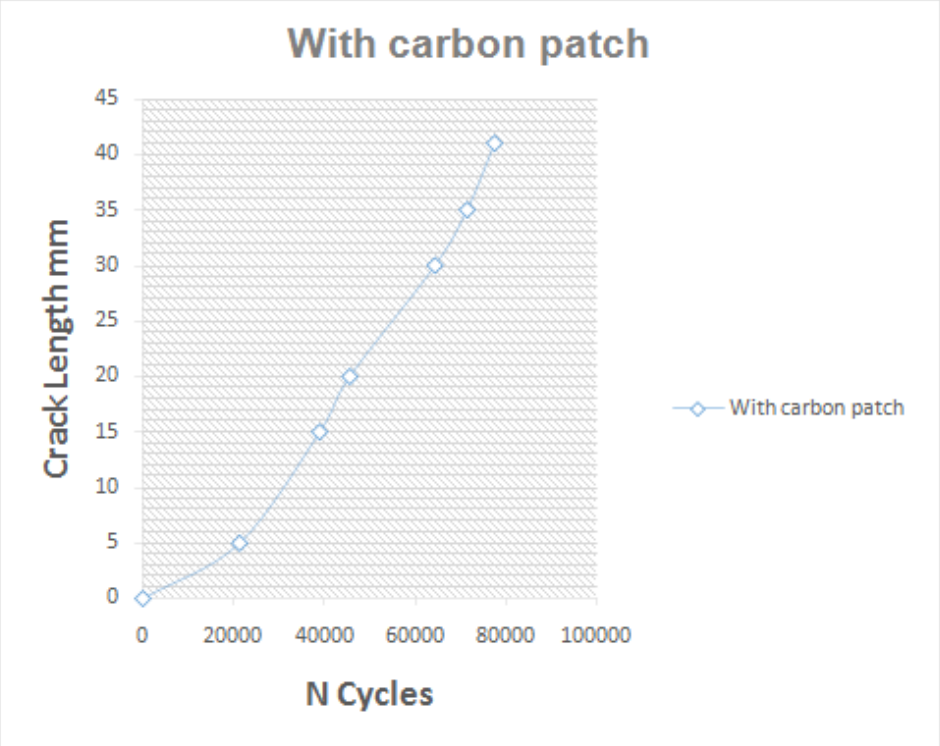


Figure 4.29: Fatigue life of notched/ repaired plate with carbon fibre epoxy composite patch.

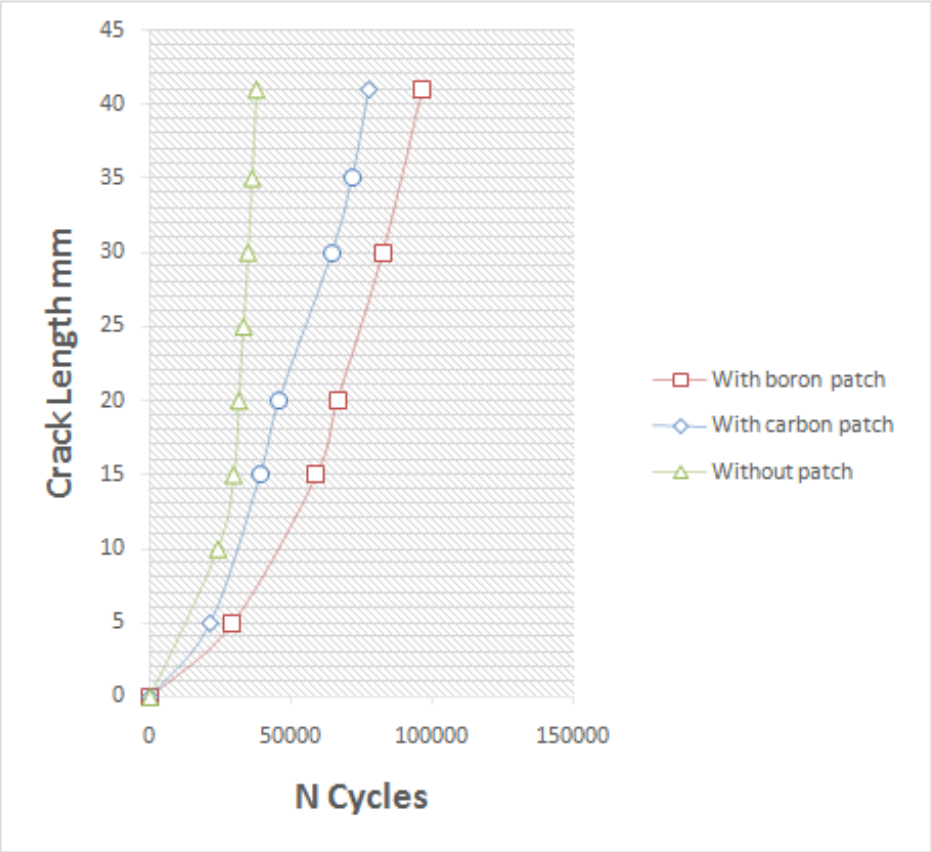


Figure 4.30: Fatigue life of notched/ repaired plate with boron fibre epoxy and carbon fibre epoxy composite patch.

Fatigue loads are the major contributors to the service induced damages in the aerospace structures. Therefore, bonded patch repair reduces stresses near the crack by bridging between the cracked plate and composite patch, and therefore retards or completely stops the crack growth. Figure 31 shows the stress intensity factor in plate without a patch was higher than a plate with a patch. This behaviour attributed to there being stress transfer from the cracked aluminium plate to the composite patch throughout the adhesive layer. A decrease in the number of cycles with crack length growth in the patched case compare to unpatched samples is associated with the repair having a direct influence on the absorption of the stress fields at the crack tip.

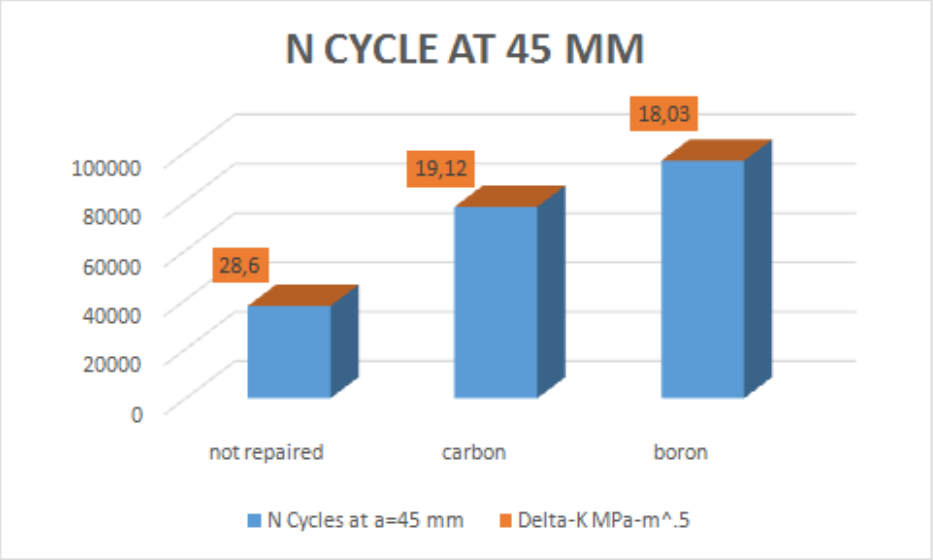


Figure 4.31: Comparison of the repair material on Fatigue life of notched/ repaired plate.

4.2.3.2 Fatigue crack growth behaviour (FCG)

Table 5 presents the variation of the FCG rate as a function of the SIF for the unpatched plate and patched plates with both repair materials. The value of da/dN is affected by the crack length growth. They are slightly lower in small cracks growth in the unpatched plate. In addition, we can remark that after the patch presence, the values of da/dN are lower than that of an unpatched one.

a mm	$\triangle K(MPa - m^{\cdot}5)$	$da/dN(mm/cyc)$
Boron Patch		
5	5,42	0,00030
10	8,049	0,00048
15	10,031	0,00051
20	11,28	0,00058
30	14,47	0,00068
35	16,11	0,00072
Carbon Patch		
5	6,39	0,00024
10	8,14	0,00051
15	10,37	0,00052
20	11,33	0,00046
30	14,27	0,00127484
35	16,78	0,00081
Unpatched plate		
5	15,80	0,00054
10	18,85	0,00070
15	22,44	0,0017
20	25,35	0,0042
30	31,86	0,0052
35	35,45	0,0083

Table 4.5: The FCG rate & $\triangle K$ variation of unpatched and patched plate.

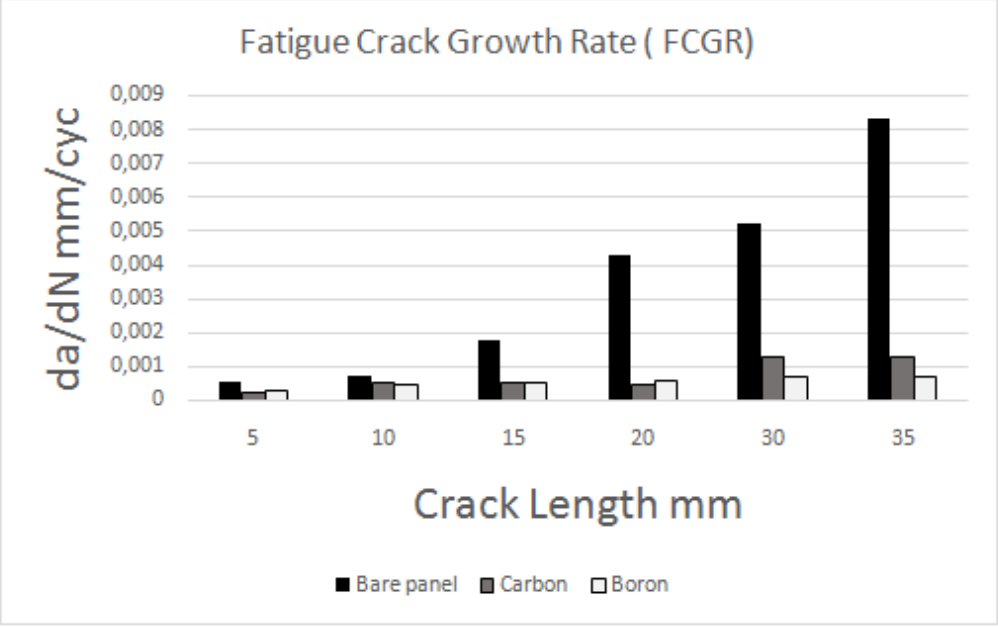


Figure 4.32: Comparison of the crack propagation rate between unpatched and patched plates.

Figure 32 shows a comparison of the crack propagation rate for a repaired sample with carbon/epoxy and boron/epoxy composite patch and unrepaired plates. The crack growth rate in unpatched plate is higher compare to the patch ones. The crack growth rate increases as the crack length increases due to higher stress concentration at the crack tip. The crack growth rate in the patched plate with boron fibre/epoxy composite patch shows a lower crack growth rate than the one repaired with the carbon fibre/epoxy. The crack growth rate using the boron fibre/epoxy composite patch showed stable crack growth rate as the crack length increases compare to the use of the carbon patch, especially in bigger crack length. This behaviour on the effect of various crack length repaired with composite patches is investigated in the following section. The retardation effect of the crack growth rate is due to the presence of the patch on the AA7075-T6 plate. This can be seen by the decrease in crack growth rate between unpatched and patched plates. The interval variation of the two curves is very different: for the unrepaired plates, $\triangle K$ varies between 16 and 35MPavm while, for the repaired plates, $\triangle K$ varies between 5 and 16MPavm. This is due to the fact that the stress fields in the crack tip for the repaired plate is minimized by the presence of the patch which delays its spread. As a result, the stress intensity factor is less significant in the repaired plates and its value can be down to 3 times than that of the unrepaired plates.

4.2.3.3 Tension test after the fatigue testing

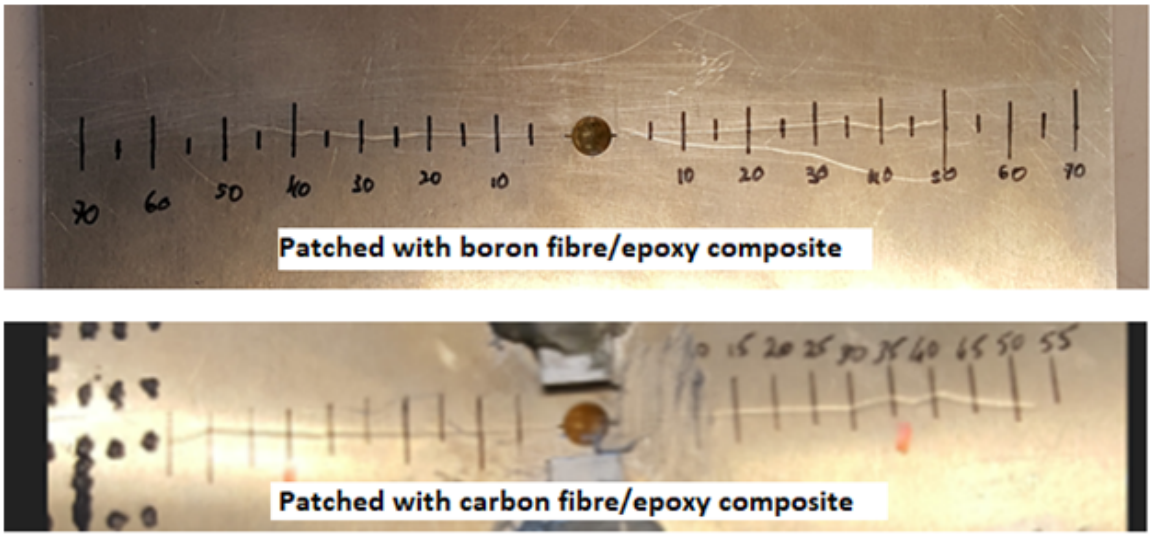


Figure 4.33: Crack propagation path behaviour of patched plates.

The fatigue crack propagation rate described by the evolution of da/dN(mm/cycle) as a function of the amplitude of the stress intensity factor $\triangle K$ is affected by the crack length. Crack propagation path in both patched samples with carbon/epoxy and boron/epoxy patches showed similar propagation path behaviour that is presented in Figure 33.

During the fatigue test of the patched plates, the crack growth exceeds the patch edge. The crack continues growing towards the end of the aluminium plate. While kept under

continuous cycling loading after the crack passed the edge of the plate; adebondingline between the patch and plate observed without a sign of detachment on the patches from the plate after long time cycling as seen in Figure 34. On completion of these fatigue tests, 2 specimens namely unpatched, patched with boron composite and patched with carbon composite were loaded to failure in tension in order to determine the strength of the damaged repair. For comparison, it was done at the same crack length growth in order to analyse the effect of the strength of the assembly on the repair efficiency and the results of these tests are presented in Figure 35.

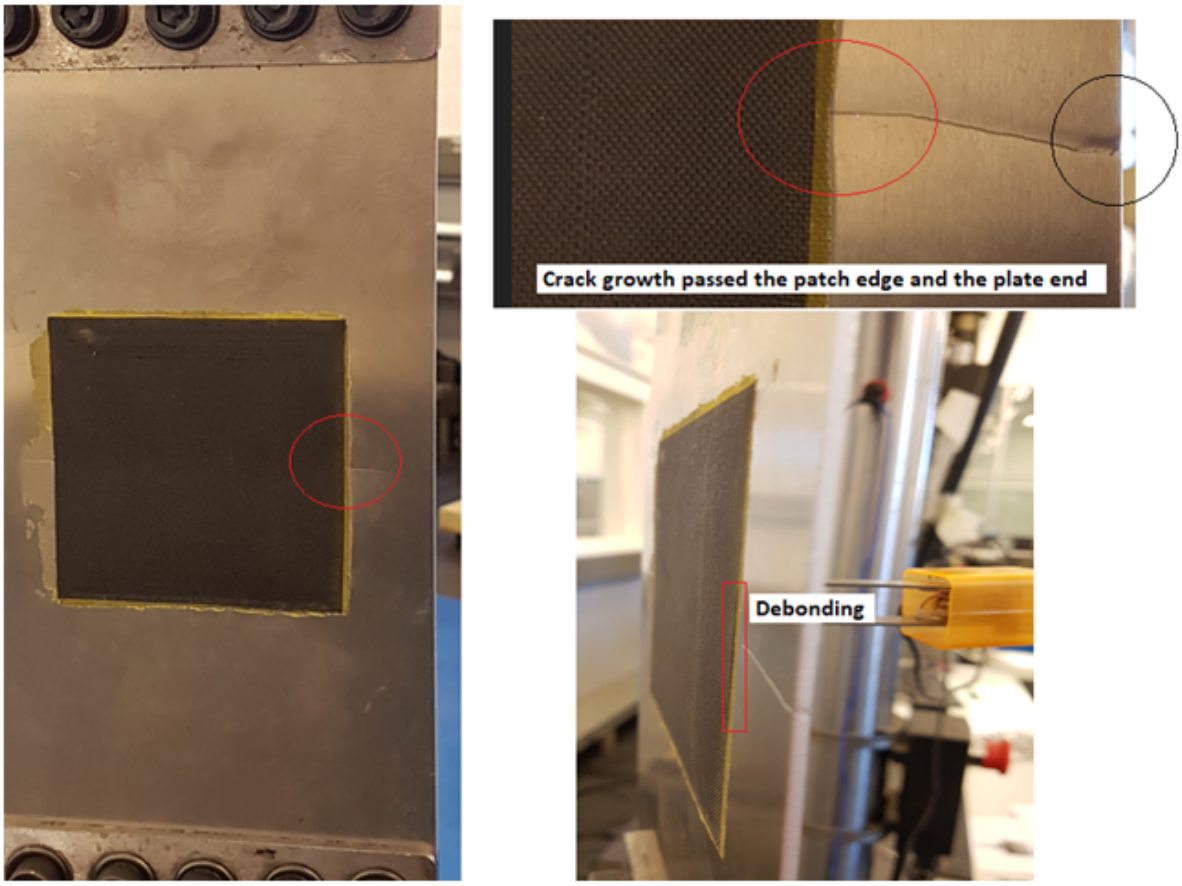


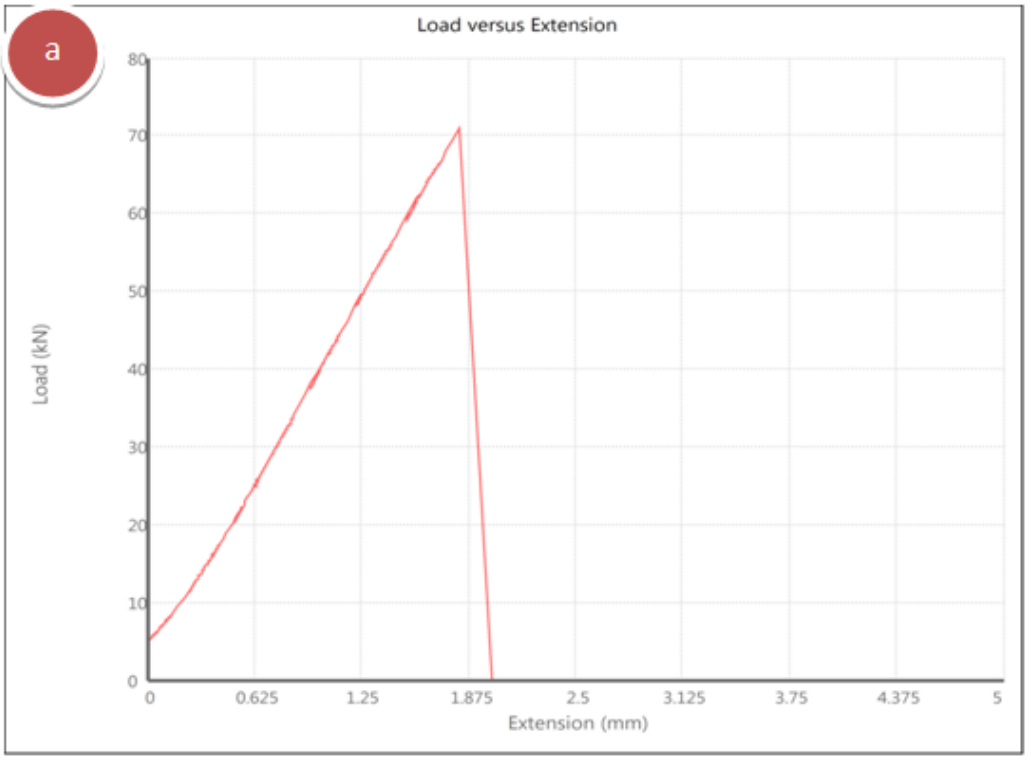
Figure 4.34: Effect of patch strength after plate failure.

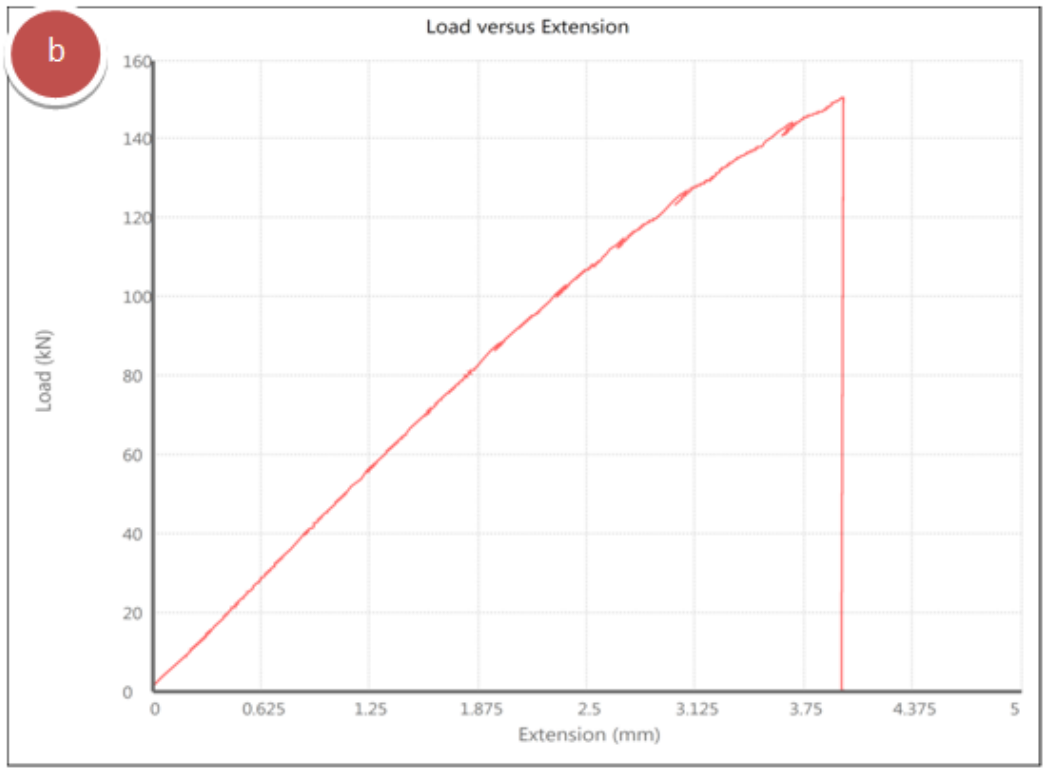
Tensile results in Figure 35 showed the plate specimens with boron/epoxy and carbon/epoxy composite patch gave higher extension compared to the unpatched as represented in Table 6. The patched sample with carbon/epoxy gave less fracture energy rate in comparison to the boron/epoxy patch. The rigidity of a plate in the presence of a crack from a notch is weakened. Under repeated strain, the crack propagates rapidly towards the free edge of the plate.Repair by patch composite increases its strength of the damaged structure and retards crack propagation, therefore ensuring a long life of the structure. From Figure 35, we can note that the repaired specimens exceeded the fracture strength of the unrepaired specimen.This explains that the strength of the assembly is highly increased with the composite patch, which explains the previous results in the improvement of the fatigue life with the bonded composite repair. The load transfer between the plate and the composite allows higher displacements in the aluminium plate

after repair. The lower fracture strength of patched specimen with the carbon compared to the boron/epoxy patched specimens was expected due to the lower stiffness of the carbon fibres. Despite this, the fracture strength of these fatigued specimens still exceeded the strength of unpatched specimens.

Display Name	Value			Unit
Test Run Name	Unpatched	Patched with Carbon	Patched with Boron	
Stress	546,015	494,865	341,005	MPa
Area (square)	130,000	304,000	487,920	mm^2
Test Rate	2,000	2,000	2,000	mm/min
Peak Load	70,982	150,439	166,383	kN

Table 4.6: Tensile test data summary of unpatched and patched plate after the fatigue testing.





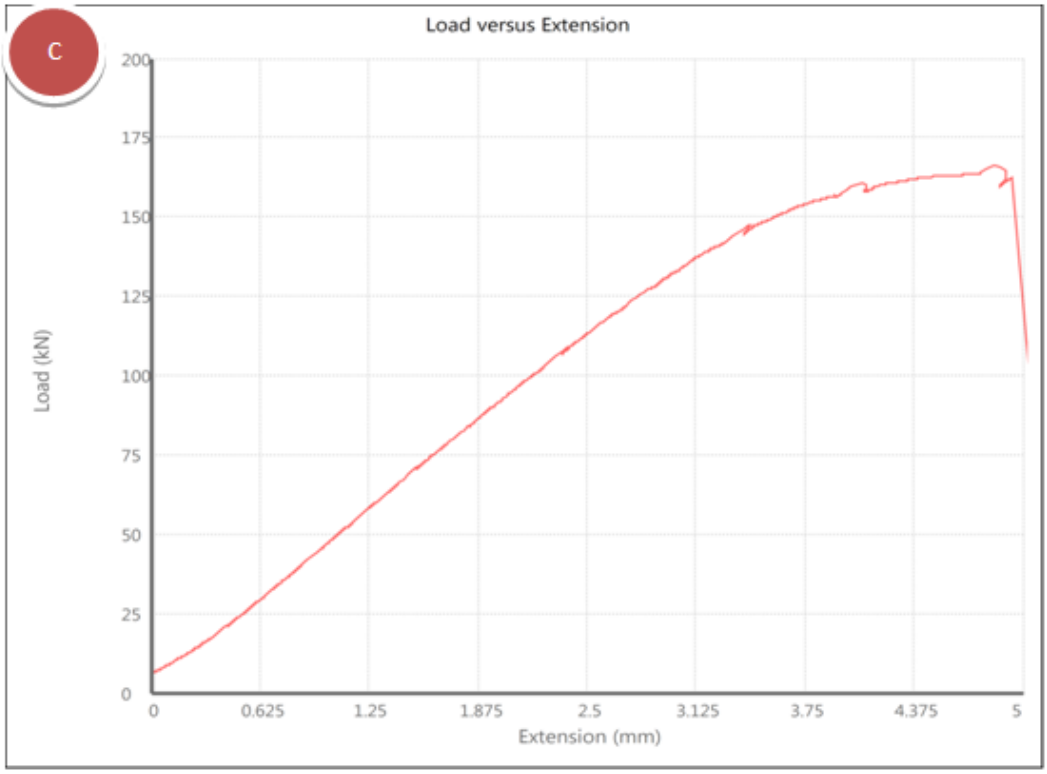


Figure 4.35: Tensile test of patched samples after the fatigue testing (a)Unpatched sample, (b) Sample patched with carbon fibre/epoxy composite and (c) Sample patched with boron fibre/epoxy composite.

The comparison between the boron/epoxy and carbon/Epoxy for repairing aircraft structures showed an increase in the number of cycles as shown in Figure 30. This was explained due to the reduction of the stress intensity factor at the crack tip where it is more significant for the boron/epoxy patch. Also, the strength was highly improved by the presence of the latter. This showed significant role of repair material selection on repair efficiency. Carbon fibres are more available in a wide range of properties forms. Thus, carbon fibres are high strength, yet not as strong as boron fibres.Specimens patched with boron patch failed in the fractured crack region across the width area with horizontal delamination between the fibres, while specimens patched with carbon composite failed by debonding/ delamination and tearing of the composite patch. This can be seen in the fracture status of boron fibres and carbon fibres in Figure36 with tearing of fibres in the case of carbon epoxy patch compare to delamination of the boron/epoxy.

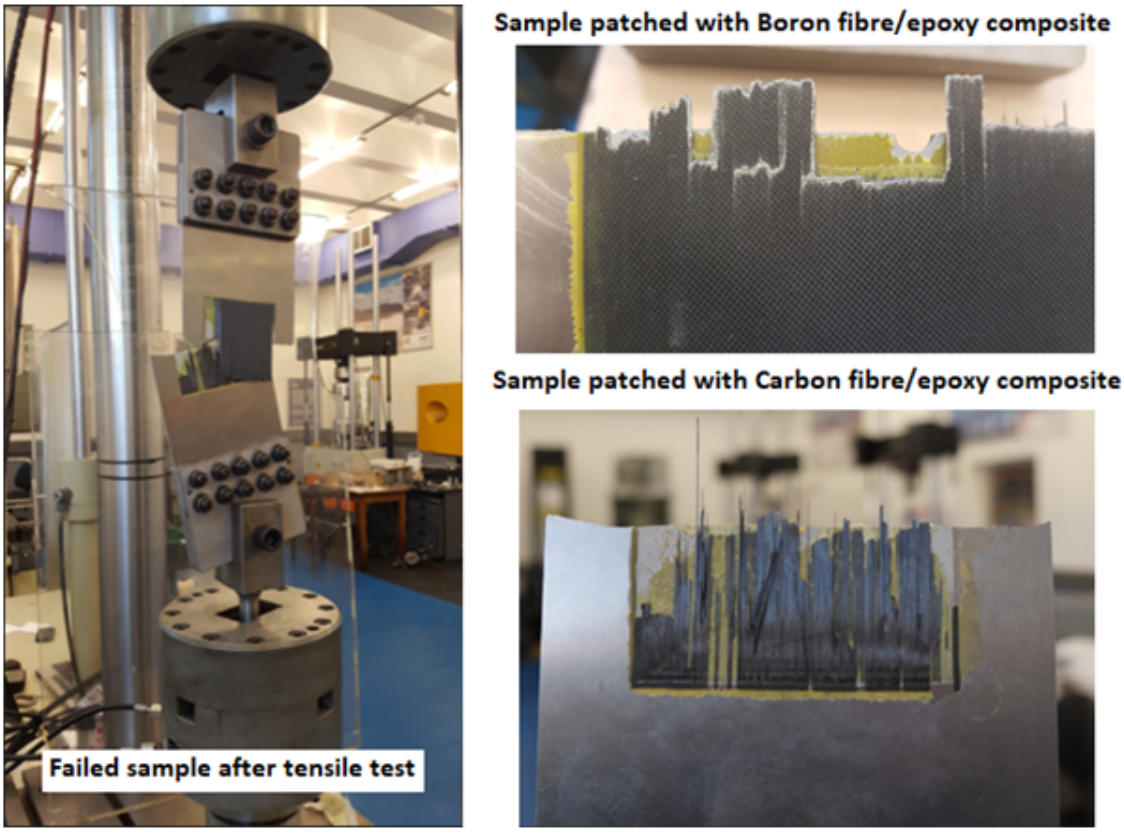


Figure 4.36: Fractured samples after tension test.

The following section deals with the study of crack behaviour through the study of fatigue life of AA7075-T6 aircraft plate cracked with various crack length and repaired with a boron/epoxy composite patch.

4.2.4 Fatigue life of AA7075-T6 aircraft plate pre-cracked with various crack length repaired with a boron/epoxy composite patch (B)

Recent prognostic programs to evaluate the extended fatigue life of ageing aircraft have revived research on the formation, detection and prediction of fatigue damage in high-strength aluminium alloys. The prediction of potential crack/damage size and crack growth rates, as well as the fatigue life, is critical for safety evaluation and strategic decision-making. In a critical component or structure of an aircraft, a millimetre crack could cause catastrophic fatigue failure under a severe overload in service. Knowledge of fatigue crack formation, early crack growth and damage size as a function of service conditions is very important for selecting appropriate non-destructive testing procedures and for designing a specific repair to arrest the crack. The presence of cracks can significantly reduce the strength of structures. Even small flaws can by time develop and become a real threat to a structure. Under cyclic loading, a small crack of an initially negligible size can grow until it reaches the critical size for brittle fracture. This section deals with the effect of different crack sizes length repaired with composite patches. The fatigue tests were done to pre-crack the plates with various crack length. Then the plates were repaired with boron fibre/epoxy composite patch. Fatigue crack growth analysis was carried out on small crack sizes and larger crack sizes. In the case of bigger crack size is to evaluate the influence of larger damages in critical emergency condition where repair need to be done temporarily. The Pre-cracking samples in various crack length was done using cycling loading.

4.2.4.1 Effect of the patch presence on FCG & SIF

In Figure 37, AA7075-T6 aluminium plate with and without patch represents fatigue life cycles using boron fibre/ epoxy composite patch as a material choice of repair. When comparing the repaired structures to unrepaired, the patched structure had a fatigue life over 3 times longer. This increase in the fatigue life performance from unrepaired to repaired is due to the boron/epoxy patch bridging the crack as discussed previously. The increased performance is again due to increased stiffness in the vicinity of the crack, but is mostly accounted for by decreased stress intensity at the crack tip due to the bonded repair. The reduction of $\triangle K$ from unrepaired to the repair can be up to 55%. The Reduction in Stress Intensity Factor with repair from unpatched to patched plate is shown in Figure 38. This shows the beneficial effect of the patch. One can thus confirm that the use of a composite patch offers an enormous advantage for repaired cracks to increase the service life as shown in Figure 37. The stress intensity factor was determined experimentally during the fatigue test from the data by using equation (4.1) as per ASTM E647 [44]:

$$\triangle k = \frac{\triangle P}{B}\sqrt{\frac{\pi\alpha}{2W}\sec\frac{\pi\alpha}{2}}, \tag{4.1}$$

where P is the load in MPa, B and W are the thickness and width of the sample in mm, respectively, a is the crack length; and $\alpha = 2a/W$.

The fatigue crack growth rate curves ($da/dN = f(\triangle K)$ where $\triangle K = K_{max} - K_{min}$ of both unpatched and patched plates were calculated experimentally and are presented

in Figure 39. It is important to note that the plot of log da/dN variation as a function of the log SIF for repaired specimen versus unrepaired specimens is not convenient to plot together because of the big difference of the SIF values. Also, it does not allow proper fatigue crack growth comparison. However, from Figure 39 the fatigue crack growth rate (FCGR) of AA7075-T6 sample with and without patch is summarized in Figure 40 to illustrated the level of change in the FCGR between patched and unpatched plate.

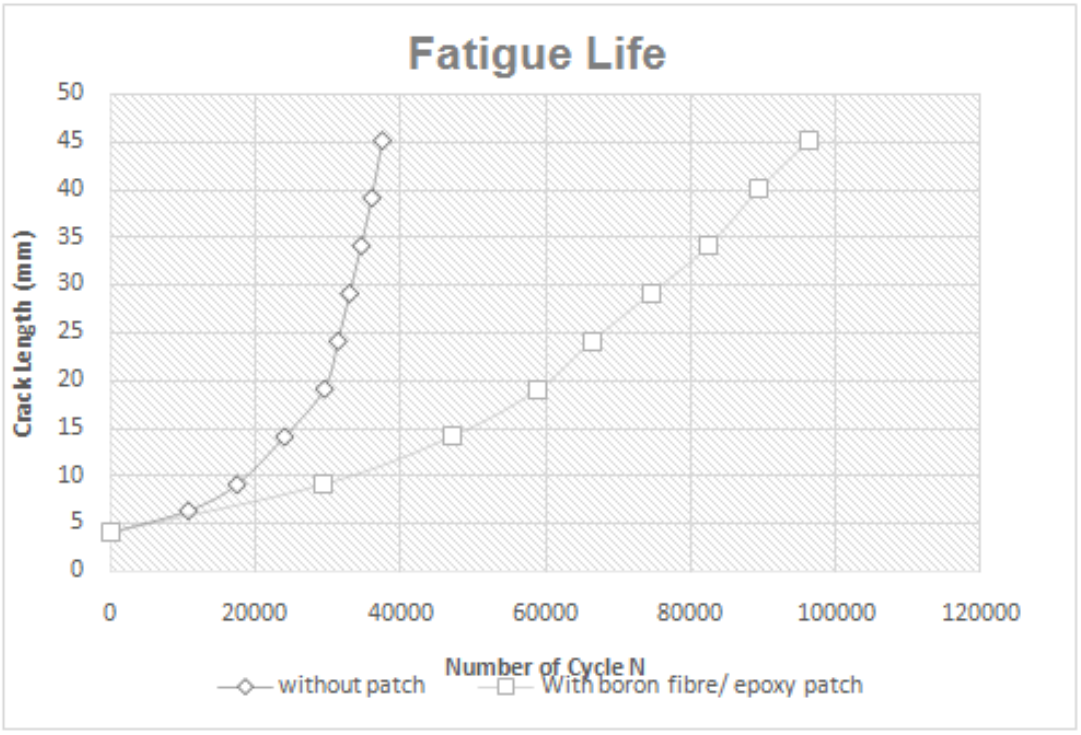


Figure 4.37: Effect of the patch presence on Fatigue Life.

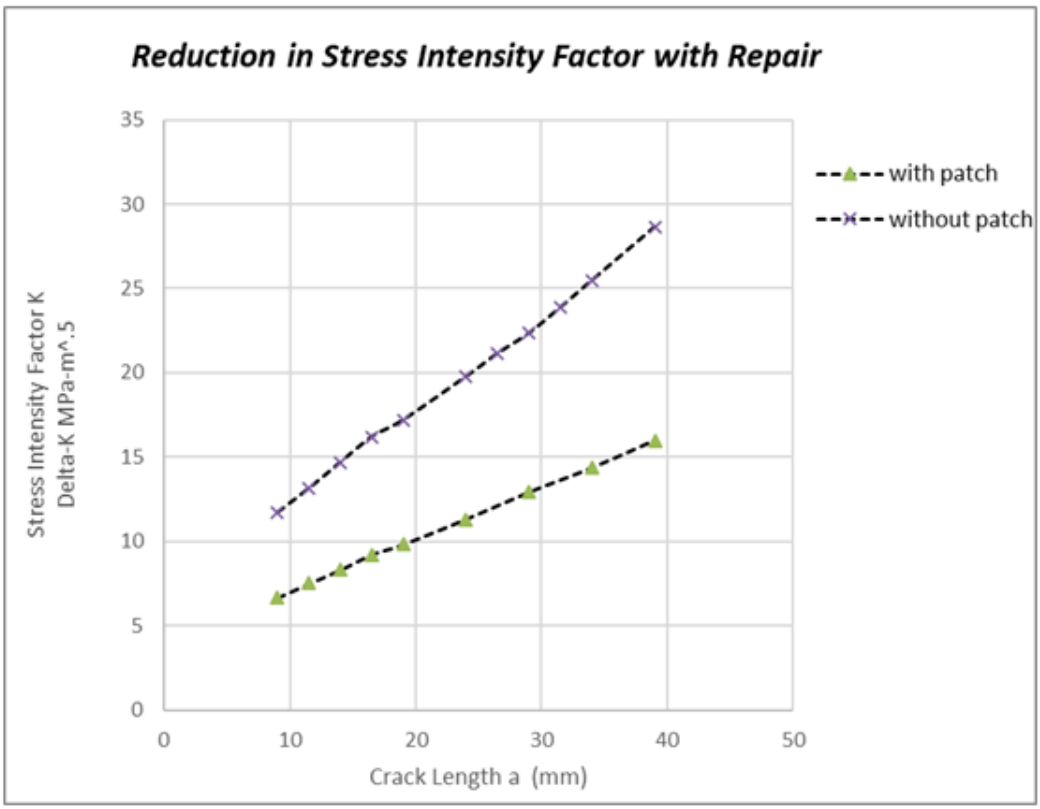


Figure 4.38: Reduction in Stress Intensity Factor with Repair.

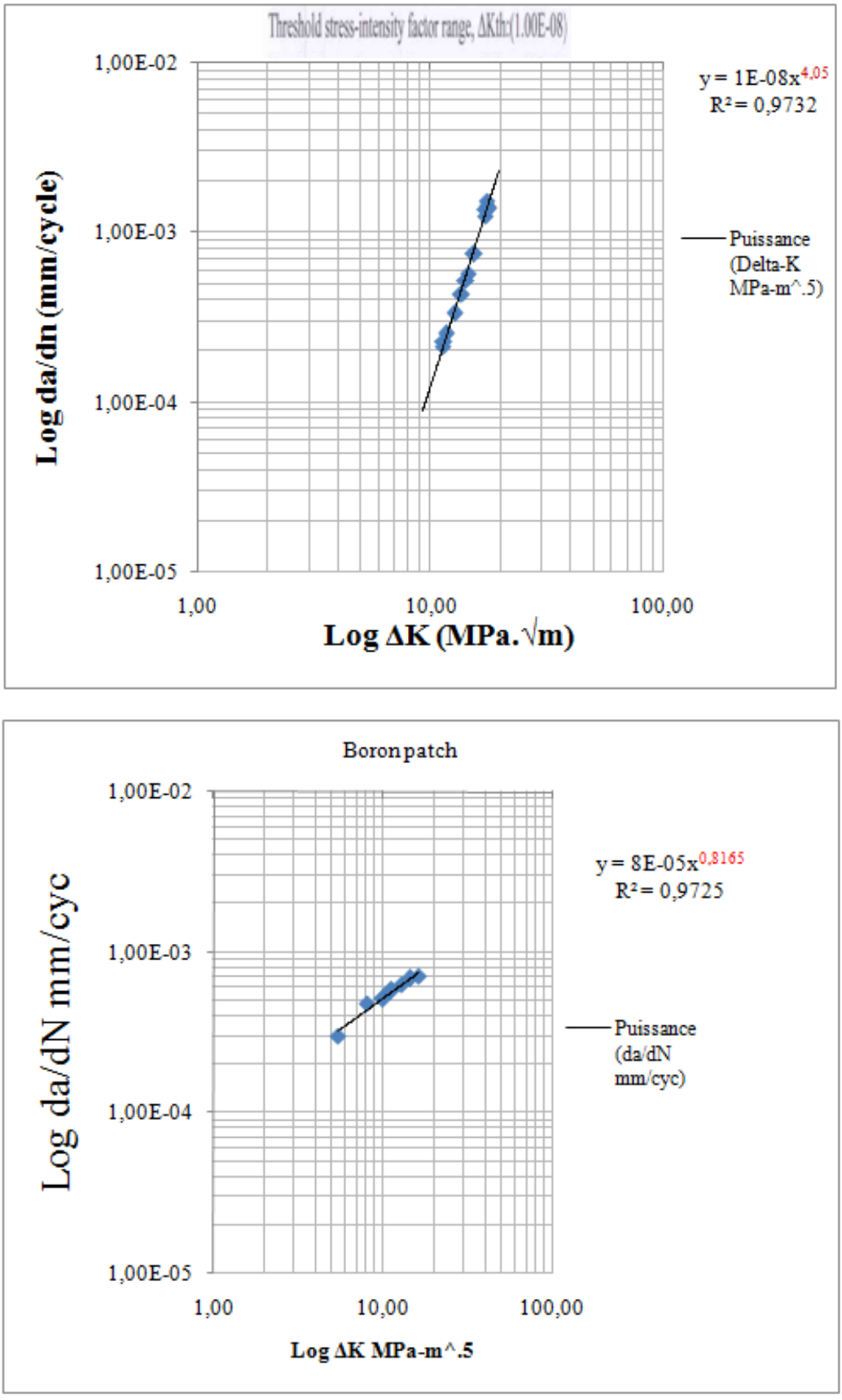


Figure 4.39: Fatigue crack growth FCG of AA7075-T6sample with and without patch.

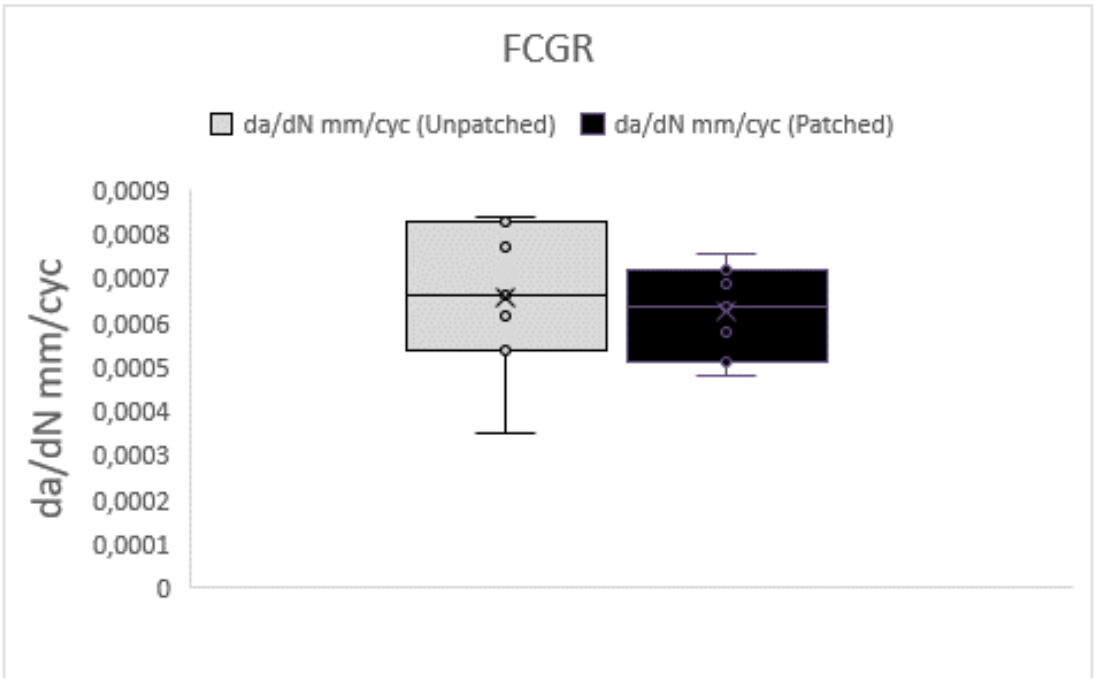


Figure 4.40: Fatigue crack growth FCG of AA7075-T6 sample with and without patch.

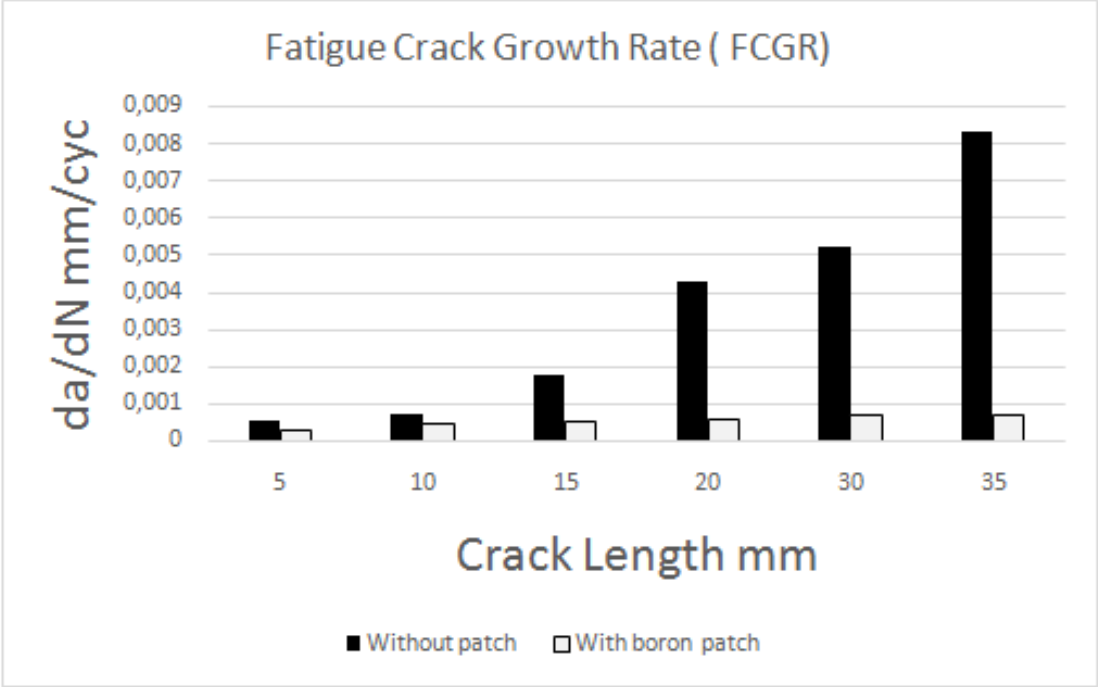


Figure 4.41: Variation of the crack growth rate of unpatched and patched plates.

Figure 41 presents the variation of the crack growth rate (da/dN mm/cyc) as a function of the crack length for patched and unpatched plate. From this Figure, it can be noted that the patch presence highly decrease the crack growth rate. Figure 40 shows the fatigue

crack growth (FCG) of AA7075-T6 sample with and without patch. An average of 85 % decrease in the crack growth rate from unpatched plate to the patched one is found. This means that, for the patched plate, the stress concentration at the vicinity of the crack tip is highly reduced by the presence of the patch.This shows that the larger the stress intensity factor, the higher the crack growth rate. Therefore reducing the stress intensity factor is the key to accomplishing a successful aircraft repair. Minimizing stress intensity factor and slowing down or stopping its growth is the aim of all crack repair techniques.It can be seen that results in the patched plates give lower values of the crack growth rate compared to unpatched plate, this is due to the high strength of the boron/epoxy compared to the aluminium plate alone. This confirms the advantage of the use of composite patches and the positive effect on the crack growth behaviour.

4.2.4.2 Effect of the boron/epoxy patch repair on the fatigue life with various crack length

Different crack lengths were pre-cracked and repaired with boron/epoxy to evaluate the effect of the crack length size on the repair efficiency and fatigue life cycle. The different crack lengths that were pre-cracked and then repaired are illustrated in Figure 42.

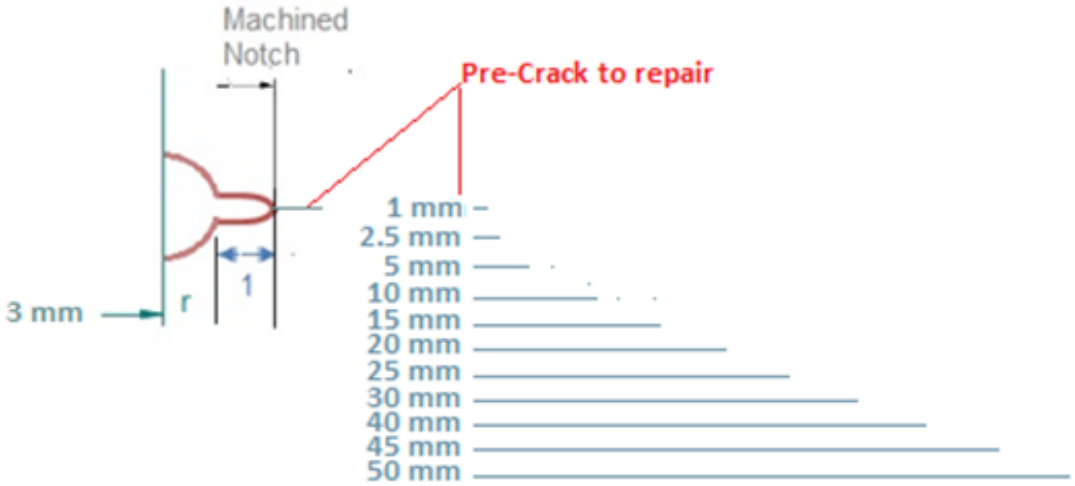


Figure 4.42: Different crack length pre-cracked to be repaired.

Figure 43 shows the crack growth passed the patch and the plate. It has been observed during the test that the crack grows until the aluminium plate rupture into two halves. However, the patch kept holding the plate together; in other words, the crack had propagated across the entire width of the specimen. Therefore, from a reference point of view and comparisons in the following data, all fatigue tests in the repair case have stopped at the crack length of 70 mm just prior to reaching the plate edge (20mm passed the patch edge). Figure 44 represents an example of the fatigue life results of pre-cracked plate of a= 1 mm crack length after the notch repaired with patch composites.

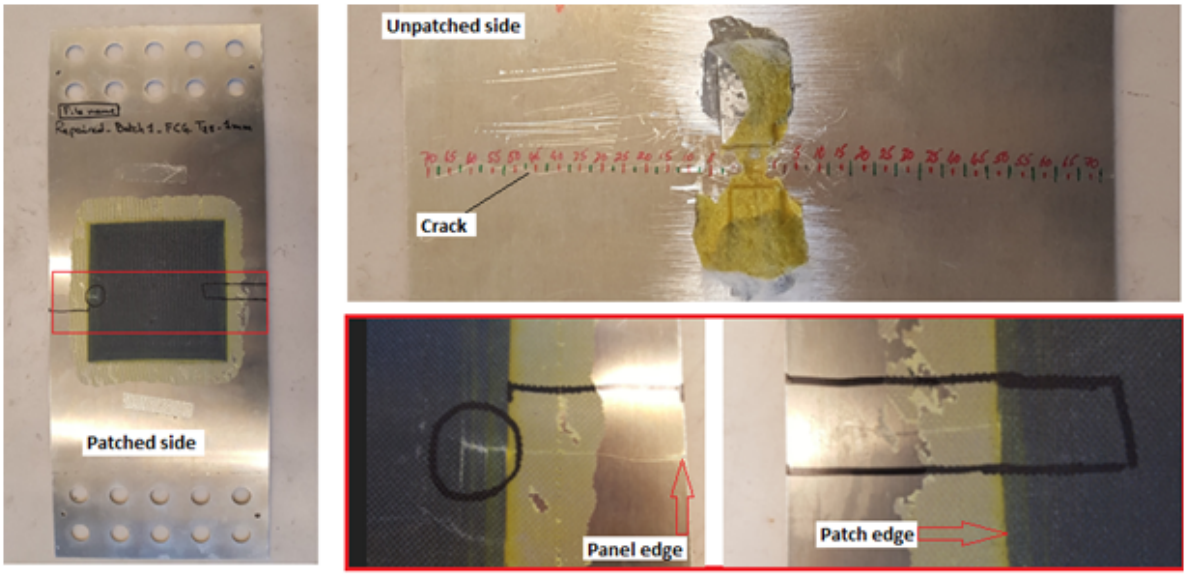


Figure 4.43: Crack growth behaviour in the repaired plate.

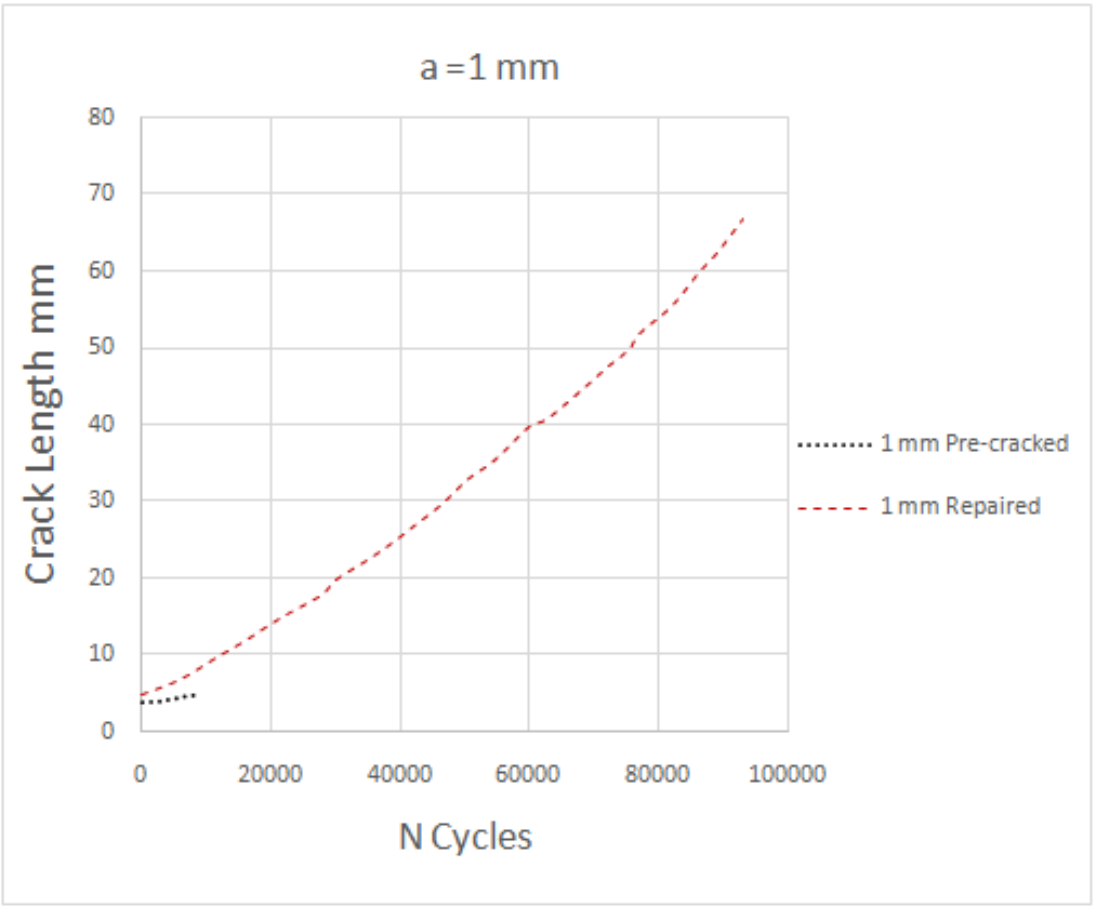
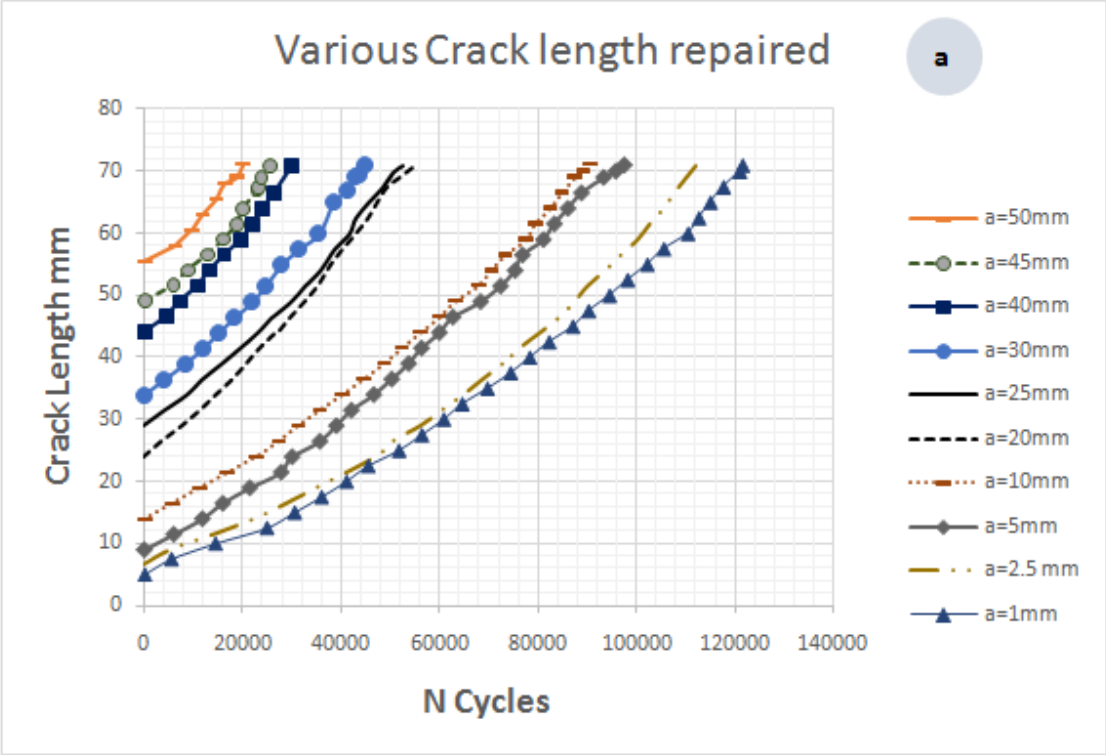


Figure 4.44: Fatigue life of 1 mm pre-cracked plate repaired with composite patch.

Figure 45 (a) represents the fatigue life of different crack length repaired where (b) shows the same graphs however, it was shifted to zero as the origin for comparison purposes. The small crack length up to 30 mm showed repair efficiency between 2 and up to 3 times from unrepaired to repaired structure with a repair percentage increase between 18 to 92% in a=[1-25] mm. It was noted that from 30 mm crack length the repair was notsuccessful.As shown in Figure 46,the small crack sizes give lower values of the FCGR compared to the larger crack sizes, however, both are higher compare to the repaired ones. Those results are shown in correlation with Figure 47 where the number cycles decreases as the crack length size increases. The fatigue crack propagation takes place according to two different regimes:

1. Below a certain critical size (depending on the structure and its loading), the crack propagates "slowly", in a progressive and stable manner, during the loading cycles.

2. Once the critical size is reached, the crack propagates suddenly in an unstable manner, which generally leads to rupture. By adding the conclusion mentioned in the previous paragraph, it is useless to repair cracks of size more than 30 mm. The repair by the patch in larger crack length reduces its efficiency.



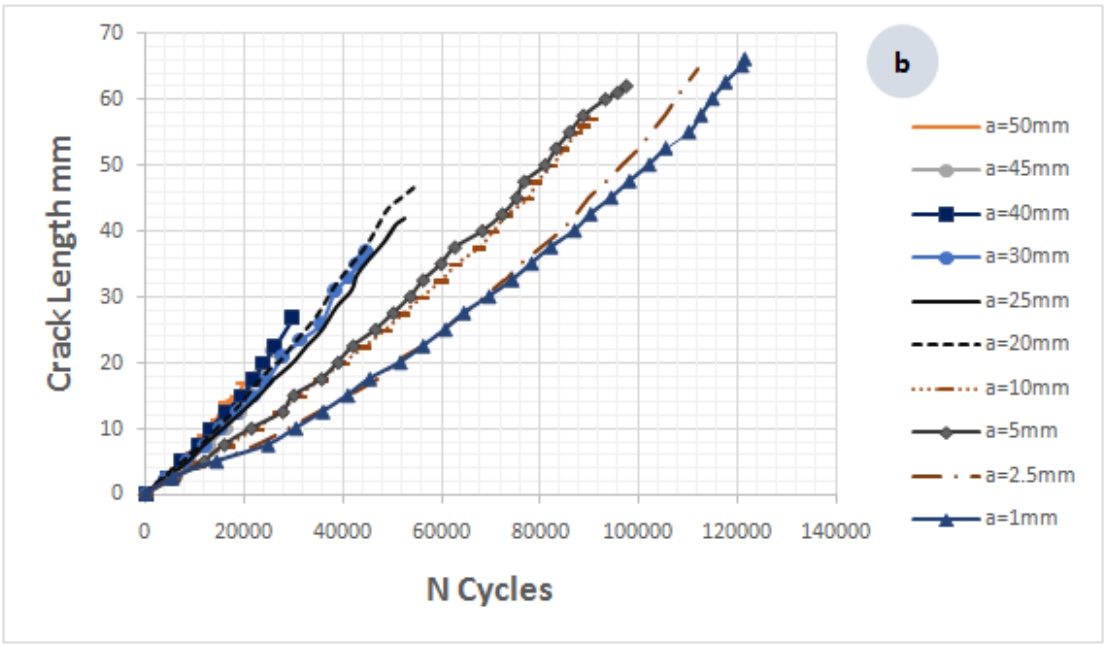


Figure 4.45: Fatigue life of AA7075-T6 repaired with composite patch of different pre-crack length size(a) Crack size length repaired and (b) Crack length size shifted to zero as the origin for comparison.

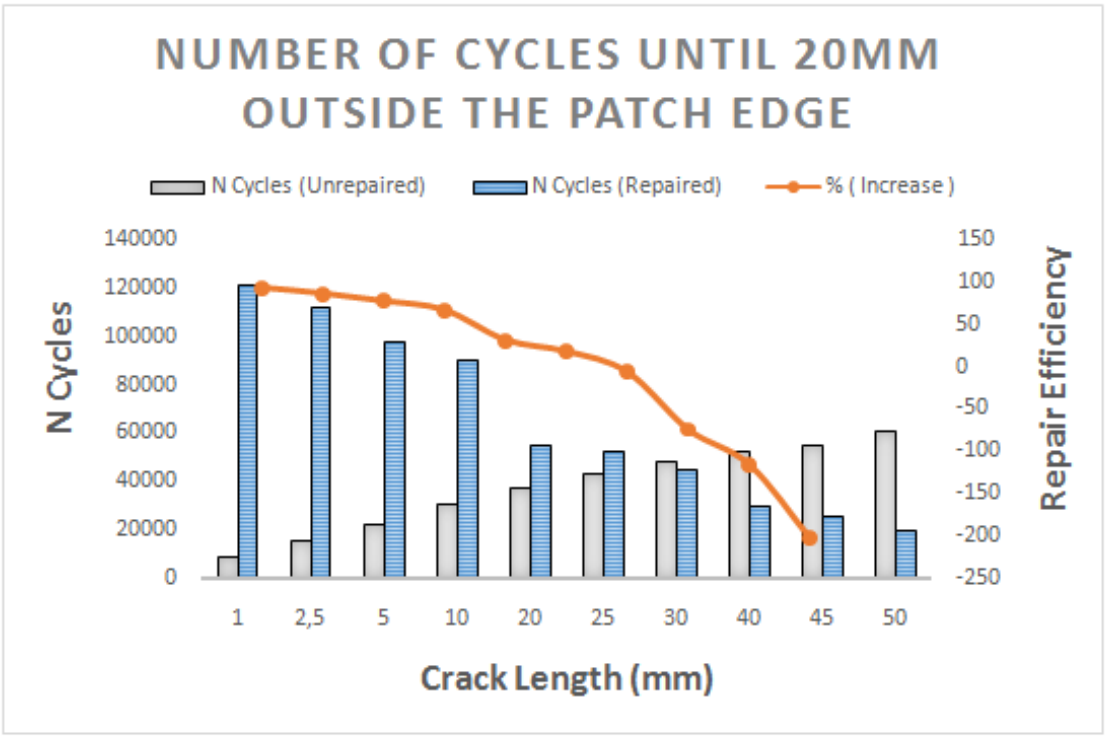


Figure 4.46: Fatigue life cycles variations of different crack length damage repaired with patch composite.

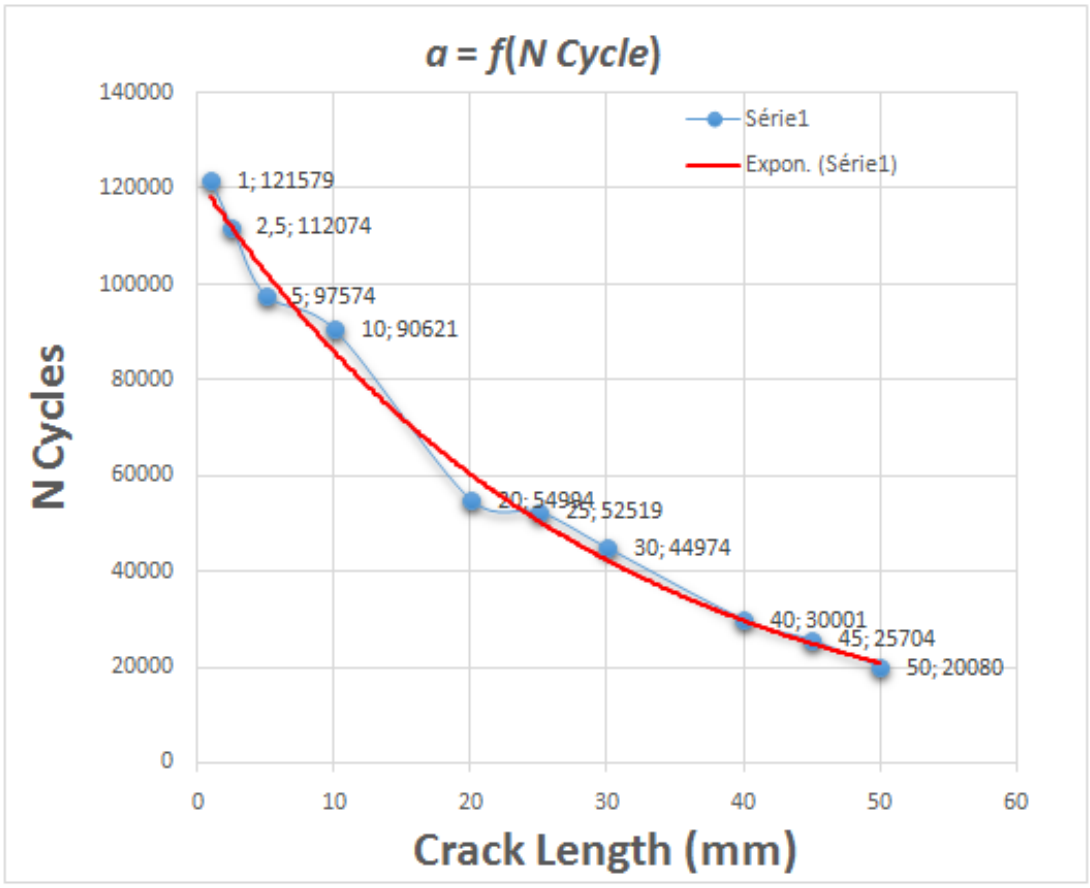


Figure 4.47: Fatigue life cycle behaviour of small and large cracks repaired with patch composite.

4.2.5 Effect of curing cycle on fatigue life of cracked AA7075-T6 aircraft plate reinforced with a boron/epoxy composite patch (C)

Summary

4.2.5.1 Introduction

Bond integrity plays a big role in the repaired structure fatigue life. The bond line is the most important part of the patch repair structure since it is the key to transferring stress from the cracked plate to the patch. Significant efforts have been devoted to the repair or reinforcement repairs by CPBR for various applications of the damaged or weakened part of a structure to restore the structural effectiveness and thereby ensure the safety of the structures. Analysis of metallic aircraft structures development and qualification indicated secondary bending structures curvature of single side in the repair area. This secondary bending structures curvatureinfluenced the mechanical behaviour and further detailed investigations were recommended [34]. The CPBR method includes the following steps: surface preparation before bonding, the application of Composite Patch, a cure cycle of the composite and the adhesive film, and the final cooling [35].Adhesive joints experience a decrease in volume as they cure, causing either an accumulation of residual stresses inside the adhesive joint or a distortion of the bonding substrates [36]. Residual stresses in composites arise during the curing stage, due to resin shrinkage, the anisotropy in thermal expansion coefficients and the tool-part interaction [37, 38]. Process induced residual stresses commonly occur in composite structures composed of dissimilar materials. In addition, the initial stress state has to be known accurately in order to be taken into account when considering the in-service performance of the parts. Experimental studies have demonstrated the influence of residual stresses on mechanical properties of laminates. The different stress states have been obtained either by varying the cure cycle, or by changing the dwell temperature and time, the cooling rate, the cooling pressure, or by applying different post-cure regimes [39]. Past research has been focusing on residual stresses induced during the manufacturing of composites material. However, there are limited investigations available on the influence of different curing parameters of structural epoxies on fatigue life of repaired metallic plates; with bi-material structures composed of a composites and aluminium, strength is a matter of concern. The purpose of this research is to define the influence of the curing time and temperature on the fatigue life of bond repaired structures of 7075-T6 with boron fibre epoxy composite patches. Including the study of fatigue life to address those various cycle conditions on the formation of distortion levels and how that is influenced by cure cycles where the time and temperature cycle combination in various conditions are discussed.

The tests were performed with four types of cure conditions as describe din the methodology section 4.4 in Chapter 3.The first (A) was subjected to a cure cycle as per the data plate - heating up in 30 minutes and held at $120^{\circ}C$ for 60 min [40]. The second (B) was subjected to a combination of low temperature of 80 $^{\circ}C$ over a long cycle time of 480 min and post cured at $120^{\circ}C$ for 60 min as the recommended cycle as per the data plate to ensure adhesion at elevated temperatures. Other researchers [41] favoured a lower curing temperature curing because of the differences in coefficients of thermal expansion between the parent and the bond material. However, a reduction in the cure temperature may

affect important properties of the bonded repair integrity. The third (C) was subjected to a lower temperature of 80 $^{\circ}C$, held for a longer time of 480 min. For processes conducted at elevated temperatures, the cure cycle temperature design can also influence residual stress build-up and shape distortions. Implementation of long cure cycles at elevated temperatures are known to minimize residual stress levels during moulding [4,43]. This is due to the viscoelastic nature of the rmosets at elevated temperatures (especially in the rubbery state) where large stress relaxation can occur [44]. Therefore, the fourth (D) was subjected to a prolonged cure cycle, heating to 120$^{\circ}C$ for 105 minutes, to evaluate the prolonged curing time. Three plates were made for each cure condition to ensure repeatability and traceability.

The full plate with patch was heated in a convection furnace. The ramp rate to all the curing temperatures was 2 $^{\circ}C/min$. Plates were taken out of the furnace and left to cool at ambient conditions after curing. To ensure repeatability, the curing process was controlled by a computer while the temperature of the workpiece (composite plate) was continuously monitored by a thermocouple fixed to the aluminium base plate. The four curing conditions that were used are summarised as follows:

- **(A)** 120 $^{\circ}C$ for 60 minutes (as per manufacturer specification).
- **(B)** 80 $^{\circ}C$ for 480 minutes with post-curing ramped to reach 120 $^{\circ}C$ in 60 minutes.
- **(C)** 80 $^{\circ}C$ for 480 minutes.
- **(D)** 120 $^{\circ}C$ for 105 minutes (60 + 45).

Initial curvatures (due to residual thermal stresses from bonding repairs), fatigue crack growth rates, were obtained in the experiments. Also, experiments with notched yet unpatched panels were conducted to establish baseline data for comparison. Our objective through this study isto provide input data for future finite element analysis (FEA) to investigate the failure modes and more in-depth the mechanics of materials in the repair field. Such as FEA analysis on the effect of different curing conditions in order to find a model that gives the optimum combination of temperature and duration for suitable cure cycles to eliminate secondary bending and formation of distortion of repaired structures after curing. Also, quantifying residual stresses that may be formed and its relation on the issue of the resulting distortion with single-sided reinforcement. There are several examples in literature on the prediction of residual stress and strain in bi-material geometries,for example, White and Hahn [4, 46] developed a process modelling approach for composites. The process model follows the temperature history of the composite and follows the transition of a composite from its uncured to cured state. Expanding upon the principles of process modelling of composites, Tavakol et al. [47] developed a three-dimensional coupled the rmomechanical process model that determines the mechanical properties on an element based on the temperature and cure kinetics (i.e. degree of cure). Timoshenko [48] predicts the curvature and displacement of a one-dimensional bi-material beam using an analytical solution that is driven by the coefficients of thermal expansion and the material stiffness. However, the approach given by Timoshenko is limited to

material properties that are uniform and constant. That is not to say the equations proposed by Timoshenko cannot predict the residual stress of a bi-material beam comprised of a thermoset polymer composite reasonably well, but they are unable to account for the entire curing process of a composite and important factors, such as polymer shrinkage. Because cure kinetics of a thermoset polymer composite can be quite impactful to the residual stress seen in a composite, Full process modelling of thermoset polymer composites often requires many input parameters to fully define: the cure kinetics, thermal, and mechanical properties, which may be difficult or costly to obtain.Furthermore, analysis on the failure mechanism of the bonded repair with a focus on debonding or delamination behaviour between the bonded material (cohesive or adhesive) using damage criteria. Such experimental investigations can become more complex by including effects such as the thermal residual strains and the secondary bending moment effects generated by the out-of-plane displacement of a one-sided repair. FEA represents a potential cost and time savings in the design process of components including composite materials this also confirmed by Xiaocongin his effort on a review of finite element analysis of adhesively bonded joints. Xiaocong concluded, that the finite element analysis of adhesively bonded joints will help future applications of adhesive bonding by allowing system parameters to be selected to give as large a process window as possible for successful joint manufacture. This will allow many different designs to be simulated in order to perform a selection of different designs before testing, which would currently take too long to perform or be prohibitively expensive in practice [49].

4.2.5.2 General observations

All plates showed distortion after curing. Distortion culminated in warping along the length of the plate. Figure 48 shows an example of a warped plate and it can be seen that the warp curvature is towards the aluminium's side. The observed distortion may be explained by the following reasoning. The adhesive and the boron/epoxy prepreg tape do not cure at room temperature when it is installed on the plate, but when the composite structure is heated to the curing temperature in the furnace; all the materials expand according to the heating procedure. The adhesive and the boron/epoxy prepreg also do not cure during the heating stages, meaning that there is no constraint up to the curing temperature. At the curing temperature with the composite fully expanded, the adhesive and boron/epoxy prepreg cure make a ridged bond to the aluminium.Then upon cooling, the aluminium is constrained by the stiffer patch, which results in the composite area bending on cooling towards the material with the highest coefficient of thermal expansion, i.e. the aluminium.

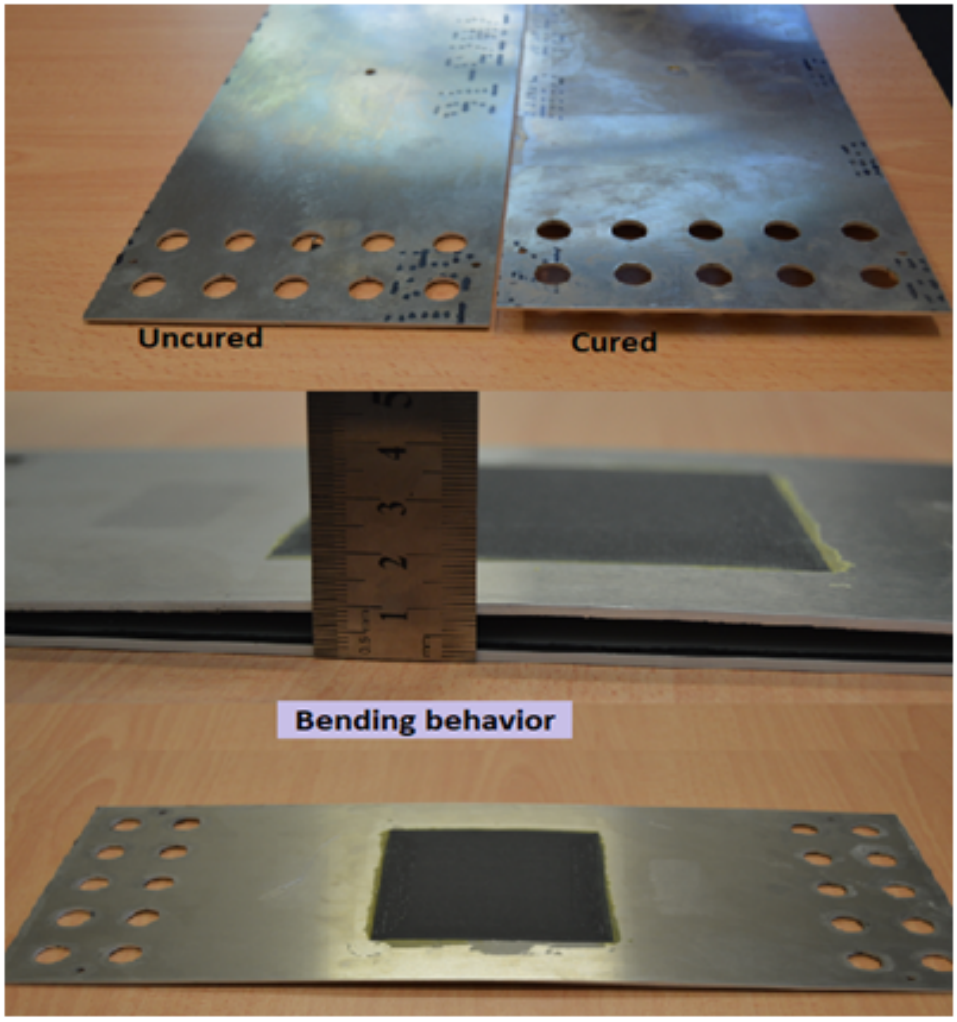


Figure 4.48: Curvature of plate s results in bending due to curing process.

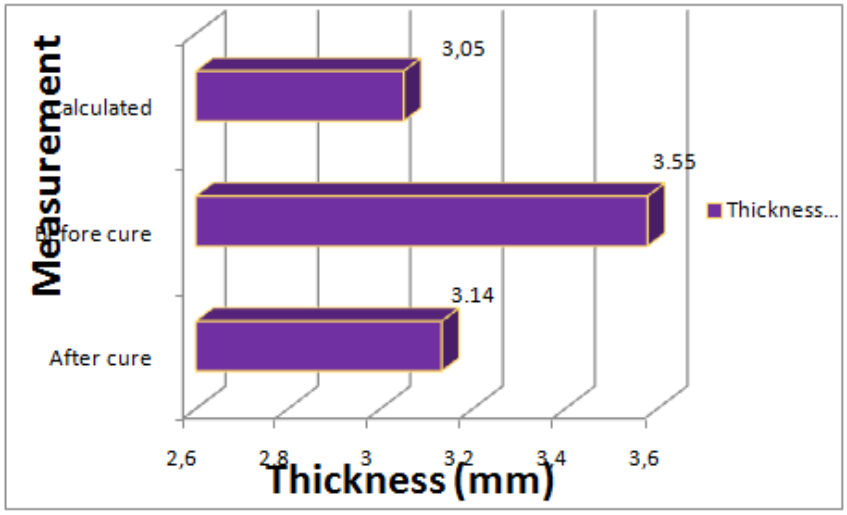


Figure 4.49: Summary of the measured thicknesses.

4.2.5.3 Plate distortion

An inverted plot of the profile from the data for curing condition A, as an example, is shown in Figure 50. It can be seen from Figure 50 that the profile is symmetric around the apex, which is the centre of the patch by way of the lay-up geometry. The distortion is significant but it should be noted that the distortion between the apex and any baseline is exaggerated due to the aspect ratio scaling between the physical plate and the graph. The aluminium ligament up to both sides of the patch baseline are straight as expected because they were unconstrained during cooling and their contraction must be linear. The two linear-fit trend lines reinforce this fact. The distortion of the plate is given as the distance from the ligament baseline to the apex.

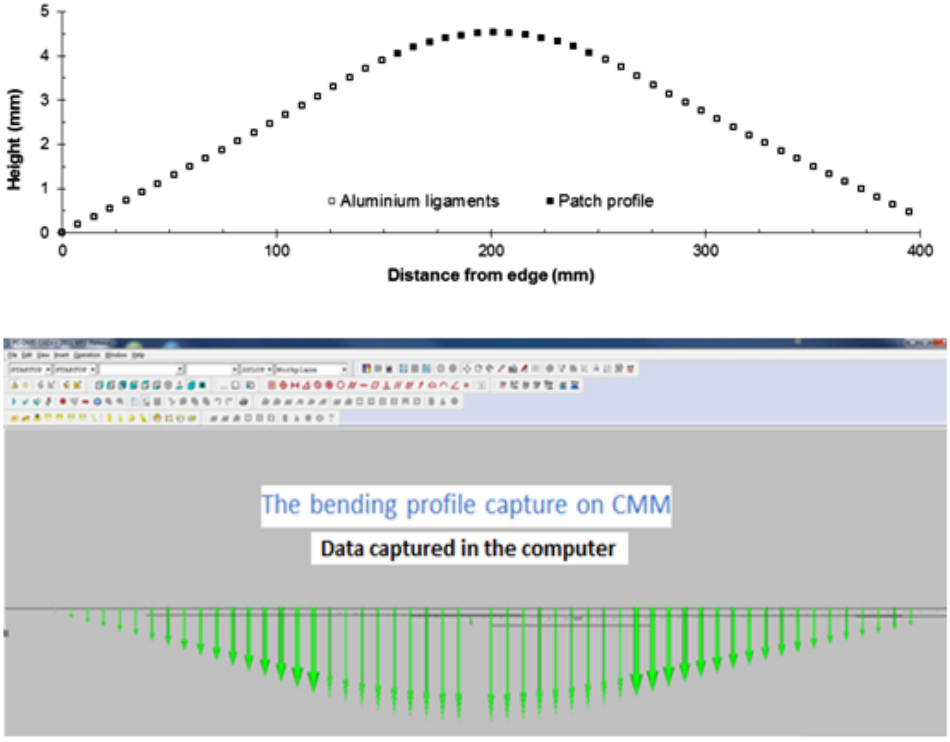


Figure 4.50: Bending profile from the data for curing condition D.

Figure 51 shows the comparison of the measured distortion of bonded repaired samples in different curing conditions.The observed distortion can be explained by the following reasoning. Before the composite patch is bonded to the component to be repaired, each part, plate and patch, is stress-free. The adhesive and the boron/epoxy prepreg tape do not cure at room temperature when it is installed on the plate. They also do not cure during the heating stages, meaning that there is no constraint up to the curing temperature; but during the curing process of the bonding, each part must be heated to an elevated cure temperature in the furnace; all the materials expand according to their coefficient of thermal expansion (CTE). The mismatch in CTE results in the plate being in tension and the patch being in compression after cooling down to ambient conditions. The aluminium is constrained by the stiffer patch that results in the composite area bending on cooling towards the material with the highest coefficient of thermal expansion, i.e. the aluminium. The contraction is related to the expansion of the materials according to the

cure temperature range chosen. As seen cure condition A and D have higher distortion levels with the use of higher curing temperatures (120 $^{\circ}C$) compared to B and C (80 $^{\circ}C$). The variation in distortion levels in relation to the temperature and time of the different cure cycles can be seen in Table 7. We can conclude that the effect of different coefficient thermal expansion on the distortion levels is more pronounced as a change in temperature $\triangle T$ increases during curing or in the cool down. Zhu et al [45] also concluded that curing-induced distortion in adhesive bonding of AA6061 and steel was due to differences in CTE of the two materials, as well as curing temperature. They suggested that differences in CTE intrinsically induced the distortion, while the high curing temperature extrinsically generated the distortion. Finally, the cured adhesive then captured the distortion. It was found that the peak curing temperature was the main contributor to the distortion.

Curing condition	Cure label	Distortion	
		Plate mm	Patch µm
120 °C / 60 min:	A	4.912	617.75
80 °C / 480 min // 120 °C / 60 min:	B	4.129	484.25
80 °C / 480 min:	C	3.735	374.75
120 °C / 60+45 min:	D	4.923	625.50

Table 4.7: Distortion variation of bonded repaired samples in different curing conditions.

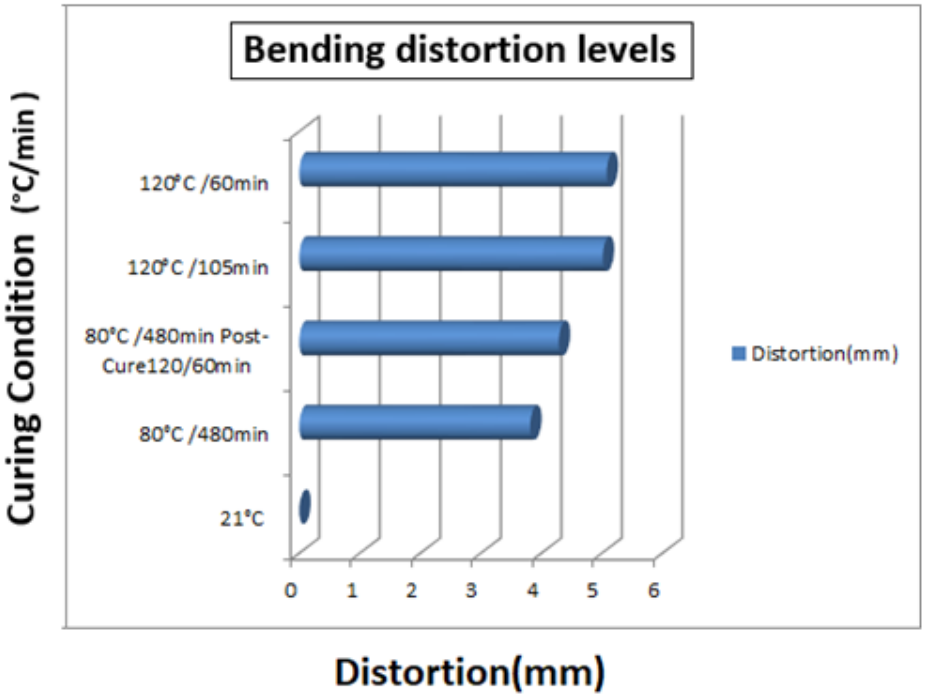


Figure 4.51: Distortion comparison of bonded repaired samples in different curing conditions.

4.2.5.4 Fatigue life

As discussed, during the curing process, adhesively bonded composite/metal laminate structures that are held at elevated temperatures for material processing, develop high residual stresses because of the difference in coefficients of thermal expansion (CTE) for different materials. This thermal mismatch can result in delamination or debonding of materials, which facilitates fatigue crack growth at the polymer/metal interface. Further, the influence of different curing temperatures and times investigation was carried out on the fatigue life of the repaired structure. Fatigue crack growth tests were performed on unrepaired and repaired panels.

The plate is symmetrical along its length with respect to the hole and notch. The crack length starts at 4 mm, which is the 3 mm radius of the hole plus 1 mm for the machined notch.The crack length measurement is calibrated at the start of each test and measurement data was automatically logged. The fatigue propagation curves terminate at 45 mm which was chosen and defines fatigue life, in the case of the repaired plates to prevent the crack from growing past the edge of the half-patch at 50 mm. The fatigue crack growth curve for unrepaired and repaired plate is presented respectively in Figure 52 and Figure 53. Curves for all cure conditions start at the same point, but Figure 53 shows the propagation portion of the curve.

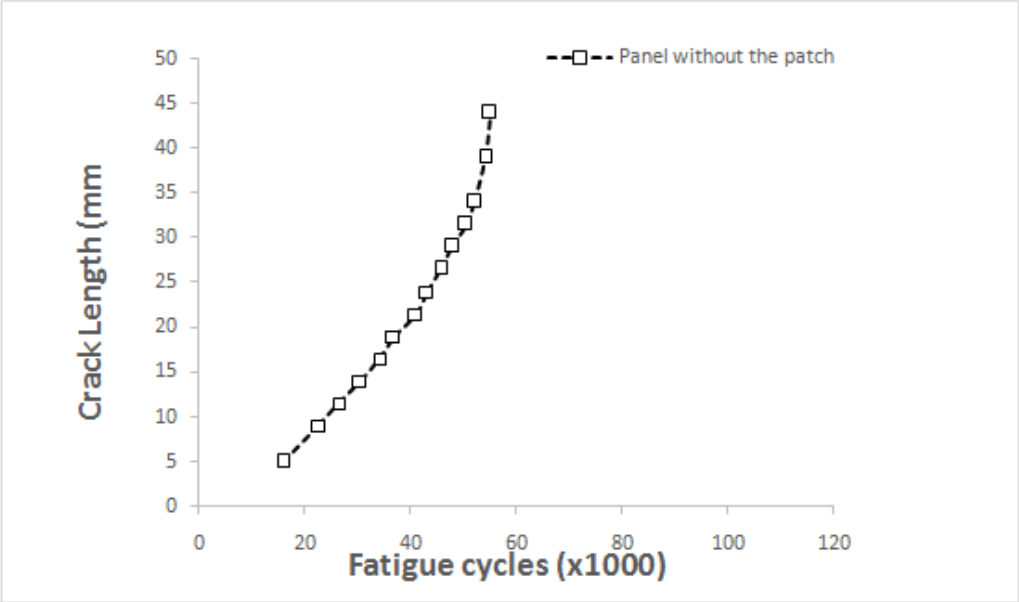


Figure 4.52: Fatigue life of cracked/ unrepaired plate.

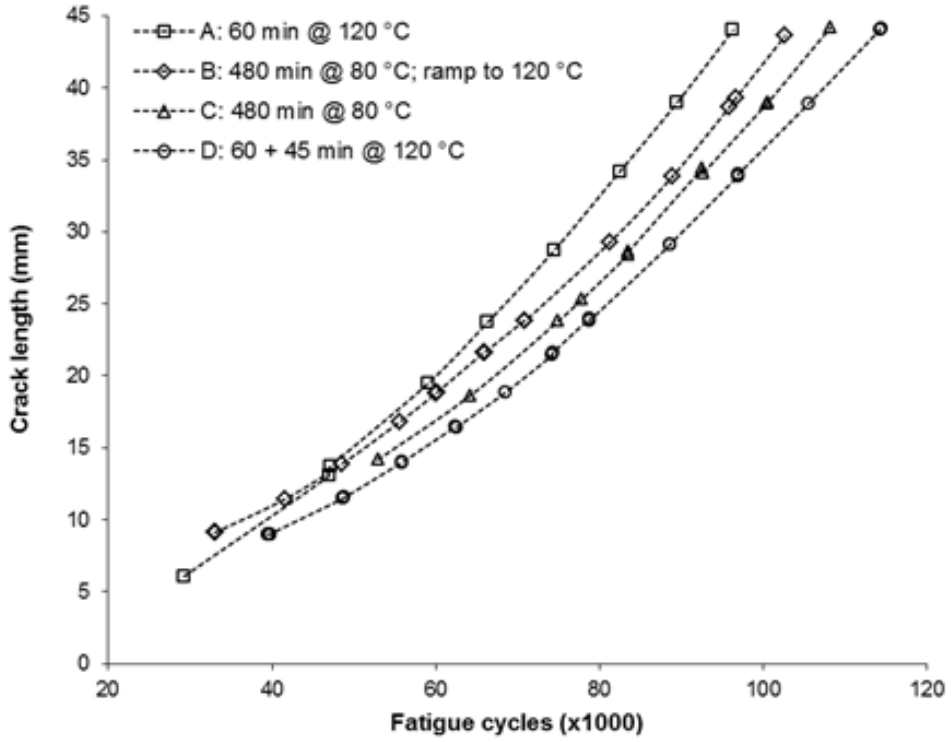


Figure 4.53: Effect of curing condition on crack propagation in the repaired plates.

Results from fatigue tests to determine the effect of various cure parameters on the repair efficiency and fatigue life of the structure showed that the fatigue life of the unrepaired structure according to Figure52is decreased compared to the repaired structures in Figure 53. The repairing of the cracked plate using bonded composite patches is effective in decreasing the crack growth rate and thereby enhancing the fatigue life of the structure. The plates with the bonded composite repair shows an increase in the number of cycles in an average of 70% when compared to the non-repaired plate, which improves considerably the fatigue life. This is achieved by the patch absorbing some of the stresses at the crack tip thereby reducing the magnitude of stress intensity factor and resulting in a significant reduction of the crack propagation as discussed earlier. This shows the beneficial effect of the patch. One can thus confirm again that the use of a composite patch offers an enormous advantage for repaired cracks to increase the service life as shown in Figure 53.The general trend in Figure 53 is uneventful with a gradual stabilisation of the crack growth rate. The final order of fatigue life cycles for cure condition is $A < B < C < D$.

4.2.5.5 Fatigue-distortion relationship

Figure 54 shows the relationship between plate and patch distortion and fatigue life. It is expected that increasing distortion would reduce fatigue life, presumably due to induced bending stresses that could contribute to the crack growth. It can be seen that fatigue life decreases as the plate distortion increases, that is, for cure conditions A, B, and C. The exception is cure condition D, which has the largest plate distortions, but the longest fatigue life.

The plates under cyclic tension-tension fatigue test are subjected to compression because of the bending curvature. In order to perform the fatigue crack growth test as the results can be only logged under tension-tension mode. Curved plate were constrained to eliminate the plate offset in order to see the effect of curing conditions on the fatigue life of the repaired structure. Elevated curing temperatures enhance the adhesion between the metal surface and the composite patch and overall durability. However, elevated cure temperature may result in distortion with bending curvature. Distortion is not suitable for aero structure skins due to aerodynamics purposes in-service conditions.

The fatigue life follows a straight-line relationship with the plate distortion (for cure conditions A, B and C), and the curve fitting coefficient has an $R^2 = 0.9779$ which indicates a very strong correlation. The fatigue life in the case of the patch distortion also follows a linear relationship with an $R^2 = 0.9991$, which indicates even a stronger relationship. Cure condition D does not follow the straight-line relationship in Figure 54 and the reasons for its unique behaviour is discussed in the following section.

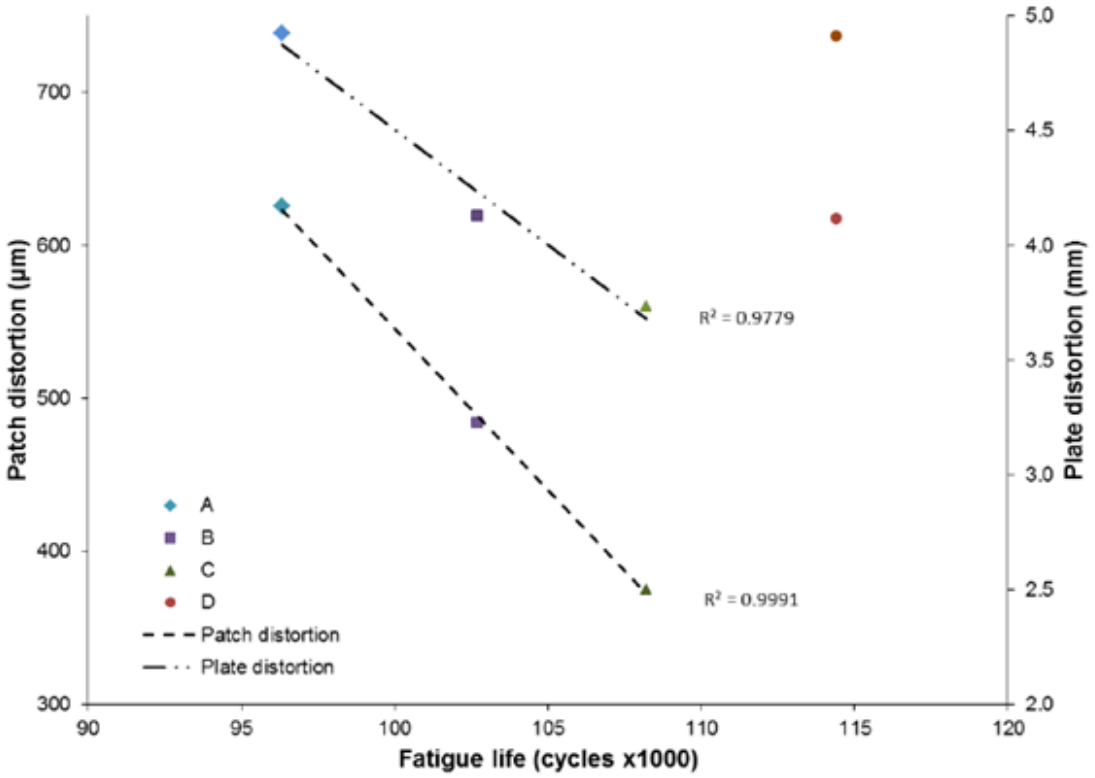


Figure 4.54: The relationships between fatigue life and plate distortion.

4.2.5.6 Failure mechanism

Adhesive bonding is defined as the process of joining parts using a non-metallic substance (adhesive) which undergoes a physical or chemical hardening reaction causing the parts to join together through surface adherence (adhesion) and internal strength (cohesion). Ideally, the joining surfaces should be clean and generally require proper prepa-

ration of the aluminium surface. It is recommended by the adhesive supplier that the adhesive be cured at $120^{\circ}C \pm 3^{\circ}C$. A cure process at 80 $^{\circ}C$ would probably result in an insufficiently cured adhesive layer. There are also questions related to the interface properties of the adhesive and prepreg tape epoxy. It could be expected that the interface does not reach its optimum properties if the cure process deviates from the recommendation, especially the cure temperature. However, curing time would also be the other important factor. The longer time curing of lower temperatures as in the curing process (C) and (B) gave longer fatigue life than condition (A) - see Figure 53. In addition, the metal surface treatment could have also influenced the effects observed, as the surface preparation of the metal is an important factor influencing the strength of the bonded joint.Today, many different types of adhesives, pre-treatment methods and processing techniques are available and on-going investigations are looking for further improvements. The selection of the optimum solution involves a close consideration of multiple criteria and close contact with the respective suppliers is recommended. In some cases, extensive experimental testing may be required.

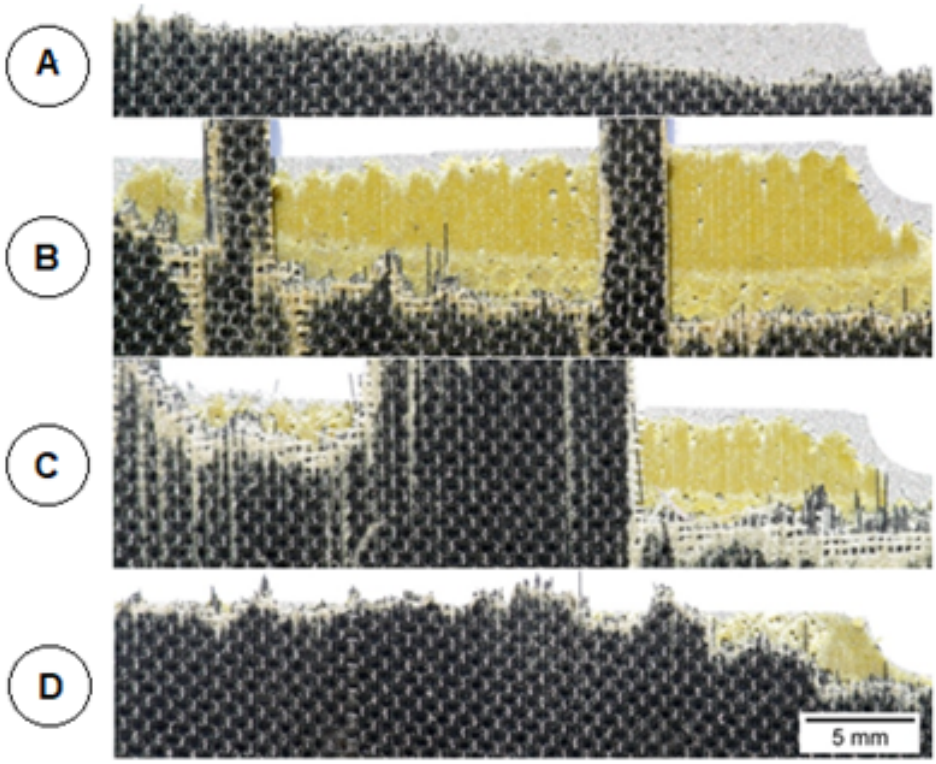


Figure 4.55: Fractured samples in shear mode after tension test.

The bending behaviour can influence the fatigue life of the structure in a cyclic mode. Tension on the repair system creates shear stress in the bond line. Also, the bond area picks up stresses because the patch is bonded to the plate surface, which can lead to possible peel and cleavage stress where bonded repairs failure occurs in shear mode. Figure 54 shows the fractured plates after the fatigue test, i.e. the fatigue test was first completed growing the crack inside the patch zone, and then the plate was subsequently pulled apart. At first, it can be seen that debonding between the aluminium base plate and the adhesive

happened in cases for cure condition A, B, and C. The same type of debonding was not evident for cure condition D. Furthermore, it can be seen from the area around the fatigue crack that the yellow adhesive is visible underneath the boron/epoxy layer, meaning that there also was debonding at the adhesive-boron/epoxy interface. The increased fatigue life of cure condition D is a good example that infers a proper interface bond between the adhesive layer and the boron/epoxy layer. That is because of the improved bonding enhanced by the elevated temperature. It has a high degree of distortion, but it also has the best fatigue life (Figure 53). However, excessive distortion due to the high thermal stresses is not suitable for aero-structure skins due to aerodynamics purposes in-service conditions. In that case, reducing the high distortion can be by reducing the effects of thermal stresses. For reducing the effect of thermal stresses is to restrict the deformation of the parent material during the curing process. An example of how this occurs would be the presence of stringers or longitudinal stiffeners restricting the bending of fuselage skin material during in situ repairs. These under structure components causing displacement constraints in the proximity of the repair result in a reduction of material expansion of the skin during the heating process. The influence of the surrounding structure in reducing the effective thermal expansion of the aluminium plate would considerably reduce the distortion and should be taken in consideration when comparing the observed distortion on the samples to what would occur in practice.

4.2.6 References

[20] Ting T, Jones R, Chiu W, Marshall I, Greer J. Composites repairs to rib stiffened sheets. Journal of Composite Structure 1999; 47: 737-743.
[21] Butkusl M, Valentin RV, Johnson WS. Fatigue analysis of cracked thick aluminum plate bonded with composite patches. Compos Struct 2004;64:79-90.
[22] Niedernhuber M, Holtmannspötter J, Ehrlich I. Fibre-oriented repair geometries for composite materials. Composites Part B: Engineering 2016; 94: 327-337.
[23] Baker A.A, Jones R. Bonded Repair of Aircraft Structures. Composite part B 1988 ; Chapter 6 ; Section 7.
[24] Khashaba UA, Aljinaidi AA, Hamed MA. Fatigue and reliability analysis of nanomodified scarf adhesive joints in carbon fibre composites. Composites Part B: Engineering 2017; 20: 103-117.
[25] Hugo C. Biscaia, Carlos Chastre, David Cruz, André Viegas. Prediction of the interfacial performance of CFRP laminates and old timber bonded joints with different strengthening techniques. Composites Part B: Engineering 2017; 108: 1-17.
[26] CSIR, internal material data sheet, AMSTS material test report.Material property of SIGRATEX KDU 1092/AXSON EPOLAM 2022unidirectional carbon fibre epoxy composite, data sheet 2017.
[27] Mohsen Heshmati, Reza Haghani, Mohammad Al-Emrani. Durability of bonded FRP-to-steel joints: Effects of moisture, de-icing salt solution, temperature and FRP type. Composites Part B: Engineering 2017; 119: 153-167.
[28] Kinloch AJ, Adhesion and Adhesives. Chapman and Hall; 1987; London and New York.
[29] Adams RD, Wake WC. Structural Adhesive Joints in Engineering. Elsevier Applied Science Publishers 1984; London.
[30] Emami Mehr M, Aghamohammadi H, Hosseini Abbandanak SN, Aghamirzadeh Gh R, Eslami-Farsani R, Siadati SMH. Effects of applying a combination of surface treatments on the mechanical behaviour of basalt fiber metal laminates. International Journal of Adhesion and Adhesives 2019; 92: 133-141.
[31] Zouambi L, Khodja M, Oudad W, Fekirini H, Moller H, Bachir Bouiadjra B. J-integral evaluation of repaired cracks in AA7075-T6 structures subjected to uniaxial tensile stresses. Polymer Testing 2019; https://doi.org/10.1016/j.polymertesting.2019.04.021.
[32] Pastor ML, Balandraud X, Grédiac M, Robert JL. The fatigue response of aluminium specimens reinforced with carbon epoxy patches. Composite Structures 2008; 83(3): 237-246.
[33] Hosseini-Toudeshky H, Mohammadi B, Bakhshandeh S. Mixed-mode fatigue crack growth of thin aluminium sheets with single-side repair using experimental and numerical methods. Fatigue & Fracture of Engineering Materials & Structures 2007;30: 629-639.
[34] Günther G, Maier A. Composite repair for metallic aircraft structures. Proceedings of 27th International congress of the aeronautical sciences ICAS2010; 19-24 September 2010; France.
[35] Heatcon Abaris Training, Advanced Composite Structures: Fabrication and Damage Repair Phase I, II, III, Training Materials, Cwmbram, UK, 2008.
[36] Kristopher Robert Doll. Characterization polymerization-induced work piece distortion in photo-activated adhesive workholding. PhD Thesis 2014; The Pennsylvania State University.

[37] Wisnom M, Gigliotti M, Ersoy N, Campbell M, Potter K. Mechanisms generating residual stresses and distortion during manufacture of polymer-matrix composite structures. Composites Part A: Applied Science and Manufacturin 2006; 37(4): 522-529.
[38] Taylor P, Antonucci V, Giordano M, Nasser J, Nicolais L. Process-Induced Residual Stresses in Polymer-Based Composites. Polymer News 2005; 30(8): 238-247.
[39] Olivier PA, Sawi I. Designing curing conditions in order to analyse the influence of process-induced stresses upon some mechanical properties of carbon / epoxy laminates at constant Tg and degree of cure. International Journal of Material Forming 2010; 3(2): 1373-1389.
[40] Cytec Engineered Materials; FM73 Epoxy Film Adhesive; Technical Data Sheet.
[41] Russel AJ. Fatigue crack growth in adhesively bonded graphite/epoxy joints under shear loading. Proceedings of the ASME Symposium on Advances in Adhesively Bonded Joints. ASME; New York 1988: 87-92.
[42] Kim KS and Hahn HT. Residual stress development during processing of graphite/epoxy composites. Composites Science and Technology 1989; 36(2): 121-132.
[43] Weitsman Y. Optimal cool-down in linear viscoelasticity. Journal of Applied Mechanics 1980; 47(1): 35-39.
[44] Montgomery TS and MacKnight WJ. Introduction to Polymer Viscoelasticity. Wiley-Interscience, 3rd edition; ISBN: 978-0-471-74045-2; Aug 2005; 336 pages.
[45] White SR, Hahn HT. Process Modeling of Composite Materials: Residual Stress Development during Cure. Part I. Model Formulation. Journal of Composite Materials 1992; 26: 2402-2422.
[46] White SR, Hahn HT. Process Modeling of Composite Materials: Residual Stress Development during Cure. Part II. Experimental Validation. Journal of Composite Materials 1992; 26: 2423-2453.
[47] Tavakol B, Roozbehjavan P, Ahmed A, Das R, Joven R, Koushyar H, Rodriguez A, and Minaie B. Prediction of residual stresses and distortion in carbon fiber epoxy composite parts due to curing process using finite element analysis. Journal of Applied Polymer Science 2013; 128(2): 941-950.
[48] Timoshenko S. Analysis of bi-metal thermostats. Journal of the Optical Society of America 1925;11(3): 233-255.
[49] Xiaocong He. A review of finite element analysis of adhesively bonded joints. International Journal of Adhesion and Adhesives 2011; 31(4): 248-264.
[50] R. Fredell, Dr. I. J. Schijve and A. Lizza. "Damage-Tolerant Repairs to Pressurized Fuselages: 'Soft-Patching' with GLARE Fiber Metal Laminates." Presented to 18th ICAFSymposium, May, Melbourne, Australia. 1995.

Chapter 5

Conclusions & Recommendations

Bonded fibre composite repair on damaged metal structures can be an effective method to restore load carrying capacity or increase damage resistance. The purpose of this method is to slow or arrest crack growth, and thus, prolong the life of the structure. The technology has been successfully used in repair applications on both military and civil aircraft for fatigue enhancement and crack patching. This research presents an experimental study on the fatigue patching efficiency for notched 7075-T6 aluminium alloy panels reinforced with single sided boron/epoxy patches. Crack propagation behaviour from a notched hole under uniaxial fatigue loading of 7075T6 aluminium alloy repaired with composite patches was investigated. Part 1: Crack propagation from a notched hole under uniaxial fatigue loading of AA7075-T6 plate; Part 2: The behaviour of cracks repaired with boron fibre / epoxy composite patch in AA 7075-T6. The latter is divided into three-section research:

1. Fatigue life a notched hole in AA7075-T6 aircraft plate repaired with a boron/epoxy and Carbon/ epoxy composite patches.
2. Fatigue life of AA7075-T6 aircraft plate cracked with various crack length repaired with a boron/epoxy composite patch.
3. Effect of curing cycle on fatigue life of cracked AA7075-T6 aircraft plate repaired with a boron/epoxy composite patch.

Findings are summarised as follows:

- Fatigue loads are the major contributors to the service induced damages in the aerospace structures.
- For unpatched plate, our study results has shown a good agreement with data in the literature.
- Bonded patch repair reduces stresses near the crack tip by bridging between the cracked plate and composite patch, and therefore retards or completely stops the crack growth.

- The FCGR decrease is achieved by the patch absorbing some of the stresses at the crack tip, thereby increasing the number of cycles to failure.

- The repair patch is effective in decreasing the crack growth rate and thereby enhancing the fatigue life of the structure.

- The crack had propagated across the entire width of the specimen to failure. However, the patch kept holding the panel together.It was also shown that the static strength of the damaged repairs exceeds that of an uncracked specimen.

- The stress intensity factor in plate without a patch was higher than a plate with a patch.

- The repair with patch composites leads to significant crack growth retardation in aluminium alloy 7075-T6, but it is different from a repair material to another. The retardation effect due to the presence of the patch was seen by the decrease in crack growth rate between unpatched and patched panels.

- The reduction of the stress intensity factor at the crack tip with boron/epoxy patches repair of thin structures compare to carbon/ epoxy patches is more significant.

- The thermal expansion coefficient of boron/epoxy makes it a better choice. Thus, carbon fibres are attractive yet not as strong as boron fibres.

- The principal advantages of the composite materials is their high stiffness which allow the use of these materials as thin patches with minimal weights and with a minimal effect on the aerodynamic properties.

- The effect of boron/epoxy and carbon/ epoxy composite patches on the rigidity of the aircraft metallic structures showed this rigidity was highly improved by the presence of the boron/epoxy patch repair.

- The rigidity of the repair structure was highly improved by the presence of the boron/epoxy patches. This showed a significant role of repair material selection on repair efficiency.

- Compared to carbon/epoxy, boron/epoxy is much more costly, less readily available and because of the large fibre diameter less formable. Thus carbon/epoxy is used whenever it is more cost effective or where very high formability is required.

- The composites satisfy most of the requirements; however, their main disadvantage which gives rise to undesirable residual tensile stresses in the repaired component is their low thermal expansion coefficient compare to the adherent.

- Bonded patch repairs, however, require a stringent cleaning, preparation and curing procedure, but when properly designed, fabricated and used, they extend the life of cracked components with high structural efficiency at an economical cost.

- The crack growth rate in unpatched panel is higher compare to the patch ones. The crack growth rate increases as the crack length increases due to higher stress concentration rate at the crack tip.

- The variation of crack growth rate ($da/dN, mm/cyc$) as a function of the crack length for patched and unpatched panel of different crack length repair showed that the patch presence highly decrease the crack growth rate.

- An average of 85% decrease in the crack growth rate from unpatched panel to the patched one.

- The crack growth rate in the patched panel with boron/epoxy composite patch shows 23% lower crack growth rate than the one repaired with the carbon/ epoxy patch.

- The crack growth rate using the boron /epoxy composite patch showed stable crack growth rate as the crack length increases compare to the use of the carbon patch especially with bigger crack lengths.

- Small crack sizes give lower values of the SIF compared to the larger crack sizes. This is due to the high rigidity of the boron/epoxy presence for repair compared to the unrepaired. However, both short and long crack sizes values are higher compare to the repaired ones.

- The small crack length up to 30 mm showed repair efficiency between 2 to 3 times from unrepaired to repaired structure with a repair percentage between 18 to 92% in a crack length between 1 to 25 mm.

- The repair by the patch in larger crack length reduces its efficiency.

- The number of cycles corresponding to the propagation of the crack to rupture of the damaged structure is low compared with a repaired structure especially in large

crack from up to 30mm. That means in such cases, the number of cycles decreases as the crack length size increases.

- It is inefficient to repair cracks of size more than 30mm crack length.

- In various crack length repaired with composite patches, it is not recommended to repair large cracks for long-term maintenance as it can lead to a catastrophic results. However, it is effective to repair large crack for emergency conditions.

To enhance the repair bonding durability and reliability, curing conditions and processes of bonded composite patches repair with structural adhesive were investigated.

- Elevated curing temperature for the repair materials induces thermal stress in the metal structure. Overall, curing condition affects the structural distortion and secondary bending influences the behaviour in single side reinforcement/repair.

- The structure's curvature in the repair area would require detailed investigation if applied to fleet aircrafts.

- Elevated curing temperatures may result in thermal stresses which is a matter of concern on the durability of the repair.When designing a bonded composite repair it must be understood that adhesives and composite materials, like all materials, have an operating temperature envelope. Outside the limits of this envelope, the repair may not perform adequately.

- Localized bending (due to neutral load axis shift) and thermally induced curvatures (generated during patch adhesion) on the fatigue crack growth rate were also examined. In some cases, extensive experimental testing may be required.

- The boron fibre/epoxy composite patch gives less distortion because of the close thermal expansion coefficient to aluminium structure, but is used less due the delicate handling and the expensive cost.

- The higher coefficient of thermal expansion differences (CTE), the less residual stress problems will be encountered.

- The optimal curing cycles (temperatures vs durations) of bonded repair structures can be evaluated by investigating separately the thermal expansion values of each material to match during the curing cycle as the patch cure cycle and the bonding process is carried out in one operation. It also enables conditions to be adjusted to optimise bond durability in low cure temperature operations.

- Before implementing any repair it is necessary to understand the load transfer mechanisms.

- Factors affecting crack growth after bonded patched repair specific to the present patch repair problem, are:Thermal Residual Stresses, Bond Quality and Secondary Bending.

Future work

Recommendations for further research and future technology requirements in this topic area include:

- A detailed research for aircraft structures repairs durability with patch composite that considers repair materials characterisation is needed. This includes the optimum curing time and temperature of epoxies in different composite patches and the adhesive layer.

- Thermomechanical Analysis (TMA) and Dynamic Mechanical Analysis (DMA) for more detailed dynamic material data to understand the thermal and mechanical behaviour of the involved material are required.

- Additional understanding of material properties should not be depended only on manufacturer data sheet, as it is insufficient for full behaviour analysis and each material batch manufactured is different from the other.

- For aircraft damage repair with composites patches and aircraft composites components, it is more important to perform those analyses starting from low temperatures ranges ($< 18^{\circ}C$). That will conform to the in-service aircrafts temperatures and environment to predict the failures before maintenance to detect defects or damage during inspection. It can be beneficial to a more precise performance evaluation and service life prediction of aviation metallic structures. Therefore, extensive studies on bond failure needs to be performed.

- In a curved repair, stresses are developed normal to the plane of the repair (peel stresses) and current understanding of these and the associated design allowable for various repair materials is poor. The problem is that it is difficult to estimate accurately the degree of thermal expansion of the repaired area due to the constraint that is provided by the surrounding cooler structure.

- There is scope to optimise the stress distributions within repairs using routines within Finite Element Packages. Validation of these experimental data is required and further development of them to ensure that they can accurately predict the

effect of various parameters such as the nature of the loading, the influence of residual stresses and environmental effects. The ability to calculate more readily the magnitude of any thermally generated residual stresses would be advantageous for some repairs.

- Numerical validation of models can be used when repairs are required to different crack sizes in different materials thicknesses.

- Considerable work is required to ensure that acceptable levels of bond durability can be achieved. It may be necessary to prove that the repair will not be compromised by durability problems caused by the dissimilar materials and surface preparation methods.

- A parametric study into the effect of bending on repaired plates stiffness would provide the necessary insight for minimizing thermal bending effects.

- The effects of thermal residual stresses and deformations due to laboratory fatigue loading need to be quantified using numerical methods to extend the findings of this study to the repair conditions of restrained structure (with stringers etc.) on aircraft.

- The prediction of fatigue crack growth behaviour in metallic structures is a highly involved and complex process. The complexity is further compounded when these structures are repaired with bonded composite patches. Despite this complexity, there is a need to establish a generalized approach, of sufficient accuracy, to model the crack growth rates after repair.

Effect of Curing Cycle on Fatigue Life of Cracked AA7075-T6 Aircraft Sheet Repaired with a Boron/Epoxy Composite Patch

Malika Khodja, Hamida Fekirini, Gonasagren Govender & Belabbes Bachir Bouiadjra



Iranian Journal of Science and Technology, Transactions of Mechanical Engineering

ISSN 2228-6187

Iran J Sci Technol Trans Mech Eng
DOI 10.1007/s40997-020-00400-6

Springer

Your article is protected by copyright and all rights are held exclusively by Shiraz University. This e-offprint is for personal use only and shall not be self-archived in electronic repositories. If you wish to self-archive your article, please use the accepted manuscript version for posting on your own website. You may further deposit the accepted manuscript version in any repository, provided it is only made publicly available 12 months after official publication or later and provided acknowledgement is given to the original source of publication and a link is inserted to the published article on Springer's website. The link must be accompanied by the following text: "The final publication is available at link.springer.com".



Author's personal copy

Iranian Journal of Science and Technology, Transactions of Mechanical Engineering
https://doi.org/10.1007/s40997-020-00400-6

RESEARCH PAPER

Effect of Curing Cycle on Fatigue Life of Cracked AA7075-T6 Aircraft Sheet Repaired with a Boron/Epoxy Composite Patch

Malika Khodja[1,2] · Hamida Fekirini[1] · Gonasagren Govender[2] · Belabbes Bachir Bouiadjra[1]

Received: 7 February 2020 / Accepted: 26 September 2020
© Shiraz University 2020

Abstract
Fatigue failure cracking under fluctuating stressing has been a major concern of engineering for many years. Fatigue failures have become more common and thereby significantly influencing engineering design as sources of vibration and dynamic loading of components have increased. The repairs of damaged metallic structures using bonded composite patch have been proved as an economical method to increase the fatigue life of damaged structures. This paper deals with an experimental study on the fatigue life of AA7075-T6 aluminium alloy plates reinforced with single-sided boron/epoxy patches. The study focuses on the effect of different curing cycles on the fatigue life of reinforced/repaired structure. Bonds were tested with four different curing cycles. Distortion behaviour of bonded repair is studied. The influence of the curing cycle conditions on fatigue life is presented. Also, the repair patch effectiveness on the stress state at the crack tip in increasing the fatigue life of the unrepaired structure is shown. The obtained results show that repair of the cracked plate can be done effectively by using bonded composite patches. It is shown to decrease the crack growth rate to enhance the fatigue life of the structure by 70%. Elevated curing temperatures enhance the adhesion between the metal surface and the composite patch and overall durability. However, elevated cure temperature may result in distortion with bending curvature. The optimum combination of temperature and duration for the curing cycle is of importance to reduce the distortion.

Keywords Cure behaviour · Fatigue life · Boron/epoxy · AA7075-T6 · Crack · Failure

1 Introduction

The technology of advanced composites has been utilised for a long time in repair/reinforcement procedures of metallic aero-structures. The bonded patch composite is an example of these advances in aviation and other disciplines such as civil engineering. The repair reliability remains an engineering concern from a technology and design evaluation perspective. It can be dangerous if the repair is applied and implemented badly in comparison to an unrepaired structure (Gwo-Chung and Shyan 2004). A commonly used repair/reinforcement method is mechanical fastening, by attaching the metallic patch with bolting or riveting over the damaged part to rearrange the load distribution and minimise the stress concentration. This mechanical method is less appealing, due to non-uniformity of the load transfer and the formation of stress concentration areas nearby the fasteners. The main advantages of composite materials for repair applications are their high strength, relatively low weight and good corrosion resistance (Niedernhuber et al. 2016). Bonded patch repair to metallic structures has been used before by the Aeronautical Research Laboratory (ARL) in Australia as another option to bolted repairs in their development programs. The application of the bonded composite repair in some of the ARL fleet repair programs exhibited a significant improvement of the durability of aerostructures in-service environments (Baker and Jones (1988). The Composite Patch Bonded Repair [CPBR method consists of the following steps: (1) preparation of the metal surface, (2) the composite patch application, (3) Curing of the repair materials, and (4) a cooling step (Heatcon Abaris Training 2008)].

Structural fatigue cracking is inevitable in all ageing aircraft because of usage. High strength aluminium alloys such as AA7075-T6 are particularly susceptible to fatigue

✉ Hamida Fekirini
fekirini.hamida@gmail.com; hamida.fekirini@univ-sba.dz

[1] LMPM Laboratory Mechanics and Physics of Materials, Mechanical Engineering Department, University of Sidi Bel Abbes, Sidi Bel Abbès, Algeria

[2] Materials Science and Manufacturing, Council for Scientific and Industrial Research, Pretoria 0184, South Africa

Published online: 16 October 2020

Author's personal copy

cracking. AA7075-T6 is our material of choice for this study. This alloy is used due to its high strength, but this compromises crack initiation/growth performance and corrosion resistance, especially susceptibility to stress corrosion cracking. Structural profiles also notably affect the fatigue existence. For examples, square holes or sharp corners lead to high local stresses where fatigue cracks can initiate (Chakherlou and Vogwell 2003). Aircraft parts may also be exposed to multiaxial stress states of fluctuating amplitude, which can lead to fatigue crack initiation and growth and catastrophic failure (Bureau of Accident Investigation 1988). When subjected to cyclic loading, the presence of defects reduces the operating life of structural components. This can cause damage to the interior structure of laminated composites, which are barely visible in many instances. Hence, fatigue and reliability analyses and prediction of structural performance are important to investigate, especially with the development of composite materials. Such studies have been done in various cases by using different techniques (Khashaba et al. 2017; Biscaia et al. 2017; Heshmati et al. 2017).

Bonded Repair technology is becoming more advanced and accepted, but additional research is needed to help predict how well bonded composite patches perform. The are many parameters that influence the success of the repair of damaged metal components, such as the mechanical and geometrical properties of the repair materials (composite and adhesive) (Benyahia et al. 2014; Baker et al. 2009; Chung 2003; Bouanani et al. 2013). Important characteristics are the shape of the patch, the fibre orientation, number of plies in the composite, as well as the thickness of the adhesive (Mall and Schubbe 2009; Riccio et al. 2014; Brighenti et al. 2006; Zouambi et al. 2019).

The repair's performance is influenced by some other parameters too, including the thermal residual stresses caused by the adhesive during curing, debonding of the adhesive due to fatigue loading and bending curvature due to curing.

Numerous studies have been conducted on the repair or reinforcement repairs by CPBR for various applications of damaged components (Klemczyk and Belason 1994; Ouinas et al. 2007; Wang et al. 2006; Alderliesten 2009; Datla et al. 2011; Mall and Conley 2009; Günther and Maier 2010; Benyahia and Bouiadjra 2014; Benyahia et al. 2015; Sohail et al. 2017; Albedah et al. 2018). The most suitable materials have been identified as boron/epoxy and carbon/epoxy composites. The thermal expansion coefficient of boron/epoxy is advantageous, because it is closer to the thermal expansion coefficient of aluminium than carbon/epoxy (Gwo-Chung and Shyan 2004). Klemczyk and Belason (1994) showed that the rigidity of aircraft metallic structures was improved by using boron/epoxy patch repair or reinforcement. Ouinas et al. (2007) used boron/epoxy, as well as graphite/epoxy for the repair of aircraft structures. The boron/epoxy patch resulted in a more significant reduction in the stress intensity factor at the tip of the crack tip. The thermal residual stresses were also found to be less for the boron/epoxy due to the adhesive curing. Various researchers Wang et al. (2006), Alderliesten (2009) and Datla et al. (2011) have performed several other studies experimentally and numerically. Mall and Conley (2009) investigated repaired cracks in thin and thick panels and observed significant bending. Asymmetric repair also resulted in non-uniform crack growth between the unpatched and patched faces of the thick panels. Analysis of metallic aircraft structures development and qualification showed secondary bending structures curvature of single-sided repairs in the repair area. This secondary bending structures curvature influenced the mechanical behaviour and further detailed studies were recommended (Günther and Maier 2010).

Expanding upon bonded repair, especially the evaluation of the fatigue behaviour of aluminium alloy sheets repaired with composite patches, the results indicated that the aircraft structures fatigue life could be improved with the patch repair. For example, Benyahia and Bouiadjra (2014) and Benyahia et al. (2015) conducted experimental fatigue tests on bonded composite repairs of aluminium alloy AA2024-T3 cracked sheet repaired with carbon/epoxy patch. The obtained results show the fatigue life of the damaged structure was increased, especially if the patch repair is performed at relatively small crack sizes. In the paper of Sohail et al. (2017), the behaviour of fatigue cracks in AA2024-T3 and AA7075-T6 under two levels of applied fatigue stresses plates repaired with the bonded composite patch was analysed. It was reported that the patches improved the fatigue properties, but this improvement was less when the applied load was increased. The AA2024-T3 had better fatigue properties in both repaired and unrepaired cases. Another study by Albedah et al. (2018) performed fatigue tests on repaired specimens bonded with elevated temperature curing. They evaluated the effects of thermal residual stresses and showed that fatigue life was greatly reduced by an initial adhesive debond. Their results also showed that the stress intensity factor increased with an increase in the debond width and due to thermal residual stresses.

There are two phenomena that decrease the performance of this technique; the adhesive debond by fatigue loading and the thermal residual stresses due to the adhesive curing. Many different curing effects of composites and adhesive curing have been reported. Studies showed that adhesive bonds experience a decrease in volume while they cure, causing either an accumulation of residual stresses inside the adhesive joint or a distortion of the bonding substrates (Wisnom et al. 2006; Taylor et al. 2005). Residual stresses in composites occur during the curing stage due to resin shrinkage (Olivier and Sawi 2010; White and Hahn 1992).

Author's personal copy

Timoshenko (1925) predicted the curvature and displacement of a one-dimensional bi-material beam using an analytical solution that is influenced by the coefficients of thermal expansion and by the material stiffness. However, the approach given by Timoshenko is limited to material properties that are uniform and constant. Also, it does not account for the entire curing process of composite and other important factors such as polymer shrinkage. To use a composite repair, it must be considered that, like all materials, adhesives and materials have an operating temperature range. Outside these limits, the repair could be unsuccessful. Other researchers have only used a single set of curing parameters, mostly those that have been recommended by the composite patch manufacturers. The novelty of our research is that we have tested different curing parameters to determine differences in terms of bond strength, distortion effects and fatigue life of repaired AA7075-T6 sheets. There are limited investigations available on the influence of curing parameters of structural epoxies on fatigue life of repaired metallic panels considering different curing regimes in a single study. Our study gives an overview on the fatigue life of bonded repair difference using four different curing process parameters. This will assist with applications of adhesive bonding by allowing implementation of smart cure cycles that give a large process window for successful bond repair.

In this investigation, the performance of composite patch repairs on AA7075-T6 sheets was compared to unpatched AA7075-T6 sheets. The fatigue life behaviour of aluminium sheets repaired with a composite patch with four configurations of curing cycles was studied. Through this study, we also aimed to provide more understanding of the composite bonding repair process for future finite element analysis (FEA) to investigate the failure modes and more in-depth the mechanics of materials in the repair field. Experimental data will support more in-depth FEA analysis such as the effect of different curing conditions to find a model that gives the optimum combination of temperature and duration for suitable cure cycles to reduce secondary bending and formation of distortion of repaired structures after curing. FEA will allow numerous designs to be simulated before testing, which would otherwise be time-consuming and expensive to carry out experimentally.

2 Materials and Methods

2.1 Materials

Three panels were made for each cure condition. The layers per panel were: an aluminium AA7075-T6 plate, an adhesive layer and a structural composite layer. The dimension of the AA7075-T6 aluminium plate was 400 mm × 160 mm × 2 mm and contained a 6 mm size rivet hole with a wire-cut notch of 1 mm length perpendicular to the length of the plate emanating from the hole's edge. Figure 1 illustrates the geometry of the plate and composite repair patch. Cytec/Solvay FM73 (Cytec Engineered Materials 2011) knitted epoxy film of 0.25 mm thick was used as an adhesive between the aluminium and the boron/epoxy patch. The structural composite layer was composed of six plies, 5.2-mil thick, of type B4/5521 unidirectional 4-mil Boron/epoxy prepreg tape from Specialty Materials, Inc. (Boron 5521 Prepreg Tape

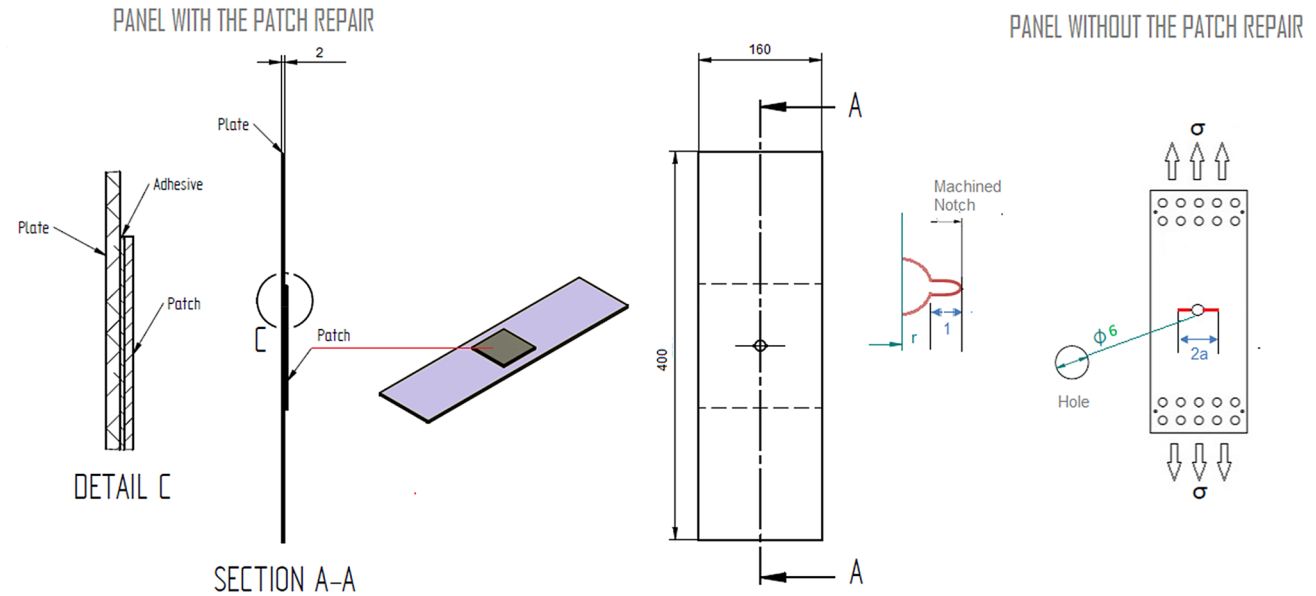


Fig. 1 Geometry of the plate and composite repair patch

Author's personal copy

2013). The materials properties from the supplier materials datasheet utilised in this study for the bonded repair of structures are summarised in Table 1. The chemical composition (in weight %) given for the AA7075-T6 sheet by the manufacturer was 5.6%Zn, 2.6%Mg, 1.4%Cu, 0.19%Cr, 0.13%Fe, 0.07%Si, and aluminium the balance.

2.2 Surface Treatment

An important factor influencing the strength of the bonded joint is the surface preparation of the metal. Many studies have been done on the influence of surface treatment on the mechanical behaviour of bonded assemblies (Kinloch 1987; Adams and Wake 1984; Emami Mehr et al. 2019). To prepare the bonded area, the metal surface was subjected to mechanical abrasion (grit blasting) with uniform alumina grit. The process employed 210 µm alumina grit at a pressure of 300 kPa and a working distance of 10–15 cm to obtain the required surface roughness without inducing compressive stresses. The surface was then cleaned with trichloroethylene to remove impurities. Figure 2 shows the bare prepared aluminium panel.

2.3 Composite Panel Manufacturing

The lay-up is shown schematically in Fig. 3 and it shows the sequence that was employed. The AA7075-T6 aluminium was the base plate; it was bonded with the FM73 adhesive to six layers, Type B4/5521 unidirectional prepreg tape. In the case when the crack position is perpendicular to the loading, the best stacking layup is with the fibres in the unidirectional orientation, as the fibres will inhibit the opening of the notch in fatigue (Zouambi et al. (2019)). An orientation of the fibres for the composites layups in the perpendicular direction to the crack direction is used to obtain optimum results, as also noted by other authors (Pastor et al. 2008; Hosseini-Toudeshky et al. 2007). Plies were all stacked with the unidirectional orientation running perpendicular to the crack and parallel to the fatigue loading direction, the rivet hole being at the centre of the patch. The following conventional layers were applied over the prepreg tape: peel ply, release film, breather fabric cloth, and then the vacuum bag. The vacuum was sealed off at the edge of the vacuum bag with Cytec sealant tape. The full panel with the layup was put in an air convection furnace for curing while maintaining a vacuum of 80 mbar drawn over the layup. An overview of the final samples is shown in Fig. 4 and the steps to the final patched test panel. The patch and adhesive had dimensions of 100 mm × 100 mm × 1.03 mm (thickness including six prepreg tape layers and the adhesive layer).

2.4 Curing Parameters

The tests were performed with four types of curing conditions. The first (A) was subjected to a curing cycle as per the manufacturer's datasheet—heating up in 30 min and held at 120 °C for 60 min (Cytec Engineered Materials 2011). The second (B) was subjected to a combination of low temperature of 80 °C over a long cycle time of 480 min and post cured at 120 °C for 60 min as the recommended cycle as per the datasheet to ensure adhesion at elevated temperatures. A reduction in the cure temperature may affect important

Table 1 Mechanical properties of different repair materials

Properties	Materials		
	AA7075-T6	Boron patch	Adhesive (FM73K)
Longitudinal young modulus (GPa)	68.5	200	2.778
Transverse young modulus (GPa)	68.3	18	2.778
Poison ratio	0.33	0.23	0.38
Hardness (HV20)	186	–	–
Elongation at break (%)	12	–	–
Shear modulus (GPa)	–	5.59	1.006
Thickness (mm)	2	0.13/layer	0.25



Fig. 2 Surface preparation by grit blasting

Author's personal copy

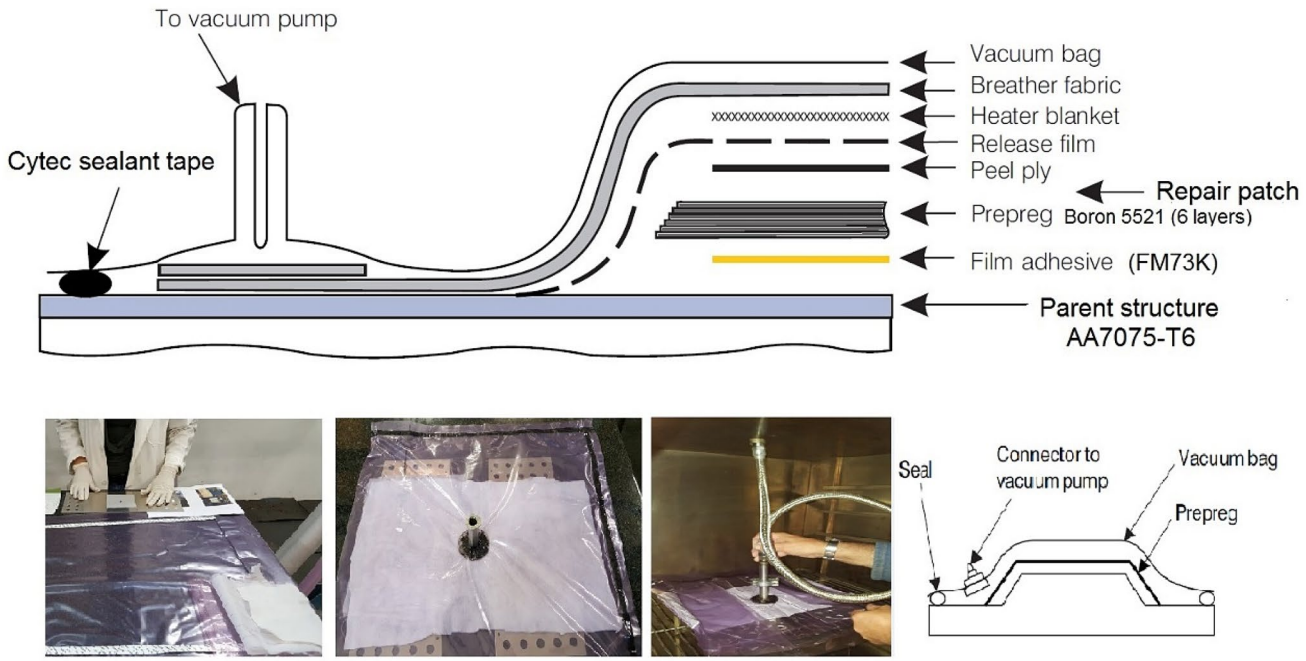


Fig. 3 Detail of vacuum bag lay-up

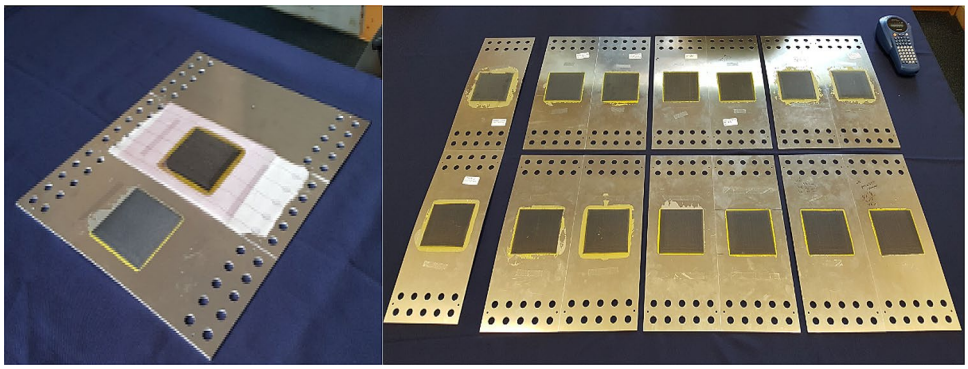


Fig. 4 The steps to the final patched test panel

properties of the bonded repair integrity. Other researchers (Baker and Jones 1988) favoured a lower curing temperature curing because of the differences in coefficients of thermal expansion between the parent and the bond material. The third (C) was subjected to a lower temperature of 80 °C, held for a longer time of 480 min. At elevated temperatures, the curing temperature influences residual stress build-up and shape distortion. Long curing cycles at elevated temperatures minimise residual stress levels during moulding (Kim and Hahn 1989; Weitsman 1980). This is due to the viscoelastic nature of thermosets at elevated temperatures (especially in the rubbery state) where large stress relaxation can occur (Montgomery and MacKnight 2005). Therefore, the fourth (D) was subjected to a prolonged cure cycle, heating to 120 °C for 105 min, to evaluate the prolonged curing time.

The full plate with patch was heated in a convection furnace. The ramp rate to all the curing temperatures was 2 °C/min. Plates were taken out of the furnace and left to cool at ambient conditions after curing. To ensure repeatability, the curing process was controlled by a computer while the temperature of the workpiece (composite panel) was continuously monitored by a thermocouple fixed to the aluminium base plate. The four curing conditions that were used are summarised as follows:

A. 120 °C for 60 min (as per manufacturer specification).
B. 80 °C for 480 min with post-curing ramped to reach 120 °C in 60 min.
C. 80 °C for 480 min.
D. 120 °C for 105 min (60 + 45).

Author's personal copy

2.5 Distortion Measurement

Distortion measurements were done accurately with metrology equipment controlled by computer at NMISA (National Metrology Institute of South Africa, (www.nmisa.org). Distortion was measured by a laser metrology scanning technique using a laser scanner. The distortion measurement was through scanning the length of the plate (400 mm) and took place over 83 intervals. Composite plates were placed convex-side down with the bare aluminium surface facing up for measurement.

2.6 Fatigue Testing

Fatigue crack growth tests of repaired and unrepaired structures were performed on an Instron 1342-H1314 servo-hydraulic system with a dynamic capacity of 25 kN following the ASTM E647 standard (ASTM International 2013). Samples dimensions were chosen to closely match dimensions where the width of the panels is wider to comply with the spacing in stiffened structures of aeroplane skin. The test panel setup is illustrated in Fig. 5. All fatigue tests were conducted at cyclic constant-load amplitude, frequency of 10 Hz and stress ratio of 0.1. Crack length data and number of cycles were logged on a computer. Panels were fatigued until crack lengths approached the edge of the patch.

3 Results and Discussion

3.1 General Observations

All plates showed distortion after curing. Distortion culminated in warping along the length of the plate. Figure 6 shows an example of a warped plate and it can be seen that the warp curvature is towards the aluminium's side.

3.2 Plate Distortion

An inverted plot of the profile from the data for curing condition A, as an example, is shown in Fig. 7. It can be seen from Fig. 7 that the profile is symmetrically around the apex, which is the centre of the patch by way of the lay-up geometry. The distortion is significant but it should be noted that the distortion between the apex and any baseline is exaggerated due to the aspect ratio scaling between the physical plate and the graph. The aluminium ligaments up to both sides of the patch baseline are straight as expected, because they were unconstrained during cooling and their contraction must be linear. The two linear-fit trend lines reinforce this fact. The distortion of the plate is given as the distance from the ligament baseline to the apex. Table 2 represents the measured distortion of bonded repaired samples in different curing conditions.

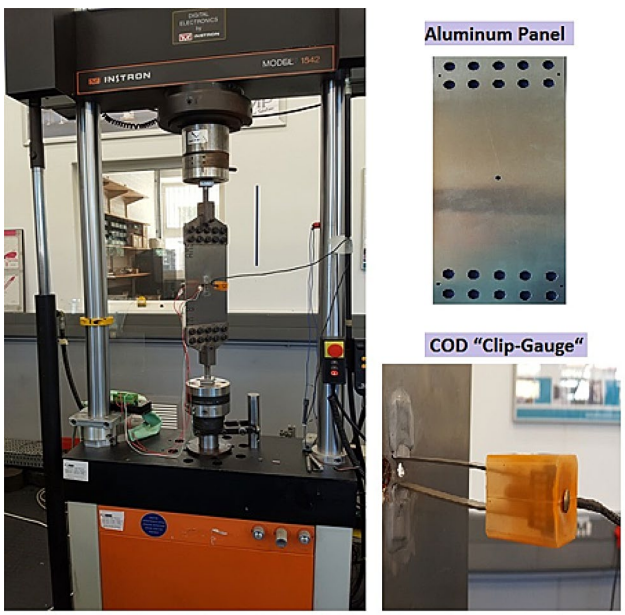


Fig. 5 AA7075-T6 Aluminium Panel fitted with COD "Clip-Gauge"

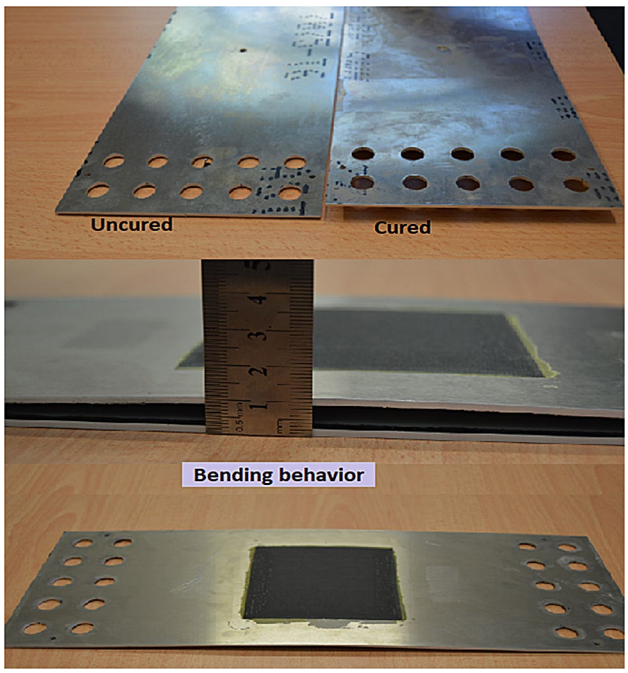


Fig. 6 Curvature of panels results in bending due to the curing process

Author's personal copy

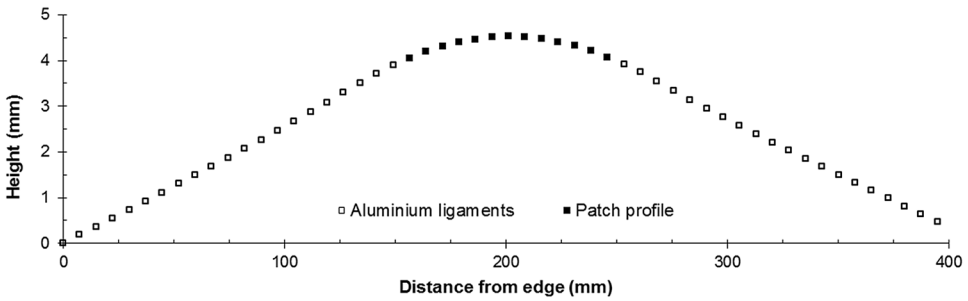


Fig. 7 Bending profile from the data for curing condition A

Table 2 Distortion variation of bonded repaired samples in different curing conditions

Curing condition	Cure label	Distortion	
		Plate (mm)	Patch (μm)
120 °C/60 min	A	4.912	617.75
80 °C/480 min // 120 °C/60 min	B	4.129	484.25
80 °C/480 min	C	3.735	374.75
120 °C/60 + 45 min	D	4.923	625.50

The observed distortion can be explained by the following reasoning. Before the composite patch is bonded to the component to be repaired, each part, plate and patch, is stress-free. The adhesive and the boron/epoxy prepreg tape do not cure at room temperature when it is installed on the plate. They also do not cure during the heating stages, meaning that there is no constraint up to the curing temperature; but during the curing process of the bonding, each part must be heated to an elevated cure temperature in the furnace; all the materials expand according to their individual coefficient of thermal expansion (CTE). The mismatch in CTE results in the plate being in tension and the patch being in compression after cooling down to ambient conditions. The aluminium is constrained by the stiffer patch that results in the composite area bending on cooling towards the material with the highest coefficient of thermal expansion, i.e. the aluminium. The contraction is related to the expansion of the materials according to the cure temperature range chosen. As seen cure condition A and D have higher distortion levels with the use of higher curing temperatures (120 °C) compared to B and C (80 °C). The variation in distortion levels in relation to the temperature and time of the different cure cycles can be seen in Table 2. We can conclude that the effect of different coefficient thermal expansion on the distortion levels is more pronounced as a change in temperature ΔT increases during curing or in the cooldown. Zhu et al. (2013) also concluded that curing-induced distortion in adhesive bonding of AA6061 and steel was due to differences in CTE of the two materials, as well as curing temperature. They suggested that differences in CTE intrinsically induced the distortion, while the high curing temperature extrinsically generated the distortion. Finally, the cured adhesive then captured the distortion. It was found that the peak curing temperature was the main contributor to the distortion.

3.3 Fatigue Life

As discussed, during the curing process, adhesively bonded composite/metal laminate structures that are held at elevated temperatures for material processing, develop high residual stresses because of the difference in coefficients of thermal expansion (CTE) for different materials. This thermal mismatch can result in delamination or debonding of materials, which facilitates fatigue crack growth at the polymer/metal interface. Further, the influence of different curing temperatures and times investigation was carried out on the fatigue life of the repaired structure. Fatigue crack growth tests were performed on unrepaired and repaired panels.

The panel is symmetrical along its length with respect to the hole and notch. The crack length starts at 4 mm, which is the 3 mm radius of the hole plus 1 mm for the machined notch. The crack length measurement is calibrated at the start of each test and measurement data were automatically logged. The fatigue propagation curves terminate at 44 mm which was chosen and defines fatigue life, in the case of the repaired panels to prevent the crack from growing past the edge of the half-patch at 50 mm. The fatigue crack growth curves for unrepaired and repaired panels are presented in Fig. 8.

Results from fatigue tests to determine the effect of different cure parameters on the repair efficiency and fatigue life of the structure showed that the fatigue life of the unrepaired structure according is decreased compared to the repaired structure. The general trend in Fig. 8 is uneventful with a gradual stabilisation of the crack growth rate. The final order of fatigue life cycles for cure condition is A < B < C < D.

The repairing of the cracked plate using bonded composite patches is effective in slowing down the crack growth rate and thereby enhancing the fatigue life. The panels with the bonded composite repair show an increase in the number of cycles in an average of 70% when compared to the unrepaired panel, which improves the fatigue life considerably.

Author's personal copy

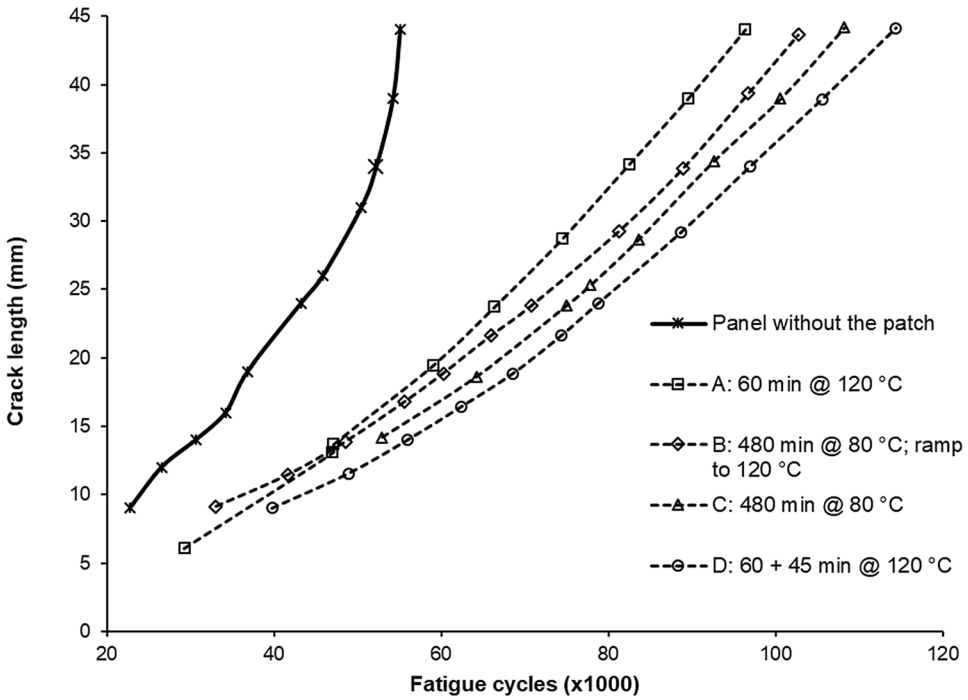


Fig. 8 Effect of curing condition on crack propagation in the repaired panels

The use of a composite patch offers advantages for repaired cracks to increase the service life as shown in Fig. 8.

The increase in the fatigue life performance from unrepaired to repaired is due to the decrease in stress intensity factor due to the boron/epoxy patch bridging the crack. This is confirmed in Fig. 9 representing the reduction in stress intensity factor (SIF) between panels with and without patch. Figure 9 shows that the stress intensity factor increases as the crack length increases both in panels with and without patch, this due to higher stress concentration at the crack tip. The stress intensity factor was determined experimentally during the fatigue test from the data by

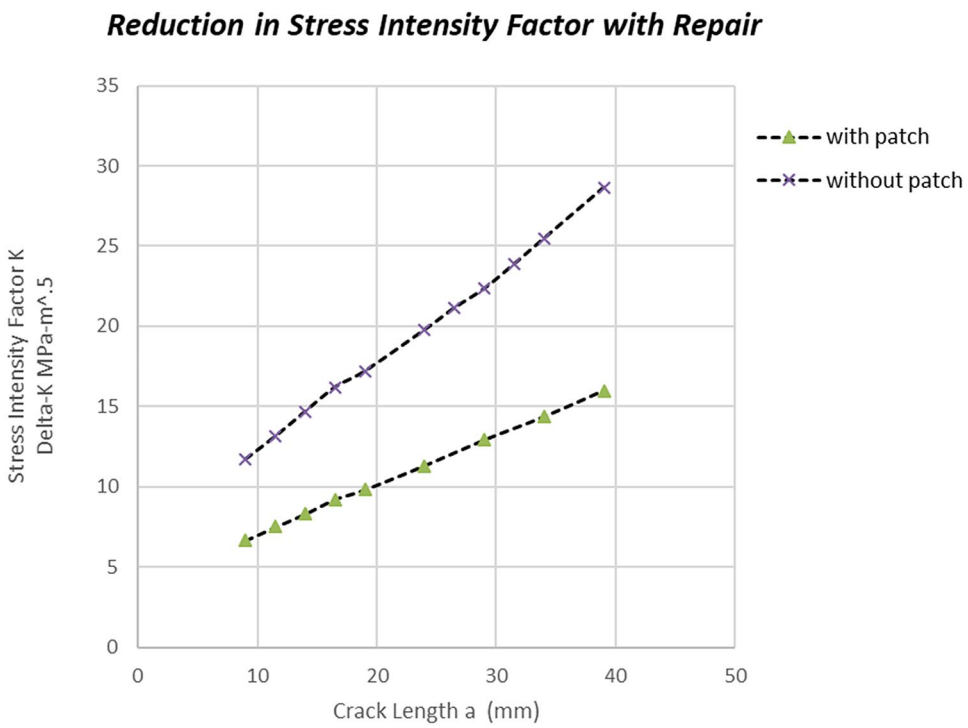


Fig. 9 Reduction in stress intensity factor with repair (Curing condition A)

Author's personal copy

using Eq. (1) as per ASTM E647 (ASTM International 2013):

$$\Delta k = \frac{\Delta P}{B}\sqrt{\frac{\pi\alpha}{2W}\sec\frac{\pi\alpha}{2}} \quad (1)$$

where P is the load in MPa, B and W are the thickness and width of the sample in mm, respectively, a is the crack length; and $\alpha = 2a/W$.

3.4 Fatigue–Distortion Relationship

Figure 10 shows the relationship between plate and patch distortion and fatigue life. It is expected that increasing distortion would reduce fatigue life, presumably due to induced bending stresses that could contribute to the crack growth. It can be seen that fatigue life decreases as the plate distortion increases, that is, for cure conditions A, B, and C. The exception is cure condition D, which has the largest plate distortions, but the longest fatigue life.

The fatigue life follows a straight-line relationship with the plate distortion (for cure conditions A, B and C), and the curve fitting coefficient has an $R^2 = 0.9779$, which indicates a strong correlation. The fatigue life in the case of the patch distortion also follows a linear relationship with an $R^2 = 0.9991$, which indicates even a stronger relationship. Cure condition D does not follow the straight-line relationship in Fig. 10, and the possible reason for its unique behaviour is the use of the elevated curing temperatures for a longer time. This enhanced the adhesion between the metal surface and the composite patch and overall durability.

3.5 Failure Mechanism

Adhesive bonding can be described as a procedure of joining parts with a non-metallic material (adhesive) which go through a hardening reaction causing the parts to bond with surface adherence (adhesion) and internal strength (cohesion). Under overstress loads, the interfaces between different adhering materials can suffer adhesive failure. Ideally, the joining surfaces should be clean and generally require proper preparation of the aluminium surface. It is recommended by the adhesive supplier that the adhesive be cured at 120 °C ± 3 °C. A curing process at 80 °C would probably result in an insufficiently cured adhesive layer. There are also questions related to the interface properties of the adhesive and prepreg tape epoxy. It could be expected that the interface does not reach its optimum properties if the curing process deviates from the recommendation, especially the cure temperature. However, curing time is also the other important factor. The longer time curing of lower temperatures as in curing process (C) and (B) gave longer fatigue life than condition (A). The use of cure condition (A) parameters for an extended curing time as in cure condition (D) also showed the effect of curing time on the fatigue life. In addition, the metal surface treatment could have influenced the effects observed, as the surface preparation of the metal is an important factor influencing the strength of the bonded joint. Different types of adhesives, pre-treatment

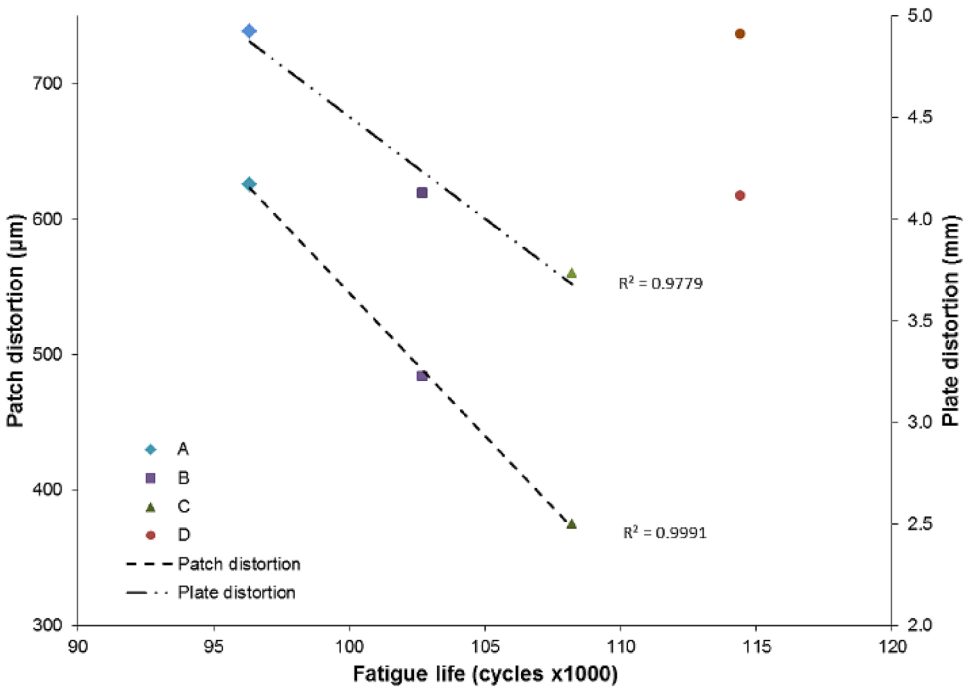


Fig. 10 The relationships between fatigue life and plate distortion

Author's personal copy

methods and processing techniques are available, but there is still a need for further improvements (Duncan et al. 2003; Broughton and Lodeiro 2002). The best solution involves close consideration of multiple criteria and seeking advice from the respective suppliers is recommended. In some cases, extensive experimental testing may be required.

During the fatigue test of the patched plates, the crack growth exceeds the patch edge. The crack continues growing towards the end of the aluminium plate as shown in Fig. 11; in other words, the crack had propagated across the entire width of the specimen without any sign of detachment of the patches from the plate after long time cycling. While kept under continuous cycling loading after the crack passed the edge of the plate, a debonding line between the patch and plate was observed as seen in Fig. 12. The fatigue test was first completed by growing the crack inside the patch zone, and then the panels were subsequently pulled apart. Specimens, namely unpatched and patched with boron composite patch in cure condition (A, B, C and D) were loaded to failure in tension in order to observe the failure mechanism of the damaged repair. For the purpose of comparison, it was done at the same crack length growth (70 mm) in order to analyse the effect of the strength of the assembly on the repair efficiency. Results observed explains that the strength of the assembly is highly increased with the composite patch

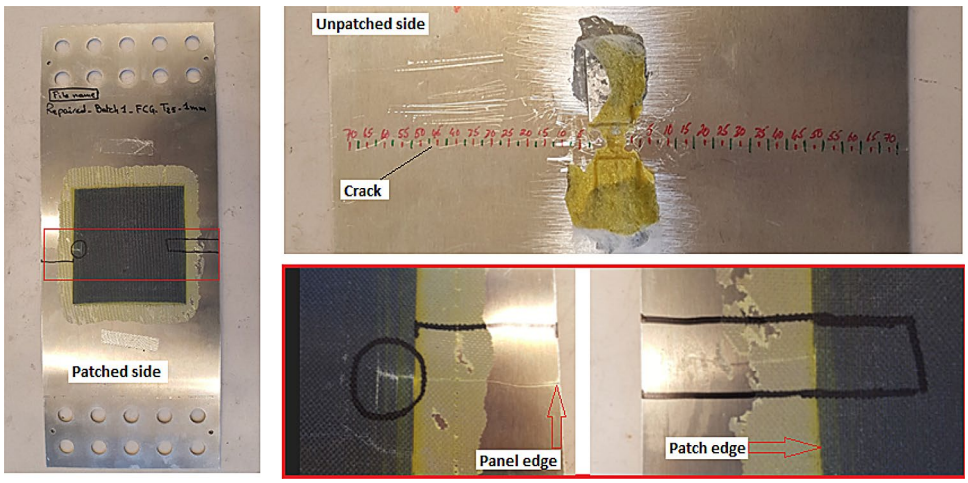


Fig. 11 Crack growth behaviour in the repaired plate

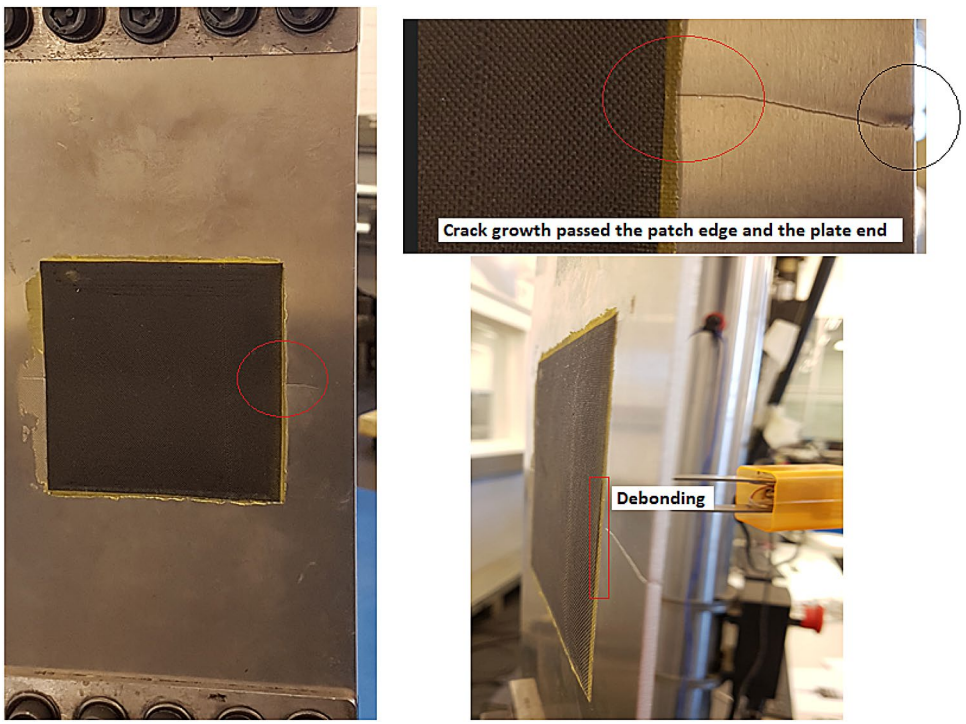


Fig. 12 Effect of patch strength after plate failure

Author's personal copy

presence, which confirms the results in the improvement of the fatigue life with the bonded composite repair.

The observed debonding in Fig. 12 creates unbound interfaces. Such a defect can give rise to preferential sites at the initiation of rupture or detachment. Similarly, when the stress supports are not on the same plane a bending moment appears, thus creating in the joint area and the supports stress normal to the bonding, which is superimposed on the shear stresses. If the rigidity of the supports is low, the latter will flex under the action of bending moment, which increases the normal stresses at the ends of the bond line and risks endangering the bonded area by initiating the separation. Stresses in the adhesive bond layer are increased due to the mismatch between the patch and plate that can lead to possible peel and cleavage stresses. Tension on the repair system creates shear stress in the bond line. The composite fibre breakage in a sliding failure along a plane that is parallel to the loading direction can be seen in Fig. 13.

Figure 13 shows the fractured panels after the tension test. At first, it can be seen that debonding between the aluminium base plate and the adhesive happened in cases for cure condition A, B, and C. The same type of debonding was not evident for cure condition D. Furthermore, it can be seen from the area around the fatigue crack that the yellow adhesive is visible underneath the boron/epoxy layer, meaning that there also was debonding at the adhesive-boron/epoxy interface. The increased fatigue life of cure condition D is a good example that infers a proper interface bond between the adhesive layer and the boron/epoxy layer. That is because of the improved bonding enhanced by the elevated temperature. It has a high degree of distortion, but it also has the best fatigue life (Fig. 8). However, excessive distortion may not be suitable for aero-structure skins due to aerodynamics purposes for in-service conditions. The bending behaviour might also influence the fatigue life of the structure in a cyclic mode.

4 Conclusions and Recommendations

This paper presents an experimental study on the fatigue patching efficiency for notched AA7075-T6 aluminium alloy panels reinforced with single-sided boron/epoxy patches. The novelty of this paper is that the patches are adhesively bonded to the panels with four different chosen cure regimes. To enhance the repair bonding durability and reliability, curing conditions and processes of bonded composite patches repair with structural adhesive were investigated. The findings are summarised as follows:

- The repair of the cracked plate by means of bonded composite patches is effective in decreasing the crack growth rate and thereby enhancing the fatigue life of the structure.

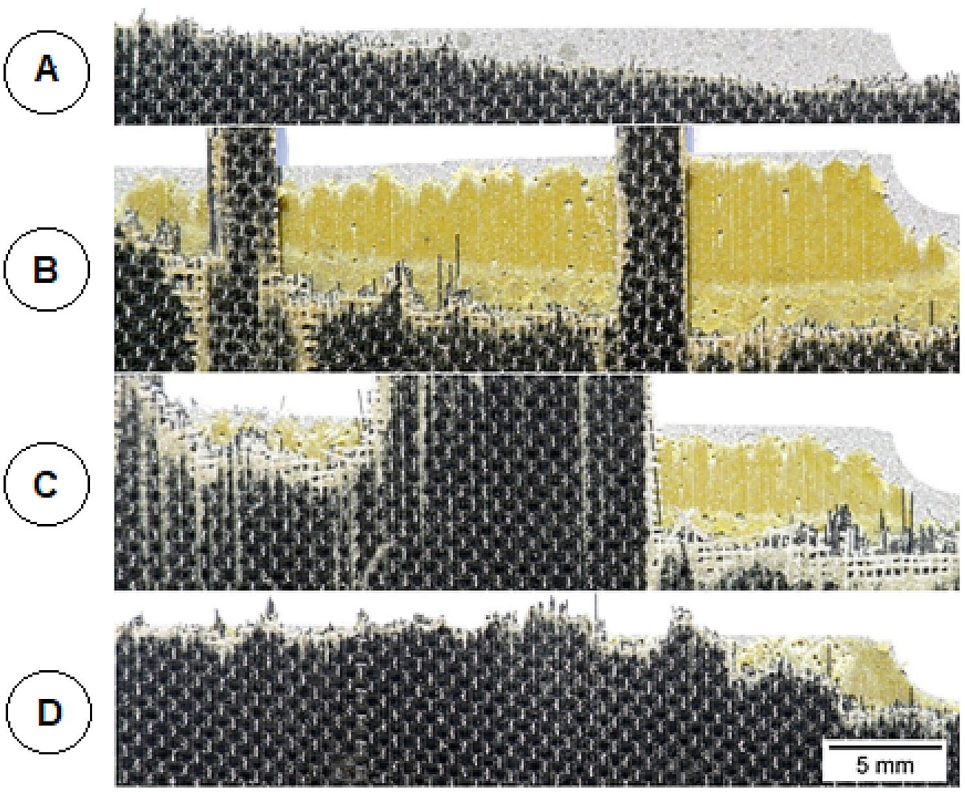


Fig. 13 Fractured samples in shear mode after tension test

Author's personal copy

- The various cure parameters affect the repair efficiency and fatigue life of the structure. The order of fatigue life cycles for cure conditions is A < B < C < D.
- The fatigue life with a composite patch is greater than without a patch.
- Bonded patch repair reduces stresses near the tip of the crack by bridging between the cracked plate and composite patch to slow down or stop the crack growth.
- An average of 70% decrease in the crack growth rate from unpatched panel to the patched one was found
- The increased fatigue life of cure condition D is a good example of the enhanced adhesion between the metal surface and the composite patch due to the use of elevated temperature and duration.
- The curing condition parameters affect the structural distortion and secondary bending influences the behaviour in single side reinforcement/repair.
- The fatigue life decreases as the plate distortion increases, that is, for cure conditions A, B, and C. The exception is cure condition D, which has the largest plate distortions, but the longest fatigue life.
- The optimum combination of temperature and duration is of importance to reduce the distortion.
- The selection of the best cure parameters involves close consideration of multiple criteria.

5 6. Recommendations

Behaviour prediction of the fatigue crack growth in metallic structures is a highly involved and complex process. When these structures are repaired with bonded composite patches, the complexity is (summy text).

- The behaviour prediction of the fatigue crack growth in metallic structures is a highly involved and complex process. When these structures are repaired with bonded composite patches, the complexity is further compounded. Despite this complexity, there is a need to establish a generalised approach, of sufficient accuracy, to model the crack growth rates after repair
- Thermomechanical analysis (TMA) and dynamic mechanical analysis (DMA) are required to understand more in-depth the thermal and mechanical behaviour of the involved material.
- For aircraft damage repair with composites patches and aircraft composites components, it is also important to perform those analyses starting from low temperatures ranges (< 18 °C). That will conform to the in-service aircrafts temperatures and environment to predict the failures before maintenance to detect defects or damage during inspection.
- There are opportunities to optimise the stress distributions within repairs using finite element packages. Validation of these experimental data is required and further development to ensure that they can predict the effect of various parameters accurately. The ability to accurately calculate the magnitude of thermally generated residual stresses would be an advantage for certain repairs.
- The structure's curvature in the repair area would require detailed investigation if applied to fleet aircraft.

Acknowledgements The Council for Scientific and Industrial Research (CSIR) supported this work through funding by South Africa's Department of Science and Innovation (DSI), A grant under the National Exceptional Program (PNE) from the Algerian Ministry of Higher Education and Scientific Research and Directorate General of Scientific Research and Technological Development is also acknowledged. Dr Kevin Slattery and Selina Cobbs at Boeing and Dr Leonard Mac Adams at Cytec/Solvay Group-USA are also acknowledged for assistance with acquiring materials. We express gratitude to Pierre Rossouw for logistical management and technical and practical support; Danie Wilkins, Martin Williams, and Marius Grobler for assistance with sample preparation; Chris McDuling for assisting with mechanical testing; and Dr Heinrich Möller and Dr Ulyate Curle are acknowledged for comments on the manuscript.

References

Adams RD, Wake WC (1984) Structural adhesive joints in engineering. Elsevier, London

Albedah A et al (2018) Effects of adhesive disbond and thermal residual stresses on the fatigue life of cracked 2024-T3 aluminum panels repaired with a composite patch. Int J Adhes Adhes 87:22–30

Alderliesten RC (2009) Damage tolerance of bonded aircraft structures. Int J Fatigue 31:1024–1030

ASTM International (2013) ASTM E647 − 13$^{\epsilon 1}$, Standard test method for measurement of fatigue crack growth rates

Baker AA, Jones R (1988) Bonded Repair of Aircraft Structures. Springer, Berlin

Baker A, Rajic N, Davis C (2009) Towards a practical structural health monitoring technology for patched cracks in aircraft structure. Compos Part A Appl Sci Manuf 40:1340–1352

Benyahia F, Bouiadjra BB (2014) Bonded composite repairs of aluminum alloy 2024T3 cracked plates. In: Advances in aerospace technology, vol 1

Benyahia F, Albedah A, Bouiadjra BAB (2014) Elliptical and circular bonded composite repair under mechanical and thermal loading in aircraft structures. Mater Res 17:1219–1225

Benyahia F et al (2015) Experimental and numerical analysis of bonded composite patch repair in aluminium alloy 7075 T6. Mater Des 73:67–73

Biscaia HC, Chastre C, Cruz D, Viegas A (2017) Prediction of the interfacial performance of CFRP laminates and old timber bonded joints with different strengthening techniques. Compos Part B Eng 108:1–17

Boron 5521 Prepreg Tape (2013) Specialty Materials; INC; Technical data sheet

Bouanani FM, Benyahia F, Albedah A, Aid A, Bouiadjra BB, Belhouari M, Achour T (2013) Analysis of the adhesive failure in the bonded composite repair of aircraft structures using modified damage zone theory. Mater Des 50:433–439

Author's personal copy

Brighenti R, Carpinteri A, Vantadori S (2006) A genetic algorithm applied to the optimisation of patch repair for cracked plates. Methods Appl Mech Eng 196:466–475

Broughton WR, Lodeiro MJ (2002) Review of surface characterisation techniques for adhesive bonding. NPL Report. MATCv (A) 66

Bureau of Accident Investigation (1988) Aircraft accident report: Aloha, Airlines Flight 243; Boeing 737-200; N73711; Near Maui; Hawaii; April 28; 1988. Report No. NTSB/AAR-89/o3; National Transportation and Safety Office 14 June 1989; Washington

Chakherlou TN, Vogwell J (2003) The effect of cold expansion on improving the fatigue life of fastener holes. Eng Fail Anal 10(1):13–24

Chung HK (2003) YangWH, Mixed-mode fatigue crack growth in aluminium plates with composite patches. Int J Fatigue 25:325–333

Cytec Engineered Materials (2011) FM73 Epoxy film adhesive; Technical data sheet

Datla NV, Papini M, Ulicny J, Carlson B, Spelt JK (2011) The effects of test temperature and humidity on the mixed-mode fatigue behaviour of a toughened adhesive aluminium joint. Eng Fracture Mech 78:1125–1139

Duncan BC et al (2003) A review of adhesive bonding assembly processes and measurement methods.NPL Report. MATCv (A) 135

Emami Mehr M, Aghamohammadi H, Hosseini Abbandanak SN, Aghamirzadeh GhR, Eslami-Farsani R, Siadati SMH (2019) Effects of applying a combination of surface treatments on the mechanical behaviour of basalt fibre metal laminates. Int J Adhes Adhes 92:133–141

Günther G, Maier A (2010) Composite repair for metallic aircraft structures. In: Proceedings of 27th international congress of the aeronautical sciences ICAS2010; 19–24 September 2010, France

Gwo-Chung T, Shyan BS (2004) Fatigue analysis of cracked thick aluminium plate bonded with composite patches. Compos Struct 64:79–90

Heatcon Abaris Training (2008) Advanced composite structures: fabrication and damage repair phase I, II, III; Training materials, Cwmbram; UK

Heshmati M, Haghani R, Al-Emrani M (2017) Durability of bonded FRP-to-steel joints: effects of moisture, de-icing salt solution, temperature and FRP type. Compos Part B Eng 119:153–167

Hosseini-Toudeshky H, Mohammadi B, Bakhshandeh S (2007) Mixed-mode fatigue crack growth of thin aluminium panels with single-side repair using experimental and numerical methods. Fatigue Fract Eng MaterStruct 30:629–639

Khashaba UA, Aljinaidi AA, Hamed MA (2017) Fatigue and reliability analysis of nano modified scarf adhesive joints in carbon fibre composites. Compos Part B Eng 20:103–117

Kim KS, Hahn HT (1989) Residual stress development during processing of graphite/epoxy composites. Compos Sci Technol 36(2):121–132

Kinloch AJ (1987) Adhesion and adhesives. Chapman and Hall, London

Klemczyk C, Belason EB (1994) Analysis of maximum stresses associated with a boron/epoxy doubler bonded to aluminium. Science 6:11–71

Mall S, Conley DS (2009) Modelling and validation of composite patch repair to cracked thick and thin metallic panels. Compos Part A Appl Sci Manuf 40:1331–1339

Mall S, Schubbe J (2009) Bonded composite patch geometry effects on fatigue crack growth in thin and thick aluminium panels. Struct Longev 2:25–48

Montgomery TS, MacKnight WJ (2005) Introduction to Polymer Viscoelasticity ISBN 978-0-471-74045-2, 3rd edn. Wiley, Hoboken, p 336

Niedernhuber M, Holtmannspötter J, Ehrlich I (2016) Fibre-oriented repair geometries for composite materials. Compos Part B Eng 94:327–337

Olivier PA, Sawi I (2010) Designing curing conditions to analyse the influence of process-induced stresses upon some mechanical properties of carbon/epoxy laminates at constant Tg and degree of cure. Int J Mater Form 3(2):1373–1389

Ouinas D, Bouiadjra BB, Serier B, Said BM (2007) The effects of disbands on the stress intensity factor of aluminium panels repaired using composite materials. Compos Struct 78:278–284

Pastor ML, Balandraud X, Grédiac M, Robert JL (2008) The fatigue response of aluminium specimens reinforced with carbon epoxy patches. Compos Struct 83(3):237–246

Riccio A, Di Felice G, Scaramuzzino F, Sellitto A (2014) A practical tool for the preliminary design of bonded composite repairs. Appl Compos Mater 21:495–509

Sohail MA, Khan B et al (2017) Fatigue crack propagation in aluminium plates with composite patch including plasticity effect. J Aerosp Eng 232:2122–2131

Taylor P, Antonucci V et al (2005) Process-induced residual stresses in polymer-based composites. Polym News 30(8):238–247

Timoshenko S (1925) Analysis of bi-metal thermostats. J Opt Soc Am 11(3):233–255

Wang QY, Sriraman MR, Kawagoishi N, Chen Q (2006) Fatigue crack growth of bonded composite repairs in the gigacycle regime. Int J Fatigue 28:1197–1201

Weitsman Y (1980) Optimal cool-down in linear viscoelasticity. J Appl Mech 47(1):35–39

White SR, Hahn HT (1992) Process modeling of composite materials: residual stress development during cure. Part I. Model formulation. J Compos Mater 26:2402–2422

Wisnom M, Gigliotti M, Ersoy N, Campbell M, Potter K (2006) Mechanisms generating residual stresses and distortion during manufacture of polymer–matrix composite structures. Compos Part A Appl Sci Manuf 37(4):522–529

Zhu XB Yong Bing Li, Guan Long C, Pei-Chung W (2013) Curing-induced distortion mechanism in adhesive bonding of aluminum AA6061-T6 and steels. J Manuf Sci Eng 135(5):1–11

Zouambi L, Khodja M, Oudad W, Fekirini H, Moller H, Bouiadjra BB (2019) J-integral evaluation of repaired cracks in AA7075-T6 structures subjected to uniaxial tensile stresses. Polym Test 77:105874