Basic Research Grants Scheme 2002

A Study of Damage Initiation and Growth in Composite Bolted Joints

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Test Specifications for Material Tests and Open-Hole Tests

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Section 1: Introduction

The project “A Study of Damage Initiation and Growth in Composite Bolted Joints” is funded under the Basic Research Grants Scheme 2002, jointly administered by Enterprise Ireland and the Irish Research Council for Science, Engineering and Technology. It runs from October 2002 to September 2005.

The goal of the project is to develop computational models for prediction of the initiation and growth of damage in composite bolted joints. Two approaches will be investigated. The first will be based on a stiffness reduction scheme. The second will be based on continuum damage mechanics. The two approaches will be compared against experimental data generated within the project and also from a previous EU research project, and critically assessed.

This report (D3.1a) outlines the proposed specification for the first part of the experimental test series, material characterisation tests and open-hole tests, to be carried out during the course of the project. The proposed specification for the second part of the experimental test series, pin-bearing tests, will be outlined in a future report (D3.1b). This specification has been developed following a review of previous work carried out on the subject of damage initiation and growth in fibre reinforced composite material. It should be noted that the specification will undergo continuous review and may be modified as results are evaluated during the test programme. Section 2 specifies the tests to be performed, Section 3 describes details of specimen manufacture and test. Appendix A describes previous work on the subject of damage initiation and growth in fibre reinforced composite materials.

Section 2: Test Matrices

2.1 Test Series 1: Material Characterisation Tests

This series will involve coupons of material loaded quasi-statically in tension or compression, with varying lay-up, instrumentation and load level. As two material systems are to be examined in this project, Test Series 1 has been split into two sub-series, 1a and 1b. Each sub series will involve similar tests, but different material systems. The objective is to determine key material properties needed to implement finite element models being investigated as part of the project. The material properties to be determined are:

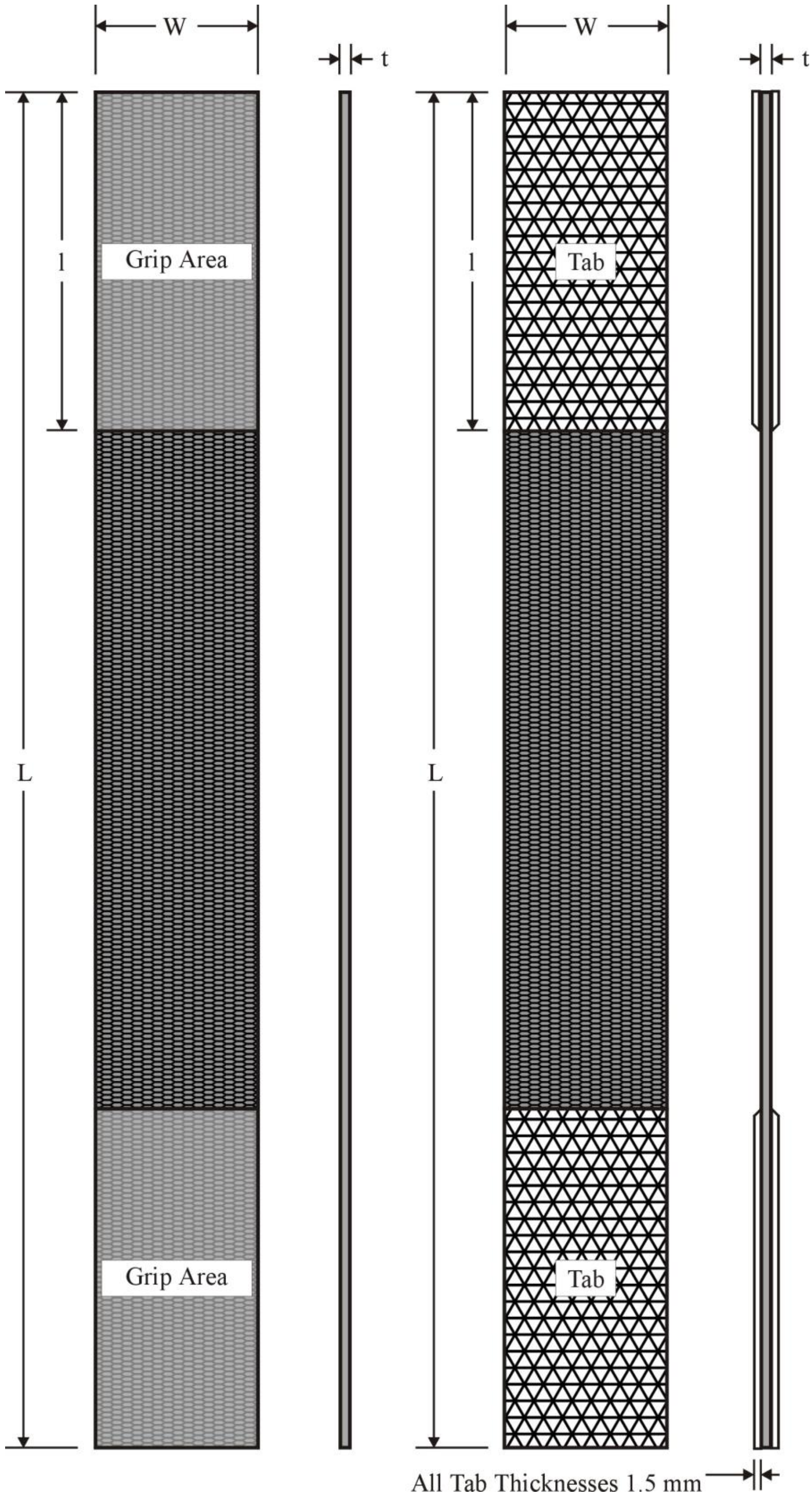
1. Young’s Modulus in the 1 and 2 material coordinate directions (E_1, E_2)
2. Shear Modulus in the 1-2 material coordinate plane (G_{12})
3. Poisson’s Ratio in the 1 and 2 material coordinate directions (v_{12}, v_{21})
4. Ultimate Strength in the 1, 2 and 1-2 material coordinate directions (S_{11}, S_{22}, S_{12})
5. Ultimate Strain in the 1, 2 and 1-2 material coordinate directions (ε_{11}, ε_{22}, ε_{12})
6. Compressive values for E_1, v_{12} and S_{11}
7. Damage Parameters

All tests involve rectangular coupons, with variable dimensions and test procedure depending on the material property to be determined. The procedure in the American Society of Testing and Materials (ASTM) standard D3039/D3039M – 00, “Standard Test Method for Tensile Properties of Polymer Matrix Composite Materials” [1] will be followed when determining values for E_1, E_2, v_{12}, v_{21}, S_{11}, S_{22}, ε_{11} and ε_{22}. The procedure in the ASTM standard D3518/D3518 – 94, “Standard Test Method for In-Plane Shear Response of Polymer Matrix

Composite Materials by Tensile Test of a ± 45° Laminate" [2] will be followed when determining G_{12}, S_{12}, ε_{12} and the damage parameters. The procedure in the Composites Research Advisory Group (CRAG) Standard 401, "Method of Test for Longitudinal Compression strength and Modulus of Multi-Directional Fibre Reinforced Plastics" [3] will be followed when determining compressive values for E_1, v_{12} and S_{11}.

Specimen Geometry: The basic geometries of the coupon test specimens are shown in Fig. 1 and the dimensions of each coupon specimen are given in Tables 1 and 2.

Material: Two material systems will be examined in this test series, each in its own sub-series. The material to be examined in Test Series 1a is Hexcel Materials Ltd. 6376C-HTA(12K)-5.5-29.5% carbon fibre reinforced plastic (CFRP). The material to be examined in Test Series 1b is Cytec Engineered Materials Ltd. FM94-27%-S2-187-460 glass fibre reinforced plastic (GFRP). Both of these material systems are mainly used in aerospace structural applications. The CFRP material was used by the University of Limerick for bolted joints in the EU Project BOJCAS [4] (basic materials tests were not performed), so this material is used again here to allow comparisons with results from BOJCAS. The GFRP material is of particular interest as it is the fibre reinforcing material used in GLARE. GLARE is a material containing layers of aluminium alloy and layers of GFRP and is to be used as the upper fuselage skin on the new Airbus widebody jet, the A380, as well as in other future aerospace applications. Details on specimen lay-ups are given in Tables 1 and 2.



(a) **(b)**

Notes: **1.** l – grip length or tab length, this dimension is dependent on the standard to which the coupon is manufactured to.
2. Dimensions for the specimens are given in Tables 1 and 2

Figure 1 Material characterisation specimen geometry (a) without tabs, (b) with tabs

			Specimen Geometry						Test Loading
Code	**Lay-up**	**Loading**	**L (total)**	**w**	**t**	**Tabs**	**Instrumentation**	**Primary Output**	**Failure**
MT_C_0_T#	$(0)_8$	Tension	250	15	1.04	Yes	Strain Gauged/ Extensometers	E_1, ν_{12}, S_{11}, ε_{11}	5
MT_C_90_T#	$(90)_{16}$	Tension	175	25	2.08	Yes	Strain Gauged/ Extensometers	E_2, ν_{21}, S_{22}, ε_{22}	5
MT_C_PM45_T#	$(45/-45)_{8s}$	Tension	250	25	2.08	No	Strain Gauged/ Extensometers	G_{12}, S_{12}	5
MT_C_0_C#	$(0)_{16}$	Compression	250	25	2.08	No	Strain Gauged/ Extensometers	E_1, S_{11}, ε_{11} (compressive)	5
								Total	**20**

- All test specimens are manufactured from 6376C-HTA(12K)-5.5-29.5% CFRP Prepreg
- All specimen dimensions are given in millimetres
- Symbols: # - Test Number; *E* – Young's Modulus; *G* – Shear Modulus; *ν* – Poisson's Ratio; *S* – Ultimate Strength; ε - Ultimate Strain; L – Length; w – Width; t - thickness
- Subscripts: $(\_)_\_$ - Number of plies in Laminate; *1, 2, 3* – Material Principal Axes; $(\_)_{\_s}$ – Laminate is symmetric

Table 1 Test Matrix for Test Series 1a

			Specimen Geometry						Test Loading		
Code	**Lay-up**	**Loading**	**L (total)**	**w**	**t**	**Tabs**	**Instrumentation**	**Primary Output**	**Failure**	**Linear Elastic Region**	**Total**
MT_G_0_T#	$(0)_8$	Tension	250	15	1	Yes	Strain Gauged/ Extensometers	E_1, ν_{12}, S_{11}, ε_{11}	5	0	5
MT_G_90_T#	$(90)_{16}$	Tension	175	25	2	Yes	Strain Gauged/ Extensometers	E_2, ν_{21}, S_{22}, ε_{22}	5	0	5
MT_G_PM45_T#	$(45/-45)_{4s}$	Tension	250	25	2	No	Strain Gauged/ Extensometers	G_{12}, S_{12}, Damage Parameters	3	2 cyclic (5 cycles)	5
MT_G_P45_T#	$(45)_8$	Tension	250	25	1	No	Strain Gauged/ Extensometers	Damage Parameters	0	2 cyclic (5 cycles)	2
MT_G_PM67_T#	$(67.5/-67.5)_{4s}$	Tension	250	25	2	No	Strain Gauged/ Extensometers	Damage Parameters	3	2 cyclic (5 cycles)	5
MT_G_0_C#	$(0)_{16}$	Compression	250	25	2	No	Strain Gauged/ Extensometers	E_1, S_{11}, ε_{11} (compressive)	5	0	5
								Total	**21**	**6**	**27**

- All test specimens are manufactured from FM94-27%-S2-187-460 GFRP Prepreg
- All specimen dimensions are given in millimetres
- Symbols: # - Test Number; *E* – Young's Modulus; *G* – Shear Modulus; *v* – Poisson's Ratio; *S* – Ultimate Strength; ε - Ultimate Strain; L – Length; w – Width; t - thickness
- Subscripts: $(\_)_{\_}$ - Number of plies in Laminate; *1, 2, 3* – Material Principal Axes; $(\_)_{\_s}$ – Laminate is symmetric

Table 2 Test Matrix for Test Series 1b

2.2 Test Series 2: Open Hole Tests

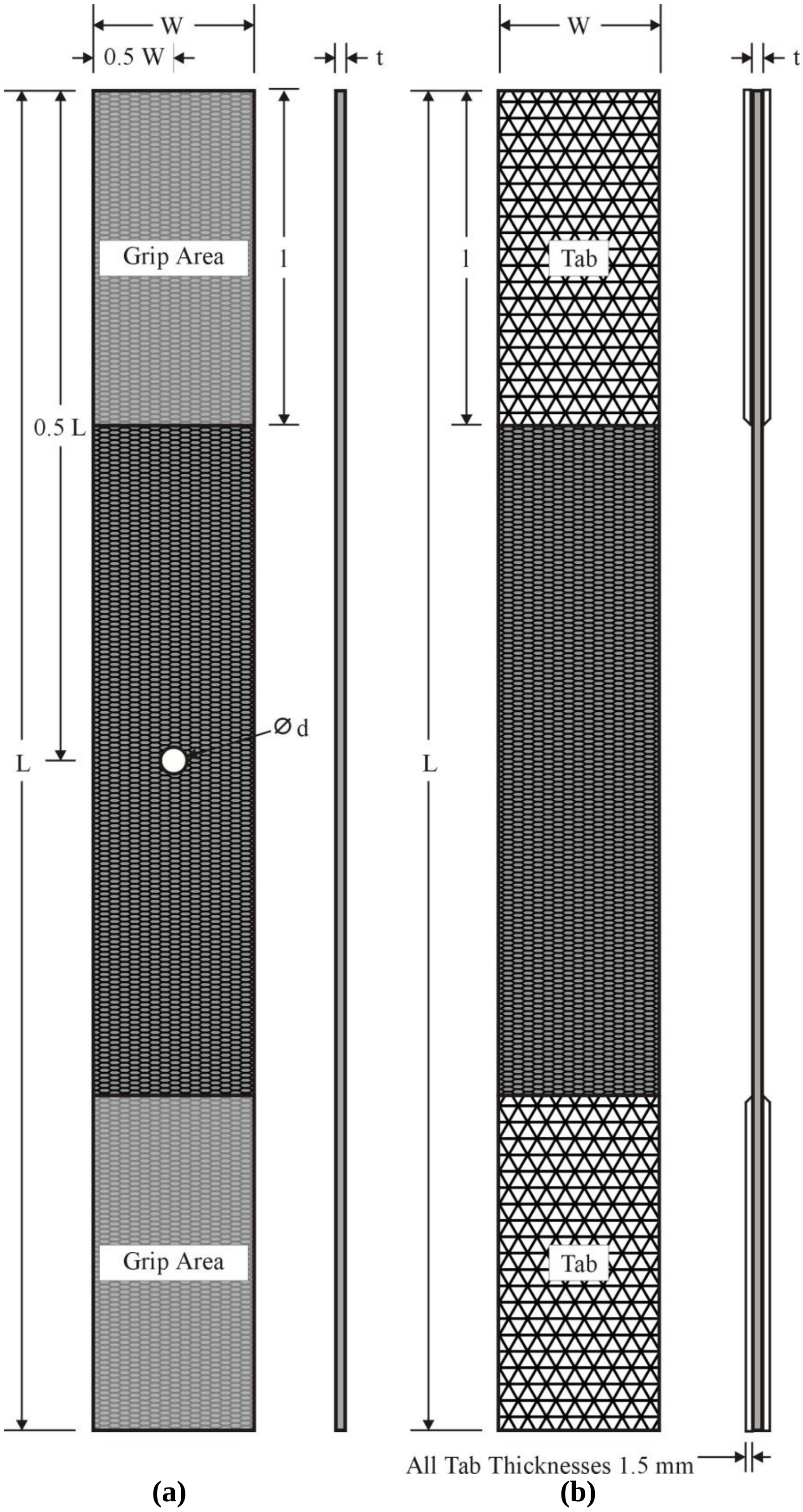
This series will involve open hole, filled hole and unnotched coupon specimens loaded quasi-statically in tension, with varying lay-up, instrumentation and load level. As with Test Series 1 two material systems are to be examined so the test series has been split into two sub-series, 2a and 2b. Each sub-series will involve similar tests, but different material systems. The objective is to study:

1. The load-deflection characteristics of the coupon specimen
2. The initiation and progression of damage at varying load levels
3. The strains at various locations in the coupon
4. The effects of varying lay-up
5. The difference between the carbon and glass reinforced material systems
6. The effects of filling the hole with a bolt

All test specimens are rectangular coupons. The test procedure for the open hole specimens will follow (with a few exceptions documented below) the procedure in the ASTM standard D5766/D5766M – 02, "Standard Test Method for Open Hole Tensile Strength of Polymer Matrix Composite Laminates" [5]. Unnotched control specimens will be tested in accordance with ASTM Standard D3039 [1] and filled hole tests will be carried out in accordance with the procedure in the ASTM standard D6742/D6742 – 01, "Filled Hole Tension and Compression Testing of Polymer Matrix Composite Laminates" [6].
There are many notched specimen configurations that could be investigated for this project (e.g. through slits rather than circular holes). The circular hole configuration was chosen because, firstly, it is well standardised, not only by ASTM but also by the aircraft manufacturer Airbus in its standard AITM 1.0007-3F [7]. Secondly, advanced fibre reinforced polymer matrix composite material systems, of the type being studied in this project, are used most by the aerospace industry and circular holes are the most commonly found notches in aerospace structures, so the work carried out in this project will be relevant to the primary intended application area.

Specimen Geometry: The basic geometries of the open hole test and unnotched control test specimens are shown in Fig. 2. In the ASTM standards D5766 [5] and D3039 [1] a baseline geometry is given for each specimen, but variations are allowed, if fully documented. For the open hole specimens, a baseline hole diameter of 6mm is given in the ASTM standard [5]. In Test Series 2a three hole diameters are used, 3mm, 6mm and 8mm, but all ratios (w/d, d/t) are maintained within the limits allowed by the standard. Dimensions of each coupon specimen are given in Tables 3 and 4.

Material: As with Test Series 1, two material systems will be examined in this test series. The material to be examined in test series 2a is Hexcel Materials Ltd. 6376C-HTA(12K)-5.5-29.5% carbon fibre reinforced plastic (CFRP). The material to be examined in test series 2b is Cytec Engineered Materials Ltd. FM94-27%-S2-187-460 glass fibre reinforced plastic (GFRP).



Notes: **1.** l – grip length or tab length, this dimension is 75 mm for all Test Series 2 specimens.
2. Dimensions for the specimens are given in Tables 3 and 4

Figure 2 Test Series 2 Specimen Geometry (a) Open hole test specimen, (b) Unnotched control test specimen

			Specimen Geometry									Test Loading		
Code	**Layup**	**Loading**	**L (total)**	**w**	**t**	**d**	**w/d**	**d/t**	**Tabs**	**Instrumentation**	**Primary Output**	**Failure**	**% of Failure Load**	**Total**
UN_C_QI_#	QI	Tension	300	36	2.08	-	-	-	Yes	Extensometers	S_{UT}	5	0	5
OHT_C_QI_#	QI	Tension	300	36	2.08	6	6	2.9	No	Extensometers	S_{OHT}	5	3	8
UN_C_ZD_#	ZD_1	Tension	300	36	2.6	-	-	-	Yes	Extensometers	S_{UT}	5	0	5
OHT_C_ZD_#	ZD_1	Tension	300	36	2.6	6	6	2.3	No	Extensometers	S_{OHT}	5	3	8
OHT_C_ZD_D3#	ZD_1	Tension	300	18	2.6	3	6	1.2	No	Extensometers	S_{OHT}	3	3	6
OHT_C_ZD_D8#	ZD_1	Tension (C1 hole)	300	48	2.6	8	6	3.1	No	Extensometers	S_{OHT}	3	3	6
FHT_C_ZD_D8#	ZD_1	Tension with **bolt** in C1 hole	300	48	2.6	8	6	3.1	No	Extensometers	S_{FHT}	3	3	6
OHT_C_ZD_SS2#	ZD_2	Tension	300	36	2.6	6	6	2.3	No	Extensometers	S_{OHT}	3	3	6
UN_C_CP_#	CP	Tension	300	36	2.08	-	-	-	Yes	Extensometers	S_{UT}	5	0	5
OHT_C_CP_#	CP	Tension	300	36	2.08	6	6	2.9	No	Extensometers	S_{OHT}	5	3	8
											Total	**42**	**21**	**63**

- All test specimens are manufactured from HTA/6376 CFRP Prepreg
- All specimen dimensions are given in millimetres
- Layups*: QI – $(45/0/-45/90)_{2s}$ ZD_1 - $(45/0/-45/90/0/0/45/0/-45/0)_s$ ZD_2 - $(45/0/0/-45/90/0/45/0/-45/0)_s$
  CP - $(90/0)_{4s}$ ** - See Table 1 for layup notation definitions*
- Symbols: # - Test Number L – Length w – Width t – Thickness
  d – Hole Diameter *S* – Ultimate Strength C1 – 0 μm Nominal Diameter Bolt-Hole Clearance
- Subscripts: UT – Unnotched Tension OHT – Open Hole Tension FHT – Filled Hole Tension

Table 3 Test Matrix for Test Series 2a

			Specimen Geometry									Test Loading		
Code	**Layup**	**Loading**	**L (total)**	**w**	**t**	**d**	**w/d**	**d/t**	**Tabs**	**Instrumentation**	**Primary Output**	**Failure**	**% of Failure load**	**Total**
UN_G_QI_#	QI	Tension	300	36	2	-	-	-	Yes	Extensometers	S_{UT}	5	0	5
OHT_G_QI_#	QI	Tension	300	36	2	6	6	3	No	Extensometers	S_{OHT}	5	3	8
UN_G_CP1_#	CP_1	Tension	300	36	2	-	-	-	Yes	Extensometers	S_{UT}	5	0	5
OHT_G_CP1_#	CP_1	Tension	300	36	2	6	6	3	No	Extensometers	S_{OHT}	5	3	8
OHT_G_CP2_#	CP_2	Tension	300	36	0.5	6	6	12	No	Extensometers	S_{OHT}	3	3	6
OHT_G_CP3_#	CP_3	Tension	300	36	1	6	6	6	No	Extensometers	S_{OHT}	3	3	6
OHT_G_CP4_#	CP_4	Tension	300	36	1	6	6	6	No	Extensometers	S_{OHT}	3	3	6
											Total	**29**	**15**	**44**

- All test specimens are manufactured from FM94-27%-S2-187-460 GFRP Prepreg
- All specimen dimensions are given in millimetres
- Layups: QI – $(45/0/-45/90)_{2s}$ CP_1 - $(90/0)_{4s}$ CP_2 – $(90/0)_s$ CP_3 – $(90/0)_{2s}$ CP_4 – $(90_2/0_2)_s$ ** - See Table 1 for layup notation definitions*
- Symbols: # - Test Number L – Length w – Width t – Thickness d – Hole Diameter *S* – Ultimate Strength
- Subscripts: UT – Unnotched Tension OHT – Open Hole Tension FHT – Filled Hole Tension

Table 4 Test Matrix for Test Series 2b

Section 3: Specimen Manufacture and Test

This section describes procedures that will be involved in the specimen manufacture and test. These procedures have been developed from previous work carried out at the University of Limerick on composite test specimen manufacture and test.

3.1 Manufacture of Flat Panels

The procedure for manufacturing flat panels from 6376C-HTA(12K)-5.5-29.5% and FM94-S2-27-140-360 composite prepregs has been developed by consulting ASTM standard D 5687/D 5687M – 95, "Standard Guide for the Preparation of Flat Composite Panels with Processing Guidelines for Specimen Preparation" [8], as well as communications with the prepreg manufacturers, Hexcel (U.K.) for the 6376C-HTA(12K)-5.5-29.5% and Cytec Engineered Materials Ltd. for the FM94-S2-27-140-360, and the consumables' supplier, Aerovac Systems Ltd.

Ply cutting and lay-up will be carried out in the usual manner in a dedicated 'clean room', using templates to produce consistently sized and oriented plies. For the ±45° plies, joints within plies will be of the butt joint type and will be staggered so as to vary the in-plane location throughout the thickness of the laminate. Debulking will be employed after every 10 plies have been laid-up.

The stack will be laid-up according to Fig. 3 and cured in an autoclave. Temperature and pressure profiles will be determined from prepreg supplier recommendations. A steel dam of height 6 mm and width 25 mm will be placed around the unconsolidated composite. This has proven successful during previous work in maintaining dimensional accuracy of the final panel by preventing excessive transverse flow of the fibres (and associated fibre washing) near the edges of the panel. The peel ply adjacent to the surface plies will be a synthetic fabric, but not coated with any release solutions. This ensures it will not contaminate the composite part, which makes the cleaning process for subsequent strain gauging considerably easier. The release plies will be 'solid', thus preventing excessive resin loss in the net resin prepreg system to be used. The base plate and caul plate have been carefully machined such that the surfaces are parallel and of a low surface roughness.

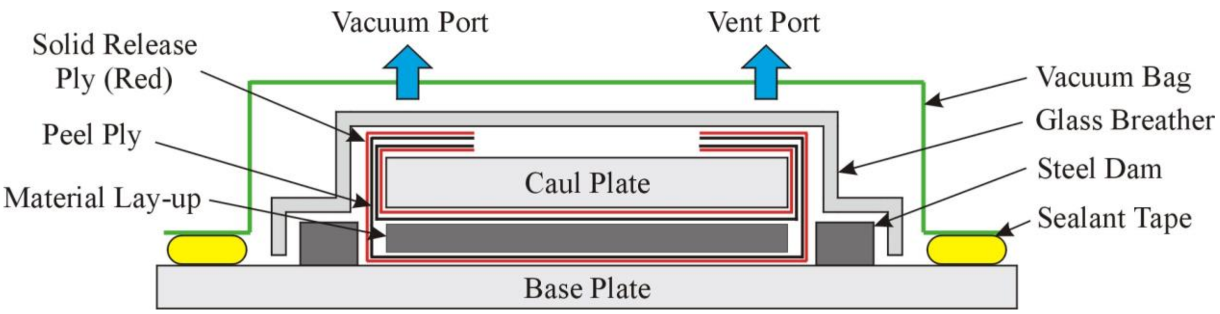


Figure 3 Schematic of Lay-up for Consolidation of the Test Material Prepreg.

3.2 NDT

Upon removal of the consolidated composite panel from the autoclave, the panel will be visually inspected for resin rich areas, excessive resin bleed etc. Additionally a C-scan will be performed using an Ultrasonic Sciences Ltd. scanning machine.

3.3 Cutting of Specimens

Specimen cutting will be carried out on a dedicated composite cutting machine with diamond coated cutting wheel. Two datum backstops are provided for positioning the panel. One is fixed, normal to the direction of cut, the other is movable and determines the width of the cut. The backstops have been calibrated in previous test series, by using a specially made template of high dimensional accuracy. The width of a number of cut specimens was measured at several points along their length. In all cases, variations were within the tolerance allowed in the ASTM standard [1].

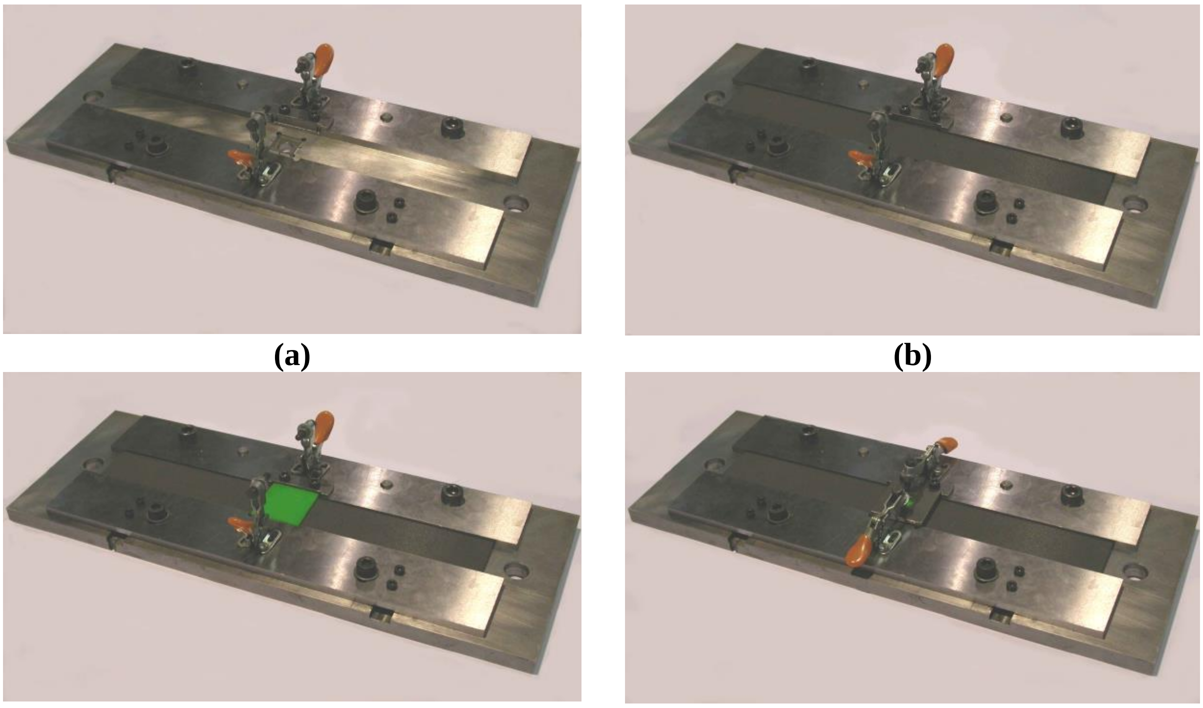
3.4 Drilling

In the preparation of the open hole tension (OHT) specimens, drilling is a crucial part of the process, since in order to examine the initiation and growth of damage around the hole in the specimen, high quality holes must be generated as free as possible from damage associated with solid-tool drilling of composite materials such as chip-out, surface delamination, internal delamination and fibre/resin pull-out. It is recommended by the ASTM standard [5] that holes should be drilled undersized and reamed to their final dimension. Solid carbide tooling has been obtained and is illustrated in Fig. 4. It consists of undersized drills of diameter 7.53mm, 5.7mm and 2.8mm together with reamers of the same nominal size as the three hole sizes that will be examined in the test series. All reamers are within the tolerance allowed by the ASTM standard [5]. Three pieces of each tool have been obtained to allow for tool wear.

In order to drill as straight and perpendicular a hole as possible through the centre of the specimen, a special jig has been manufactured to hold the specimen in place during drilling (Fig. 5). The jig has been manufactured from ground flat stock steel, and after assembly, was re-ground to produce highly accurate final dimensions. As shown in Fig. 5, the steps involved in mounting the specimen are: (a) insert small square perspex backing piece, (b) insert specimen, located accurately with pins, (c) add perspex piece on top and (d) add steel plate with hole and clamp.

Drilling trials with variable speeds and feeds have been performed. In all trials a perspex backing piece has been used to prevent breakout on the exit side. However, there were also some problems with small degrees of chip-out on the *entry* side. These were eliminated by the use of the perspex piece on the entry side of the joint as well. Clearly this is not a practical production method, but it has produced high quality holes for research purposes, as evidenced by x-rays and microscopy of the holes.

Figure 4 Carbide Tooling for Hole Drilling, Left to Right: Spiral Reamer, Drill, Straight Reamer, Tin Coated Carbide Drill.



(a) **(b)**

(c) **(d)**

Figure 5 Drilling Jig Showing Steps in Mounting of Specimens

3.5 Testing

Testing will be performed on either a 100 kN or a 300 kN Zwick/Roell universal straining frame. Grips on both machines are hydraulic. Two Epsilon extensometers will be used to record the strain across the open hole area of the specimens. Strain gauges and photoelastic coatings will be used in some tests to record the strain distribution around the hole and these strains will be recorded, together with the instantaneous displacement and load by a dedicated National Instruments data acquisition system. Both still and video pictures will be taken in some tests.

Acknowledgements

The authors would like to gratefully acknowledge Enterprise Ireland and the Irish Research Council for Science, Engineering and Technology for funding this work.

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Appendix A

A.1 Review of Previous Work on Notched Composite Specimens

This appendix briefly surveys the work of other researchers on the subject of notched composite specimens. It is by no means an exhaustive treatment, but provides a list of references, which may be of interest. Further references can be obtained from these papers.

Awerbuch & Madhukar [9] carried out a comprehensive review of existing theoretical fracture models for notched strength of fibre reinforced composite laminates loaded quasi-statically in tension. Both circular holes and elongated notches (slits) were considered. The theoretical predictions obtained from the models were compared with over 2800 experimental notched strength data sets for seventy different laminate configurations, collected from the open literature. Notched strength data for three different material systems, (graphite/epoxy, boron/aluminium, graphite/polyamide), were analysed. Theoretical failure criteria reviewed included Linear Elastic Fracture Mechanics (LEFM), Modified LEFM, Point and Average Stress Criteria, Modified Point Stress Criterion and Strain Criterion. The authors found that there was very good agreement between all the fracture models reviewed and the experimental notched strength data, provided that the fracture model parameters were properly determined. However, as all the models reviewed were semi-empirical, they can only be applied in cases where certain notched strength data parameters are known, as well as the unnotched strength of the laminate and the lamina elastic properties. These parameters are strongly dependent on the laminate configuration and material system as well as a variety of intrinsic and extrinsic variables, consequently they must be determined experimentally for each material system, laminate configuration etc.

Due to their semi-empirical nature, all the fracture models reviewed ignore the actual characteristics of the notch tip damage zone, but rather simulate the damage zone by some 'effective' notch tip damage zone and assume that it grows self-similarly, giving no indication of how the damage actually initiates or grows.

From their review of the experimental results presented in the open literature, the authors found that all the laminates reviewed were highly notch sensitive. In some cases the notched strength reduced by as much as 50% for a notch-length-to-width ratio of 0.2-0.3. It was also found that most laminates reviewed did not exhibit a notch insensitive region, i.e. the laminate strength dropped sharply with the introduction of the smallest discontinuity. The review indicated that the notched strength of laminates was dependent on a variety of intrinsic and extrinsic variables, and that a comprehensive evaluation of the effects of these variables on the notch sensitivity of composite laminates is still lacking.

Harris & Morris [10] studied the effect of thickness on the fracture behaviour of laminated graphite/epoxy composites. Three fracture toughness specimen configurations were studied, namely the centre-cracked tension, compact tension and three-point bend specimen configurations. Three lay-ups, $(0/\pm45/0)_{ns}$, $(0/90)_{ns}$ and $(0/\pm45)_{ns}$ were studied in each configuration with varying thickness from 8 to 120 plies. The authors found that fracture toughness was a function of laminate thickness and lay-up. The fracture toughness of $(0/\pm45/0)_{ns}$ and $(0/90)_{ns}$ laminates was found to decrease with increasing thickness, whereas the fracture toughness of $(0/\pm45)_{ns}$ increased sharply with increasing thickness until reaching a plateau at about 30 plies thickness. The authors found that that the fracture of thick laminates was relatively smooth and self-similar, while thin laminates were found to fail in a ‘deviate’ manner. Thick laminates exhibited a surface boundary layer in which the crack tip matrix splitting and delaminations were similar to those found in thin laminates. However, inside this boundary layer the interior region of the thick laminates exhibited self-similar fracture that was uniform and free of delamination. The authors found that the strain criterion fracture model developed by Poe [25] was the only thin laminate fracture model that was successful in using thin laminate parameters in predicting the fracture of thick laminates. As a final point, in their literature review, the authors noted that other authors have found that the notch root radius has no obvious effects on the toughness values obtained.

Harris & Morris [11] studied the influence of ply stacking sequence on the strength of graphite/epoxy notched laminated composites. Four groups of three laminates with similar lay-up, but different stacking sequence were studied. Group A contained laminates with 0° and ±45° ply orientations, Group B contained laminates with 0° and ±60° ply orientations, Group C contained laminates with 90° and ±30° ply orientations and finally Group D contained laminates with quasi-isotropic lay-ups. All test specimens were of centre-cracked configuration and were tested quasi-statically in tension. The authors found that that laminate notched strength varied considerably with both lay-up and stacking sequence, due to the way these parameters affected the formation of the damage zone at the notch tip. It was found that laminates in which a notch tip damage zone develops before failure tended to have higher final fracture loads. The damage zone provided stress relief as it reduced the notch-tip stress concentration. Laminates where no damage zone was found to develop were found to have lower final fracture loads. These laminates tended to fail due to the formation of delaminations, which brought about ply uncoupling followed immediately by catastrophic failure of the laminate. The authors also found that the percent of laminate fracture by broken fibres was in direct correlation with the laminate notched strength. Laminates, which were not characterised by damage zones containing major delaminations exhibited higher percentages of failure by broken fibres and higher notched strength.

Harris & Morris [12] investigated the use of a critical crack-tip-opening displacement $(CTOD)_c$ fracture criterion, based on a modified Dugdale model, for predicting the notched strength of graphite/epoxy laminates. It had been noticed in previous work carried out by the authors that for a variety of laminates with significantly different thickness, only minor changes occurred in the value of crack opening displacement (COD) at failure, in contrast to other associated failure parameters, such as load or stress, which varied significantly. Graphite/epoxy centre-cracked specimens of various lay-ups, including quasi-isotropic and cross-ply, were loaded quasi-statically to failure. Most laminates exhibited significant notch-tip damage prior to ultimate failure. Predictions of notched laminate strength were made using the modified Dugdale model and experimentally measured values of COD. It was found that the modified Dugdale model predicted the notched laminate strength for most lay-ups quite accurately, except for the cross-ply laminate. However, it was found that an Irwin-corrected linear elastic fracture mechanics expression for critical CTOD accurately predicted the notched failure strength of the cross-ply laminate.

In a series of four papers [13-16], Kortschot *et.al.* studied damage growth and strength of double-edge notched (DEN) cross-ply graphite/epoxy laminates. In an experimental study [13], they found that the damage zones in these laminates were characterised by splits in the 0° plies, transverse ply cracks in the 90° plies and triangular delamination zones at the 0/90 interfaces. They found a clear relationship between the split length to notch length ratio at fracture (defined as the terminal damage state {TDS}) and the fracture strength. They also found that lay-up only affected notched strength indirectly, through the effect it had on damage growth. In [14], they conducted a finite element study that revealed that DEN specimens failed when the maximum tensile strength in the 0° ply exceeded the strength of that ply. Weibull analysis was used to independently determine the strength of the 0° ply in the vicinity of the notch tip. A semi-empirical model was developed that used knowledge of terminal damage state and independently measured material properties and Weibull parameters to accurately predict notched strength of cross-ply graphite/epoxy laminates.

In [15], Kortschot *et al.* further developed this model. A function was derived using linear elastic fracture mechanics (LEFM) that related split length to applied stress for $(0/90)_s$ laminates. This function combined with the existing model allowed notched strength to be predicted from first principles. Notched strength dependence on notch length in cross-ply graphite/epoxy laminates could then be modelled without the need for empirical parameters. Finally, in [16] they derived a function that related split length to applied stress for $(90_j/0_j)_{ns}$ laminates. Combined with the existing model, this allowed the dependence of notched strength on lay-up to be modelled without the need for empirical parameters.

In another series of papers [17-20], Spearing *et. al.* studied the tension-tension fatigue damage and post fatigue properties of graphite/epoxy centre-notched specimens. Cross-ply and quasi-isotropic specimens were evaluated, with particular emphasis on the cross-ply specimens. It found that notch-tip damage in centre-notched specimens, under cyclic tensile loading, consisted of splits, delaminations and transverse ply cracks, which are the same damage components as observed by Kortschot & Beaumont [13] for quasi-static loading. It was also found that fatigue damage at notches in composite materials could be characterised by the split length at the notch tip. In [18] they developed a model for fatigue damage growth at the notch-tip, where the damage is modelled as a series of interacting phenomena including splitting and delamination. The model was an extension of the existing damage growth model for quasi-static loading developed by Kortschot & Beaumont [16]. Good agreement was found between the model predictions and experimental results, although the model is limited in its application as it can only predict growth in laminates where knowledge of the fatigue damage pattern that will occur in that laminate is known. In addition, the effects of residual stresses and transverse ply cracks on the damage processes have been omitted.

In [19], Spearing & Beaumont further developed the fatigue damage model to predict post-fatigue residual strength for graphite/epoxy laminates. It was found that the model could be accurately applied for graphite epoxy cross-ply laminates, where residual strength increases throughout the fatigue test duration and final rupture does not occur. The model could also be used to accurately predict the fatigue life of laminates where damage reduces the residual strength, the lifetime being reached when the applied maximum stress reached the current residual strength. Finally in this series, Spearing *et al.* [20] developed the model further to predict the stiffness of cross ply graphite/epoxy laminates containing a notch from which damage has grown. The model was based on a finite element representation of the damaged specimen, containing all the experimentally observed damage modes in simplified form. Agreement was found between the model predictions and experimental data.

Dimant *et al.* [21] evaluated the damage models for notched strength of graphite/epoxy cross-ply laminates, developed by Kortschot *et al.* [13-16] for static loading and Spearing *et al.* [17-20] for tension-tension fatigue loading, for use on Kevlar fibre reinforced plastic (KFRP) and glass fibre reinforced plastic (GFRP). Centre-cracked specimens of various cross-ply lay-ups were manufactured from both material systems. Specimens were tested under static loading in tension and in tension-tension fatigue loading. It was found that the KFRP specimens exhibited similar damage initiation and growth behaviour, in both static and fatigue loading, to that observed in graphite/epoxy. It was found that the notched strength model developed for cross-ply graphite/epoxy laminates could be applied to cross-ply KFRP laminates. However, differently to graphite/epoxy, the notched residual strength of KFRP after fatigue loading was not strongly correlated to the terminal damage state for the conditions and lay-ups investigated. Hence, the fatigue model could not be applied satisfactorily. It was found that the cross-ply GFRP exhibited significantly different failure characteristics in static and fatigue loading compared with KFRP and graphite/epoxy. GFRP specimen failure was characterised by splits developing at the notch root, followed by fibres breaking in bundles in the 0° ply, effectively extending the notch. This was not initially catastrophic as secondary splits formed, blunting the notch, however, specimen failure occurred soon after. This difference in damage initiation and growth characteristics was strongly influenced by the statistical variability in strength, and relative weakness of glass fibres compared to Kevlar and carbon fibres. As a consequence, the graphite/epoxy damage models could not meaningfully be applied GFRP laminates.

Cowley & Beaumont [22] examined the combined effects of stress and temperature on cross-ply open-hole tension specimens manufactured from two different carbon fibre reinforced composites, one with a thermoplastic matrix, the other with a thermoset matrix. Each specimen was quasi-statically loaded to a predetermined stress at temperatures between 20°C and 250°C. Damage initiation and growth was expected to be similar to that observed by Kortschot & Beaumont [13]. It was found the rate of split growth at 20°C in the thermoplastic matrix material was 50% lower than in the thermoset matrix material, reflecting the difference in toughness of the two matrices. For both materials, an increase in temperature, particularly at temperatures close to the glass transition temperature, promoted a higher rate of split growth. The delamination angle for the thermoplastic matrix material remained constant, but the delamination angle for the thermoset matrix material was observed to increase, with increasing stress and temperature. The threshold stress for split initiation was seen to reduce with increasing temperature for the thermoset matrix material, but remained unaffected for the thermoplastic matrix material. Overall, it was observed that increasing temperature had a weakening effect on both materials as the strength of the matrix was reduced, the shear modulus of the matrix was reduced, the strength of the fibre/matrix bond and interlaminar shear strength decreased, impairing the ability on the matrix to transmit load to the fibre.

Daniel & Ishai [23] observed from a comparison of experimental results from similar graphite/epoxy specimens containing open holes and elongated slits that strength reduction, as a function of hole radius or crack length, was almost independent of notch geometry.

Finally, Coats & Harris [25] investigated translaminate fracture behaviour of graphite/epoxy centre-cracked laminates with the aim of developing a progressive damage methodology for notched residual strength predictions. Centre-cracked laminates of three different widths and two notch-length-to-laminate-width ratios were loaded quasi-statically to failure. Specimens were made from four different graphite/epoxy material systems, all with the same $[\mp 45/0/90/\mp 30/\bar{0}]_S$ lay-up. It was observed that the damage displayed was similar for each material system. As the tensile load on the specimen was monotonically increased, a zone of micro-crack damage developed at the notch tips and progressed steadily with increasing load. Fibre fracture in the 0° and 30° plies extended a significant distance from the notch before final catastrophic laminate failure took place. The pattern of fibre fracture was collinear with the notch length and resembled the crack extension due to stable tearing observed in ductile alloys prior to fracture. Ply bridging was also evident in the laminates.

A non-linear, damage dependent constitutive finite element (FE) model was developed by the authors. The model predicted the formation of intraply matrix cracks and fibre fracture for monotonic tensile loading and for tension-tension fatigue, as well as the associated ply level stress and strain states, and the residual strength of the laminate. The analysis did not have the capability to model three dimensional stress states such as those that occur at the free edges of laminates, a capability that is needed to predict initiation and growth of delamination in the laminate. The constitutive model used internal state variables to represent the average effects of local deformation due to the various modes of micro-crack damage. The stable damage growth predicted by the FE model was qualitatively similar to that observed in the experiments. A comparison of experimental R-curve data with analytical predictions showed that the trend was similar. The progressive damage growth methodology predicted damage growth resistance well. Most of the analytical residual strength predictions were within ± 10% of the experimental average values. Since there was no consistent overprediction or underprediction trend exhibited by the model the simple failure criterion for each damage mechanism appears to be reasonable for tensile fracture tests. The progressive damage model was also found to predict the fracture strength of the test specimens more accurately than traditional LEFM methods, particularly for wide panels where the fracture resistance effects are dominant. Toughening mechanisms exhibited by wide panel laminates were correctly predicted by the progressive damage model.

Lamina material property sensitivity studies were carried out using the progressive damage model developed. It was found that varying the values of Young's Modulus in the longitudinal fibre direction, E_{11} and the critical strain in the longitudinal fibre direction, ε_{11}^{cr} had a significant effect on the predicted residual strength. Not using shear strain to initiate mode II matrix cracking was found to significantly increase the predicted residual strength. This is because the longitudinal strain in the fibre direction, ε_{11}, is dependent on the laminate resultant shear force, which includes the effects of mode II matrix cracking. These results suggest the importance of accounting for mode II matrix cracking as well as fibre fracture.

Experimental observations provided strong evidence that delamination in the specimens tested was primarily surface ply delamination and that fibre fracture was the dominant failure mechanism. It was also observed that accurate prediction of residual strength for these types of laminates is primarily dependent on the ability to model fibre fracture and mode II matrix cracking. Delamination was not a dominant failure mechanism.

A.2 Concluding Remarks

The above brief overview shows that several researchers, working in the aerospace industry and other industries have studied notched composite specimens in the past. Many of the intrinsic and extrinsic variables that affect the strength of notched composite specimens have been studied. Most of the work has been carried out on centre cracked tension specimens, although according to Daniel & Ishai [23] similar results can be obtained from open hole tension specimens. In addition, Harris & Morris [10] found that notch root radius had little effect on toughness values. Most of the work involved graphite/epoxy material systems, which demonstrated similar damage initiation and growth characteristics. Work carried out on GFRP showed that it behaved quite differently to graphite/epoxy.

Several semi-empirical models have been developed to predict the notched strength of composite specimens. All attain reasonable agreement with experimental results for the material systems and lay-ups for which they were developed, but few are universally applicable. The progressive damage methodology developed by Coats & Harris [25] had good agreement with experimental results and in theory could be applied to any material system, using basic material properties as the input parameters.