Investigation of Low-Velocity Impact Damage in Fibre-Metal-Laminates

by

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A thesis submitted to
the Faculty of Graduate Studies and Research
in partial fulfilment of the requirements for the degree of

Doctor of Philosophy

Ottawa-Carleton Institute for Mechanical and Aerospace Engineering

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Investigation of Low-Velocity Impact Damage in Fibre-Metal-Laminates

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Dedicated to the loving memory of my Grandfather, Norman Richardson.

Per Ardua Ad Astra

ABSTRACT

Fibre-metal-laminates (FMLs) represent a significant evolution in airframe material technology. This new family of materials combines low density, high strength and excellent damage tolerance through the use of metal layers strengthened with fibre-reinforced polymer layers. When subjected to low-velocity impact these laminates like traditional composites, develop internal delamination damage, matrix cracks and limited fibre fractures. Also, as in traditional composites, this damage is hidden within the laminate. A method for predicting the amount of internal damage would reduce the experimental testing requirements for the certification of new laminates.

This thesis describes the development of a modelling methodology that makes use of a new material subroutine based on continuum damage mechanics in the explicit finite-element code LS-DYNA. This subroutine was verified using the experimental data from low-velocity impact tests of various types of GLARE (GLAss REinforced) aluminum laminates, a common type of commercially available fibre-metal-laminate. Static characterization tests were also conducted on GLARE coupons to provide basic property data for the development of the model. These included static tensile tests and double cantilever beam delamination tests. The modelling methodology was used to improve simulations of low-velocity impact on GLARE laminates. The simulations demonstrated that intralaminar damage has a greater effect on the impact response of the panels than interlaminar damage. Parts of this thesis were components of a multi-year collaborative FML Durability Project between Carleton University, Bombardier Aerospace and the National Research Council Canada.

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NOMENCLATURE

GLARE and ARALL designation used throughout this thesis:

FML-X-Y/Z-(t_a) where:

FML – a fibre-metal-laminate (e.g. GLARE, ARALL, etc)

X – type designation for GLARE or ARALL (e.g. GLARE-2)

Y – number of alloy layers

Z – number of prepreg layers

t_a – thickness of alloy sheets in mm (commonly omitted)

Abbreviations:

AEI	Aviation Equipment Industries
AFRP	Aramid Fibre Reinforced Polymer
ALLIC	ALuminum LITHium Carbon reinforced laminate
ARALL	Aramid Reinforced ALuminum Laminate
BRALL	Bamboo Reinforced ALuminum Laminate
BVID	Barely Visible Impact Damage
CARALL	CArbon Reinforced ALuminum Laminate (also CARE)
CARE	CArbon REinforced laminate (also CARALL)
CDM	Continuum Damage Mechanics
CFRP	Carbon Fibre Reinforced Polymer
DCB	Double Cantilever Beam

DCLS	Double Crack Lap Shear
FAA	Federal Aviation Administration
FAR	Federal Airworthiness Regulation
FML	Fibre-Metal-Laminate
GFRP	Glass Fibre Reinforced Polymer
GLARE	GLAss REinforced aluminum laminate
HTCL	Hybrid Titanium Carbon Laminate (see also TiGr)
PEEK	PolyEther Ether Ketone
SEN	Single Edge Notched specimen
SLC	Structural Laminates Corporation
SLI	Structural Laminates Industries
TiGr	Titanium-graphite fibre-metal-laminates (see also HTCL)
TU Delft	Technical University of Delft
UD	Uni-directional

Latin Symbols:

a	Distance from centre of panel, crack length
a_{ijkl}	Fourth order elasticity stiffness tensor
a_o	Initial crack length
b	Cantilever beam width, damage coupling parameter
d, d'	Damage coefficients
d_i	Deflection of coupon
d_i, d_{i-1}	Deflection of plate at time i and $i-1$, respectively

g	Gravitational constant
h	Half beam thickness, enthalpy
k	Contact rigidity
m	Mass of the crosshead
n	Power index
q	Power index
r	Radius of panel
s	Strength limit of a material
t	Thickness, time
t_{al}	Aluminum layer thickness
Δt	Time increment
u	In-plane deflection
w	General deflection in z-direction
w_o	Deflection of centre of panel in z-direction
A_{ij}	Laminate in-plane stiffness matrix
C	Compliance matrix
D_a	Empirical constant for delamination initiation
$(E_o)_i$	Absorbed energy at time i
E_{11}	Local 11 elastic modulus
E_{22}	Local 22 elastic modulus
E_e	Elastic material strain energy
E_D	Damaged material strain energy
E_f	Prepreg modulus

E_{imp}	Impact energy
E^o	Elastic (undamaged) Young's modulus
E_x	Global longitudinal elastic modulus
E_y	Global transverse elastic modulus
F	Contact force
G_{12}	Local 12 shear modulus
G_{Ic}	Critical fracture toughness/strain energy release rate for Mode I delamination
G_{IIc}	Critical fracture toughness/strain energy release rate for Mode II delamination
G_{xy}	Shear modulus in xy
J	Jacobian matrix
J_2	Von Mises equivalent stress
ΔK_{eff}	Effective stress intensity factor
L	Beam length
P	Applied load
P_i, P_{i-1}	Load at times i and $i-1$, respectively
Q	Elasticity matrix
R	Ratio of maximum and minimum stress during fatigue cycling
S	Unloading rigidity
U	Energy (elastic)
V_i, V_{i-1}	Instantaneous velocity at time i and $i-1$, respectively
$\overline{V}_i$	Average velocity at time i

Y	Damaged strain energy release rate
Y_o, Y_o'	Damage development law variables
Y_c, Y_c'	Damage development law variables
Y_r, Y_s	Damage development law variables

Greek Symbols:

α	Indentation depth
α_m	Indentation depth corresponding to start of impact unloading
α_p	Critical indentation depth
α_o	Permanent indentation depth
$\gamma_{r\theta}$	In-plane shear strain
δ, ∂	Differential operators
ε_f	Fibre strain
ε_r	Radial strain
ε_θ	Circumferential strain
ζ	Parametric z co-ordinate
η	Normalized radial position (r/a), parametric y co-ordinate
θ	Fibre angle, thermodynamic temperature
φ	Scalar field variable for damage
ρ	Density
ρ_a	Areal density
σ_y	Yield stress
$\sigma_1, \sigma_{13}, \sigma_2, \sigma_{23}$	Normal and shear stresses in local co-ordinate directions

σ_{12}^r	Shear rupture stress
ν	Poisson's ratio
ν_{l2}	Local longitudinal Poisson's ratio
ν_{2l}	Local transverse Poisson's ratio
ν_{xy}	Global longitudinal Poisson's ratio
ν_{yx}	Global transverse Poisson's ratio
ϕ	Plasticity threshold, shape functions
ξ	Normalized deflection (w/w_o), parametric x co-ordinate
ψ	Free energy density

CHAPTER 1

INTRODUCTION

Fibre-metal-laminates (FMLs) are a new class of hybrid materials consisting of thin metallic layers laminated with fibre-reinforced polymer (FRP) layers (Figure 1). The resulting laminates possess lower density, higher strength and improved fatigue resistance as compared to traditional monolithic aircraft alloys. These laminates can be produced with a wide variety of alloys and prepregs (fibres that are pre-impregnated with resin prior to being cured). This thesis presents the results of an extensive impact testing program. A methodology for modelling the impact damage in these laminates is also proposed and applied to examine the impact response of FML panels.

Figure 1: Basic FML configuration (adapted from Asundi and Choi, 1997).

1.1 Development of FML Technology

One of the critical damage modes in modern metallic aircraft structures is that of fatigue crack growth. A tremendous amount of research has been devoted to the study of this damage mode since the crashes of de Havilland Comet jetliners caused by metal fatigue in the 1950s. More recently, the Aloha Airlines near-disaster caused a renewed emphasis to be placed on the study of fatigue damage in riveted metallic structures. A significant amount of research has been devoted to developing damage tolerant materials that may be applied to aircraft structures to combat this costly problem.

As new materials are developed there is a constant struggle between the desire to reduce weight and the necessity of ensuring passenger safety. Traditional composite materials have been applied to aircraft structures in an attempt to address both of these issues. These materials provided reduced density and improved fatigue damage tolerance over metallic materials. In many cases, traditional composites also provided increased stiffness- and strength-to-weight ratios. However, there remain significant barriers to the widespread application of these materials in aircraft structures. For example, composite materials are sensitive to low-velocity impacts. Impact damage in composites is often widespread within the laminate but there may be little surface evidence of damage, this is referred to as “barely visible impact damage” or BVID.

Even as composite materials were being developed, research continued into techniques for improving the fatigue crack growth resistance of metallic materials. Some of this work resulted in the development of improved alloy formulations or heat treatments. Mechanical methods for improving the crack growth resistance were also investigated. Early research demonstrated, that by adhesively laminating sheets of metal

together, significant increases in the fracture toughness were possible (Kaufman, 1967, Bijlmer, 1978 and Johnson, 1983).

Kaufman observed that the fracture toughness of laminated plates of 7075-T6 made up of eight 1.6 mm thick layers, was double that of monolithic plates of the same total thickness as shown in Figure 2. The laminated plates possess critical stress-intensity factors that approach those of the constituent monolithic layers. In thin materials the amount of plastic constraint at the crack tip is smaller and plane-stress conditions are dominant (Hertzberg, 1996). Therefore, the thin sheets comprising the laminated plates exhibit increased toughness. Damage tolerance of the laminated plates is improved because of lower stress-intensity factors in cracked layers due to stress transfer to undamaged layers and high resistance to inter-ply crack transfer (Johnson, 1983).

Figure 2: Variation in critical stress-intensity factors for layered and monolithic sheets (adapted from Kaufman, 1967).

Fatigue crack growth resistance was improved further by including fibre-reinforced prepreg layers between the metal layers to produce fibre-metal-laminates. Unbroken fibres bridge the crack tip and slow its growth (Bijlmer, 1978). Early FML variants developed at the Technical University of Delft (TUDelft) in the Netherlands consisted of unidirectional aramid prepreg layers laminated with 2024-T3 or 7075-T6 aluminum sheets (Marissen and Vogelsang, 1981, Vogelsang *et al.*, 1981, Bucci *et al.*, 1989 and Gregory and Roebroeks, 1991). Aramid reinforced polymer was selected due to its high blunt-notch strength. These Aramid Reinforced Aluminum Laminates (ARALL) are only manufactured in a unidirectional variant. Carbon and E-glass fibres, among others, were eventually used as well.

Currently, S2-glass based FMLs are available commercially under the trade name GLARE aluminum laminates. Structural Laminates Corporation (SLC) manufactures these products in Europe through a licensing agreement with Fokker Aerostructures, and Aviation Equipment Industries (AEI) manufactures them in North America through a similar agreement. Another company, MCGill Corp., manufactures woven glass and aramid FML panels for use as cargo bay liners. Comtek Ltd., a Canadian composite materials company, is currently installing these laminates as a retrofit into certain types of regional aircraft. FMLs using carbon fibres with aluminum-lithium (ALLIC), graphite fibres with titanium (TiGr) and bamboo or vinylon fibres with aluminum are also being developed.

Fibre-metal-laminates incorporate some of the benefits of traditional composites, such as reduced density and fatigue damage tolerance. Many of the desirable properties of monolithic metals including formability and plastic deformation prior to failure are

also retained by FMLs. The combination of metal and prepreg layers also provides crack growth resistance, impact damage resistance and good impact damage visibility due to formation of plastically deformed dents. This material system seems to meet the requirements of improving passenger safety while decreasing the weight of the aircraft. However, as with any new material, a detailed examination of the properties of FMLs is required before selecting them for an aircraft design.

1.2 Mechanical Properties of Typical FMLs

Table 1 summarizes the composition of ARALL, GLARE and two woven fibre-metal-laminates. The manufacturers' names are shown in parentheses under the commercial designation of the FML. In addition to the resins quoted in the table, newer varieties of GLARE are available which use FM94 resin from Cytec instead of the usual F-185 from Hexcel or the older AF-163-2 from 3M.

Typical mechanical properties of several FMLs can be found in Table 2. In this table, A refers to ARALL, G refers to GLARE, L is the longitudinal direction parallel to the rolling direction of the aluminum (0°) and LT is the longitudinal-transverse (90°) direction perpendicular to the rolling direction of the aluminum. As can be seen from the data, GLARE typically has a higher strength than monolithic aluminum. The lower stiffness of GLARE must be considered when replacing aluminum in vibration prone areas of an aircraft. The density per unit area, or areal density, is also lower in GLARE laminates than monolithic aluminum. If GLARE laminates are used to replace equal thickness aluminum sheets in an aircraft, an immediate reduction in structural weight can be realized.

To improve the overall fatigue performance of the laminates, a plastic post-cure stretch is sometimes applied during the manufacturing process. The standard plastic strain of 0.4% reverses the as-cured stress-state in the laminae by inducing a compressive stress in the aluminum and a tensile stress in the prepreg. The compressive residual stress in the aluminum layers increases their resistance to fatigue crack initiation. None of the panels used in this project received this treatment.

Table 1: Summary of some commercially available FMLs.

Commercial Designation	Basic Description	Metal Alloy and Thickness	Prepreg Thickness	Fibre	Adhesive	Post-cure Stretch
ARALL-2 (SLC)	120°C cure prepreg, 50% by volume UD fibres	2024-T3, t= 0.2 - 0.4 mm	0.21 mm	Aramid	AF-163-2	with or without 0.4%
ARALL-3 (SLC)	120°C cure prepreg, 50% by volume UD fibres	7475-T76, t=0.3 - 0.4 mm	0.21 mm	Aramid	AF-163-2	0.4%
ARALL-4 (SLC)	175°C cure prepreg, 50% by volume UD fibres	2024-T8, t=0.3 - 0.4 mm	0.21 mm	Aramid	AF-191	with or without 0.4%
GLARE-1 (SLC)	120°C cure prepreg, 60% by volume UD fibres	7475-T6, t=0.3 mm	0.25 mm	S2-glass	F-185	0.3-0.5%
GLARE-2 (SLC)	120°C cure prepreg, 60% by volume UD fibres	2024-T3, t= 0.2 - 0.4 mm	0.25 mm	S2-glass	F-185	no pre-strain
GLARE-3 (SLC)	120°C cure prepreg, 60% by volume 50/50 cross ply fibres	2024-T3, t= 0.2 - 0.4 mm	0.25 mm	S2-glass	F-185	no pre-strain
GLARE-4 (SLC)	120°C cure prepreg, 60% by volume fibres with 70% in rolling direction	2024-T3, t= 0.2 - 0.4 mm	0.375 mm	S2-glass	F-185	no pre-strain
GLARE-5 (SLC)	120°C cure prepreg, 60% by volume 50/50 cross ply fibres	2024-T3, t= 0.4 - 0.51 mm	0.51 mm	S2-glass	F-185	no pre-strain
GILLFLOOR 5033 (MCGill)	Woven aramid	2024-T3		Woven aramid		no pre-strain
GILLFLOOR 5433 (MCGill)	Woven glass	2024-T3		Woven S-glass		no pre-strain

Table 2: Summary of properties of several FMLs and 2024-T3 aluminum (Asundi and Choi, 1997 and Department of Defense, 1998).

Parameter	Direction	Material (A - ARALL, G - GLARE)						
		2024-T3	A-2	A-3	G-2	G-3	G-4	G-5
Ultimate tensile strength (MPa)	L	441	717	821	1213	745	1027	683
	LT	434	317	780	317	745	607	683
0.2% Offset yield strength (MPa)	L	324	366	607	393	303	352	-
	LT	290	228	331	228	303	25	-
Tensile elastic modulus (GPa)	L	72.4	66	66	66	58	57	70
	LT	-	53	49	50	58	50	70
Ultimate strain (%)	L	15.0	2.5	2.2	4.7	4.7	4.7	-
	LT	-	12.7	8.6	10.8	4.7	4.7	-
0.2% offset compressive yield strength (MPa)	L	269	255	345	413	310	365	283
	LT	310	234	360	236	310	285	283
Compressive elastic modulus (GPa)	L	73.8	65	64	67	59	60	-
	LT	-	53	50	52	59	54	-
0.2% offset shear yield strength (MPa)	L-LT	-	111	117	-	-	-	-
	LT-L	-	111	117	-	-	-	-
Shear modulus (GPa)	L-LT	27.6	17	17	-	-	-	-
	LT-L	-	16	16	-	-	-	-
Areal density (kg/m ²)	N/A	2.82				2.17	2.42	2.69

Another benefit of using FMLs to replace monolithic aluminum in airframe structures is the increase in fatigue life. Figure 3 is a comparison of fatigue crack propagation in GLARE and aluminum riveted lap joints. The graph shows the aggregate length of all of the cracks growing from rivet holes in a lap joint fatigue specimen. Both specimens are subjected to the same load and frequency and use the same rivets. The crack lengths were normalized to zero to simplify comparison of the data. As can be seen, cracks in the GLARE specimen grew at a steady rate. However, in the aluminum specimen, the crack growth accelerated sharply before failure. Inspection intervals for monolithic airframes must be short enough to find such damage before it reaches the stage of rapid growth to failure. Inspection intervals for GLARE structures can be relaxed since crack growth does not proceed as rapidly.

Figure 3: Aggregate multi-site fatigue crack propagation in GLARE-3-2/1 compared to monolithic aluminum (Laliberté *et al.*, 1999e) [crack lengths were normalized to zero cycles].

1.3 The FML Durability Project

The FML Durability Project, a multi-year collaborative research program, was established between Carleton University, Bombardier Aerospace and the Institute for Aerospace Research (IAR-SMPL) of the National Research Council Canada (NRC) to examine the effect of various types of loading on GLARE FMLs.

Phase I (1998-1999) examined the impact damage tolerance, post-impact strength, multi-site fatigue damage and burn-through resistance of GLARE. Some of the results of the impact experiments have been published elsewhere (Laliberté, *et al.*, 1999a, 1999b, 1999c, 1999d, 1999e, 2000a, 2001a, 2002b). Important results from these tests will be highlighted in the following paragraphs.

Tension-after-impact (TAI) tests were conducted on GLARE-3-2/1, GLARE-4-2/1, GLARE-5-2/1 and aluminum samples (Laliberté *et al.*, 1999e). The test method was based upon a procedure described by Mattousch, 1993. The results showed a small decrease in the static strength of GLARE-3 and GLARE-5 following impact. However, the GLARE-4 coupons maintained the greatest fraction of the undamaged static strength. Recall from Section 1.2 that GLARE-4 has more fibres in the loading direction (0°).

Another property of interest to designers and certification authorities is that of post-impact fatigue damage growth. Two sets of post-impact fatigue tests were conducted as part of the FML Project (Laliberté *et al.*, 1999e, 2000b, 2002a). In the first set of tests, impacted panels of GLARE-3-2/1, GLARE-4-2/1, GLARE-5-2/1 and aluminum were subjected to tension-tension fatigue. Cracks were observed to initiate at the edges of the impact induced dent. As in the TAI tests, the GLARE-4-2/1 specimens

showed the highest resistance to the growth of fatigue cracks due to the higher amount of fibres in the loading direction. The second, more specific series of tests, examined off-axis post-impact fatigue crack growth in GLARE-4-3/2 laminates. These results showed that fatigue cracks in impacted FML panels manifested first on the front (impacted) face of the laminate. This observation is very significant; it means that in practice, fatigue cracks resulting from impact dents on FML skins would be easily detectable. Also, hidden fatigue cracks on the back (unimpacted) face lagged behind the development of damage on the front face. Figure 4 shows sample crack growth curves from the testing conducted at IAR-SMPL. The aggregate crack growth for the front and back faces of the specimen was measured using a travelling microscope. When the cracks on the front face reached the edge of the specimen, growth stopped and the cracks in the back face began to grow.

Figure 4: Sample post-impact fatigue results (Laliberté, *et al.*, 2002)

The final series of post-impact experiments measured the effect of impact damage on the shear-buckling behaviour of GLARE panels (Laliberté *et al.*, 1999e and 2001b). Impacted specimens were installed in a “picture-frame” fixture that introduced a shear load along the specimen edges. Strain gauges were used to detect the onset of buckling in the panels. The buckling experiments built upon earlier work on the buckling of undamaged FML panels at IAR-SMPL.

Multi-site fatigue crack growth was also evaluated during Phase I. This damage mode occurs because the stresses on rivets in a lap-splice are approximately equal along the joint. Cracking will tend to initiate at the same time at each of the rivets. The cracks grow simultaneously until they reach a critical aggregate length and the entire structure fails catastrophically as happened to an Aloha Airlines Boeing 737-200 aircraft in 1988. This problem is worsened by corrosion damage, which is still prevalent in modern aircraft. As part of the FML Durability Project, several simulated fuselage lap-splices were constructed using GLARE-3-2/1 (Laliberté *et al.* 1999e). The results showed that MSD damage in GLARE splices grew at a steady rate. In aluminum splices, the crack growth rates increased sharply near the end of the test. Also, when the aluminum layers in the GLARE splices had failed, the prepreg layers continued to carry the fatigue loads.

Phase II (2000-2003) consisted of investigations of off-axis fatigue, off-axis post-impact fatigue, static lap joint strength, bonded splice fatigue, and the buckling and post-buckling of large stiffened panels and cylinders (Jodoin *et al.*, 2001a, 2001b, 2002; Shi and Poon, 2001). Tests of the delamination response and static tensile strength of GLARE were also performed during the first two years of Phase II.

1.4 Thesis Motivation

Fibre-metal-laminates will be employed in next-generation civilian and military aircraft. For example, GLARE (GLAss REinforced) aluminum laminates have been selected as a fuselage material for the new Airbus A380 transport aircraft and TiGr (pronounced “tiger”) laminates of titanium and graphite reinforced polymer are being considered for parts of the F-35 Joint Strike Fighter. ARALL (Aramid Reinforced ALuminum Laminates) and GLARE are already being used in secondary airframe structures that are subjected to impact loads such as cargo bay liners and wing flaps. Other FMLs are being installed in Bombardier RJ and European ATR regional aircraft as cargo bay liner retrofits.

As shown in Figure 5, there are many areas of civil transports that can suffer impact damage. The impactor could vary in size and mass from a small stone to a galley service vehicle. The energy levels have a similar variability depending on the relative speeds of the aircraft and the impactor. Relatively low-energy impacts can introduce significant internal damage in traditional composites that is not immediately apparent to the naked eye as shown in Figure 6. In contrast, impact damage in monolithic alloys is clearly indicated by the formation of a plastic dent.

Impact testing remains important as new FMLs continue to replace monolithic aluminum in impact-sensitive primary airframe structures. Impact damage experiments are time consuming and expensive; a limited test program together with finite element simulations would reduce both the cost and time required to certify new FMLs for aircraft.

Figure 5: Areas of civil transport aircraft exposed to impact hazards.

Figure 6: Formation of impact damage in a traditional composite (Majeed, 1995).

1.5 Thesis Objectives

Given the above motivation, the objectives of this thesis project were:

- to conduct a detailed review of the state-of-the-art in FML technology and FML impact testing,
- to plan and carry out an extensive series of impact experiments on different varieties of GLARE, to improve the level of understanding of the impact damage modes that occur in these materials,
- to develop a practical impact damage modelling methodology for GLARE to predict the formation of damage and, if required, to use material characterization experiments to support this development, and
- to plan and carry out a series of low-velocity impact simulations on aluminum and GLARE and to compare the modelling results with the experimental tests.

1.6 Thesis Overview

This thesis presents the results of an experimental and numerical investigation into the formation of impact damage in GLARE laminates. The need for a damage modelling methodology was identified during Phase I of the FML Durability Project. Work on the methodology was carried out during Phase II of the Project. The development of this methodology was not within the scope of the collaborative agreement between IAR-SMPL and Bombardier Aerospace. Persons wishing to employ the methodology should contact the author through his supervisor at Carleton University.

Where appropriate, each Chapter contains a review of the relevant literature and the state-of-the-art. This method for presenting the background information was chosen to follow the same path as the development of the experimental program and the modelling methodology. The thesis is organized into the following chapters:

Chapter 2 – review of impact tests on FMLs and results of impact testing.

Chapter 3 – review of modelling techniques for FMLs and composites; overview of the proposed damage modelling methodology; review of the LS-DYNA finite element code.

Chapter 4 – results of low-velocity impact simulations of aluminum.

Chapter 5 – delamination characterization test results.

Chapter 6 – development of a prepreg material subroutine in LS-DYNA.

Chapter 7 – results of simulations of low-velocity impacts on GLARE FMLs.

Chapter 8 – discussion of results of the experimental and modelling work.

Chapter 9 – conclusions from the project.

Chapter 10 – recommendations for future work and follow-on projects.

CHAPTER 2

IMPACT TESTING OF FMLS

Any material, particularly laminates such as FMLs and traditional composites, must be subjected to stringent qualification programs before being used in airframes. One area of concern is the susceptibility of materials to impact damage from in service hazards as described in Section 1.4. To determine the extent of internal damage in GLARE from impact, a series of low-velocity impact tests was conducted. A review of historical impact test methods and results was conducted to guide the development of the test method applied in the present project.

2.1 Review of Impact Testing of FMLs

A large range of “standard” test procedures for composites and FMLs is described in the literature by groups that have developed their own methods or variations on existing methods. This review will present test methods and test data from a variety of these sources. Figure 7 shows two typical impact test geometries with definitions.

Ideally, the impact test fixture used should employ a coupon size and geometry that is representative of typical aircraft structures. The coupon may represent a generic region of fuselage skin between a pair of stringers and a pair of frames, or a specific component such as a radar bulkhead. Small coupons may not accurately represent such structures. Large coupons are more expensive and require more material, so fewer can be tested. Thus, there is a trade-off between the size of the specimen and the number of specimens. A choice must also be made as to what shape the opening in the test fixture should have. Square fixture openings may cause undesirable stress concentrations in the corners. Circular specimens produce deformation patterns that are relatively free of fixture induced effects but may not be entirely representative of all aircraft structures.

Figure 7: Basic impact test fixture geometries.

2.1.1 Coupon Impact Test Methods and Results

The amount of published work on impact testing of FMLs is steadily increasing as is evident from Table 3. Much of the early work focused on ARALL (Johnson, 1986, Vlot, 1987, Hoogsteden, 1992, Wu and Sun, 1993; and Sun *et al.*, 1993). Later, the focus shifted to GLARE (Vlot, 1987, 1993a, 1993b, 1996, Vlot *et al.*, 1998, Poston, 1994; Liaw, 2001), aluminum-lithium and carbon fibre (ALLIC) (Freischmidt, 1994), TiGr laminates (Li and Johnson, 1998) and carbon fibre-reinforced aluminum (CARE or CARALL) (Vlot, 1993b, Gregory and Roebroeks, 1991). Low-velocity impact tests are typically conducted using a drop tower or a pendulum apparatus. Pressurized gas guns are used to perform tests requiring velocities approaching ballistic levels. There are also tests that examine airframe response to specific types of impactors, such as birds.

Table 3: Summary of impact tests conducted to date on FMLs.

Reference	Materials Tested	Test Type	$r_{impactor}$	Coupon Size	Test Area Dimensions
			(mm)	(mm)	(mm)
Gregory and Roebroeks, 1991	H	Drop and gas gun	6.35	101.6 x 101.6	Not available
Vlot, 1993a	A, C, E, F, G	Drop	7.5	100 x 100	ϕ 80
Vlot, 1993b	A, C, D, E, G	Drop	7.5	150 x 150	100 x 100
Vlot, 1993b	A, B, C, D, E, F, G, H, J	Drop	5 and 7.5	100 x 100	ϕ 80
Li <i>et al.</i> , 1996	M	Charpy	N/A	12 x 55	Not applicable
Vlot, 1987	A, C, E, F, G	Drop	7.5 and 0.5	120 x 120	ϕ 80
Johnson, 1986	C, G	Drop	12.7	76 x 406	ϕ 50.8
Hoogsteden, 1992	G	Pendulum	6.35	127 x 254	Not available
Wu and Sun, 1993	G	Gas gun	6.35	101.6 x 25.4	Not available
Sun <i>et al.</i> , 1993	G	Gas gun	6.35	101.6 x 25.4	Not available
Vlot, 1996	A, C, F, G	Pendulum	7.5	125 x 125	100 x 100
Poston, 1994	A, C, D, E, F, K, L, N	Drop and bird	8.0	100 x 150	75 x 125
Vlot <i>et al.</i> , 1998	A, B, C, D, E, F, G, H, J	Drop	7.5	100 x 100	ϕ 80
Liaw, 2001	A, F, G	Drop	8	100 x 100	ϕ 76

Note that the materials in Table 3 are represented as follows: A-2024-T3, B-7075-T6, C-carbon fibre reinforced polymers (CFRP), D-glass fibre reinforced polymers (GFRP), E-aramid fibre reinforced polymers (AFRP), F-GLARE, G-ARALL, H-CARALL, I-ALLIC, J-monolithic polycarbonate, K-composite plus honeycomb, L-2024-T3 plus honeycomb, M-BRALL, N- 2219-T62. Table 4 summarizes the relevant impact test results from several of the researchers listed in Table 3. The following paragraphs discuss some of these results in greater detail. The first cracking energy is the impact energy required to produce a crack in the unimpacted aluminum face of the FML panel.

Table 4: Review of key impact test results.

Reference	Material	First Cracking Energy	Notes
		(J)	
Gregory and Roebroeks, 1991	GLARE-3-3/2	18.3	
	2024-T3	17.3	
	ARALL-2-3/2	8.5	
Vlot, 1993a	GLARE-2-2/1	10.3	
	GLARE-2-3/2	>14.2	
Vlot, 1993b	GLARE-3	18	Quasi-static
		37	10 m/s
		95	100 m/s
Vlot, 1993b	ARALL-2-2/1	4	Circular clamped coupon
Hoogsteden, 1992	ARALL-1-3/2	14.1-15.9	Appearance of partial through crack
Vlot, 1993b	ARALL-2-2/1	16.4	Stiffened panel test compared to circular coupon

Gregory and Roebroeks (1991) described results from impact tests of GLARE. It was reported that the minimum energy required to initiate cracking in GLARE-3-3/2- was higher than for either monolithic 2024-T3 or for ARALL-2-3/2. No description of the shape of the opening in the impact fixture was provided.

Vlot (1993a) compared the impact properties of monolithic clad and unclad 2024-T3 aluminum, ARALL-2-3/2, GLARE-2-2/1, GLARE-2-3/2 and two types of carbon/PEEK laminate. In these tests a square test area was used. The impact force

versus panel deflection curves were recorded for the tests and the energy required to initiate cracking was determined. The minimum cracking energy reported for GLARE-2-2/1 was higher than that reported for the thicker GLARE-2-3/2 laminates. The GLARE-2-2/1 result exceeded the minimum cracking energies reported for several ARALL variants by 40-50%. No cracking was observed in any of the monolithic 2024-T3 samples. It was observed that GLARE-3 had a smaller internal damage size than monolithic carbon/PEEK laminates with the same thickness and areal density, defined as mass per unit area. Also, GLARE-3 had a higher minimum cracking energy than ARALL and carbon/PEEK laminates with the same areal density. The impact dent depth for the GLARE-3 specimens was the same as that of the monolithic aluminum samples. The GLARE-3-3/2 material absorbed more impact energy than monolithic 2024-T3 for the same thickness. GLARE laminates also performed comparably to cross-ply glass-reinforced composites of the same thickness. First failure occurred at the same deflection but at a much lower force in the composite (4 kN versus 2.75 kN).

Vlot (1993b) used a circular impact fixture to test a variety of materials including monolithic aluminum alloys, composites, CARALL, ARALL and GLARE. It was reported that GLARE-3 demonstrated a minimum cracking energy that increased as the impact velocity was increased (Table 4). The reason for this velocity dependence is that glass fibres have strain rate-dependent mechanical properties (Vlot, 1993b). Therefore, to a greater degree than observed in ARALL, the velocity and energy level of impact both have an effect on energy absorption. Interestingly, there was no difference in dent depth observed for GLARE-3 when impacted at low and high velocities for the same energy levels.

Vlot (1996) later published data from another investigation of the impact properties of FMLs and monolithic aluminum alloys. The tested materials included monolithic 2024-T3 and 7075-T6 aluminum, several types of ARALL, several types of GLARE, CARALL, R-glass epoxy and carbon epoxy composites. The overall specimen size and impact test set-up were the same as described previously by Vlot (1993a). The impact test support fixture used circular openings with a variety of diameters and a 7.5 mm radius impactor. It was found that the proximity of the impact site to the edge of the opening in the test fixtures influenced the damage modes by introducing additional deformation at the edge of the dents. This behaviour was most pronounced in small diameter specimens.

There are two standard composite impact test methods from Boeing and NASA that have also been applied to FMLs. Boeing (1988) suggested a specimen-clamping fixture for impact testing as shown in Figure 8. This test uses a 152.4 mm x 101.6 mm coupon clamped by four rubber clamp tips over a 127 mm x 76.2 mm rectangular hole. A 12.7 mm diameter hemispherical impactor is specified. This type of constraint is quite different from the continuous edge clamping used by Vlot (1993a, 1993b, 1996) and does not work well. The thin FML panels can buckle between the rubber clamps since this fixture was intended for thicker composite coupons.

* Clamp tip centered 6.35 mm from edge of cut out

Figure 8: Boeing impact fixture (reproduced from Boeing Company, 1988).

Figure 9: NASA 1092 impact fixture (reproduced from NASA, 1983).

The other widely used test method for determining the impact resistance of composites is described by NASA (1983). This method employs larger coupons, ranging from 178 mm x 254 mm to 178 mm x 318 mm. The specimen is clamped using a tie down plate over a 127 mm x 127 mm square opening (Figure 9). For this test, a 12.7 mm diameter steel impactor is specified.

The variation of the NASA arrangement was used at the Air Force Research Laboratory (AFRL) to investigate the compression-after-impact strength of ARALL-1 (Hoogsteden, 1992). A total of eight specimens with two control specimens were impacted. A 3/2 lay-up was used, with three layers of aluminum 7075-T6 and two unidirectional plies of aramid fibre prepreg. The specimens were manufactured to a size of 254 mm x 127 mm, modified from the NASA specification of 254 mm x 178 mm. An instrumented pendulum impact apparatus (Figure 10) with a 12.7 mm diameter steel impactor was used by Hoogsteden with the NASA impact fixture.

Figure 10: Wright-Patterson impact apparatus (reproduced from Hoogsteden, 1992).

All specimens were C-scanned after impact to measure the extent of the damaged area. In comparing the ARALL-1 energy absorption data to that of graphite/epoxy and fibre glass/epoxy, Hoogsteden observed that the ARALL laminates required more impact energy to achieve the same energy absorption as the composites. The ARALL laminates absorbed a large portion of the impact energy to deform the aluminum layers. The fibres deformed elastically and thus returned a portion of the impact energy to the impactor on rebound. Also, to induce damage of a similar size, the ARALL laminates required higher impact energy than the composite materials.

2.1.2 Component Level Impact Testing

Vlot (1993b) described a series of impact tests conducted on stiffened panels as shown in Figure 11. A 2/1 lay-up of ARALL was used for the skin ($t=0.8$ mm) and a 3/2 lay-up was used for the stringers ($t=1.3$ mm). These panels more accurately represented actual aircraft structures. The ARALL stringers were manufactured using standard aluminum forming techniques, with the fibres oriented in the longitudinal direction of the stringer. These stringers were bonded to the skin using AF-163-2 adhesive. The panels were then impacted at the centre between two stiffeners. These larger panels required more impact energy (Table 4) than circular clamped panels to induce cracking. The circular panels were tested as part of the study by Vlot (1993b) described previously.

Figure 11: ARALL stiffened impact test panel (reproduced from Vlot, 1993b).

Other component level tests were carried out to select a material for the forward radome bulkhead for the Learjet 45 (Poston, 1994). In these tests several bulkhead configurations were developed using a variety of common materials. The projectile used for the impact testing was a 4 kg gelatinous “bird” fired from a gas gun. All test articles were 565 mm in diameter, with a doubler added to the periphery of some panels. Based on the deflection and energy absorption measurements, GLARE-3-5/4 was selected.

2.1.3 Impact Damage Measurement Techniques

Before one can model the impact damage formed in FMLs, it is necessary to understand the various damage modes. The following techniques have been used to examine the damaged areas in FMLs (Fahr *et al.*, 1999, Fahr *et al.*, 2000 and Armstrong and Fawaz, 2001):

- Ultrasonic C-scan,
- Eddy current,
- Penetrant enhanced x-ray,
- Destructive inspection of the impacted specimens,
- Removal by chemical milling of the aluminum layers to expose delamination and matrix cracking in the fibre-reinforcing layers, and
- Infrared thermography.

Each of the above methods has specific advantages and disadvantages. The first two methods are non-destructive techniques. However, there is a general agreement among researchers that C-scan (Vlot, 1993b; Sun *et al.*, 1993; and Fahr *et al.*, 1999) is ineffective for impacted coupons due to the large plastically deformed dent that scatters the sound waves. Eddy current techniques are not affected by the shape of the specimen but are only useful for detecting cracking in the aluminum layers.

The next three techniques are all to some degree destructive. Penetrant enhanced x-ray is the least destructive of the three. This process involves drilling a small hole through the top layers of the specimen and then injecting an x-ray opaque penetrant. The x-radiographs then clearly show the extent of the delamination. However, with this two-

dimensional imaging technique it is not possible to determine which layer has the most damage. Destructive inspection uses standard metallographic techniques to mount, cut and then polish impact specimens. Scanning electron microscopy or conventional optical microscopy can then be used to inspect the damage. This technique gives a good picture of the through-thickness damage pattern.

Another damage characterization technique developed for FMLs consists of removing the aluminum layers with a nitric or hydrochloric acid solution. The aluminum layers in GLARE are removed without damaging the epoxy/glass prepreg layers (Vlot, 1993a). The prepreg can then be examined to determine the levels of matrix cracking and fibre damage.

Infrared thermography has been used with limited success for detecting delamination in impacted FML panels (Armstrong and Fawaz, 2001) and also in FML panels subjected to post-impact fatigue (Laliberté, *et al.*, 2001). This method proved to be qualitative and not suitable for providing detailed measurements of the damaged area.

Delfosse and Poursartip (1997) described another method used to measure the damage in impacted composite panels. A high temperature furnace was used to burn off the epoxy layers. The measured length of the broken fibres was used to determine the approximate amount of energy absorbed through fibre fracture in impacted composite panels. It was noted that this method was labour intensive and was appropriate for a limited number of specimens, rather than a large sample of impacted specimens. Also, it would not be possible to place FML panels in the furnace due to the presence of the alloy layers. The metal layers would have to be removed using the acid etch method first. This would increase the labour requirement further.

2.2 Low-Velocity Impact Testing of FMLs

Based on the review of impact data and test methods, it was determined that up-to-date impact resistance information using a realistic specimen size was required for GLARE. This section presents the results of low-velocity impact tests of several variants of GLARE and monolithic 2024-T3. The impact tests were conducted with the Dynatup instrumented drop weight impact tower located at IAR-SMPL.

2.2.1 Preliminary Impact Testing

Prior to the formal impact tests a series of preliminary tests were completed using only GLARE-3-2/1. These tests established the procedures and test matrix to be used for the formal impact test program. The NASA impact fixture described in Section 2.1.1 was initially used to support the GLARE specimens. As can be seen in Figure 12a, the square opening causes stress concentrations at the corners of the specimen that can influence the formation of damage. Therefore, a new impact fixture with a circular opening was developed that resulted in the impact damage geometry shown in Figure 12b. The NRC impact fixture that was developed is shown in Figure 13. This fixture reduced the effects of the boundary conditions on the impact damage patterns and provided a clamped and symmetrical boundary condition.

Details of the results of the comparative tests using the NASA and NRC fixtures were published by Laliberté *et al.* (1999b). Appendix A is a summary of preliminary impact results from the NRC impact fixture with various impactors. Appendix B contains the impact results from the NASA and NRC fixtures using 25.4 mm and 12.7

mm diameter impactors. Appendix C contains graphical comparisons of the impact test results from the NASA and NRC test fixtures.

Figure 12: Comparison of damage geometry from the (a) NASA and (b) NRC impact fixtures.

Figure 13: NRC impact fixture.

2.2.2 Formal Impact Testing

Following the completion of the preliminary tests, a formal series of impact tests was carried out using three varieties of GLARE and one type of aluminum. The NRC impact fixture was used for these tests. The following materials were tested:

- **aluminum 2024-T3** $t=1.02$ mm, $\rho_a=2.82$ kg/m²,
- **GLARE-3-2/1-(0.3)** $t=0.85$ mm, $\rho_a=2.17$ kg/m²,
- **GLARE-4-2/1-(0.3)** $t=0.98$ mm, $\rho_a=2.42$ kg/m², and
- **GLARE-5-2/1-(0.3)** $t=1.11$ mm, $\rho_a=2.69$ kg/m².

The GLARE-3-2/1 material was clad with a thin layer of pure aluminum on the impacted face. The remaining GLARE types were chromated for corrosion resistance. The prepreg layers consisted of Hexcel F185 epoxy with cross-pplies of S2-glass unidirectional fibres providing reinforcement. Figure 14 shows the GLARE variants that were tested.

Figure 14: GLARE-2/1 variants subjected to impact testing.

2.2.3 Experimental Procedure

Test Matrix

The test matrix employed for the impact testing is shown in Table 5. The undamaged control specimens provided baseline data for post-impact strength tests. Energy levels were selected based on the preliminary impact test results. The energies represent impact events that would produce the following progressive damage states:

- plastically deformed dent with no delamination (typically for $E_{imp} \leq 25$ J),
- plastically deformed dent with delamination (typically for $25 \text{ J} < E_{imp} < 55$ J), and
- puncture (typically for $E_{imp} > 55$ J).

During the preliminary testing, specimens were x-rayed to determine levels of internal damage to help establish these energy levels.

Table 5: Test matrix for formal FML impact tests.

Material	Control	Impact Energy Level (J)							Total
		15	25	35	45	55	60	65	
2024-T3	2	0	6	6	6	6	6	0	32
GLARE-3	2	6	6	6	6	6	0	0	32
GLARE-4	2	0	6	6	6	6	6	0	32
GLARE-5	2	0	6	6	6	6	0	6	32
Total	6	6	24	24	24	24	12	6	

Drop Tower Data Collection Methods

Poon and Benak (1986) prepared a complete description of the drop tower (Figure 15), its capabilities and its limitations. A block diagram of the data acquisition system is shown in Figure 16.

Figure 15: NRC Dynatup drop-weight impact tower with the NRC fixture.

Figure 16: Block diagram for Dynatup Model 730-I impact system.

The Dynatup drop tower is interfaced with an IBM XT personal computer through a commercial data acquisition card. The data acquisition software is triggered when a metal flag passes through a timing gate. The sampling rate is determined by the user and is based on the duration of the impact event. Event durations can range from 10 ms to 100 ms with 1024 data points recorded regardless of the event duration. The velocity at impact is measured by the infrared timing gate triggered by the passage of a metal flag mounted on the crosshead. The location of the timing gate is adjusted so that it records the velocity of the crosshead at the moment of impact. The instrumented load cell records impact force and then uses this information to calculate deflection, instantaneous velocity and absorbed energy with the following equations (Poon and Benak, 1986):

the displacement of the tup, d_i :

$$d_i = d_{i-1} + \bar{V}_i \Delta t \quad (1)$$

average velocity:

$$(\bar{V}_i) = \frac{1}{2}(V_i + V_{i-1}) \quad (2)$$

instantaneous velocity at any time interval i :

$$V_i = V_{i-1} + \left(g - \frac{P_{i-1} + P_i}{2m} \right) \Delta t \quad (3)$$

where m is the impactor mass and P is the force. The absorbed energy is thus:

$$(E_o)_i = (E_o)_{i-1} + \frac{P_{i-1} + P_i}{2} (V_i) \Delta t \quad (4)$$

All panels were clamped between the top and bottom plates of the fixture with 3/8 in. machine bolts. The bolts were tightened and loosened in a uniform pattern with a torque of 10.2 N-m to provide a consistent clamping force for all of the samples.

Several panels from each sample group were instrumented with strain gauges to collect instantaneous strain data from each test. Table 6 summarizes the energy levels at which the instrumented panels were impacted. A Nicolet model Pro-20 four-channel digital oscilloscope was employed to capture the strain data. Pacific model 1073 strain gauge amplifiers were used to amplify and condition the signals from the strain gauges.

Table 6: Summary of instrumented and impacted GLARE specimens.

Material	Impact energy level (J)							Total
	15	25	35	45	55	60	65	
2024-T3	0	0	0	0	0	0	0	0
GLARE-3	1	1	1	1	1	0	0	5
GLARE-4	0	1	1	1	1	1	0	5
GLARE-5	0	1	1	1	1	0	1	5
Total	1	3	3	3	3	1	1	15

Impact Damage Inspection

Table 7 contains the post-impact damage characterization matrix. Specimens were x-rayed using the penetrant enhanced technique described in Section 2.1.3. Additional specimens were selected from each energy level for destructive inspection. The results of the post-impact examinations are presented in Section 2.2.4.

Table 7: Summary of inspected specimens.

Inspection Method	Impact Energy Level (J)							Total
	15	25	35	45	55	60	65	
Total X-Rayed	2	6	6	6	6	2	2	30
GLARE-3	2	2	2	2	2	0	0	10
GLARE-4	0	2	2	2	2	2	0	10
GLARE-5	0	2	2	2	2	0	2	10
Total Destructively Inspected	1	3	3	3	3	1	1	15
GLARE-3	1	1	1	1	1	0	0	5
GLARE-4	0	1	1	1	1	1	0	5
GLARE-5	0	1	1	1	1	0	1	5
Overall Totals	3	9	9	9	9	3	3	45

2.2.4 Impact Test Results

The impact test results are presented in several forms. First, in terms of the amount of energy absorbed by the panel as a function of impact energy. Next, the impact data was normalized in terms of the areal density of the panels to rank the laminates on their relative performance. Another method for displaying the impact data is maximum impact force versus impact energy. Several force versus time and strain versus time traces for representative impact tests are presented. Complete tabulated results of the impact tests can be found in Appendix D.

Absorbed Energy

Figure 17 shows the progression of absorbed energy with increasing impact energy. Prior to puncture the amount of absorbed energy is below 100%, indicated by the straight line on the graph. Energy is absorbed first by plastic deformation of the panel and by matrix cracking. Then, delamination damage occurs and absorbs a portion of the impact energy. Finally, the panels are punctured by the impactor and energy is absorbed by fibre and aluminum fracture. The unabsorbed energy is transferred back to the impactor or converted to sound and heat. Typically, the energy converted to heat and sound is quite small (Vlot, 1993b). It is not obvious from the experiments what fraction of energy is absorbed by these mechanisms. Impact simulations are required to determine this.

The GLARE-3 and GLARE-5 laminates puncture in a similar manner; cracks develop in the 0° and 90° fibre directions. However, in punctured GLARE-4 a single unidirectional crack is formed in the 0° direction rather than a complete puncture. As the impact energy increases so does the length of the crack. The GLARE-4 panels have 30

percent of the fibres perpendicular to the rolling direction and 70 percent parallel to the rolling direction creating difference in stiffness that leads to the observed crack geometry. Once the GLARE-4 panel is punctured a portion of the energy is still returned to the impactor. In the GLARE-3 and GLARE-5 panels there is almost no rebound of the impactor following puncture. At impact energies exceeding the puncture threshold, the percentage of absorbed energy decreases. This occurs because the impactor passes through the specimen and retains some of its kinetic energy. The aluminum 2024-T3 panels were not punctured during this test program, so direct comparison of puncture initiation energies is not possible.

Figure 17: Absorbed energy versus impact energy for GLARE and 2024-T3.

Figure 18 shows the absorbed energy normalized by areal density as a function of the impact energy for various laminates. It can be seen that GLARE-5 absorbed the least amount of energy for its areal density, and therefore suffered the least amount of damage. This result was expected since GLARE-5 was reportedly optimized for impact loading.

Figure 18: Normalized absorbed energy versus impact energy. Each data point in the above graph is an average of 5 tests.

Peak Impact Force

The peak impact force is shown in Figure 19 as a function of the impact velocity. The impact force varies almost linearly with the velocity of the crosshead. However, at higher velocity levels the impact force begins to level off in the GLARE-3 and GLARE-4 punctured panels. The capacity of the panels to resist the impactor was exceeded once puncture occurred. The GLARE-5 panels did not have the same levelling off of peak impact force at higher velocities.

Figure 19: Impact force versus impact velocity.

Figures 20 to 23 are sample load traces for each of the materials tested. The initial cyclic variations observed in all the load traces are the result of oscillations in the panel induced by impact. The load traces for aluminum shown in Figure 20 are smooth and well defined. The shape of the impact force response curves remains consistent as the impact energy increases. However, the overall event time decreases. For GLARE-3 and GLARE-5 the load drops off sharply (Figures 21 and 22) once the panel is fully penetrated.

Figure 20: Aluminum 2024-T3 sample load traces.

Figure 21: GLARE-3 sample load traces.

Figure 22: GLARE-5 sample load traces.

The impact force response of GLARE-4 panels shown in Figure 23 is significantly different. The load did not drop off as sharply or by as great a magnitude when the specimen was punctured (specimens GL4-24 and GL4-25). The impact-induced crack formed in a stable manner and dissipated the energy uniformly. This resulted in the crack geometry described previously. The crack always formed parallel to the major fibre direction (i.e. major loading direction). One implication of this behaviour is that a cyclic operating load would not cause the crack to grow unless the crack was at an angle to the load. This damage mode was also observed in thicker GLARE-4-3/2 laminates (Laliberté *et al.*, 2001a).

Figure 23: GLARE-4 sample load traces.

Impact Strain Data

Figure 24 shows the locations of the strain gauges relative to the impact point. All gauges were located 38.1 mm from the centre of the panel. Figures 25 to 27 show impact strain trace data for GLARE-3, GLARE-4 and GLARE-5. For most of the instrumented specimens strain data was collected from locations 1 and 4 and for several specimens data was collected at all locations.

Figure 24: Strain gauge locations for four-gauge (a) and three-gauge (b) configurations.

The strain traces for GLARE-3 in Figure 25 show relatively smooth behaviour for energy levels below that required for penetration. At energy levels above the penetration threshold, the curves follow a shape similar to that of the load traces. The strain level dropped off sharply once the panel has been punctured. The peak strains measured by the gauges for each energy level are similar for both directions. This result is expected, since the GLARE-3 has the same amount of fibres in the 0° and 90° directions. The curves shown in Figure 25 for the 16.4 J (GL3-36) energy level are skewed and smoothed because the strain gauge amplifiers were set with excessive noise filtering.

Figure 25: GLARE-3 impact strain traces for locations 1 and 4.

Unlike the GLARE-3 strain curves, the peak values for gauges 1 and 4 are significantly different for the GLARE-4 impacted panels. There are more than twice as many fibres oriented parallel to gauge 1 (0°) than there are for gauge 4 (90°). As can be seen in Figure 26, the strains recorded by gauge 4 are approximately twice as high as the values recorded by gauge 1. This indicates that the panels are twice as stiff in the direction of gauge 1 (0°) than in the direction of gauge 4 (90°). After fracture, strain measured by gauge 1 drops off sharply. However, the strain measured by gauge 4 does not drop off after fracture. The fibres parallel to gauge 4 did not fracture and thus continued to carry loads when the crack formed.

Figure 26: GLARE-4 impact strain traces for locations 1 and 4.

The GLARE-5-2/1 strain traces shown in Figure 27 are smooth, like the GLARE-3 and GLARE-4 traces. Only one GLARE-5 specimen failed during this portion of the testing, GL5-30. The strain exhibited the same sharp drop at punctures as was observed for the GLARE-3 panels. The strain values for gauge 1 and gauge 4 are nearly identical for all the load traces as was also observed for GLARE-3.

Figure 27: GLARE-5 impact strain traces for locations 1 and 4.

Impact Damage Inspection

Internal impact damage was visualized using the penetrant enhanced x-ray technique developed at IAR-SMPL (Fahr *et al.*, 1999). Sample x-rays of the internal damage are shown in Figure 28. The effect of increasing impact energy is clearly shown in the figure as the extent of the damaged zone also increases. The small white circles in the centre of the images are the holes drilled to provide a penetrant path to the damage.

Figure 28: Penetrant enhanced x-ray images of impact damage.

Sample punctured panels were also photographed using visible light to record the damage patterns present in the panels as shown in Figures 29 to 31. The damage observed for GLARE-3 and GLARE-5 was similar. Cracks that formed did not follow a preferential direction during growth. However, as stated earlier, the single crack formed in the GLARE-4 panels was always parallel to the major fibre direction.

Figure 29: Puncture damage in GLARE-3.

Figure 30: Puncture damage in GLARE-4.

Figure 31: Puncture damage in GLARE-5.

Damage in the specimens was also examined using a destructive inspection technique. These results were used to determine the efficacy of the penetrant enhanced x-ray process. Seven impacted GLARE-3 and GLARE-5 specimens, which had previously been x-rayed, were cold-mounted in epoxy and then polished to reveal the internal damage. The specimens were cut from the impacted panels using a slow speed saw to

prevent the introduction of additional damage. A further four GLARE-4 specimens that had not been x-rayed were inspected. Figures 32 to 34 show unimpacted samples of GLARE-3, GLARE-4 and GLARE-5, provided for reference. As can be seen in these figures, there is no evidence of delamination from the mounting and polishing operations. Table 8 summarizes the delamination size measurements that were taken from both the x-rays and the polished specimens.

Figure 32: Cross-section of an unimpacted GLARE-3 specimen.

Figure 33: Cross-section of an unimpacted GLARE-4 specimen.

Figure 34: Cross-section of an unimpacted GLARE-5 specimen.

Table 8: Delamination measurements from the x-rays and cross-sections.

Sample	E_{imp}	Delamination on x-ray (mm)		Delamination in 0° direction (mm)
	(J)	0°	90°	
3-31	15.73	X-ray showed no delam		21.0
3-32	26.19	X-ray showed no delam		19.5
3-33	36.42	N/A	16.0	27.5
3-34	46.72	N/A	17.0	28.0
4-16	26.44	No x-rays taken, information provided for reference		8.0
4-27	36.88			10.5
4-08	47.03			16.0
4-19	53.27			21.0
5-11	25.82	X-ray showed no delam		13.5
5-12	36.47	X-ray showed no delam		22.0
5-04	57.74	X-ray showed no delam		26.0

Matrix cracking was observed in all of the specimens as indicated in Table 9. Delamination damage was also observed in all of the impacted specimens. However, as indicated in Table 9 this damage was not visible in all of the x-rays. Only GLARE-4 specimens containing through-cracks were selected for destructive inspection. It was in these specimens that fibre fracture was observed. No x-rays were available for these GLARE-4 specimens. Complete results of the cross-sectioning are presented in Appendix E. Typical results are shown below.

Table 9: Summary of observed damage in the cross-sectioned GLARE panels.

Sample	E_{imp}	Delamination Visible		Matrix Cracking Visible	Fibre Fracture Visible	Fibre Bridging Visible	Aluminum Fracture Visible
	(J)	In X-Sect.	In X-Ray				
3-31	15.73	Yes	No	Yes	No	Yes	No
3-32	26.19	Yes	No	Yes	No	Yes	No
3-33	36.42	Yes	Yes	Yes	No	Yes	No
3-34	46.72	Yes	Yes	Yes	No	Yes	No
4-16	26.44	Yes	N/A	Yes	No	Yes	No
4-27	36.88	Yes	N/A	Yes	Yes	Yes	Yes
4-08	47.03	Yes	N/A	Yes	Yes	Yes	Yes
4-19	53.27	Yes	N/A	Yes	Yes	Yes	Yes
5-11	25.82	Yes	No	Yes	No	Yes	No
5-12	36.47	Yes	No	Yes	No	Yes	No
5-04	57.74	Yes	Yes	Yes	No	Yes	No

Figures 35 to 46 show the typical damage modes observed in the cross-sections of the GLARE specimens. The x-ray of sample GLARE-3-31 (15.73 J), shown in Figure 35a, did not indicate the presence of delamination damage. However, the cross-section, Figure 35b, revealed delamination damage throughout the impacted area. The prepreg layers near the impacted surface of the panel were intact and adhering to the aluminum. This prevented adequate flow of the penetrant to the damage.

Figure 35: Sample GLARE-3-31 (15.73 J) (a) x-ray and (b) cross-section.

The GLARE-3-31 specimen shown in Figures 36 and 37 developed matrix cracking from the impact. Also, some of the individual fibres separated from the rest of the prepreg layer, with some adhesive left on the aluminum surface (Figure 36). This fibre-bridging was typical of impacted GLARE specimens.

Figure 36: GLARE-3-31 (15.73 J) fibre delamination.

The x-ray of GLARE-3-33 (36.42 J) in Figure 37a, revealed delamination damage from the impact. In this case the cross-section (Figure 37b) agreed with the x-ray. The sample also had matrix cracking (Figure 38) that is not visible in the x-ray. This cracking occurred between the fibres of the 90° layer and no damage is apparent in the 0° layer.

Figure 37: GLARE-3-33 (36.42 J) (a) x-ray and (b) cross-section.

Figure 38: Matrix cracking in sample GLARE-3-33 in the 90° layer.

Sample GLARE-4-08 (47.03 J) shown in Figure 39, was impacted with approximately the same energy level as sample GLARE-3-34 (46.72 J). However, the GLARE-4 sample developed a through-crack in addition to the delamination damage as shown in Figure 40. There was also matrix cracking as in GLARE-3-34. The fibres in the 0° direction were fractured when the through-crack formed. The panel suffered extensive delamination (Figure 40a) and matrix cracking (Figures 40b and 41). The

offset between the two halves of the crack occurred after the specimen was cut from the impacted panel; there was a release of stored elastic energy. It was not possible to determine if this release of energy could have increased the size of the delaminated region in these specimens.

Figure 39: GLARE-4-08 (47.03 J) cross-section.

Figure 40: Through-crack in GLARE-4-08 (47.03 J).

Figure 41: Matrix cracking in GLARE-4-08 (47.03 J)

Sample GLARE-5-04 (57.74 J) showed considerable delamination damage (Figures 42a and 42b). In this case the x-ray (Figure 42a) and the cross-section (Figure 42b) agreed. Where the dent starts to flatten out, there is significant delamination (Figures 43a and 43b). This sample also contained matrix cracks (Figure 44).

Figure 42: GLARE-5-04 (55 J) (a) x-ray and (b) cross-section.

Figure 43: Delamination and fibre-bridging in GLARE-5-04 on (a) left and (b) right of dent.

Figure 44: Matrix cracking in GLARE-5-04.

Sample GLARE-5-12 (36.47J) shown in Figure 45 developed delamination damage that was not apparent in the x-rays. This specimen provides a clear example of the type of matrix cracks that were typical of all the specimens that were inspected (Figure 46).

Figure 45: GLARE-5-12 (35 J) (a) x-ray and (b) cross-section.

Figure 46: GLARE-5-12 matrix cracking and top 90° layer adhesive failure.

Interlaminar shear forces between the aluminum and prepreg layers are responsible for the onset of delamination damage. As the impactor rebounds, the plastically deformed aluminum layers pull away from one another, causing regions of delamination to open up around the dent. Mode I delamination dominates the propagation of damage in impacted GLARE panels (Vlot, 1993b). However, initiation of the damage is controlled by mode II fracture strengths. This initial damage provides the initiation point for the development of mode I delamination.

Table 10 summarizes the key results of the impact testing of GLARE-3, GLARE-4 and GLARE-5. Aluminum 2024-T3 is not presented in this table since puncture did not occur in any of these specimens. The GLARE-5 laminates required higher energy for the onset of initial damage than the other two GLARE varieties. This material also showed higher energy for the onset of complete puncture.

A large number of experiments (162 total) were required to generate the impact characterization results described in the preceding sections. Future FML materials should be screened using impact simulations to reduce the testing requirements

Table 10: Summary of impact test results for GLARE laminates.

	Units	Material		
		GLARE-3	GLARE-4	GLARE-5
Areal density	kg/m²	2.168	2.418	2.688
Impact force to initiate internal damage	kN	5.5	5.8	8.5
Energy of initiation of internal damage	J	25	25	45
Peak strain at initiation (X-dir gauge 4)	μϵ	4500	4500	7050
Peak strain at initiation (Y-dir gauge 1)	μϵ	4200	2800	6400
Impact force to initiate partial puncture	kN	No partial puncture	7.1	9.6
Energy of initiation of partial puncture	J		35	58
Peak strain at initiation (X-dir gauge 4)	μϵ		12000	8200
Peak strain at initiation (Y-dir gauge 1)	μϵ		5400	7800
Impact force for complete puncture	kN	7.9	8.2	10.3
Initiation energy for complete puncture	J	46	51	65
Peak strain at puncture (X-dir gauge 4)	μϵ	4800	10000	6300
Peak strain at puncture (Y-dir gauge 1)	μϵ	7800	6500	6300

CHAPTER 3

FML IMPACT MODELLING

METHODOLOGY

Based on the results of the low-velocity impact tests described in the previous Chapter, it was decided to develop a methodology to model low-velocity impacts on FML panels. This Chapter presents an overview of the planned development of this methodology. A review of modelling techniques that have previously been applied to FMLs is first presented. This review revealed the capabilities and limitations of existing modelling techniques. Next, a conceptual overview of the proposed modelling methodology and its components is provided in Section 3.2. Finally, a description of some of the key features of the explicit finite element code LS-DYNA is presented in Section 3.3. This code was chosen to model low-velocity impact in the present work. Subsequent Chapters will describe the implementation of each component of the methodology, culminating with its application to predict the impact response of GLARE presented in Chapter 7.

3.1 Review of Modelling of FMLs

Efforts have been made by researchers over the last decade to model the impact response of FMLs using either analytical or finite element methods. Table 11 presents a summary of recent modelling efforts and also lists their applications, capabilities and limitations. Several of the more important and potentially useful methods will be discussed in greater detail. However, none of these models is comprehensive. For example, some do not account for delamination while others neglect the loss of stiffness through the formation of microcracks in the prepreg. The experimental results presented in the previous Chapter showed that both of these damage modes occur in impacted FML panels. Therefore, a representation of these damage modes is required to predict the impact response of FMLs. Finally, a review of delamination prediction models for FMLs and composites will be presented. These delamination methods were not all developed specifically for impact simulations.

Table 11: Summary of modelling techniques applied to FMLs.

Reference	Application	Type	Capabilities	Limitations
Hurez <i>et al.</i> , 1993	Static loading simulations	FEA	Damage in the prepreg,	No delamination
Hashagen <i>et al.</i> , 1995	Static loading simulations	FEA	Elastic-plastic behaviour of aluminum	No delamination
Hashagen <i>et al.</i> , 1999	Delamination fatigue	FEA	Delamination included	No plasticity, prepreg elastic up until failure,
Vlot, 1998b	Impact	Non-linear Analytical	Elastic plastic behaviour of aluminum	No delamination, assumed elastic prepreg
Vlot, 1993a	Impact	Linear Analytical	Can model contact forces	No delamination, assumed elastic prepreg
Sun <i>et al.</i> , 1993	Impact	Non-linear Analytical	Can model contact forces, can model dent depth	No delamination, assumed elastic prepreg

3.1.1 Finite Element Modelling of FMLs

In 1993, Hurez *et al.* described an elastic-plastic model incorporating damage for static loading cases. A membrane-bending finite element was employed that included transverse stresses. Each layer in the laminate was then modelled either as elastic-plastic with strain hardening (metal) or transversely isotropic with damage (prepreg). The behaviour of the aluminum layers was modelled using the following threshold for the onset of plasticity:

$$\bar{\sigma} - \sigma_{y\text{current}} = 0 \quad (5)$$

where $\bar{\sigma} = \sqrt{3J_2}$ is the equivalent von Mises stress and $(\sigma_{y\text{current}})$ is the current yield stress taking into account isotropic strain hardening. Incorporating a damage prediction for the prepreg layers into the model is somewhat more problematic. Hurez *et al.* based their approach upon the elementary ply damage mechanics theory developed by Ladevèze and Le Dantec (1992) for use with composite materials. This theory is described in greater detail in Section 6.1 covering continuum damage mechanics for FMLs.

Hashagen *et al.* (1995) described a geometrically and materially non-linear thick-shell element that was developed for FMLs. This element employs 16 nodes at the top and bottom surfaces. The von Mises isotropic yield criterion is employed for the aluminum layers. The prepreg layers are modelled using the Hoffman yield criterion. This criterion employs terms that vary linearly with the stress so that differences in tensile and compressive response can be properly described. Several benchmark tests from literature were modelled by Hashagen *et al.* (1995) using their shell element. The benchmark results agreed with those obtained using conventional thin-shell elements.

The main shortfall of the previous two models is the lack of delamination damage in the predictions. Hashagen *et al.* (1999) described a technique for modelling delamination in spliced FML panels. Spliced FMLs have been designed to reduce the requirement for mechanically fastened (i.e. riveted) joints in large diameter aircraft fuselages. The splice (Figure 47) overcomes the limitations placed on the production of FML panels by the available width of aluminum sheets. The aluminum layers are reinforced with continuous prepreg layers, resulting in an efficient structural joint.

Figure 47: A basic FML splice concept (not to scale).

Hashagen *et al.* (1999) used this model to predict delamination growth in spliced panels tested previously by de Vries *et al.* (1999). Once again, Hashagen *et al.* used thick-shell elements to model the laminates. A layer of interface elements was inserted between the layers of aluminum and prepreg. A Hoffman-like delamination model was introduced that used a yield function to predict the failure of the interface. This approach assumed propagation of delamination from an initial flaw. It was also assumed that the delamination was dominated by mode II crack growth. With this assumption, the numerical results agreed well with experiments for double-sided splice joints. However, there were some differences between the results for single-sided, asymmetrical splices. There was significant out-of-plane bending experienced by this configuration. This

resulted in mixed-mode delamination where the mode I contribution was no longer negligible. Due to this geometrically non-linear behaviour, the model could not accurately predict delamination growth in the single-sided splice.

3.1.2 Analytical Modelling of Impact on FMLs

In addition to finite element methods there are also several closed-form analytical techniques that have been applied to impact in FMLs. Vlot (1996 and 1998) developed a model to describe the deflection of a GLARE panel under low-velocity impact loading. One of the key assumptions was that the deflection-energy curves are approximately equal for quasi-static and low-velocity impact loading. At low strain levels the deformation of the panel is dominated by bending. Higher order non-linear terms that relate to the stretching of the material must be taken into account for larger deflections. In Equations 6 to 8, Vlot employed the Von Kármán relationships for the assumption of small strains and moderate deflection levels in a circular plate.

$$\varepsilon_r = \frac{du}{dr} + \frac{1}{2} \left[\left(\frac{du}{dr} \right)^2 + \left(\frac{dw}{dr} \right)^2 \right] - z \frac{d^2w}{dr^2} \quad (6)$$

$$\varepsilon_\theta = \frac{u}{r} - \frac{z}{r} \frac{dw}{dr} \quad (7)$$

$$\gamma_{r\theta} = 0 \quad (8)$$

where, w is the general out-of-plane deflection of the circular area, u is the in-plane displacement, and r , z and θ are the radial, out-of-plane and angular directions. The deflection shape function that Vlot employed, Equation 9, satisfies the fully clamped boundary conditions of the circular impacted area.

$$\xi = (1 - \eta^2)^2 (1 - 1.2\eta + 1.2\eta^2) \quad (9)$$

where $\xi = w/w_o$ is the normalized deflection and $\eta = r/a$ is the normalized relative position from the centre of the panel. In this case w_o is the central deflection of the panel. Now, it is shown in Equations 10 to 12 that the Von Kármán relations can be rewritten in terms of normalized co-ordinates.

$$\varepsilon_r = \frac{w_o^2}{2a^2} \left(\frac{d\xi}{d\eta} \right)^2 - \frac{zw_o}{a^2} \frac{d^2\xi}{d\eta^2} \quad (10)$$

$$\varepsilon_\theta = -\frac{zw_o}{a^2\eta} \left(\frac{d\xi}{d\eta} \right) \quad (11)$$

$$\gamma_{r\theta} = 0 \quad (12)$$

where, ε_r is the in-plane strain in the radial direction, ε_θ is the in-plane strain in the circumferential direction and $\gamma_{r\theta}$ is the in-plane shear strain. Vlot employed the effective strain and a bi-linear stress-strain curve for the aluminum alloy to calculate the elastic-plastic energy by integrating over the aluminum volume of the circular plate. The effective strain in the metallic layers is given as:

$$\bar{\varepsilon}^2 = \frac{4}{3} [\varepsilon_r^2 + \varepsilon_\theta^2 + \varepsilon_r \varepsilon_\theta] \quad (13)$$

The elastic-plastic energy stored in the metallic volume is given in two parts, the energy from elastic deformation and plastic deformation:

$$\begin{aligned} dU_e &= \frac{1}{2} E \bar{\varepsilon}^2 \text{ for } \bar{\varepsilon} < \frac{\sigma_y}{E} \\ dU_p &= \frac{1}{2} \alpha \bar{\varepsilon}^2 + \left[1 - \frac{1}{2} \frac{\alpha}{E} \right] \frac{\sigma_y}{E} - \frac{1}{2} \frac{\sigma_y^2}{E} \left[1 - \frac{\alpha}{E} \right] \text{ for } \bar{\varepsilon} > \frac{\sigma_y}{E} \end{aligned} \quad (14)$$

In Equation 14 α is the central deflection of the panel. The total strain energy in the metal layers is obtained by employing the integral shown in Equation 15. The integration is then performed through a numerical summation of the strain energy for small elements over the entire plate. First, the mean radial and tangential strains are calculated from Equations 10 to 12 with the known shape of the dent.

$$U_p = \int_z \int_r dU_p 2\pi r dr dz \quad (15)$$

Next, the strain in the prepreg layers is given in Equation 16.

$$\varepsilon_f = \varepsilon_r \cos^2 \theta + \varepsilon_\theta \sin^2 \theta \quad (16)$$

The key assumptions that Vlot (1996) made at this point are that the prepreg layers deform elastically and that all of the strain energy is stored in the glass-fibres. The elastic deformation assumption appeared to be valid since the fibres yield at a significantly higher stress level than the aluminum. The resulting integration required to find the total strain energy stored in the fibre layers is shown in Equation 17.

$$\begin{aligned} U_f &= \int_V \int \int dU_f r d\theta dz = \int_V \int \int \frac{1}{2} E_f [\varepsilon_f(\theta)]^2 r d\theta dz \\ &= \int_A \int \frac{1}{2} \pi E_f \left[\frac{3}{4} \varepsilon_r^2 + \varepsilon_r \varepsilon_\theta + \frac{3}{4} \varepsilon_\theta^2 \right] r dr dz \end{aligned} \quad (17)$$

where E_f is the modulus of the prepreg layer in the fibre direction.

The integral in Equation 17 is computed in the same manner as that shown in Equation 15 for the aluminum layers. The contact force between the impactor and the panel is then found to be:

$$F = \frac{dU}{dw_0} \approx \frac{U(w_0 + \delta) - U(w_0)}{\delta} \quad (18)$$

where $U = U_p + U_f$ and δ is a small value, e.g., $0.0001w_0$ as suggested by Vlot (1998). No explanation for the selection of this value was provided. This model was implemented in

a computer program called EPIM (Elastic-Plastic Impact Model) and the results were compared to numerous experiments, first with traditional composites (Vlot, 1996) and later with FMLs (Vlot, 1998). In both comparisons the model estimated the stiffness of the panels with good agreement to the tests. The failure of the prepreg was estimated using the Tsai-Wu stress criterion. The main deficiencies identified by Vlot were that the model does not incorporate a failure criterion to determine the first failure energy level. Also, it cannot be used to estimate the permanent dent depth after impact or delamination.

Another analytical approach for modelling the impact behaviour in ARALL has been developed by Sun *et al.* 1993 and Wu and Sun, 1993. A contact law for impact loading to estimate the force response of the impacted panels was proposed. They then used a series of static indentation tests to develop best-fit parameters for the relations. The contact force, F , is:

$$F = k\alpha^n \quad (19)$$

where k is the contact rigidity, α is the indentation depth and n is a power index. Next, the following method was used to account for the permanent deflection of metal panels subjected to impact.

$$F = F_m \left(\frac{\alpha - \alpha_o}{\alpha_m - \alpha_o} \right)^q \quad (20)$$

The above equation is used to model the unloading path of the impact event. The parameters n and q are typically determined from static indentation experiments. Equation 20 can be rearranged as shown in Equation 21.

$$F = S(\alpha - \alpha_o)^q \quad (21)$$

The unloading rigidity, S , is given in Equation 22.

$$S = \frac{F_m}{(\alpha_m - \alpha_o)^q} \quad (22)$$

It was also assumed that there is a critical indentation value, α_p , below which there would be no permanent deformation as shown in Equation 23.

$$\alpha_o = \begin{cases} S_p (\alpha_m - \alpha_p) & \alpha_m > \alpha_p \\ 0 & \alpha_m \leq \alpha_p \end{cases} \quad (23)$$

The parameters k , q , S_p and α_p were determined using a 19.05 mm x 25.4 mm specimen loaded statically. These values are read directly from loading and unloading force deflection curves from static indentation tests. This method is very limited in its applicability and has no capability for predicting delamination or other internal damage in FMLs. It can only be applied to estimate the contact force in the impacted panel.

3.1.3 Delamination Damage Modelling for FMLs

Current FML delamination research is mainly focused on delamination damage associated with spliced and notched FML coupons. At present there appears to be no work published describing delamination models for application to impact in FMLs. Delamination in FMLs can be influenced by geometry and material discontinuities, and consists of mode I, II and III fracture behaviour (Figure 48).

Figure 48: Schematic of fracture modes.

The main focus of this research has been delamination growth associated with the formation of fatigue cracks in the aluminum layers of FML sheets. This type of delamination is governed primarily by the mode II fracture toughness of the FML material. Table 12 summarizes current research. Initiation of delamination in GLARE was not covered in any of these methods. Lin and Kao (1996) and Guo *et al.* (1997) present in-depth approaches for predicting delamination growth. Other researchers have examined aspects of delamination behaviour in FMLs and their work will also be discussed in this review.

Table 12: Summary of FML delamination research to date.

Reference	Material	Delamination Type	Specimen Type	Numerical Model Type	Comments
Hashagen <i>et al.</i> (1999)	GLARE-2	Delamination growth in a splice	Numerical only	FE model with interface elements	Growth only – mode II
Vlot <i>et al.</i> (1998)	GLARE and ARALL	Delamination growth in bonded FML doublers	Doubler specimen (cyclic) DCB and ENF	None used	Effect of post-stretching on delamination and mixed mode
Guo <i>et al.</i> (1997)	GLARE-2	Characterization of delamination growth	DCLS	Strain energy release rate	Growth only – Mode II
Lin and Kao (1996)	CARALL	Characterization of delamination growth	SEN (cyclic)	Strain energy release rate	Growth only – mode II
Yeh <i>et al.</i> (1988)	ARALL	Fracture mechanics of delamination	Numerical only	Stress singularity at delamination tip	Numerical only, examined energy release rates – Mode I and Mode II separately

Lin and Kao (1996) examined the effects of delamination on the propagation of fatigue cracks in CARALL specimens. They used a strain energy release approach to model the growth of the delamination front around the cracked region with a single-edge notched (SEN) specimen. This specimen is divided into several regions corresponding to the physical state of the panel under load. To predict the delamination they developed an expression for the effective stress intensity range (ΔK_{eff}) for the CARALL material. This model was used to predict the crack growth behaviour of CARALL sheets with an initial crack (a_o) of 6 mm. Experiments were employed to determine the delamination size corresponding to this initial flaw size. With a known crack growth rate for a_o , the corresponding effective stress intensity range can be found from the Paris Equation. The results of the analytical simulations compared well to experiments using SEN specimens in several different laminate configurations. This research demonstrated that the strain energy release approach for predicting delamination growth can be applied to FMLs.

A slightly different approach was employed by Guo *et al.* (1997) for predicting the delamination growth behaviour in a double crack lap shear (DCLS) specimen of GLARE-2-2/1. The specimen was manufactured with an electric-discharge machined slit in each of the two aluminum layers. Cracks propagate through the aluminum layers to the fibre core. The damage then grows in the form of a delamination front along the interface between the aluminum and fibre layers. Guo *et al.* found that the compliance of the DCLS specimen increased linearly with the delamination size. They also observed that the growth of the delamination was predominately controlled by mode II behaviour, with some mode I influence. The energy release rate method was chosen to predict the delamination growth behaviour. Once this expression for the energy release rate was developed, a series of fatigue tests with the DCLS specimen for several stress ratios ($R=0.1, 0.3$ and 0.5) was performed. The tests showed good agreement between the experimental data and the energy release rate approximations.

In 1988, Yeh *et al.* examined the fracture mechanics of delamination in ARALL laminates. Singular finite elements were used to model the delamination crack tip. This allowed the determination of the stress intensity factors and energy release rates for mode I and mode II delamination. It was found that mode II was the dominant form of delamination growth for the DCLS configuration. The stress intensity factors and energy release rates were independent of the size of the delamination. The predictions were compared to the experimental results of Awerbuch (1987) with good agreement.

Hashagen *et al.* (1999) carried out numerical investigations of the delamination behaviour of spliced FMLs based on the characterization experiments carried out by Vries *et al.* (1999) as described in Section 3.1.1.

3.1.4 Delamination Modelling for Composites

There are also numerous models and approaches in literature for the prediction of delamination initiation in fibre-reinforced composites. Several of these models are reviewed below. These methods are different from the ones described previously in that they predict the initiation of delamination damage, not its growth.

Brewer and Lagace (1988) developed the Quadratic Delamination Criterion (Equation 24) for estimating the initiation of delamination damage in laminated composites. This model uses an average stress criterion to compare the calculated out-of-plane interlaminar stresses to the appropriate material strength parameters.

$$\left(\frac{\bar{\sigma}_{13}}{s_{13}}\right)^2 + \left(\frac{\bar{\sigma}_{23}}{s_{23}}\right)^2 + \left(\frac{\bar{\sigma}_{33}^t}{s_{33}^t}\right)^2 + \left(\frac{\bar{\sigma}_{33}^c}{s_{33}^c}\right)^2 = 1 \quad (24)$$

The superscripts t and c in the above equation denote tensile and compressive stresses.

Another strength of materials approach was developed by Choi and Chang (1992). Their criterion assumes that delamination initiates from matrix cracks within the layers. They predict the occurrence of a critical matrix crack with a criterion of the form:

$$\left(\frac{\bar{\sigma}_{22}}{s_{22}}\right)_n^2 + \left(\frac{\bar{\sigma}_{23}}{s_{23}}\right)_n^2 \geq 1 \quad (25)$$

where the subscript n refers to the n^{th} ply layer in the specimen. When the critical matrix crack size in a given layer has been reached, Choi and Chang proposed the following delamination criterion:

$$D_a \left[\left(\frac{\bar{\sigma}_{23}}{s_{23}}\right)_m^2 + \left(\frac{\bar{\sigma}_{13}}{s_{13}}\right)_m^2 + \left(\frac{\bar{\sigma}_{22}}{s_{22}}\right)_m^2 \right] \geq 1 \quad (26)$$

where the value D_a is an empirical constant determined through experiments.

A method for predicting the failure of an interface based upon a strength-of-materials approach was described by Chang and Springer (1987). This formula, shown in Equation 27, compares the normal and shear stresses applied to the interface with the interlaminar normal and shear strength parameters (s_n and s_s). The strength parameters are obtained from either manufacturer's data or interface strength tests.

$$\left(\frac{|\sigma_n|}{s_n}\right)^2 + \left(\frac{|\sigma_s|}{s_s}\right)^2 \geq d^2$$

if $d < 1$ then there is no failure

if $d \geq 1$ then there is complete failure

(27)

3.2 Proposed Modelling Methodology

The experimental testing described in Chapter 2 demonstrated the impact response of GLARE under specific conditions. However, these experiments were limited to only three types of GLARE, with only a single boundary condition. In service these materials could be subjected to many types of impact events. As well, the lay-up and boundary conditions could vary from location to location on an aircraft. The number of experiments required to adequately characterize the response of these materials would be large. Accurate impact damage simulations could reduce this testing requirement.

The review presented in the previous section showed that there is, as yet, no practical methodology for predicting impact induced damage in FMLs that incorporates all of the important damage modes. Such a methodology must be able to predict not only the measured response (absorbed energy, force and dent depth) but also the geometry of the damage developed in the panel. However, some aspects of the previously developed

models can be applied to the problem of predicting impact damage in FMLs. For example, finite element methods offer advantages over analytical techniques. Finite element methods can be used to model many types of geometries and boundary conditions by changing the input parameters. As well, finite element models can be scaled up to simulate entire components and structures. Multi-layered models can be constructed using finite elements with a damageable interface between the layers. Some commercial finite element codes, such as LS-DYNA, also make it possible to implement user-defined material models directly as part of the code. The individual materials that compose the GLARE laminates can thus be characterized separately before incorporation in the finite element model. The analytical models presented previously, required characterization of each new laminate type. As well, analytical models are usually developed for a specific geometry and loading condition and this limits their usefulness. However, analytical models are generally computationally efficient for simple problems.

The proposed damage modelling methodology requires four components to predict the impact response of the FML panels as shown in Figure 49. The aluminum material model must be able to accurately predict the elastic and plastic response of the metal layers under impact loading. The next component is the damage model for the prepreg layers. This model is required to capture the effect of matrix cracking on the impact response. The final two components, the delamination initiation and post-delamination models, are closely related. Without an accurate initiation model, delamination damage may be predicted too early, too late or not at all in the impact simulations. The post-delamination behaviour model is required to predict the effects of the fibre-bridging that was observed in the experiments described in Section 2.2.4.

Figure 49: Required components of the FML damage modelling methodology.

Matrix plasticity and strain-rate sensitivity were shown to have a minor effect on the response of FMLs subjected to low-velocity impact (Vlot, 1993b). These effects will not be incorporated at this time. Also, the current objective is to model FMLs below the puncture threshold. Since fibre fracture was only observed in punctured specimens, a prediction for this damage mode will not be employed as part of this methodology.

Following the development and validation of the individual components, the damage modelling methodology was used to model low-velocity impacts on GLARE as described in Chapter 7. These simulations used energy levels identical to those employed for the experiments described in Chapter 2. Figure 50 shows schematically the relationship between the various tasks completed as part of the development of this methodology. As indicated in the flowchart, some of the required components already exist in the explicit finite element code LS-DYNA.

Figure 50: Modelling methodology components and task relationships.

3.3 Overview of the LS-DYNA Explicit

Finite Element Code

To develop a practical methodology for modelling the damage formed in FMLs, a commercial finite element code was used instead of developing a stand-alone, single purpose program. One commercial code that is well suited for modelling impact and non-linear problems is LS-DYNA. This code is a general purpose, vectorized, explicit finite element program. It has the capability to model 2D and 3D problems with large deformations. LS-DYNA was developed at the Livermore National Laboratory (Hallquist, 1998) for explosives and crashworthiness simulations. Its applicability has expanded to include low-velocity impact and metal forming among other applications. This code has been successfully applied to model the behaviour of composites under impact by other researchers (van Hoof, 1999, Hung, 1995, Majeed, 1994, Williams and Vaziri, 1995 and Williams and Vaziri, 2001).

The central difference method is used to solve the equations of motion with respect to time. This explicit finite element method uses time steps rather than load steps. Each time step must be smaller than a critical value to ensure the stability of the solution. The critical time step is a function of a key dimension of the element such as the length of the shortest side, and the speed of sound in the material. The sound wave should not cross the element in one time step. The smallest time step calculated from all of the elements is used as the critical time step for the integration. This time step is recalculated with each subsequent solution of equations.

There are three types of elements that were used in this thesis and each has distinct properties. The four-node thin-shell (Belytschko-Tsay) elements require the least amount of CPU time and are useful for simulating thin panels. The eight-node thick-shells are actually a hybrid of thin-shells and solid-brick elements. However, because they are 3D they are particularly useful in modelling multi-layered thin panels with discrete interlaminar interfaces. Eight-node solid-brick elements are also useful in modelling multi-layered panels but require larger amounts of CPU time. They are not suited for thin panels unless a large number of elements is used to keep the aspect ratio below 10:1. Each element type is discussed in more detail below.

Another important feature of LS-DYNA is its use of contact algorithms to model the interfaces between parts. These can be tied-, tiebreak- or sliding-interfaces. Static and kinetic coefficients of friction can also be defined for the interface. All of these types of interface are used in this thesis. Version 950e of LS-DYNA was employed in this project and, unlike older versions, it makes use of keywords to activate the different features. Table 13 gives the keywords and presents how they are referred to in this thesis. Each of these features is discussed in greater detail in the sections to follow.

Table 13: Important features of LS-DYNA used in the simulation of GLARE and aluminum subjected to low-velocity impact.

Keyword	Referred to as:
*MAT_ELASTIC	elastic (type 1)
*MAT_ORTHOTROPIC_ELASTIC	orthotropic-elastic (type 2)
*MAT_RIGID	rigid (type 20)
*MAT_PIECEWISE_LINEAR_PLASTICITY	elastic-plastic (type 24)
*MAT_USER_DEFINED_MATERIAL_MODELS	UMAT
*CONTACT_TIED_SURFACE_TO_SURFACE	tied-interface
*CONTACT_TIEBREAK_SURFACE_TO_SURFACE	tiebreak-interface
*CONTACT_AUTOMATIC_SURFACE_TO_SURFACE	
*ELEMENT_TSHELL	thick-shell
*ELEMENT_SHELL	thin-shell
*ELEMENT_SOLID	solid-brick

As with any finite element code, it is important to use consistent units for all the input parameters. The basic units that are selected by the user should be on the same order of magnitude as the basic dimensions of the structure under analysis. Table 14 lists the units for each of the major parameters used in this thesis.

Table 14: Units used in LS-DYNA simulations.

Length	millimeter (mm)
Time	millisecond (ms)
Mass	kilogram (kg)
Force	kiloNewton (kN)
Stress	$\text{kN/mm}^2 = \text{GPa}$
Velocity	$\text{mm/ms} = \text{m/s}$
Acceleration	mm/ms^2
Energy	$((\text{kg} \cdot \text{mm})/\text{ms}^2) \cdot \text{mm} = \text{J}$
Density	kg/mm^3

3.3.1 Thin-Shell Elements

The Belytschko-Lin-Tsay (Belytschko *et al.* 1984) element is the default thin-shell formulation implemented in LS-DYNA. This element, which is based upon a combined co-rotational and velocity-strain formulation, was developed to optimize computational efficiency. An embedded element coordinate system is defined in terms of the reference surface of the element (Hallquist, 1998). This coordinate system deforms with the shell element. With this formulation the element strain is restricted to small values but the out-of-plane deformations are not. The velocity-strain displacement relations are also restricted to small strains. The Belytschko-Lin-Tsay element follows the Mindlin theory of plates and shells to partition the displacement of any point in the shell to a mid-surface displacement (nodal translation) and a displacement associated with the rotation of the element's fibres (nodal rotations).

In LS-DYNA, contact interfaces are created by defining master and slave surfaces. Thin-shell elements only have one layer of nodes; therefore, nodes must be defined as being part of a master surface and a slave surface simultaneously. This can cause numerical problems that can be avoided if thick-shell elements or solid-brick elements are used.

As in most finite element codes, thin-shells in LS-DYNA can be used to represent composite plies by defining multiple integration points in the element. In most codes, the orientation and material properties of each ply can be assigned to model multi-directional laminates. In LS-DYNA one can only assign different ply angles to each integration point. It is not possible to assign different material properties to each point. This reinforces the unsuitability of the thin-shells in LS-DYNA for modelling FMLs.

3.3.2 Thick-Shell Elements

The thick-shell element is recommended for applications where thick-walled shell structures are to be modelled. Also, this element permits the modelling of multi-layered structures so that failure of the interface between different layers can be predicted. These elements also minimize potential problems with defining the contact interfaces that can arise when using thin-shell elements. Thick-shell elements are geometrically identical to traditional 8-node solid-bricks with four nodes on the top and bottom surfaces. Each group of four nodes can be assigned to a separate contact segment for the top and bottom surfaces of the layer. This avoids the problem of assigning one node to two different contact segments simultaneously.

The geometry of the element is interpolated in the same manner as the eight-node solid element (Hallquist, 1998):

$$x_i(X_\alpha, t) = x_i(X_\alpha(\xi, \eta, \zeta), t) = \sum_{j=1}^8 \phi_j(\xi, \eta, \zeta) x_i^j(t) \quad (28)$$

The shape functions for this element are defined as:

$$\phi_j = \frac{1}{8} (1 + \xi \xi_j)(1 + \eta \eta_j)(1 + \zeta \zeta_j) \quad (29)$$

A schematic of this element is shown in Figure 51. The co-ordinates of the nodes take on the values of $(\pm 1, \pm 1, \pm 1)$ with x_i^j as the nodal co-ordinate of the j th node in the i th direction. The interpolation matrix, N , for the solid-like shell element is similar to that for solid elements:

$$N(\xi, \eta, \zeta) = \begin{bmatrix} \phi_1 & 0 & 0 & \phi_2 & 0 & \dots & 0 & 0 \\ 0 & \phi_1 & 0 & 0 & \phi_2 & \dots & \phi_8 & 0 \\ 0 & 0 & \phi_1 & 0 & 0 & \dots & 0 & \phi_8 \end{bmatrix} \quad (30)$$

The stress vector, σ , for the element is given by:

$$\sigma^t = (\sigma_{xx}, \sigma_{yy}, \sigma_{zz}, \sigma_{xy}, \sigma_{yz}, \sigma_{zx}) \quad (31)$$

Finally, the 6 x 24 strain-displacement matrix, B , is expressed as:

$$B = \begin{bmatrix} \frac{\partial}{\partial x} & 0 & 0 \\ 0 & \frac{\partial}{\partial y} & 0 \\ 0 & 0 & \frac{\partial}{\partial z} \\ \frac{\partial}{\partial y} & \frac{\partial}{\partial x} & 0 \\ 0 & \frac{\partial}{\partial z} & \frac{\partial}{\partial y} \\ \frac{\partial}{\partial z} & 0 & \frac{\partial}{\partial x} \end{bmatrix}^N \quad (32)$$

Next the terms within the strain-displacement matrix are found as:

$$\begin{bmatrix} \frac{\partial \phi_i}{\partial \xi} \\ \frac{\partial \phi_i}{\partial \eta} \\ \frac{\partial \phi_i}{\partial \zeta} \end{bmatrix} = \begin{bmatrix} \frac{\partial x}{\partial \xi} & \frac{\partial y}{\partial \xi} & \frac{\partial z}{\partial \xi} \\ \frac{\partial x}{\partial \eta} & \frac{\partial y}{\partial \eta} & \frac{\partial z}{\partial \eta} \\ \frac{\partial x}{\partial \zeta} & \frac{\partial y}{\partial \zeta} & \frac{\partial z}{\partial \zeta} \end{bmatrix} \begin{bmatrix} \frac{\partial \phi_i}{\partial x} \\ \frac{\partial \phi_i}{\partial y} \\ \frac{\partial \phi_i}{\partial z} \end{bmatrix} = J \begin{bmatrix} \frac{\partial \phi_i}{\partial x} \\ \frac{\partial \phi_i}{\partial y} \\ \frac{\partial \phi_i}{\partial z} \end{bmatrix} \quad (33)$$

As with typical finite element methods we must then invert the Jacobian matrix, J , and solve for the desired terms to give:

$$\begin{bmatrix} \frac{\partial \phi_i}{\partial x} \\ \frac{\partial \phi_i}{\partial y} \\ \frac{\partial \phi_i}{\partial z} \end{bmatrix} = J^{-1} \begin{bmatrix} \frac{\partial \phi_i}{\partial \xi} \\ \frac{\partial \phi_i}{\partial \eta} \\ \frac{\partial \phi_i}{\partial \zeta} \end{bmatrix} \quad (34)$$

Figure 51: Nodal locations for the LS-DYNA thick-shell element.

A reference surface is located midway between the upper and lower surfaces of the element and is constructed in a manner identical to the Belytschko-Lin-Tsay thin-shell element in LS-DYNA (Belytschko *et al.* 1984). A local co-ordinate system is constructed for the reference surface that deforms with the element. Similar to other elements that use single-point integration in LS-DYNA, thick-shell elements require the use of hourglass control. This will be discussed in greater detail in Section 3.3.4.

3.3.3 Solid-Brick Elements

The basic formulation of the standard LS-DYNA solid-brick element is similar to that of the thick-shell element as described in Equations (28-34). The main difference is that this element does not use a reference surface. The element uses a single-point integration scheme that significantly reduces computation time over fully-integrated elements (Hallquist, 1998). However, this reduced integration scheme creates the requirement for hourglass control.

3.3.4 Hourglass Control

Hourglass control is necessary whenever element formulations are employed that are not fully integrated, as is the case with the thick-shells and solid-bricks in LS-DYNA. Hourglass modes are zero-energy deformation modes that have a period much shorter than the structural response period. These modes are orthogonal to the strain calculations in the element and as such there are no internal forces to resist the deformations (Flanagan and Belytschko, 1981). They arise due to the use of single point integration in the finite elements.

There are two methods for resisting the formation of these modes in LS-DYNA. The first, referred to as the “stiffness method”, employs a small elastic stiffness that resists the anomalous modes. The stiffness method has been identified as being suitable for low-velocity simulations. This method slows, stops and undoes the hourglass deformation modes. In large deformation problems, this method may over-stiffen the model. The second method, known as the “viscosity method”, is the default hourglass control method in LS-DYNA. With this method, the hourglass deformations are slowed and stopped but not undone. However, the viscosity method is recommended in high-velocity problems with large deformation and was used in this thesis.

Despite the problems caused by hourglassing modes, single point integration can result in substantial computational savings. For example, compared with full integration, single-point integration reduces computational time by a factor of 16 for strain and nodal force computations and by a factor of 8 for stresses (Hallquist, 1998). The computational cost of implementing hourglass control measures is minor compared to the cost of using fully-integrated elements in explicit finite element analysis (Hallquist, 1998).

3.3.5 Interface Models

The LS-DYNA code incorporates four distinct methods of modelling the contact between two surfaces. Hallquist (1998) described these algorithms. The first method of contact definition is referred to as the “kinematic constraint method”. This algorithm is currently only employed in tied-interfaces, as in the interface between the prepreg and the aluminum layers of GLARE. In this method the displacements of the slave nodes are tied to the master surface through a global transformation of the nodal displacements. This effectively eliminates the normal component of displacement for these nodes. Momentum is conserved by imposing impact and release constraints as described by Hughes *et al.* (1976). The kinematic constraint method is used for both the tied- and tiebreak-interfaces in LS-DYNA.

The second method of contact modelling is the “penalty method”. The interpenetration of the surfaces in contact is resisted by inserting springs between the surfaces. The springs resist penetration and force the surfaces apart. This method of contact simulation is used in surface-to-surface contact with friction, as takes place between the impactor and the panels.

The remaining two contact algorithms are the “surface to surface constraint method” and the “distributed parameter method”. Both are used in very limited applications. For example, the distributed parameter method is used to model only sliding in fluid-structure or gas-structure interaction problems.

These contact algorithms are used to model the interfaces between parts in several ways. Groups of elements can be connected using a **tied-interface** that constrains rotations and displacements of the slave nodes with the master surface. It is also possible

to model a sliding interface between two surfaces with or without friction. This type of interface can be used to model the contact between the impactor and the panels.

It is possible to model the failure and propagation of interlaminar failures in LS-DYNA using tiebreak contact based on the kinematic constraint method. The failure of the interface is predicted using the relationship developed by Chang and Springer (1986) as presented in Equation 27 of Section 3.1.4. Once the nodes are released, contact between the interfaces is accounted for using the penalty method. Tiebreak contact has been used to successfully model the failure of interfaces between layers of composites (van Hoof, 1999 and Hung, 1995). Post-failure separation of the two surfaces can be modelled with a user-defined piecewise-linear stress-displacement curve. This could be used to simulate the effect of fibre bridging on the delamination growth. The basic, uncorrected interface is referred to as the **simple-tiebreak interface** in this thesis. The term **corrected-tiebreak interface** indicates the use of the stress-displacement curve.

3.3.6 Material Models

The elastic-plastic behaviour of metals is simulated in LS-DYNA using the **elastic-plastic (type 24)** material model in LS-DYNA. The stress-strain curve for the material is divided into discrete linear portions and entered into the code. The prepreg layers can be represented using the standard LS-DYNA **orthotropic-elastic (type 2)** material. This model is appropriate only for elastic deformations and does not take into account plasticity or damage effects. The constitutive matrix in terms of the material axes is (Hallquist, 1998):

$$C_l^{-1} = \begin{bmatrix} \frac{1}{E_{11}} & -\frac{\nu_{21}}{E_{22}} & -\frac{\nu_{31}}{E_{33}} & 0 & 0 & 0 \\ -\frac{\nu_{12}}{E_{11}} & \frac{1}{E_{22}} & -\frac{\nu_{32}}{E_{33}} & 0 & 0 & 0 \\ -\frac{\nu_{13}}{E_{11}} & -\frac{\nu_{23}}{E_{22}} & \frac{1}{E_{33}} & 0 & 0 & 0 \\ 0 & 0 & 0 & \frac{1}{G_{12}} & 0 & 0 \\ 0 & 0 & 0 & 0 & \frac{1}{G_{23}} & 0 \\ 0 & 0 & 0 & 0 & 0 & \frac{1}{G_{31}} \end{bmatrix} \quad (35)$$

The stress and strain matrices are related to each other as:

$$S = CE = T^t C_l T E \quad (36)$$

where T is the transformation matrix to transform the global co-ordinates into the local material system and T^t is the transpose of this matrix. For the plane-stress condition the relationship for the incremental stress update is:

$$\sigma_l^{n+1} = \sigma_l^n + \Delta \sigma_l^{n+1/2} \quad (37)$$

with

$$\Delta \sigma_l^{n+1/2} = \begin{bmatrix} \sigma_{11} \\ \sigma_{22} \\ \sigma_{12} \\ \sigma_{23} \\ \sigma_{31} \end{bmatrix} = \begin{bmatrix} \frac{E_{11}}{1-\nu_{12}\nu_{21}} & \frac{\nu_{12}E_{22}}{1-\nu_{12}\nu_{21}} & 0 & 0 & 0 \\ \frac{\nu_{12}E_{22}}{1-\nu_{12}\nu_{21}} & \frac{E_{22}}{1-\nu_{12}\nu_{21}} & 0 & 0 & 0 \\ 0 & 0 & G_{12} & 0 & 0 \\ 0 & 0 & 0 & G_{23} & 0 \\ 0 & 0 & 0 & 0 & G_{31} \end{bmatrix} \begin{bmatrix} \dot{\epsilon}_{11} \\ \dot{\epsilon}_{22} \\ \dot{\epsilon}_{12} \\ \dot{\epsilon}_{23} \\ \dot{\epsilon}_{31} \end{bmatrix} \Delta t \quad (38)$$

The out-of-plane strain, ϵ_{33} , needs to be calculated as part of the material model.

For the elastic case this strain is simply:

$$\epsilon_3 = \frac{C_{12}}{C_{33}} \epsilon_1 - \frac{C_{23}}{C_{33}} \epsilon_2 \quad (39)$$

with C_{13} , C_{23} and C_{33} as follows:

$$\begin{aligned} C_{13} &= \frac{E_1(\nu_{32} + \nu_{21}\nu_{23})}{1 - \nu_{12}\nu_{21} - \nu_{23}\nu_{32} - \nu_{13}\nu_{31} - 2\nu_{12}\nu_{23}\nu_{31}} \\ C_{23} &= \frac{E_2(\nu_{32} + \nu_{12}\nu_{21})}{1 - \nu_{12}\nu_{21} - \nu_{23}\nu_{32} - \nu_{13}\nu_{31} - 2\nu_{12}\nu_{23}\nu_{31}} \\ C_{33} &= \frac{E_1(1 - \nu_{12}\nu_{21})}{1 - \nu_{12}\nu_{21} - \nu_{23}\nu_{32} - \nu_{13}\nu_{31} - 2\nu_{12}\nu_{23}\nu_{31}} \end{aligned} \quad (40)$$

In addition to the standard material models that are part of LS-DYNA, it is also possible to incorporate user-defined subroutines (UMATs) directly into the program. The UMAT is written in FORTRAN and can be activated by the user with a keyword command in the input deck. Element stress and strain data are passed to the subroutine from the main program. The UMAT is then employed to update these values. The software also has the capability to store history variables that can be used to record damage levels or simply as step counters.

3.3.7 Application of LS-DYNA to Model Impact Damage in FMLs

To summarize, LS-DYNA was selected to model the impact response of FML panels. First, simulations of monolithic aluminum panels subjected to low-velocity impact were conducted to compare the performance using thin-shell, thick-shell and solid-brick elements. Additionally, mesh sensitivity, boundary conditions, geometry effects and the sensitivity of the model to the material input parameters were examined.

Using the selected element type and the basic mesh geometry, a series of low-velocity impact simulations of GLARE was conducted. The existing elastic-plastic (type 24) model was used for the aluminum layers. As observed through the destructive inspections, damage develops within the prepreg layers following impact. None of the standard material models in LS-DYNA could be applied to thick-shell elements to predict the effect of the development of damage on the impact response of the panels. Therefore, a user-defined material subroutine (UMAT) was developed to model the prepreg material. The user-defined material model was compared with the orthotropic-elastic (type 2) material model in LS-DYNA.

The interface between the aluminum and prepreg layers was modelled using the tied- and simple-tiebreak and corrected-tiebreak interfaces. The application of a post-failure correction to model the effects of fibre-bridging was also examined. All of the simulation results were compared to the experimental impact results from Chapter 2.

CHAPTER 4

SIMULATIONS OF LOW- VELOCITY IMPACT ON ALUMINUM

The elastic-plastic material (type 24) model in LS-DYNA is a simple representation of the development of plastic strain in an isotropic material. In order to qualify LS-DYNA to model low-velocity impact and to determine the basic input parameters, a series of simulations were conducted with 2024-T3 aluminum. In this Chapter, overviews of the material properties and the effect of mesh density variations are presented first. Next, a description of the preliminary results is provided with a comparison to the experimental impact data from Chapter 2. Lastly, a series of aluminum simulations with the final mesh geometry using thin-shells, thick-shells and solid-brick elements is presented.

4.1 Material Properties

The material properties for the aluminum layers came from the sheet manufacturer, ALCOA. These properties are summarized in Table 15. The aluminum alloy layers were modelled using the manufacturer's stress-strain curve for 2024-T3 aluminum as shown in Figure 52.

Table 15: Summary of baseline material properties used for 2024-T3.

Property	Value	Unit
Young's Modulus	72.39	GPa
Yield Stress	285	MPa
Mass Density	2.643×10^{-6}	kg/mm ³
Poisson's Ratio	0.33	-

Figure 52: Manufacturer's stress-strain data for aluminum 2024-T3 sheet.

4.2 Mesh Density and Geometry

The simulated geometry corresponded to the experimental set-up described in Section 2.2.3. Only the 203.2 mm diameter unsupported region of the impacted panel was modelled. The edges of the panel were either simply-supported to allow rotation or clamped to restrain all degrees of freedom. The aluminum panel thickness was 1.016 mm, which is the same as the aluminum panels used in the experiments. The number of elements and nodes was varied to study the effect of mesh density on the simulation results. The geometry and meshes were prepared using IDEAS, a 3-D solid modelling package, and FEMB, a third-party FEA pre- and post-processor. The basic geometry of the finite element simulation is shown in Figure 53.

Figure 53: Basic impact geometry for the preliminary impact simulations.

The 25.4 mm diameter hemispherical end of the impactor was modelled using 100 shell elements and rigid material (type 20). In LS-DYNA no stresses or deformations are calculated for rigid materials. However, some properties are still required for the determination of contact restoring forces. For this reason the modulus (200 GPa) and Poisson's ratio (0.33) of common steel were used. The mass of the drop weight hammer (6.45 kg) was incorporated into the model using discrete mass elements placed at each node of the hemispherical impactor. An initial velocity was applied to give the required kinetic energy for each impact event.

4.3 Simulated Events

A total of five energy levels were simulated with aluminum (25 to 65 J). All simulations were performed on an SGI Indigo2 workstation with the UNIX operating system. There were also four different mesh geometries as follows:

1. **“Rough” mesh** – an unoptimized mesh, developed using IDEAS.
2. **“Low” density mesh** – developed with the element boundaries arranged radially and circumferentially. A bias was employed to increase the local density of the mesh at the point of impact.
3. **“Medium” density mesh** – developed with the element boundaries arranged radially and circumferentially. This mesh geometry was tested with simply-supported and clamped boundary conditions.
4. **“High” density mesh** – developed with the element boundaries arranged radially and circumferentially.

The meshes are shown schematically in Figures 54 and 55. The “rough” edge-seeded mesh shown in Figure 54 was generated by specifying the number of elements along the edge of the panel. The mesh shown in Figure 55 possesses degenerated quadrilateral elements at the centre that may cause higher local stiffness.

Figure 54: Edge-seeded “rough” mesh.

Figure 55: Basic mesh used for “low,” “medium” and “high” mesh density simulations.

The characteristics of the simulations are summarized in Table 16. The boundary conditions were changed for the “medium” density mesh to examine boundary condition effects on impact response. For each of the mesh densities a total of five energy levels were simulated. These corresponded to the different energy levels that were used in the experiments as described in Section 2.2.

Table 16: Simulation matrix for simple aluminum impact events.

	Boundary Conditions	Energy	Velocity	Elements			Nodes			CPU Time
		(J)	(m/s)	Plate	Impactor	Total	Plate	Impactor	Total	(s)
Rough Mesh	Simple	26.16	2.81	1620	155	1775	769	171	940	97200
	Simple	36.32	3.31	1620	155	1775	769	171	940	97200
	Simple	46.57	3.80	1620	155	1775	769	171	940	97200
	Simple	57.43	4.22	1620	155	1775	769	171	940	97200
	Simple	61.49	4.39	1620	155	1775	769	171	940	97200
Low Density	Simple	26.16	2.81	650	155	805	651	171	822	4700
	Simple	36.32	3.31	650	155	805	651	171	822	4700
	Simple	46.57	3.80	650	155	805	651	171	822	4700
	Simple	57.43	4.22	650	155	805	651	171	822	4700
	Simple	61.49	4.39	650	155	805	651	171	822	4700
Medium Density	Simple	25.77	2.83	720	155	875	721	171	892	10500
	Simple	35.89	3.34	720	155	875	721	171	892	10500
	Simple	46.28	3.79	720	155	875	721	171	892	10500
	Simple	57.49	4.22	720	155	875	721	171	892	10500
	Simple	61.53	4.37	720	155	875	721	171	892	10500
	Clamped	31.46	3.12	720	155	875	721	171	892	10500
	Clamped	35.92	3.34	720	155	875	721	171	892	10500
	Clamped	46.30	3.79	720	155	875	721	171	892	10500
	Clamped	57.53	4.22	720	155	875	721	171	892	10500
	Clamped	61.55	4.37	720	155	875	721	171	892	10500
High Density	Simple	25.83	2.83	960	155	1115	961	171	1132	20000
	Simple	35.96	3.34	960	155	1115	961	171	1132	20000
	Simple	46.34	3.79	960	155	1115	961	171	1132	20000
	Simple	57.58	4.23	960	155	1115	961	171	1132	20000
	Simple	61.61	4.37	960	155	1115	961	171	1132	20000

4.4 Preliminary Simulation Results

Figures 56 to 58 summarize the results of the aluminum impact simulations. As can be seen in Figure 56, the simulations predicted consistently lower absorbed energy than was measured in the experiments. However, all of the simulations predicted approximately the same values for absorbed energy, peak force and dent depth. Thus, mesh density and boundary conditions had little effect on the outcome of the simulations. Complete tabulated results are provided in Appendix E.

Figure 56: Absorbed energy versus impact energy for aluminum low-velocity impact experiments and simulations.

Figure 57: Peak force versus impact energy for aluminum low-velocity impact experiments and simulations.

Figure 58: Dent depth versus impact energy for aluminum low-velocity impact experiments and simulations.

The effect of increasing the number of elements on the CPU time is presented graphically in Figure 59. As can be seen, there is an approximately linear relationship between the number of elements and the CPU time required for the simulation. There is a decreasing dependence of the percent difference in predicted absorbed energy with increasing impact energy. However, there is no obvious relationship between the percent difference in absorbed energy and the number of elements. The difference in absorbed energy is calculated relative to the experimental values. Therefore, the amount of elements used in this study does not have an effect on the accuracy of the aluminum impact simulations.

Figure 59: CPU time versus number of elements for preliminary simulations.

4.5 Comparison to Experimental Data

The results of the aluminum simulations demonstrated the validity of the boundary condition assumptions made for the impact models. They also demonstrated the effect of the mesh density on the results, and provided familiarization training with LS-DYNA.

There was no effect of varying the mesh densities on the accuracy of the solution. However, there was a slight effect from the boundary condition. It was observed that the clamped panels absorbed less energy than the simply-supported panels. The clamped boundary condition does not allow as much deflection at the specimen edge as the simply-supported boundary condition. The overall stiffness of the specimen is increased and less elastic and plastic strain energy is absorbed.

There was a discrepancy of up to 20% between the simulated absorbed energy and the experimental values. The boundary conditions could be responsible for part of the discrepancy. To assume that the boundary is either clamped or simply-supported may not be accurate. As the panel is deformed during the impact event, there is some evidence that it slides out of the clamping fixture at the edges. Scrape marks are present at the periphery of the circular unsupported region in the impacted specimens described in Chapter 2. This would result in a boundary that is neither clamped nor simply-supported. However, for the present study a clamped or fully-fixed boundary condition will be assumed.

4.6 Final Simulations of Aluminum

Following the conclusion of the preliminary modelling, a new mesh geometry was chosen for the final aluminum and GLARE simulations. This mesh was developed to eliminate the degenerated quadrilateral elements at the point of impact present in the radial meshes as shown in Figure 55.

4.6.1 Simulation Parameters

The mesh configuration (Figure 60) incorporates a region of equal sized square elements at the centre. This area was sized to correspond to the region of influence of the impactor observed in the experiments. For all of the simulations the impactor was modelled using thin-shell elements with the rigid material model (type 20) as shown in Figure 61. Friction between the steel impactor and the aluminum was assumed in the simulation. Literature values of 0.8 for the static coefficient and 0.5 for the kinetic coefficients of friction were employed (Herzberg, 1993).

Figure 60: Half-symmetry mesh geometry.

Figure 61: Impactor model geometry and mesh.

A description of the thin-shell, thick-shell and solid-brick element formulations can be found in Section 3.3. For the solid-brick simulations the maximum aspect ratio was approximately 8:1. The results of the simulations will be compared to each other and also to the experimental results. The basic aluminum properties employed for the finite element simulations were presented in Section 4.1. A detailed summary of the simulations carried out using monolithic 2024-T3 is presented in Table 17. This table also contains the final CPU times and number of cycles for each simulation. These simulations were performed on an Intel-based PC (750 MHz) running Redhat Linux version 7.1 with version 950d of LS-DYNA.

Table 17: Summary of final 2024-T3 simulation parameters.

ID Number	Element Type	Energy Level	# of Elements	# of Nodes	CPU Time (s)	Clock Time	CPU Cycles
1-AL-PW-TN-25	Thin	25	1300	1342	425	850	31251
2-AL-PW-TN-35	Thin	35	1300	1342	424	852	31250
3-AL-PW-TN-45	Thin	45	1300	1342	426	1295	31239
4-AL-PW-TN-55	Thin	55	1300	1342	431	1305	31220
5-AL-PW-TK-25	Thick	25	1401	1342	3794	79901	120901
6-AL-PW-TK-35	Thick	35	1401	1342	3840	15247	122387
7-AL-PW-TK-45	Thick	45	1401	1342	3905	15702	123364
8-AL-PW-TK-55	Thick	55	1401	1342	3911	15707	123234
9-AL-PW-SO-25	Solid	25	1401	1342	2361	7330	164655
10-AL-PW-SO-35	Solid	35	1401	1342	2455	7527	169068
11-AL-PW-SO-45	Solid	45	1401	1342	5131	15408	172040
12-AL-PW-SO-55	Solid	55	1401	1342	2358	8002	173615

4.6.2 Material Property Sensitivity

Using the mesh geometry described in Section 4.6.1, three additional simulations were conducted to examine the effect of the aluminum properties on the predicted response of the aluminum panels. Figure 62 contains two additional stress-strain curves from the ASM handbook and MIL-HDBK-5H (ASM, 1990 and DoD, 1998) that were used to investigate the sensitivity of the impact simulations to the yield stress. Each of these sources is considered “correct”; yet as seen in the figure, there are differences between the stress-strain curves.

Figure 62: Stress-strain curves for 2024-T3 from various sources.

The results of the material property sensitivity study are tabulated in Table 18 and displayed graphically in Figures 63, 64 and 65. The results show that the amount of absorbed energy increases with decreasing yield strength. This result is not surprising given that the panel simulated using the lower yield strength would suffer greater plastic deformation and thus absorb more kinetic energy. For a similar reason, the higher yield strength of the ASM material resulted in lower absorbed energy and higher peak force. The ASM panel underwent proportionally more elastic deformation than the other two panels and thus sustained a higher peak impact force.

This study revealed the effect of using material property data from a variety of sources. Due to the potential variations in the predicted impact results, it is important to use the manufacturer's data when it is available. Alternatively, tensile tests could be used to provide the required data. In all of the impact simulations in this thesis, the manufacturer's property data will be used.

Table 18: Material sensitivity study results.

	Yield Strength	Absorbed Energy	Peak Force	Dent Depth
	(MPa)	(J)	(kN)	(mm)
ASM	420	10.4	7.88	4.6
MILHDBK-5H	345	12.4	7.17	5.4
Manufacturer	285	14.4	6.89	6.1
Experimental	Not applicable	18.8	5.70	6.2

Figure 63: Impact energy results from material sensitivity study.

Figure 64: Peak impact force from material sensitivity study.

Figure 65: Dent depth results from material property sensitivity study.

4.6.3 Results for the Final 2024-T3 Simulations

The results of the impact simulations of aluminum are plotted versus the impact energy in Figures 66, 67 and 68. Complete tabulated results can be found in Appendix F. The final 2024-T3 simulation results were similar to the preliminary results described in Section 4.4. The amount of energy absorbed in the simulations was consistently less than that recorded in the experiments. This consistent deviation indicates a fundamental difference between the simulations and the experiments. The most likely sources of this error are the mechanical losses present in the drop tower impact experiments.

The predicted impact forces were higher than those measured in the experiments. The sources of this difference are the same as those that caused the difference in absorbed energy. Using rigid elements for the impactor and a perfectly fixed boundary condition for the specimen neglects the compliance of the drop tower support structure. The

remainder of the drop tower structure was not modelled since it is the response of the aluminum panel that was of interest. The production of a fully detailed impact model could be a suitable follow-on project to study the losses in the structure.

The results obtained for the permanent deflection are interesting in that they show good agreement between the experiments and all the simulations except for the solid-brick simulations. The solid-brick simulations predicted permanent deflections 1.7 mm lower than those predicted using the other element types. Therefore, it was decided to use only thick-shell elements because of the advantages when modelling layered structures. These advantages include the designation of contact interfaces and the ability to model thin multi-layered structures. These results also confirmed the applicability of LS-DYNA to problems of low-velocity impact.

Figure 66: Absorbed Energy versus Impact Energy for the final impact simulations.

Figure 67: Maximum Force versus Impact Energy for the final simulations.

Figure 68: Permanent Deflection versus Impact Energy for the final impact simulations.

CHAPTER 5

DELAMINATION DAMAGE

CHARACTERIZATION

As described in Section 3.2, one of the components required to successfully simulate the damage formed in GLARE laminates is an accurate model of delamination initiation and growth in the panels. Manufacturer's strength data can be used for delamination initiation with the tiebreak-interface in LS-DYNA. However, experiments are required to characterize the post-delamination behaviour. Section 5.1 presents a review of delamination characterization techniques for FMLs. The results of a series of double cantilever beam (DCB) tests are then presented in Section 5.2.

5.1 Review of Delamination

Characterization of FMLs

Mode I delamination damage has been identified as an important damage mode in impacted FMLs. Two groups of researchers have characterized delamination damage in FMLs previously. Vries *et al.* (1999) conducted an experimental investigation into the delamination behaviour of spliced fibre-metal-laminates. This study was part of the BRITE-EURAM-III project investigating concepts for next-generation European passenger aircraft. They examined several splice concepts and subjected test coupons to cyclic loading to measure the resistance to delamination growth, R . Two reasons were given for using this parameter, based on strain energy release, over others based on tensile and interlaminar strength:

1. Mixed mode delamination is easy to characterize with the single resistance parameter as opposed to two critical shear and tensile stresses.
2. Neither the stress concentration at the delamination tip nor the distribution of stresses over the thickness of the laminate have to be considered.

It was found that the delamination growth rates are independent of the delamination size.

Vlot and van Ingen (1998) studied delamination in unidirectional ARALL-2, ARALL-3 and GLARE-1 under static and dynamic loading. Each of the laminates was tested in the as-cured condition and in a post-stretched state (0.4% permanent stretch). As previously discussed, in as-cured FMLs there is a residual tensile stress-state in the aluminum layers. This can lead to reduced fatigue resistance in the panels. Some panels

of each laminate were also subjected to a second curing cycle following the stretching operation to simulate the final bonding step for an airframe component. Double cantilever beam (DCB) tests and end notch flexure (ENF) tests were used to study the G_{IC} and G_{IIC} energy release rate, respectively. The dynamic tests were conducted with a specimen that simulated the edge of a bonded FML doubler. This configuration is seen in aircraft that employ bonded doublers around doors and windows.

The post-stretch operation had a positive influence on the mode I static delamination resistance in ARALL and no influence on the mode II delamination resistance. In GLARE laminates the post-stretching had no influence on the mode I static delamination resistance and only a small positive effect on the mode II resistance. It was also found that the GLARE panels exhibited significantly higher overall delamination resistance than ARALL under static conditions. Equations based on classical beam bending theory combined with rotation of the two beams at the delamination tip in the DCB specimen were developed to calculate the energy release rates. A series of DCB tests using the specimen shown in Figure 69 were carried out.

Figure 69: DCB specimen (reproduced from Vlot and Ingen, 1998).

Aluminum beams of the same composition as the aluminum used in the fibre-metal lay-up of GLARE were used to support the FML coupon. The DCB specimens were loaded at constant displacement until delamination occurred. The displacement was stopped immediately and the crack length was then marked on the specimen edge. This process resulted in a load-displacement curve for each specimen.

The test method used by Vlot and van Ingen is similar to the composite DCB test methods discussed by Hashemi *et al.* (1989), Whitney (1989) and Russell and Street (1983). However, Vlot and van Ingen did not employ the same basic loading method. When loading the specimens, the crack was allowed to propagate for a set interval before unloading. However, the other researchers used an initial crosshead displacement that was held during the growth of the delamination. When the propagation of delamination ceased, the load was removed. No reason for the difference in loading method was indicated.

5.2 DCB Characterization of GLARE

To support the development of the damage modelling methodology, a series of DCB tests was conducted using several types of GLARE. These tests were intended to provide crack opening stress versus displacement data for the LS-DYNA impact model. Coupons of GLARE-3, -4 and -5 were selected for the DCB tests; four samples of each for 0° fibre orientation and four GLARE-5 coupons in the 90° fibre orientation (Table 19).

Table 19: Experimental test matrix.

Material Type	Number Tested
GLARE-3-2/1	4
GLARE-4-2/1	4
GLARE-5-2/1 (0°)	4
GLARE-5-2/1 (90°)	4

5.2.1 Preparation of Test Specimens

The test article design was based upon that of Vlot and van Ingen (1998) as shown in Figure 70. To ensure that a crack would initiate in the GLARE coupon rather than the bond line between the coupon and the aluminum bar, a notch was introduced using a 0.15 mm thick diamond saw blade. The notch was protected with Teflon tape to ensure that adhesive would not overflow into the notch. The room temperature-cure adhesive known as DP-420 from 3M was used to bond the specimens to the adherends.

5.2.2 Test Procedure

The specimens were prepared as shown in Figure 70. Each specimen was tested using an MTS hydraulic load frame under displacement control through pins inserted in the holes near the end of the specimen. They were then loaded quasi-statically at a rate of 5 mm/min under displacement control until the first evidence of cracking occurred on the load curve. This crack length was measured and the load was reduced to zero. Each specimen was again loaded until the crack began to grow again. The crack was allowed to grow to another arrest point. This procedure was repeated until a final overall crack length was reached.

Figure 70: GLARE DCB specimen configuration.

5.2.3 Delamination Test Results

A sample load versus crosshead displacement curve is shown in Figure 71. This particular load-displacement curve was considered acceptable since a total of 7 crack length measurements were possible. Figure 72 shows an unacceptable crack growth curve from a GLARE-3 specimen. Only two measurements were possible using this specimen and then the specimen failed in the bondline between the laminate and the aluminum bar. The delamination failed to propagate any further.

Figure 71: Acceptable load curve from a GLARE-5 DCB test.

Figure 72: Unacceptable load curve from a GLARE-3 DCB test.

Extensive fibre bridging was noted during examinations of the fracture surface of the specimens (Figure 73 and Figure 74a). The delamination cracks propagated in prepreg layers with the fibre oriented parallel to the delamination direction. Each of the

three types of GLARE exhibited the same behaviour. This indicates that the “path of least resistance” through the prepreg layer is parallel to the fibres rather than perpendicular. The fibres pull away from the matrix and produce the fracture surface shown in Figure 73.

Figure 73: Optical micrograph of the failure surface in a typical DCB specimen.

Although the geometry was different between the DCB specimens and the impacted specimens, the damage modes were similar. Therefore the DCB results can be applied to the impact simulations. A comparison between the DCB and impact specimen delamination fractures is shown in Figure 74. As can be seen in this figure the fibre bridging in both types of specimens are similar.

Figure 74: (a) Fibre bridging in a typical FML DCB specimen and (b) in an impacted FML specimen.

5.2.4 Application of the DCB Test Results in LS-DYNA

As described in Section 3.3.5 it is possible to incorporate a piecewise linear stress-displacement curve for modelling ply separation in LS-DYNA during delamination failure. Upon more detailed examination of the DCB test data it was determined that the load-displacement curves generated during the testing could not be directly incorporated into LS-DYNA. LS-DYNA requires the stress at the interface rather than the overall load, as was generated by the DCB tests. It was not possible to convert the load-displacement data from the experiments into stress-displacement curves. These tests, however, did provide useful information into the overall delamination behaviour in the laminates. A new specimen may be required to provide the appropriate data. A finite element package specifically suited for modelling the stress singularity at the tip of the delamination could also be used to determine the stresses for a given displacement.

CHAPTER 6

DEVELOPMENT OF A CDM- BASED SUBROUTINE

This chapter describes the development of a user-defined subroutine for the prepreg layers in the GLARE laminates. A review of continuum damage mechanics (CDM) is presented in Section 6.1. Continuum damage mechanics is a branch of materials science that has been widely used to predict the degradation and failure of engineering materials in many applications. The application of one CDM-based approach within LS-DYNA is discussed in Section 6.2. The validation of the model using coupon simulations is given in Sections 6.3 and 6.4.

6.1 Review of Continuum Damage

Mechanics for Composites and FMLs

In his seminal work, Kachanov (1958) described the cumulative effect of the deterioration of a material through the formation of microcracks by using a field variable. A dimensionless scalar field variable φ is used to describe the state of the material. In an undamaged material this quantity is $\varphi=1$ and in a completely damaged material it is $\varphi=0$. As this theory was originally applied to creep-rupture, Kachanov developed a law in which φ varied with time at elevated temperatures. The complementary quantity, $d=1-\varphi$, was defined to denote the level of damage or deterioration.

This theory was later expanded and applied to numerous other physical situations. These include fatigue damage, thermomechanical fatigue and damage in composite materials. Continuum damage mechanics can be applied to both the lamina and the interface between the laminae in composites. Lemaitre (1987, 1992) summarizes the original development of Kachanov's theory and subsequent additions and improvements. To illustrate the basic concept of the damage variable, Figure 75 shows a simple uniaxial specimen under a tensile load, F , in the normal direction, $\vec{n}$. The large volume is referred to as the Representative Volume Element (RVE) and is sized to represent an average of the microprocesses, but must also be small enough that high gradients in the material are not excessively smoothed out. This scale is approximately 1 mm^3 for composites (Lemaitre, 1992). Within the specimen there is a plane perpendicular to $\vec{n}$. Within the element there is δS , which is the area of intersection between the plane and the

RVE. There is also have δS_{Dx} , which is the effective area of intersection between all microcracks and cavities that lie in δS .

Figure 75: Representation of the micro-meso scale definition of damage (adapted from Kachanov, 1958 and Lemaitre, 1992).

There is an amount of damage associated with the RVE in the direction of $\vec{n}$ and at the point x that is given in Equation 41.

$$d(\vec{n}, x) = \frac{\delta S_{Dx}}{\delta S} \quad (41)$$

The final failure plane is that which contains the most damage. This is found by examining all planes along the x direction as shown in Equation 42.

$$d_{(\vec{n})} = \text{Max}_{(x)}[d_{(\vec{n}, x)}] \quad (42)$$

Thus, the co-ordinate x disappears and the final relation for the damage is given by Equation 43.

$$d(M, \vec{n}) = \frac{\delta S_d}{\delta S} \quad (43)$$

The value of d is bounded by 0 and 1, where 0 represents completely undamaged material and 1 represents completely failed material. In reality the specimen will fail before the value of d reaches 1 through a process of material instability that leads to rupture at the microscopic level.

There is typically a critical value of damage, d_c , that determines when failure will occur. In a purely brittle fracture the damage will be $d_c \sim 0$ and in a pure ductile fracture it will be $d_c \sim 1$. This amount of damage can be related to a critical value of the effective stress, σ_∞ , in the material as follows in Equation 44.

$$\tilde{\sigma} = \frac{\sigma}{1 - d_c} = \sigma_\infty \quad (44)$$

In practice the value of d_c is closer to 0.5, as shown by Ladevèze and Le Dantec (1992) for fibre reinforced composites. It is possible to approximate the critical stress, σ_∞ , with the ultimate stress, σ_u , which is smaller and to arrive at an estimate for d_c as shown in Equation 45.

$$(d_c)_{est} \approx 1 - \frac{\sigma}{\sigma_u} \quad (45)$$

For example, in a uniaxial monotonic stress test, σ is equal to the rupture stress of the specimen. This test can then define a value of d_c , which is then considered a material characteristic.

It has been shown (Lemaitre, 1992) that “any strain constitutive equation for a damaged material may be derived in the same way as for virgin material except that the usual stress is replaced by the effective stress.” This concept of effective stress is very

important for the application of damage mechanics. A consequence of this principle is that the elastic modulus can be measured to determine the amount of damage in a specimen. By extension, other similar material properties, such as the flexural modulus or the shear modulus, could also be measured. There is thus an influence of damage on the elastic response of the material:

$$\varepsilon_e = \frac{\sigma}{E(1-d)} \quad (46)$$

where ε_e is the elastic strain. For the Young's modulus, the relationship with damage is given by Equation 47.

$$d = 1 - \frac{E}{E^o} \quad (47)$$

This allows the direct measurement of the accumulation of damage through a series of cyclic tensile loading-unloading tests.

The damage constitutive equations are determined using a thermodynamic framework, in terms of the strains used in elasticity, plasticity and viscoplasticity. Through this method the coupling between all the different physical phenomena is obtained automatically. This allows the accumulation of damage to reduce the load carrying capacity of the material, resulting in a reduction in the modulus of the material.

The thermodynamic foundations of continuum damage mechanics have been described by numerous researchers since Kachanov (Lemaitre, 1987, Chaboche, 1988, Maugin, 1992 Germain *et al*, 1983, Coleman and Gurtin, 1967 and Krajcinovic, 1983). Chaboche (1988) and Lemaitre (1992) stated that the damage strain energy release rate is a function of the change in free energy density as shown in Equation 48.

$$\bar{Y} = \rho \frac{\partial \psi}{\partial d} \quad (48)$$

Recall from Equation 46 the principle of effective stress. As shown in Equation 49, this can be extended to three-dimensions.

$$\tilde{\sigma}_{ij} = \frac{\sigma_{ij}}{1-d} \quad (49)$$

Thus the law of elasticity coupled with the damage of the material is:

$$\sigma_{ij} = \rho \frac{\partial \psi}{\partial \varepsilon_{ij}} = a_{ijkl} \varepsilon_{kl}^e (1-d) \quad (50)$$

with a as the fourth order elasticity stiffness tensor. The overall expression for the free energy in Equation 50 was defined by Lemaitre (1992) to be:

$$\psi = \frac{1}{\rho} \left[\frac{1}{2} a_{ijkl} \varepsilon_{ij}^e \varepsilon_{kl}^e (1-d) + R_{\infty} \left[r + \frac{1}{b} \exp(-br) \right] + \frac{X_{\infty} \gamma}{3} \alpha_{ij} \alpha_{ij} \right] \quad (51)$$

where the values of R and b are related to the isotropic strain hardening of the material. Also, X_{∞} and γ are parameters characterizing the non-linear kinematic strain hardening of the material. Therefore, the associated damage variable is found by Equation 52.

$$\bar{Y} = \rho \frac{\partial \psi}{\partial d} = -\frac{1}{2} a_{ijkl} \varepsilon_{ij}^e \varepsilon_{kl}^e \quad (52)$$

It is now necessary to find a relationship between the damage variable and the elastic strain energy of the system. To simplify the development of this relation we define the value $(Y = -\bar{Y})$. By definition the increment of elastic strain energy density is given by Equation 53.

$$dE_e = \sigma_{ij} d\varepsilon_{ij}^e \quad (53)$$

Integrating this with the law of elasticity and assuming no variation of damage will give:

$$E_e = \int a_{ijkl} \varepsilon_{kl}^e (1-d) d\varepsilon_{ij}^e = \frac{1}{2} a_{ijkl} \varepsilon_{ij}^e \varepsilon_{kl}^e (1-d) \quad (54)$$

and it follows that the value of the damage variable will be given by Equation 55.

$$Y = \frac{E_e}{1-d} \quad (55)$$

The damage variable can also be written as one-half the variation of the strain energy density corresponding to the variation of damage (Chaboche, 1976). Beginning with the law of elasticity with constant stress:

$$\delta\sigma_{ij} = a_{ijkl} [(1-d)\delta\varepsilon_{kl}^e - \varepsilon_{kl}^e \delta d] = 0 \quad (56)$$

and

$$\delta\varepsilon_{kl}^e = \varepsilon_{kl}^e \frac{\delta d}{1-d} \quad (57)$$

with the definition for strain energy density from above

$$\delta E_e \Big|_{\sigma=const} = \sigma_{ij} \delta\varepsilon_{ij}^e = \sigma_{ij} \varepsilon_{ij}^e \frac{\delta d}{1-d} = a_{ijkl} \varepsilon_{ij}^e (1-d) \varepsilon_{kl}^e \frac{\delta d}{1-d} \quad (58)$$

or

$$\frac{\delta E_e}{\delta d} \Big|_{\sigma=const} = a_{ijkl} \varepsilon_{ij}^e \varepsilon_{kl}^e \quad (59)$$

thus

$$Y = \frac{1}{2} \frac{\delta E_e}{\delta d} \quad (60)$$

The above relationship for the damage variable provides a link between the strain energy and damage. This relationship was used by Ladevèze and Le Dantec (1992) to describe a model for damage accumulation in an individual lamina. A version of this

model was implemented previously by Hurez *et al.* (1993) as a stand-alone FEA code to model static tensile tests on GLARE-2.

Ladevèze and Le Dantec (1992) developed the relationship in Equation 61 for the damaged material strain energy, E_D .

$$E_D = \frac{1}{2} \left[\frac{\sigma_{11}^2}{E_{11}^0} - 2 \frac{\nu_{12}^0}{E_{11}^0} \sigma_1 \sigma_2 + \frac{\langle \sigma_{22} \rangle_+^2}{E_{22}^0 (1-d')} + \frac{\langle \sigma_{22} \rangle_-^2}{E_{22}^0} + \frac{\sigma_{12}^2}{G_{12}^0 (1-d)} + \frac{\sigma_{13}^2}{G_{12}^0 (1-d)} + \frac{\sigma_{23}^2}{G_{23}^0} \right] \quad (61)$$

where the subscripts 1, 2 and 3 denote the fibre direction, the transverse direction and the through-thickness direction, respectively. The angled brackets and the $\pm$ subscripts indicate negative or positive conditions for σ_{11} ; therefore, if σ_{22} is positive then $\langle \sigma_{22} \rangle_-^2$ is equal to zero. This allows different compressive and tensile conditions to be modelled. When σ_{22} is compressive (negative) the matrix cracks close and do not affect the modulus. E_{11} is the modulus in the fibre direction and E_{22} is the modulus in the transverse direction. The two damage variables (d and d') act upon the initial transverse modulus and the shear modulus, (E_{22}^0 and G_{12}^0) as shown in Equation 62.

$$G_{12} = G_{12}^0 (1-d) \quad \text{and} \quad E_{22} = E_{22}^0 (1-d') \quad (62)$$

Furthermore, two dissipation variables are associated with the damage terms d and d' :

$$\begin{aligned} Y_{d'} &= \rho \frac{\partial \psi}{\partial d} \bigg|_{\bar{\sigma}, d'} = \frac{\partial E_D}{\partial d} \bigg|_{\bar{\sigma}, d'} = \frac{1}{2} \frac{\sigma_{12}^2}{G_{12}^0 (1-d)^2} \\ Y_d &= \rho \frac{\partial \psi}{\partial d} \bigg|_{\bar{\sigma}, d} = \frac{\partial E_D}{\partial d} \bigg|_{\bar{\sigma}, d} = \frac{1}{2} \frac{\langle \sigma_{22} \rangle_+^2}{E_{22}^0 (1-d')^2} \end{aligned} \quad (63)$$

where ψ is the free energy density and $\bar{\sigma}$ is the effective stress as given in Equation 64.

$$\bar{\sigma} = \begin{bmatrix} \sigma_{11} \\ \frac{\langle \sigma_{22} \rangle}{1-d'} + \langle \sigma_{22} \rangle_- \\ \frac{\sqrt{2}\sigma_{12}}{(1-d)} \end{bmatrix} \quad (64)$$

Two mechanisms were identified that contribute to the development of damage: matrix micro-cracking and fibre/matrix de-bonding. The following two quantities were introduced to describe the development of the damage:

$$\begin{aligned} \underline{Y} &= (\sqrt{Y_d + bY_{d'}}) \\ \underline{Y}' &= (\sqrt{Y_{d'}}) \end{aligned} \quad (65)$$

The damage development laws developed by Ladevèze and Le Dantec (1992) follow in Equations 66 and 67:

$$d = \frac{\langle \underline{Y} - Y_0 \rangle_+}{Y_c} \quad (66)$$

if $d < 1$ and $\underline{Y}' < Y'_S$; otherwise $d=1$ and the element fails. As well:

$$d' = \frac{\langle \underline{Y} - Y'_0 \rangle_+}{Y'_C} \quad (67)$$

if $d' < 1$ and $\underline{Y}' < Y'_S$; otherwise $d'=1$ and the element fails.

The parameters Y_c , Y'_c , Y_0 , Y'_0 and b are characteristics of the material and are determined by tension tests as well as a few loading-unloading cycles on basic glass-reinforced lamina as described by Ladevèze and Le Dantec (1992). The value Y'_S is the brittle-damage threshold that determines the behaviour of the fibre-matrix interface in the element transverse direction.

Property evaluation tests are listed below along with some of the parameters that they provide. It should be noted that the testing also produces additional parameters used for plasticity modelling and compressive stiffness loss, two factors not used in the present project.

1. Tension tests of $[0^\circ/90^\circ]_{2s}$ or $[0^\circ]_8 - E_{11}^o, \nu_{12}$, rupture stress and rupture strain
2. Tension tests of $[45^\circ]_8 - E_{22}^o, G_{12}^o, Y_c$ and Y_o
3. Tension tests of $[\pm 45^\circ]_{2s} - G_{12}^o$ and Y_c'
4. Tension tests of $[\pm 67.5]_{2s} - b$

As presented previously in Section 3.1.1, Hurez *et al.* (1993) applied a version of this model to simulate GLARE-2-2/1 under static tensile loading. Using the data from these tests Hurez *et al.* constructed an elementary CDM-based model for GLARE-2 under tension. The results agreed well with experiments for static loading and also for three-point bending simulations. The Hurez *et al.* version was used to model a very specific situation - tensile loading of GLARE-2-2/1. This was a very limited application which was not useful for a dynamic event such as low-velocity impact. Therefore, a version of the Ladevèze and Le Dantec approach will be developed and implemented in LS-DYNA as a user-defined material model. This model will be used to predict the behaviour of GLARE panels subjected to impact loads.

These CDM theories have also been extended to include the prediction of delamination damage (Allix *et al.*, 1991, Allix and Ladevèze, 1992, Daudeville and Ladevèze, 1993, Daudeville *et al.*, 1995, Allix and Ladevèze 1995 and Allix *et al.*, 1998).

Another CDM-based approach that has also been employed is the MLT method (Matzenmiller *et al.*, 1995). This has been implemented in LS-DYNA by other

researchers (Williams and Vaziri, 1995 and van Hoof, 1999). The MLT method is applied to unidirectional laminates by examining damage modes in each axis separately. Similar to the Ladevèze and Le Dantec approach, the MLT model does not include the effects of delamination in the overall damage relations as it only deals with the individual lamina. Damage is assumed to take the form of cracks perpendicular or parallel to the fibre direction in the unidirectional lamina. Hashin's (1980) failure criteria for lamina are the basis for the failure surfaces that bound the elastic loading region. Damage variables based upon a Weibull-type function are used to modify the stiffness matrix of the lamina. This method was developed for unidirectional lamina in composite materials. As discussed by Williams and Vaziri (2001) there are several weaknesses in the model. For example, it is not possible to model a material that is elastic up to failure but then displays gradual strain-softening after the peak load.

Based on the above review the Ladevèze and Le Dantec method was selected over the MLT method since the input properties for the GLARE prepreg are readily available in literature. Additionally, the Ladevèze and Le Dantec method focuses on damage in the transverse directions such as matrix cracking. This is one of the main damage modes in GLARE panels subjected to impact below the puncture threshold.

6.2 Subroutine Development

As described in the preceding review, a version of the Ladevèze and Le Dantec (1992) model was implemented by Hurez *et al.* (1993) for application to GLARE-2. This model was implemented as a stand-alone program and could not be extended to other geometries or materials. Therefore, it was decided to develop the Ladevèze and Le Dantec model as a user-defined material subroutine, called a UMAT, in LS-DYNA. It is evident from Equations 66 and 67 that:

$$\underline{Y} = dY_c + Y_o = d'Y'_c + Y'_o \quad (68)$$

It is then possible to isolate d' as shown in Equation 69.

$$d' = \frac{dY_c + Y_o - Y'_o}{Y'_c} \quad (69)$$

Substituting Equations 63, 65 and 69 into Equation 67 and manipulating gives:

$$(dY_c - Y_o)^2 - \left[\frac{1}{2} \frac{\sigma_{12}^2 + \sigma_{13}^2}{G_{12}(1-d)^2} + \frac{b}{2} \frac{\sigma_{22}^2}{\left(1 - \frac{dY_c + Y_o - Y'_o}{Y'_c}\right)^2} \right] = 0 \quad (70)$$

To remain consistent with the Ladevèze method a check for $\underline{Y} - Y_o > 0$ must be performed; if this condition is not met then $d = d' = 0$. Equation 70 has up to 6 roots that were identified using the symbolic solution capabilities of Mathcad. In Fortran, this equation must be solved numerically. The only valid value of d is slightly negative when the stresses are zero. As the applied stresses increase, the value of this root increases. None of the other possible solutions exhibit the same behaviour and none of them

increase with increasing applied load between 0 and 1. Since the range of valid values for d is known to be between 0 and 1, it is best to use the Bisection Method rather than Newton's Method to find the root. The Bisection Method is also simple to implement in Fortran and converges quickly. Furthermore, when the stresses reach the rupture limit of the material the root becomes complex. At this point, it is not possible to find a root in the required range, therefore, the critical damage level has been reached. When this critical damage value is reached the element has failed.

The damage variables influence the stiffness of the orthotropic material. Thus, these terms are incorporated into the calculation of the stiffness matrix terms. Recall from Equation 38 that the incremental stresses in the orthotropic material are as follows:

$$\Delta \sigma_l^{n+1/2} = \begin{bmatrix} \sigma_{11} \\ \sigma_{22} \\ \sigma_{12} \\ \sigma_{23} \\ \sigma_{31} \end{bmatrix} = \begin{bmatrix} \frac{E_{11}}{1 - \nu_{12}\nu_{21}} & \frac{\nu_{12}E_{22}}{1 - \nu_{12}\nu_{21}} & 0 & 0 & 0 \\ \frac{\nu_{12}E_{22}}{1 - \nu_{12}\nu_{21}} & \frac{E_{22}}{1 - \nu_{12}\nu_{21}} & 0 & 0 & 0 \\ 0 & 0 & G_{12} & 0 & 0 \\ 0 & 0 & 0 & G_{23} & 0 \\ 0 & 0 & 0 & 0 & G_{31} \end{bmatrix} \begin{bmatrix} \dot{\epsilon}_{11} \\ \dot{\epsilon}_{22} \\ \dot{\epsilon}_{12} \\ \dot{\epsilon}_{23} \\ \dot{\epsilon}_{31} \end{bmatrix}_l \Delta t \quad (71)$$

It should also be noted that the Poisson's ratios are calculated as follows (Hallquist, 1998):

$$\begin{aligned} \nu_{21} &= \frac{E_{22}}{E_{11}} \nu_{12} \\ \nu_{13} &= \frac{E_{33}}{E_{11}} \nu_{31} \\ \nu_{32} &= \frac{E_{33}}{E_{22}} \nu_{23} \end{aligned} \quad (72)$$

Since the damage variables d and d' modify the transverse and shear stiffnesses of the material, it is simple to include them in the stress update.

$$\Delta \sigma_i^{n+1/2} = \begin{bmatrix} \sigma_{11} \\ \sigma_{22} \\ \sigma_{12} \\ \sigma_{23} \\ \sigma_{31} \end{bmatrix} = \begin{bmatrix} b_{11} & b_{12} & 0 & 0 & 0 \\ b_{12} & b_{22} & 0 & 0 & 0 \\ 0 & 0 & G_{12}(1-d) & 0 & 0 \\ 0 & 0 & 0 & G_{23} & 0 \\ 0 & 0 & 0 & 0 & G_{31} \end{bmatrix} \begin{bmatrix} \dot{\epsilon}_{11} \\ \dot{\epsilon}_{22} \\ \dot{\epsilon}_{12} \\ \dot{\epsilon}_{23} \\ \dot{\epsilon}_{31} \end{bmatrix}_L \Delta t \quad (73)$$

With the components as follows in Equations 74, 75 and 76.

$$b_{11} = \frac{E_{11}}{1 - \nu_{12}^2 \frac{E_{22}(1-d')}{E_{11}}} \quad (74)$$

$$b_{12} = \frac{\nu_{12} E_{22}}{1 - \nu_{12}^2 \frac{E_{22}(1-d')}{E_{11}}} \quad (75)$$

$$b_{22} = \frac{E_{22}(1-d')}{1 - \nu_{12} \nu_{12}^2 \frac{E_{22}(1-d')}{E_{11}}} \quad (76)$$

These components are then used to update the stresses as per Equation 73.

The basic damage model was first implemented in Mathcad to test the formulation and structure of the subroutine. The calculations were performed assuming only σ_{12} loading for the F185/S2 glass prepreg in GLARE and the IM6/914 carbon prepreg tested by Ladevèze and Le Dantec (1992). This loading would occur in a prepreg layer oriented at 45° . The results of these validation calculations and the input parameters are shown in Table 20. Rupture occurs at a value of $d_c < 1$ as anticipated by Kachanov. The experimental rupture stress of IM6/914 as measured by Ladevèze and Le Dantec (1992) is also provided. As can be seen in the table, the calculated rupture stress for the IM6/914 composite agrees very well with the experimental value reported by Ladevèze and Le Dantec (1992). Ladevèze and Le Dantec also quote a predicted value of 79 MPa. No

data was available for the shear strength of the F185/S2 glass prepreg. However, Johnson (1975) gives 41 MPa as the in-plane shear strength of a generic unidirectional, fibreglass/epoxy composite. This value is indicated with an asterisk (*) in Table 20.

Table 20: Summary of shear rupture estimates and material properties for F185/S2 and IM6/914 prepregs.

Property	Units	F185/S2	IM6/914
$(\sigma'_{12})_{prediction}$	MPa	45	83
$(\sigma'_{12})_{experiment}$	MPa	41*	82
d_c		0.470	0.473
Y_o	MPa ^{1/2}	0.12	0.15
Y_o'	MPa ^{1/2}	0.0	0.24
Y_c	MPa ^{1/2}	2.0	2.77
Y_c'	MPa ^{1/2}	2.3	3.78
b	none	3.4	2.5
Y_s	MPa ^{1/2}	1.1	0.7
E_{11}	GPa	53.98	170
E_{22}	GPa	9.412	10.8
E_{33}	GPa	9.412	
G_{12}	GPa	3.310	5.8
ν_{12}	none	0.330	0.34
ν_{13}	none	0.058	
ν_{23}	none	0.058	

The main assumption with this material model is that damage forms due to the in-plane shear and in-plane transverse stresses. The result of this assumption is that there is no stiffness loss in the fibre direction since no damage is predicted in this direction. Rupture in the fibre direction can be predicted using a strength limit with the assumption that the deformation is elastic up until rupture. The impact experiments and subsequent damage characterization described in Section 2.2 showed that there was no fibre damage below the puncture threshold in the impacted FMLs. Damage consisted of matrix cracks, transverse cracks and delamination. Therefore, fibre damage was not incorporated into the prepreg model. One method for including these effects is to use Hashin's criteria for fibre failure as described by Matzenmiller *et al.* (1995). Fibre damage could be

incorporated provided the appropriate strength parameters were measured or available in literature. This would not be difficult, but was not required in the present application.

When the damage level predicted by the model in an element reaches the critical value the values of d and d' are immediately set to 1 and the element fails. The element can also fail if the numerical limits in Equations 66 and 67 are exceeded. Two options are available for handling the failed element. The first is to use the built-in element deletion capability of LS-DYNA to completely remove the element from the mesh. If this approach is used, the element is not included in any further stress calculations and load carrying capacity in all directions is lost. However, this is not realistic; in fact, when compressive loads are applied, matrix cracks would close and load would continue to be transmitted. An additional problem with deleting the elements is that the aluminum layers that had been separated by the prepreg layer can then come into contact. This situation, of course, does not occur in the experiments. Even after delamination and cracking have occurred in the prepreg layers, they still carry loads and keep the aluminum layers separate. Furthermore, in LS-DYNA it is impractical to establish multiple contact definitions for layers that might come into contact if the elements between them fail.

Another method for handling the failed element is to set the stresses in the tensile 22 and shear 12 directions to zero. This is performed over 100 time steps to prevent a stress wave from the sudden loss of stiffness. Stresses in the tensile and compressive 11, 22 and 33 directions are still calculated. This allows the element to continue to carry compressive loads in all directions and tensile loads in the fibre direction. This technique was incorporated into the UMAT.

6.3 Validation of Elastic Model

A pair of coupon simulations were conducted using the CDM-based user-defined subroutine with damage deactivated and the standard orthotropic-elastic (type 2) model. The purpose of these simulations was to validate the portions of the code not related to the calculation of damage in the prepreg material. The formulation of the elastic portions is identical to the definition of the orthotropic-elastic (type 2) material in LS-DYNA. The coupon stresses and strains predicted by the user-defined material model (UMAT) should be identical to values predicted using the orthotropic-elastic (type 2) material model. The coupon used for these simulations is shown in Figure 76.

Figure 76: Prepreg elastic response verification tensile coupon (half-geometry).

The first simulation used the standard LS-DYNA orthotropic-elastic material model and the second used the UMAT with the damage calculation portions deactivated. The lay-up of the coupon was chosen as $[15^\circ]_8$. The elastic properties of the prepreg material are summarized in Table 21. The stress-strain response of the two simulations is shown in Figure 77. There is very good agreement between the two material models.

Table 21: Elastic properties of the F185/S2 prepreg material.

Property	Units	Value
E_{11}	GPa	53.98
E_{22}	GPa	9.412
E_{33}	GPa	9.412
G_{12}	GPa	3.310
ν_{12}	none	0.330
ν_{13}	none	0.058
ν_{23}	none	0.058

Figure 77: Comparison of simulated stress-strain response for prepreg coupon using the orthotropic-elastic model and the UMAT without damage.

6.4 Validation of the Damage Model

This section describes the results of finite element simulations and experiments carried out on GLARE coupons. The purpose of these tests was to verify the UMAT prior to implementing it for impact simulations. A series of static tensile tests was conducted using various types of GLARE-3. These tests were conducted to provide data to support other ongoing tasks on the FML Project at IAR-SMPL. As will be described below, some of these data were used for the CDM-based model validation. Validating the damage portions of the CDM-based model prior to application to impact simulations was necessary to eliminate the possibility of problems with the model.

6.4.1 Static Tensile Testing of GLARE

Coupons of GLARE-3-3/2, GLARE-3-4/3 and GLARE-3-5/4 were subjected to static tensile loading to measure their tensile properties. The test methodology and specimen design were conducted in accordance with the FML tensile test method described by Mattousch (1993). Specimens were cut from large GLARE panels to the dimensions shown in Figure 78.

Figure 78: Static tensile strength coupon.

Specimens were tested in an Instron load frame at IAR-SMPL with a loading rate of 2 mm/min. Displacement data was collected using an extensometer attached to the centre of the specimen. Results from the tests are summarized in Table 22. These results are to be published in an IAR-SMPL laboratory memorandum at a future date.

Table 22: Summary of static tensile results for the GLARE-3 test coupons.
Each data point is an average of 5 tests.

Specimen ID	Modulus MPa	Yield Stress MPa	% Elongation at Failure	UTS MPa
GL-3-3/2 0°	55443	304	4.5	698
GL-3-3/2 15°	53198	270	6.0	521
GL-3-4/3 0°	56458	307	4.2	705
GL-3-4/3 15°	53417	269	5.9	526
GL-3-5/4 0°	51930	296	4.5	713
GL-3-5/4 15°	51534	261	6.4	532

The GLARE-3-3/2 data were then used to validate the CDM-based damage model through comparisons with coupon simulations in LS-DYNA. Typical measured stress-strain curves for the 0° and 15° directions are shown in Figure 79.

Figure 79: Typical GLARE-3-3/2 measured stress-strain curves for 0° and 15°.

6.4.2 Tensile Simulations of GLARE in LS-DYNA

Simulations were developed to provide results for comparison to those obtained from the experiments described in Section 6.4.1. Two simulated geometries were prepared. The low-density mesh coupon geometry is shown in Figure 80. This coupon has straight edges and was suggested by Ladevèze and Le Dantec (1992) for validating the CDM theory. The high-density mesh coupon is shown in Figure 81. This coupon represented the experimental configuration shown in Figure 78. As can be seen in these two figures, the high-density coupon is 10 elements wide in the test section, whereas the low-density coupon is only 5 elements wide. Both simulated coupons consisted of a test

region and “grips” that transfer the loads into the test region. Half-symmetry was used to reduce computational requirements. A complete list of the coupon simulations carried out is presented in Table 23.

Figure 80: Low-density mesh GLARE coupon geometry.

Figure 81: High-density mesh GLARE coupon geometry.

Table 23: List of prepreg and GLARE-3-3/2 coupon simulations.

SIM ID	Material Model	Lay-up	Mesh Density
1	Orthotropic-elastic	[2024/(0°/90°)/2024/(90°/0°)/2024]	Low
2	Orthotropic-elastic	[2024/(15°/105°)/2024/(105°/15°)/2024]	Low
3	UMAT	[2024/(0°/90°)/2024/(90°/0°)/2024]	Low
4	UMAT	[2024/(15°/105°)/2024/(105°/15°)/2024]	Low
5	UMAT	[2024/(0°/90°)/2024/(90°/0°)/2024]	High
6	UMAT	[2024/(15°/105°)/2024/(105°/15°)/2024]	High

These simulations used prepreg properties published by Hurez *et al.* (1993) as listed in Table 24. The aluminum properties were presented earlier in Section 4.1. The aluminum layers were 0.3 mm thick and the prepreg layers were 0.125 mm thick. The coupons were composed of 7 layers of thick-shell elements.

Table 24: Summary of prepreg constituent properties.

Property	Units	Value
Y_o	MPa ^{1/2}	0.12
Y_o'	MPa ^{1/2}	0.00
Y_c	MPa ^{1/2}	2.00
Y_c'	MPa ^{1/2}	2.30
b	none	3.40
Y_s	MPa ^{1/2}	1.10
E_{11}	GPa	53.98
E_{22}	GPa	9.412
E_{33}	GPa	9.412
G_{12}	GPa	3.310
ν_{12}	none	0.330
ν_{13}	none	0.058
ν_{23}	none	0.058

Thick-shell elements were used to represent the GLARE coupons and solid-bricks were used for the “grip” load introduction components. No delamination was modelled in these simulations. The orthotropic-elastic simulations (1 and 2) modelled the prepreg material as elastic without damage. The UMAT (3-6) simulations used the CDM-based prepreg model. In all cases the aluminum was modelled using the elastic-plastic model

described previously. The simulations were not carried to complete rupture of the specimens since the material model did not incorporate fibre fracture.

The two coupons were selected to study the effect of mesh density on the results. The effect of this increase in mesh density on the stress-strain response is shown in Figure 82. Two pairs of stress-strain curves are shown in this figure. The first pair labelled “3” and “5” have the material oriented at 0° with respect to the loading direction. Simulation 3 uses the low-density coupon and simulation 5 uses the high-density coupon. The second pair of simulations had the material oriented at 15° with respect to the loading direction. Simulation 4 used the low-density coupon and simulation 6 used the high-density coupon.

For the lower density mesh used in simulation 4, there are drops in the stress-strain curve near the end of the simulation. These occur when one of the elements fails and its load carrying capacity is eliminated. Since there are only a few elements across the width of the specimen, a significant portion of the strength of the coupon is lost suddenly. The higher density mesh used in simulations 5 and 6 did not show the same effect and the stress-strain curve is somewhat smoother. The stress drops were not observed in the 0° simulations for either mesh density.

Figure 82: Effect of mesh density on the stress-strain response of the GLARE coupons.

As shown in Figure 83, there was good agreement between the experimental stress-strain measurements and the simulated results for GLARE-3-3/2 in the 0° direction. The graph compares the results of the experiment to simulations using the low- (3) and high- (5) density meshes with the UMAT, and low-density mesh using the UMAT without damage for the prepreg (1). As would be expected, the orthotropic-elastic simulation (1) was stiffer than the experiment for the 0° case. The orthotropic-elastic model does not account for any stiffness degradation in the prepreg.

For the 0° cases (3 and 5) with damage, the results show slightly reduced stiffness compared to the 0° orthotropic-elastic simulation (1). In the 0° fibre-direction, the UMAT calculates no stiffness losses and only stiffness losses in the 90° fibre layers are calculated. Again, in practice there would be some damage formed by fibre fracture and also matrix deformation in the 0° plies, that is not accounted for in the model.

Figure 83: Comparison between simulated stress-strain response and experimental data for 0° GLARE-3-3/2.

For the off-axis 15° cases (4 and 6) the results obtained with the UMAT model are closer to the experimental results as shown in Figure 84. However, there is more strength loss predicted by the UMAT model for simulation 4 than was observed in the experiments. This is the effect of the lower mesh density as described above. The simulation results for the high-density mesh (6) show very good agreement with the experiments.

Figure 84: Comparison between simulated response and experimental data for 15° GLARE-3-3/2 loaded in tension.

The GLARE coupon simulations successfully demonstrated the application of a CDM-based model to predict the tensile response of GLARE. The UMAT was then applied to model the behaviour of the GLARE prepreg material in complete impact simulations.

CHAPTER 7

SIMULATION OF LOW- VELOCITY IMPACTS ON GLARE LAMINATES

Based upon the success of the UMAT simulations in predicting the tensile response of GLARE, the modelling methodology was applied to low-velocity impact simulations. These simulations duplicate the experiments presented in Chapter 2. The effects of several parameters were studied including mesh density, interface type and the type of material model. Section 7.1 describes the application of the damage modelling methodology and how the individual components from Chapters 4, 5 and 6 were combined. Sections 7.2, 7.3 and 7.4 discuss details of the input parameters, mesh sensitivity and simulation results.

7.1 Application of the FML Damage

Modelling Methodology

The components of the FML damage modelling methodology presented in Chapters 4, 5 and 6 were assembled to model the impact response of GLARE coupons. The most important part of the methodology is the CDM-based prepreg model. The properties described by Ladevèze and Le Dantec (1992) and Hurez *et al.* (1993) were used in this material model. The material model was implemented in LS-DYNA as described in Section 6.2. The normal and shear strength properties of the interface were provided in the manufacturer's data sheets for the adhesive. It was not possible to employ the data presented in Section 5.2.4 to model the post-delamination response. Therefore, the correction applied to the interface was a linear stress-displacement curve. The aluminum layers were modelled using the elastic-plastic (type 24) material model in LS-DYNA with material properties as presented in Section 4.1.

The components of the modelling methodology were applied as follows:

1. Elastic-plastic aluminum, orthotropic elastic prepreg and tied-interface,
2. Elastic-plastic aluminum, orthotropic elastic prepreg and uncorrected tiebreak-interface,
3. Elastic-plastic aluminum, orthotropic elastic prepreg and corrected-tiebreak interface, and
4. Elastic-plastic aluminum, UMAT prepreg and simple-tiebreak interface.

7.2 GLARE Simulation Input Parameters

The geometry of the simulations was identical to that used for the final aluminum simulations described in Chapter 4. For reference, the mesh geometry is repeated in Figure 85. The aluminum layers were modelled with the elastic-plastic (type 24) material model using properties presented previously in Chapter 3 and repeated for reference in Table 25 and Figure 86. The prepreg layers were modelled using either the orthotropic-elastic (type 2) material (Section 3.3.6) or the UMAT (Chapter 6). For reference, the prepreg properties are repeated in Table 26. All of the layers of the simulations were modelled using thick-shell elements.

Figure 85: Mesh geometry for GLARE impact simulations.

Table 25: Aluminum material properties.

Property	Value	Unit
Young's Modulus	72.39	GPa
Yield Stress	285	MPa
Mass Density	2.643×10^{-6}	Kg/mm ³
Poisson's Ratio	0.33	-

Figure 86: Stress-strain curve for aluminum 2024-T3.

Table 26: Prepreg material input properties.

Property	Units	Value
Y_o	$\text{MPa}^{1/2}$	0.12
Y_o'	$\text{MPa}^{1/2}$	0.00
Y_c	$\text{MPa}^{1/2}$	2.00
Y_c'	$\text{MPa}^{1/2}$	2.30
b	none	3.40
Y_s	$\text{MPa}^{1/2}$	1.10
Y_r	$\text{MPa}^{1/2}$	2.00
E_{11}	GPa	53.98
E_{22}	GPa	9.412
E_{33}	GPa	9.412
G_{12}	GPa	3.310
ν_{12}	none	0.330
ν_{13}	none	0.058
ν_{23}	none	0.058

Simulations were carried out with three different interface modelling techniques. These models were discussed in detail in Section 3.3.5. To review, the tied-interface connects the layers together without delamination. Next, the simple-tiebreak interface uses the tiebreak model with the manufacturer's shear and normal interface strengths of 35 MPa and 48 MPa. Finally, the corrected-tiebreak interface method uses the tiebreak

model with a stress-displacement correction curve to model the post-failure behaviour. This curve defines the delamination opening resistance stress as a function of the separation distance between the nodes at the contact interface. The simplest model is a linear decrease in opening stress as the separation distance increases. The curve starts at the normal strength of the adhesive and decreases linearly to zero over a distance of 1.5 mm. This distance was chosen based on the destructive inspections presented in Chapter 2. The scale of the delamination openings was on the order of 1 to 2 mm. The assumed stress-displacement curve is shown in Figure 87.

It was found, for both the simple- and corrected-tiebreak interface simulations with elastic prepreg, that the manufacturer's properties were too high to allow interface failure. The values were lowered until delamination occurred in the specimens. This occurred with property values at approximately 0.025 of the quoted strengths. As described in Section 5.2.4, the application of the DCB data to correct this response was unsuccessful. When the UMAT was used with the simple-tiebreak interface, the unchanged manufacturer's properties were successfully employed to predict delamination.

Figure 87: Stress-displacement correction curve.

A total of 31 simulations were carried out as shown in Table 27. This table summarizes the material models, interface types, element types, impact energy levels and simulation ID numbers.

Table 27: Summary of GLARE low-velocity impact simulations.

Simulation ID	Material	Element Type	Energy Level	Material Model	Interface Type
1	GL-3-2/1	Thick	25	Piecewise Linear/ Orthotropic	Tied
2	GL-3-2/1	Thick	35	Piecewise Linear/ Orthotropic	Tied
3	GL-3-2/1	Thick	45	Piecewise Linear/ Orthotropic	Tied
4	GL-3-2/1	Thick	55	Piecewise Linear/ Orthotropic	Tied
5	GL-4-2/1	Thick	25	Piecewise Linear/ Orthotropic	Tied
6	GL-4-2/1	Thick	35	Piecewise Linear/ Orthotropic	Tied
7	GL-4-2/1	Thick	45	Piecewise Linear/ Orthotropic	Tied
8	GL-4-2/1	Thick	55	Piecewise Linear/ Orthotropic	Tied
9	GL-5-2/1	Thick	25	Piecewise Linear/ Orthotropic	Tied
10	GL-5-2/1	Thick	35	Piecewise Linear/ Orthotropic	Tied
11	GL-5-2/1	Thick	45	Piecewise Linear/ Orthotropic	Tied
12	GL-5-2/1	Thick	55	Piecewise Linear/ Orthotropic	Tied
13	GL-3-2/1	Thick	25	Piecewise Linear/ Orthotropic	Simple
14	GL-3-2/1	Thick	35	Piecewise Linear/ Orthotropic	Simple
15	GL-3-2/1	Thick	55	Piecewise Linear/ Orthotropic	Simple
16	GL-4-2/1	Thick	35	Piecewise Linear/ Orthotropic	Simple
17	GL-4-2/1	Thick	45	Piecewise Linear/ Orthotropic	Simple
18	GL-4-2/1	Thick	55	Piecewise Linear/ Orthotropic	Simple
19	GL-3-2/1	Thick	25	Piecewise Linear/ Orthotropic	Corrected
20	GL-3-2/1	Thick	35	Piecewise Linear/ Orthotropic	Corrected
21	GL-3-2/1	Thick	45	Piecewise Linear/ Orthotropic	Corrected
22	GL-3-2/1	Thick	25	Piecewise Linear /UMAT	Simple
23	GL-3-2/1	Thick	35	Piecewise Linear /UMAT	Simple
24	GL-3-2/1	Thick	45	Piecewise Linear/UMAT	Simple
25	GL-4-2/1	Thick	35	Piecewise Linear/UMAT	Simple
26	GL-4-2/1	Thick	45	Piecewise Linear/UMAT	Simple
27	GL-4-2/1	Thick	55	Piecewise Linear/UMAT	Simple
28	GL-5-2/1	Thick	35	Piecewise Linear/UMAT	Simple
29	GL-5-2/1	Thick	45	Piecewise Linear/UMAT	Simple
30	GL-5-2/1	Thick	55	Piecewise Linear/UMAT	Simple
31	GL-5-2/1	Thick	65	Piecewise Linear/UMAT	Simple

7.3 Mesh Sensitivity

A mesh density study was conducted using GLARE-3-2/1 simulations with three mesh density levels. These simulations used the UMAT material model for the prepreg with the simple-tiebreak interface model for initiation of delamination. The basic simulation parameters are presented in Table 28. The remaining parameters were presented in the previous section in Figures 85 to 87 and Tables 25 and 26.

Table 28: Summary of GLARE mesh density simulation parameters.

Run ID	# of Elements	# of Nodes	# of Cycles	CPU Time
				(s)
1-low	1636	3437	270979	17644
2-medium	2500	5229	270609	27644
3-high	5500	11389	273158	68716

As can be seen in Figure 88, there was little effect of mesh density on the predicted levels of absorbed energy. The agreement between the three mesh densities was also apparent from the force versus time curves plotted in Figure 89. It is only for the deflection curves in Figure 90 that some difference in the response of the three panels is observed. The low-density mesh shows less permanent deflection than the medium- and high-density meshes.

Figure 88: Impact energy versus time for three different mesh densities of a GLARE-3 impact simulation.

Figure 89: Impact force versus time for three different mesh densities of a GLARE-3 impact simulation.

Figure 90: Dent depth versus time for three different mesh densities with GLARE-3.

The predicted damaged panel geometries are shown in Figure 91. The medium- and high-density meshes give similar predicted damage geometries. However, the low-density mesh showed larger amounts of delamination damage. The smaller number of nodes in the low-density simulation is likely the cause of the increased delamination. There are fewer nodes to be connected by the simple-tiebreak interface delamination criterion. A large area of delamination is created when one node is released after the failure criterion is reached. As the number of nodes decreases, the size of the delaminated region increases when a single node is released. Another effect of larger elements is that the damage is distributed through fewer elements. Therefore, a single element would have to accumulate a greater amount of damage before failing. Based on this study, the medium density mesh was selected for the remainder of the GLARE impact simulations.

Figure 91: Predicted impacted panel geometries for (a) low-, (b) medium- and (c) high-density meshes.

7.4 Simulation Results

The measured and predicted absorbed energy, dent depth and peak force results are presented in the following sections. These damage comparisons include maps of the delamination damage in the simulations and experiments. A complete tabulation of the simulation results is provided in Appendix G. Note that the steps near the end of each set of experimental data are due to puncture damage developing in some of the impacted panels.

7.4.1 Absorbed Energy

Typical energy versus time plots from the simulations and experiments conducted using GLARE-3 impacted at 25 J are shown in Figure 92. The simulations performed using the orthotropic-elastic prepreg (type 2) predicted lower absorbed energies than was measured in the experiments. However, the simulation that used the UMAT showed a definite improvement in the predicted amount of absorbed energy. One possible source of the remaining difference between the predictions and the experiments is the method by which the energy is calculated by the drop tower software. The experimental impact energy is calculated directly from the impact force (Equations 1-4). Thus, the experimental energy would include any system losses. Further differences could also result from the experimental boundary condition that is between simply-supported and clamped.

Figure 92: Typical absorbed energy versus time curves for simulations and experiments with GLARE-3 impacted at 25 J.

Figures 93 to 95 summarize the results of all of the simulations for GLARE-3, GLARE-4 and GLARE-5 and compare the values with the experiments. However, the predicted values were still lower than the experimental values. A similar discrepancy was observed for the aluminum simulations (Chapter 4). The lack of fixture losses could also be affecting the GLARE simulations. The simple-tiebreak interface simulations showed the lowest absorbed energy of all types. This is due to the rebound of the prepreg layers, an action that releases stored elastic energy from these layers.

Figure 93: Absorbed Energy versus Impact Energy for GLARE-3.

Figure 94: Absorbed Energy versus Impact Energy for GLARE-4.

Figure 95: Absorbed Energy versus Impact Energy for GLARE-5.

7.4.2 Permanent Dent Depth

A plot of the typical displacements measured for GLARE-3-2/1 impacted at 25 J is shown in Figure 96. Note that the experimental displacement curve is calculated as per Equations 1-4 from the impact force data by the drop tower software. The experimental set-up did not include equipment to measure the displacement of the panel during impact. However, the dent depths from the simulations are direct measurements of nodal displacements at the point of impact. The UMAT simulations predict higher dent depths than the other simulations. As shown below, the permanent dent depth measurements from the experiments agree favourably with the UMAT simulations.

Figure 96: Typical dent depth versus time plots for GLARE-3-2/1 impacted with 25 J.

The effects of including the UMAT subroutine on the predictions of permanent dent depth are shown in Figure 97 to 99. The simple-tiebreak interface dent depth results were significantly lower than the other simulation results and the experimental values. The permanent dent depth from the experiments was measured directly from the panels using a vernier depth gauge.

Figure 97: Dent Depth versus Impact Energy for GLARE-3.

Figure 98: Dent Depth versus Impact Energy for GLARE-4.

Figure 99: Dent Depth versus Impact Energy for GLARE-5.

7.4.3 Peak Impact Force

The last comparison between the GLARE simulation results and the experiments is the peak impact force. This is the peak force measured between the specimen and the impactor during impact. Plots of the typical impact force versus time measurements for the simulations and experiments with GLARE-3-2/1 impacted at 25 J are shown in Figure 100. The simulations conducted using the UMAT model gave a lower peak impact force than the other simulations. However, the forces were still higher than those measured during the experiments. The experiments showed less vibrations than the simulations. Damping in the structure of the crosshead, where the load is measured, could be the cause of this effect.

Figure 100: Typical impact force versus time plots for GLARE-3-2/1 impacted at 25 J.

The tied-, simple-tiebreak, and corrected-tiebreak interface models, shown in Figures 101 to 103, all over-predict the peak impact force to approximately the same degree. These models use an elastic model for the prepreg layers. In the UMAT model, the prepreg layer loses stiffness as it accumulates damage and the panel is not able to resist the penetration of the impactor to the same degree as it does with the elastic material model. While the UMAT-based simulations still over-predict the impact force, they are an improvement over the other three techniques.

Figure 101: Maximum Force versus Impact Energy for GLARE-3.

Figure 102: Maximum Force versus Impact Energy for GLARE-4.

Figure 103: Maximum Force versus Impact Energy for GLARE-5.

7.4.4 Predicted Damage Geometry

Figure 104 shows delamination damage recorded during the experimental impact testing of GLARE-3. Images of the typical predicted delamination damage for each type of simulation are shown in Figure 105a-d. Figure 105a shows the typical damage geometry predicted using the tied-interface. There is no delamination predicted using this interface model.

As shown in Figure 105b, using the simple-tiebreak interface with an elastic prepreg resulted in the largest difference in predicted geometry. These simulations were first attempted using the manufacturer's adhesive property data ($s_n=48$ MPa and $s_s=30$ MPa). However, it was found that no delamination damage was predicted in the panels using these values. Reduced strengths were simulated until it was found that a value of

approximately 0.025 of that suggested by the manufacturer resulted in the formation of delamination damage.

As described in Section 3.3.5, a correction curve can be used to modify the separation behaviour of the layers in LS-DYNA. An image of the typical corrected-tiebreak interface delamination damage is shown in Figure 105c. This correction did improve the predictions of the damage geometry but did not improve the predicted response of the panels. Figure 105d shows the typical geometry predicted using the simple-tiebreak interface model with the UMAT material model. The damage geometry is closer to the experiments compared to the simple-tiebreak interface model and the predicted response of the panels is improved as well.

Figure 104: Delamination damage in GLARE-3 impacted at ~45J.

Figure 105: Typical delamination damage predicted using the (a) tied-interface, (b) simple-tiebreak interface, (c) corrected-tiebreak interface and (d) UMAT/simple-tiebreak interface models.

Figures 106 to 113 contain maps of the delamination damage predicted in the simulations with comparisons to the damage observed in the x-rays and the destructive inspections. The damage maps were created by examining the geometry of the final simulations and locating each delaminated element. Note that each small square in the damage maps is 3.75 mm x 3.75 mm. The x-rays are shown in the same scale as the damage maps. The bars across each image are the delamination width measured from the destructive inspections. “Upper” refers to delamination between the upper (impacted) aluminum layer and the prepreg and “lower” refers to delamination between the lower aluminum layer and the prepreg.

Both the simple-tiebreak interface and corrected-tiebreak interface simulations over-predicted the amount and extent of the delamination damage for the GLARE-3. In general the predicted delamination damage from the UMAT simulations agreed well with the damage patterns from the experiments.

The GLARE-4 damage patterns show that the delamination damage is more extensive in one direction than the other. This is a consequence of the 70/30 split between the fibres in the 0° and 90° directions. The same effect was noted in the x-rays of the delaminated GLARE-4 specimens. This difference in the amount of fibres ultimately leads to fracture perpendicular to the weaker fibre direction when the impact load is high enough.

The predicted GLARE-5 damage geometries also agreed very well with the experiments. As shown in Figures 111 to 113, the shape of the predicted delamination is oval in shape. The same shape was observed in the impacted test coupons.

Figure 106: Damage maps of GLARE-3 impacted at 25 J from experiment, simple-tiebreak interface, corrected-tiebreak interface and UMAT simulations. [Note that the x-ray did not show any damage.]

Figure 107: Damage maps of GLARE-3 impacted at 35 J from experiment, simple-tiebreak interface, corrected-tiebreak interface and UMAT simulations.

Figure 108: Damage maps of GLARE-4 impacted at 35 J from experiment, simple-tiebreak interface and UMAT simulations. [Note that no x-ray was available for this specimen.]

Figure 109: Damage maps of GLARE-4 impacted at 45 J from experiment, simple-tiebreak interface and UMAT simulations.

Figure 110: Damage maps of GLARE-4 impacted at 55 J from experiment, simple-tiebreak interface and UMAT simulations.

Figure 111: Damage maps of GLARE-5 impacted at 35 J for experiment and UMAT simulations. [Note that the x-ray did not show any damage.]

Figure 112: Damage maps of GLARE-5 impacted at 45 J for experiment and UMAT simulations. [No destructive inspection was carried out for this sample.]

Figure 113: Damage maps of GLARE-5 impacted at 55 J for experiment and UMAT simulations.

CHAPTER 8

SUMMARY AND DISCUSSION OF RESULTS

A modelling methodology to predict the formation of impact damage in FMLs was developed as part of this research. An extensive series of impact and characterization tests was carried out to support the development of the methodology. This methodology was used to study the impact response and resulting damage geometry of GLARE subjected to low-velocity impact. The damage modelling methodology can be developed further to predict failure of all layers of the FML.

8.1 Experimental Impact Testing

The impact tests conducted as part of this work provided several insights into the development of impact-induced damage in FMLs. For example, damage was observed to vary with the applied impact energy as follows:

- plastically deformed dent with no delamination,
- formation of internal delaminations and matrix cracking, damage becomes visible in penetrant enhanced x-rays, and
- puncture then occurs and the geometry of the puncture depends on the GLARE lay-up as discussed below.

It was found that, when puncture occurs in the GLARE-3 and GLARE-5 laminates, the impactor did not rebound and the panel absorbed all of the impact energy. However, there is a limit to the amount of energy that can be absorbed by the panel. Beyond this limit the impactor will pass through the panel, retaining a portion of its kinetic energy. This remaining energy is absorbed by the structure of the drop tower when the crosshead is arrested by large rubber stops. The GLARE-5 laminates absorbed less energy, on an areal density basis, than GLARE-3, GLARE-4 and 2024-T3 when subjected to low-velocity impact (Figure 18). In these laminates, a greater portion of the impact energy is converted to elastic potential energy and then returned to the impactor in these laminates. This is due to the larger amount of glass-fibres in the 0° and 90° directions of the impact-optimized GLARE-5 laminates compared to GLARE-3 and GLARE-4.

The lay-up of the specimens strongly influenced the resulting damage geometry, as shown by the formation of a single crack in GLARE-4 (Figure 30). In these laminates, the crack was oriented parallel to the major fibre direction, which is also the loading axis for GLARE-4. Thus, in locations such as in fuselage or wing skins, impact-induced cracks in GLARE-4 would not grow if the load was parallel to the crack direction. In the majority of applications the applied loads on the laminate would not be uniaxial, but would have an off-axis resultant, and would cause some crack growth. For example, as was described by Laliberté *et al.* (2001a), impact-induced cracks oriented at 45° to the loading axis were observed to grow under cyclic loading. In GLARE-3 and GLARE-5, multiple cracks formed as a result of impact (Figures 29 and 31). The damage in GLARE-3 and GLARE-5 did not change appreciably in size once the panel was punctured.

Since only GLARE-X-2/1 laminates were available for these tests, other types of GLARE should be subjected to impact loading in the future. In particular, thicker laminates should be tested to examine the influence of the number of layers on the formation of damage.

The circular impact specimen developed in this research project removed the corner deformations that were observed in rectangular specimens supported in the NASA fixture. The deformation and slippage resulting from the use of the circular fixture was limited to the periphery of the specimens, well away from the central dent. This distance between the fixture-induced deformation and the central dent increased with the use of the circular fixture compared to the square fixture (Figure 12).

Overall, the impact test method used in this thesis was successful. It may be beneficial to make minor changes to improve the test set-up further. In particular, a sensor to measure the displacement of the panel as it is impacted could be incorporated into the drop tower. This could either be a contact device such as an LVDT or a non-contact sensor such as a laser-based displacement measuring system. A high-speed camera could also provide qualitative data about the impact response of the panels.

To measure the difference between the impact force recorded by the load cell in the crosshead and the actual impact force on the panel a separate force measurement system is recommended. A series of four load cells could be used in the fixture support pillars to record this load. The difference between this measurement and the crosshead measurement would provide a measure of the system losses.

The non-destructive inspection method used to examine the impact GLARE panels accurately revealed the delamination in specimens impacted at energies greater than 35 J. In the lower energy specimens, this technique did not expose the delamination damage. With some modifications to the technique, such as the use of multiple holes, it would be possible to fully infiltrate the damaged regions with penetrant. Although the destructive inspection technique accurately depicted the damage in the specimens, this method is impractical for application to actual aircraft. Additional technologies should be investigated for application to the non-destructive inspection of impacted FMLs. These could include contact c-scan ultrasonic probes and laser ultrasound.

It was noted that all of the specimens showed significant matrix cracking in the unpunctured panels. This form of damage contributed to the overall energy absorption of the impacted panels. Fibre fracture was only observed in panels that had been punctured.

8.2 Damage Modelling Procedure for FMLs

In order to rapidly examine a large number of geometries, boundary conditions and lay-ups a damage modelling methodology was required for FMLs. Following a review of pertinent literature, a damage modelling methodology for FMLs was proposed to enable the prediction of the extent and type of damage in impacted FMLs. This methodology required four components. These consisted of an elastic-plastic aluminum model, a prepreg damage model, a delamination initiation model and a post-delamination growth model. These components were each developed and validated through simulations in LS-DYNA and experimental testing.

8.2.1 Impact Damage Modelling of Aluminum

The first component of the modelling methodology investigated was the elastic-plastic model for the aluminum layers. The aluminum simulations showed that boundary conditions have an effect on the predicted response of the panels. Panels with a clamped boundary absorbed 1 J to 3 J less than those with simply-supported edges. The simply-supported panels deflected more than the clamped panels and thus absorbed more strain energy. In the experimental impact tests the boundary condition of the panels lies between the two extremes of clamped and simply-supported. There was evidence of slipping at the edges of the impacted panels. A model of the entire impact fixture could be employed to investigate this behaviour further.

The aluminum impact simulation results also showed sensitivity to the input material properties. A wide range of properties for aluminum 2024-T3 is available in

literature and it was desired to study the effect of using different “acceptable” properties on the impact response of the panels. Yield stresses of 285 MPa, 350 MPa and 420 MPa from the manufacturer, MIL-HDBK-5H (DoD, 1998) and ASM (1990) were used. In the panel with the yield stress of 285 MPa, a greater amount of plastic deformation occurred. This panel absorbed more impact energy through increased permanent deformation, as shown in Figure 63. The peak impact forces also decreased as shown in Figure 64. The impact results from the panels with the manufacturer’s properties were closest to the experimental values. Therefore, the manufacturer’s properties were used for the remaining simulations. Where possible, it is always best to have access to actual property data for the material under investigation.

In finite element analyses, studies of sensitivity of the solution to the mesh size are always recommended. In the present work, the mesh density did not influence the results of the aluminum simulations. However, the type of element did have an effect on some results. The predicted absorbed energy and peak impact force using thin-shell, thick-shell and solid-brick elements were in agreement. The predicted values of absorbed energy were consistently lower than the experimental values, by an average of 4 J. The predicted peak impact force values were also higher than the experimental values by an average of 1.2 kN. The source of these two differences may be related to the lack of fixture losses in the model. With the current experimental set-up it was not possible to quantify these losses. A fully detailed impact fixture simulation would be required to investigate the effects of fixture losses and boundary conditions on the predicted impact response.

It was noted that the solid-brick elements used resulted in predicted dent depths approximately 1.7 mm less than the thin- and thick-shell elements. The dimensions of the solid-brick elements in the simulated specimen could explain this effect. It has been indicated in the literature that these elements exhibit increased stiffness when they have high ratios of length to thickness (Hallquist, 1998). This would have produced higher peak impact forces for the solid-brick simulations as compared to the other element types, which was not the case. The formulation of the solid-brick elements is likely the source of this difference. It was decided to forgo an in-depth evaluation of the solid-brick element formulation.

8.2.2 Delamination Growth Characterization

As previously explained in Section 5.2.4, an attempt was made to use the DCB data from the tests conducted as part of this project in LS-DYNA. However, it was found that the load-displacement curves could not be converted to the required stress-displacement curves. Therefore, a simple stress-displacement curve was used within LS-DYNA instead.

The test method must be re-evaluated if it is to be used to develop a delamination correction for GLARE in the future. Direct measurement of the forces and displacements at the interface may be required. An extensometer attached directly to the specimen would improve the accuracy of the measurements recorded during the testing. A smaller specimen that has the delamination growing over a much shorter length may allow for the measurement of the fibre-bridging loads in the laminate. This specimen could be built on a scale similar to that of a single element in the finite element simulations to provide

meaningful fibre-bridging loads. A pressure or a piezoelectric sensor could be used to monitor the stresses in the interface before and after delamination. Care would have to be taken to ensure that any in-situ sensor would not influence the formation of damage. It would be possible to use the measured stress-displacement data directly in LS-DYNA.

8.2.3 Development of Prepreg Damage Model

The development of a material model based upon the theories of Ladevèze and Le Dantec (1992) and its implementation in LS-DYNA, as described in Section 6.2, was successful.

The model was initially evaluated in Mathcad. For a carbon fibre prepreg, a shear strength of 82 MPa was predicted versus an experimental value of 83 MPa and a predicted value of 79 MPa reported by Ladevèze and Le Dantec (1992). This demonstrated that the model could provide a reasonable prediction of the static strength of a composite coupon. The material model was then tested in LS-DYNA to eliminate the possibility of serendipitous agreement between the Mathcad results and the published values. The model was implemented as a user-defined subroutine, or “UMAT”. A series of tensile coupon simulations of GLARE was carried out using this model. Numerically, the model functioned well, with no computational problems such as hourglassing.

The tensile coupon simulations showed that the model could predict the elastic response of the prepreg identically to the LS-DYNA orthotropic-elastic (type 2) material model. It was found that the simulations were sensitive to the mesh density. A lower mesh density caused sudden drops in the predicted stress-strain curve. There were fewer

elements to accumulate damage. Thus, when a single element failed, a significant portion of the load-carrying capacity of the coupon was lost.

It was noted that the 0° GLARE-3-3/2 coupons with the damage simulation in the UMAT activated were stiffer than the experimental results after yielding (Figure 83). The prepreg material model only accounts for stiffness loss perpendicular to the fibres. However, stiffness is lost in the experiments in the fibre direction, through matrix plasticity and some fibre fracture. Without accounting for these effects the model would behave slightly stiffer than the measured experimental response. The predicted stress-strain curves for the 15° GLARE-3-3/2 coupons using a higher mesh density agreed very well with experimental data (Figure 84).

The earlier work of Sun and Wu (1992) successfully used quasi-static data for their impact simulations of ARALL. They demonstrated that it is possible to use this data in dynamic applications. Furthermore, Vlot (1993b) observed that under impact velocities of 10 m/s, it is possible to use quasi-static properties in analytical impact response predictions with GLARE laminates. The impact simulations presented in this thesis ranged in velocity from 2.7 m/s to 4.4 m/s; therefore, the use of static coupon data to validate the model was considered appropriate.

8.3 Impact Modelling of GLARE

The four components of the impact damage modelling methodology were assembled to study the response of GLARE laminates to low-velocity impacts. To summarize, the following configurations were used to study the effect of prepreg material model and interface type on the predicted impact response:

1. Elastic-plastic aluminum, elastic prepreg and tied-interface,
2. Elastic-plastic aluminum, elastic prepreg and uncorrected tiebreak-interface,
3. Elastic-plastic aluminum, elastic prepreg and corrected-tiebreak interface, and
4. Elastic-plastic aluminum, UMAT prepreg and simple-tiebreak interface.

8.3.1 Mesh Sensitivity

As for the aluminum simulations, mesh density study was conducted. The predicted dent depth for the lowest mesh density was lower than those of the higher mesh density simulations (Figure 90). At the same time, the damage geometry predicted with the lower mesh density did not agree with the experiments (Figure 91). These two effects were caused by the paucity of nodes connecting the interface together in the impacted region. It only took the release of two or three nodes to completely delaminate the impacted area. The medium- and high-mesh densities predicted very similar damage geometries in agreement with the experiments. However, the CPU time required for the medium-density mesh was 40% of that required for the high-density simulations. Consequently, the medium-mesh density was selected for the GLARE simulations.

8.3.2 Predicted Panel Response

The type of interface between the constituent layers was varied to study the effect on the predicted response and damage geometry of the panels. Three types of interface were examined with the elastic prepreg. The UMAT simulations were carried out using only the simple-tiebreak interface. The percent differences between the predictions and experiments for GLARE-3 are summarized in Figures 114 to 116. All of the results quoted in the following sections are from these graphs. The results from GLARE-3 are representative of the behaviour of the other laminates.

Figure 114: Percent difference between GLARE-3 absorbed energy predictions and experimental results.

Figure 115: Percent difference between GLARE-3 permanent dent depth predictions and experimental results.

Figure 116: Percent difference between GLARE-3 peak impact force predictions and experimental results.

The tied-interface fixed the layers together without failure. These simulations, with elastic prepreg, under-predicted the absorbed energy. The permanent dent depth predictions from these simulations were below the experimental values. The peak impact force was over-predicted. The same effects caused these differences. Both the plasticity of the aluminum and damage in the prepreg contributed to the permanent deformation of the experimentally impacted panels. Without the inclusion of damage in the prepreg layers, energy can only be absorbed through aluminum plastic deformation and stored elastic energy in all the layers. As well, with only the aluminum plasticity contributing to the permanent deformation, the dent depths would be lower. In fact, the stresses in the elastic prepreg layers actually try to push the aluminum layers upward but were prevented from doing so by the lack of delamination. Without the formation of damage, the effective stiffness of the tied-interface panels is higher than the experimental panels. This resulted in the higher peak impact force predictions.

The simple-tiebreak interface used the manufacturer's interface properties to predict failure. Simulations using this interface model, with the elastic prepreg, under-predicted the absorbed energy by the largest amount. The simple-tiebreak interface also resulted in lower dent depths than measured in the experiments. Finally, these simulations over-predicted the peak impact force. Unlike the tied-interface simulations, the formation of delamination damage was enabled in these simulations, permitting the rebound of the elastic prepreg layers. This released stored elastic energy in the prepreg layers, resulting in the lower absorbed energy prediction. The same action also pushed the aluminum layers upward, leading to considerably lower dent depths than the tied-interface simulations. The peak impact force difference was slightly higher than that of

the tied-interface simulations. The small difference indicates that the formation of delamination damage does not have a large influence on the impact forces in panels.

The corrected-tiebreak interface used a stress-displacement curve to represent the post-delamination behaviour of the interface. The corrected-tiebreak interface simulations, with elastic prepreg, under-predicted the absorbed energy. The permanent dent depth was also under-predicted and the impact force was over-predicted. The inclusion of the post-delamination load-displacement curve improved the absorbed energy prediction over the simple-tiebreak interface simulations by preventing the elastic prepreg layers from rebounding. The same effect lowered the dent depth prediction differences substantially. However, the peak impact force was still over-predicted with the corrected-tiebreak interface. This reinforces the fact that delamination damage does not have a great influence on the peak impact force in the impacted panels.

The UMAT simulations, using the simple-tiebreak interface, under-predicted the absorbed energy. However, the results were a substantial improvement over the previous simulations. As well, the permanent dent depth was over-predicted by a smaller percentage than the previous simulations. Finally, the peak impact force was over-predicted by a significantly lower amount than the simulations described above. Based on these results, it is postulated that the formation of intralaminar damage has a greater effect on the impact response of impacted FMLs than the formation of delamination damage. One can imagine that the combined surface area of the multitude of matrix cracks formed in the prepreg layers is much larger than the area formed by the delamination. Therefore, the energy released by formation of the matrix cracks would be proportionally larger than that released by the formation of delamination damage.

Interestingly, the inclusion of damage in prepreg layers resulted in an over prediction rather than an under-prediction of the permanent dent depth. The reasons for this are not readily discernible.

Another effect to consider is the influence of the interlaminar and intralaminar damage on the in-plane and out-of-plane properties of the panels. Due to the large deformation caused by the impact event, it is the in-plane properties that have the greatest influence on the impact response. The formation of matrix cracks degrades these properties, reducing the stiffness of the panel and thus reducing the peak impact force. The formation of delamination damage has a greater effect on the out-of-plane properties of the laminate. Based on the impact force results, delamination damage does not have a significant influence the peak impact force. Although delamination damage does not have a significant effect on the impact response of the panels, the presence of delamination can affect the post-impact behaviour of the panels. For example, the compressive buckling sensitivity of GLARE could be increased by delamination.

While the inclusion of the UMAT improved the predicted response of the panels, there were still differences. Some of these differences could be accounted for by the lack of fixture effects in the simulations. No description of the effect of increasing the complexity of the model on the absorbed energy was found in literature. However, Majeed (1995) calculated a decrease of 22% in the peak impact force predicted for composite coupons by increasing the complexity of the model. This was achieved by switching from a rigid impactor and base to a complete model of the crosshead and base of the Dynatup rig with elastic properties. This is of a similar magnitude to the over-prediction of the peak impact force by the UMAT simulations. It is also possible that the

destructive inspections of the panels missed some fibre fractures. This damage mode would contribute to the overall energy absorption of the panels as well. A more detailed inspection of impacted panels may be required to eliminate this possibility.

8.3.3 Predicted Damage Geometry

The discussion on the damage geometry presented in the following paragraphs refers to Figures 104 to 113. The tied-interface simulations, with the layers connected without failure, were not able to predict any delamination damage.

The delamination damage predicted with the simple-tiebreak interface and elastic prepreg was not like that observed during the experiments. In these simulations, the prepreg layers were elastic, therefore they rebounded after impact. This action pushed the top aluminum layer upward and reduced the depth of the resulting impact dent.

The use of the corrected-tiebreak interface, with the elastic prepreg, improved the predicted damage geometry of the panels and eliminated the rebound of the aluminum layers by introducing a restraining force between the aluminum and the prepreg layers. However, the extent of the delamination damage was still over-predicted by the corrected-tiebreak interface models compared with the experimental results.

The predicted damage geometries for the simulations with the UMAT prepreg model agreed well with the delamination damage measured following the experiments. The extent of the damage was accurately predicted as well as the general shape of the damage. For example, GLARE-4, which has a dominant fibre direction, was observed to have elliptical delamination damage following impact. The GLARE-4 simulations also predicted this damage geometry (Figures 108, 109 and 110).

Based on the predicted delamination geometries, it is clear that the formation of delamination damage is inherently linked to the formation of damage in prepreg. With the elastic prepreg model, it was necessary to use interlaminar strength values significantly lower than those quoted by the manufacturer to achieve the formation of delamination. Even then, the predicted geometry did not resemble that of the experiments (Figure 105b). However, with the UMAT activated, the resulting geometry agreed very well with that of the experimentally impacted coupons. It was possible to achieve this result using the manufacturer's interlaminar strength properties.

8.4 Limitations of the Modelling Procedure

Although the simulations show good agreement with the experiments, the prepreg model does not account for plasticity effects in the matrix material. Literature sources indicate that these effects are negligible in FMLs (Wu and Sun, 1993). However, this effect should still be investigated in future implementations of the model. The model is also limited to simulations below the puncture threshold of GLARE since the prepreg material model does not include fibre fracture.

At present there is no direct way to visualize the level of damage in the elements of the simulation. This is a problem with LS-DYNA and the post-processing software. The only indicators of the level of damage in the prepreg are the stresses that are output by LS-DYNA. Despite the success of this project, much additional work is possible to improve the capability, accuracy and most importantly, applicability of this technique.

CHAPTER 9

CONCLUSIONS

An extensive series of impact tests on GLARE laminates was completed in this research project. Also, a practical methodology for modelling the formation of impact damage in these laminates was developed and an initial evaluation was completed. This methodology is a synthesis of a CDM-based material model and a commercial finite-element code. It was used to predict the response of GLARE panels subjected to impact loads. This methodology can form the foundation of an improved impact qualification procedure for FMLs. The designers of new FMLs will benefit from the availability of such a procedure. Conclusions drawn from this project are as follows:

1. GLARE-5, with a greater volume fraction of fibres, absorbs less impact energy on an areal density basis than aluminum 2024-T3, GLARE-3 or GLARE-4.
2. The orientation of a majority of the fibres in the 0° direction results in the formation of a single crack in punctured GLARE-4 specimens. This crack grows with increasing impact energy. In the other GLARE variants the cracks developed in both fibre directions and did not increase in size with increasing impact energy.

3. The DCB test methodology attempted in this work did not provide useful data for the prediction of the post-delamination response of the impacted panels. This issue remains to be resolved in future work.
4. The aluminum impact simulations demonstrated the applicability of LS-DYNA to the problem of simulating low-velocity impacts on FML panels.
5. The aluminum simulations were not sensitive to the mesh density, but showed a small sensitivity to the boundary conditions.
6. The CDM-based material model was used to accurately predict the static tensile response of composite and GLARE coupons compared with experimental data.
7. The UMAT, with the simple-tiebreak interface delamination model, showed some sensitivity to the mesh density. Lower mesh densities resulted in an over-prediction of the extent of the delamination damage and an under-prediction of the dent depth. The mesh densities converged on a representative damage geometry.
8. As discussed, delamination does not appear to be the dominant damage mode in impacted GLARE laminates. Damage in the prepreg layers is a more influential damage mode and is responsible for the majority of the energy absorbed in low-velocity impacts. Delamination damage may need to be considered when examining the post-impact properties of GLARE laminates.
9. The geometry of delamination damage in the impacted panels was accurately predicted using the CDM-based damage model. The geometry and the size of the predicted delaminated regions agreed with the experiments. As was observed in the experiments, the predicted damage was affected by the lay-up of the coupons.

CHAPTER 10

FUTURE WORK

The successful implementation of a damage modelling methodology for FMLs is the first step in a process that will lead to a deeper understanding of the fundamental damage processes in these materials. Monolithic alloys have been employed for centuries in engineering structures and even relatively modern composite materials have been in widespread use for decades. Fibre-metal-laminates are a comparatively new material. Thus, there remains a large amount of work to develop a complete set of tools to analyze all of the damage modes in these materials. The following sections describe some suggested topics for follow-on work in the area of FML damage modelling including possible improvements to the existing methodology and additional applications.

10.1 Experimental Testing

There are numerous tests that should be conducted on FMLs to improve the understanding of their response to damage. Vlot (1993) described a method for impact testing ARALL under an applied tensile load. Similarly, GLARE laminates should be impacted under applied biaxial and uniaxial loads. This boundary condition better represents the loads that FMLs would experience in certain locations on an aircraft. These tests would be supported by an expanded series of simulations using the damage modelling methodology with applied boundary loads.

New FML material systems are being developed with different types of prepregs and alloys. For example, TiGr laminates are being considered as a possible material for application on the F-35 Joint Strike Fighter. Additional characterization tests could be conducted using the method of Ladevèze and Le Dantec (1992) on these new constituent materials. These materials would also include the most recent GLARE prepreg materials, such as Cytac FM-94. Fibre-metal-laminates produced with woven prepregs should also be characterized. This would support further development of the CDM-based prepreg model and allow its application to the new laminates. The modelling methodology could also be applied in simulations with more complicated loading conditions including shear, compression and bending.

10.2 Full Failure Prediction of FMLs

The modelling methodology could be expanded to include the failure of the fibres and the aluminum layers. This would allow for the prediction of fracture and penetration in the impacted laminates. One possible approach for modelling the failure of the prepreg in the fibre-direction is to assume that the prepreg is elastic until the ultimate strength in that direction is reached. The element would then no longer be able to carry loads in that direction. A similar approach can be used to predict failure in the aluminum layers based on strain to failure. There are existing material models for metals in LS-DYNA that could be used for this application. The CDM-based prepreg material model would have to be updated with additional code to include failure in the fibre-direction.

10.3 Improved Delamination Modelling

An improved method for predicting the failure of the interface between the layers is required. The CDM-based delamination modelling approaches described in Section 6.1 would be appropriate for implementation in a finite element framework (Allix *et al.*, 1991, Allix and Ladevèze, 1992, Daudeville and Ladevèze, 1993, Daudeville *et al.*, 1995, Allix and Ladevèze 1995 and Allix *et al.*, 1998). The current model used for predicting the interface failure in LS-DYNA does not incorporate degradation of the interface. The existing DCB data or additional delamination testing could support this development. Since it was noted that the formation of delamination damage did not significantly influence the impact response of the panels, it is not necessary to employ a more sophisticated delamination model for thin laminates. The current methodology does provide an accurate prediction of the delamination size. Predicting the size of the delamination damage is important when considering the post-impact response of the laminates. However, in thicker GLARE laminates, where the out-of-plane properties would exert more influence, an improved delamination model should still be investigated.

10.4 Miscellaneous Extensions and Additions to the Modelling Methodology

To further extend the capabilities of the modelling methodology put forward in this thesis, the CDM-based damage model should be implemented for solid-brick elements in LS-DYNA. This would allow the model to be applied to more complex problems. Additionally the strain-rate sensitivity of the fibreglass prepreg should be investigated and, if required for specific problems, incorporated into the prepreg material model. The modelling methodology could also be applied in implicit finite element codes such as ABAQUS or the implicit part of LS-DYNA. This would improve its applicability to static loading simulations.

Full simulations of the entire impact fixture should be conducted to further examine the effects of slippage between the clamping plates and the impacted specimens. These simulations could also be used to investigate losses due to compliance in the impact fixture and crosshead.

10.5 Additional Applications

There are numerous additional applications of the FML damage modelling methodology that could be investigated as listed below.

- Application of the model to the problem of residual stress development on the aluminum layers following impact. Presently the stress-state in these layers following impact is the subject of debate. Knowledge of the stress-state would aid in the prediction of post-impact fatigue behaviour.
- Application of the model to the problem of forming of GLARE laminates. Although the current trend is towards the preparation of built-up components from GLARE, forming may still be required to minimize costs or to produce simple parts.
- The National Research Council, Bombardier Aerospace and Carleton University are currently investigating the problem of buckling and post-buckling of stiffened GLARE panels and cylinders. The finite element models use a single layer of shells with the material properties “smeared” across the laminate. A multi-layered approach using the CDM-based prepreg model could improve the accuracy of these simulations.
- A parametric study of different impactors and specimen configurations should be conducted in LS-DYNA. The model could also be scaled up to simulate actual aircraft components such as the leading edge of a wing.

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APPENDICES

A - PRELIMINARY IMPACT TEST RESULTS WITH NRC FIXTURE

Table A1: Summary of preliminary results using the NRC impact fixture and various impactors.

Sample #	Impactor Diameter (mm)	V_{imp} m/s	m_{imp} kg	E_{imp} J	E_{abs} J	F_{max} kN	% Energy Absorbed	Comments
2	12.7	1.682	6.255	8.865	3.711	3.149	41.9	Good
38	12.7	1.9	6.345	11.550	7.295	3.527	63.2	Good test, sample failed
5	15.875	1.694	6.285	9.109	8.119	2.876	89.1	Not good
36	15.875	2.3	6.375	16.870	8.276	4.7738	49.1	Good test, sample failed
37	15.875	2.44	6.375	18.900	12.27	5.0661	64.9	Good test
3	Large dia	1.677	6.621	9.309	3.86	3.95	41.5	Good
12	25.4	0.84	6.415	2.25	1.219	1.2538	54.2	Good
15	25.4	1.1	6.415	3.89	2.18	1.7864	56.0	Good
10	25.4	1.325	6.415	5.644	3.425	2.306	60.7	Good
11	25.4	1.328	6.415	5.666	3.073	2.313	54.2	Good
9	25.4	1.329	6.415	5.679	3.939	2.222	69.4	Good
29	25.4	1.333	6.415	5.769	3.26	2.3547	56.5	Repeat test
8	25.4	1.672	6.415	8.988	5.93	3.021	66.0	Good
4	25.4	1.674	6.325	8.898	5.234	3.104	58.8	Good
7	25.4	1.675	6.415	9.019	6.139	3.013	68.1	Good
18	25.4	1.95	6.415	12.2	7.607	3.7631	62.4	Good
21	25.4	2.14	6.415	14.64	9.419	4.2291	64.3	Good
24	25.4	2.4	6.415	18.47	12.88	4.74	69.7	Good
25	25.4	2.61	6.415	21.79	14.58	5.3392	66.9	Good
26	25.4	2.87	6.415	26.37	19.58	5.879	74.3	Good
27	25.4	3.35	6.415	35.9	26.26	7.12	73.1	Good
31	25.4	3.54	6.415	40.18	32.21	7.214	80.2	Good
33	25.4	3.56	6.415	40.61	32.34	7.5277	79.6	Good
34	25.4	3.58	6.415	41.12	32.84	7.6008	79.9	Good
35	25.4	3.59	6.415	41.33	41.5	7.5794	100.4	Good, sample failed
32	25.4	3.603	6.415	41.64	41.75	7.5853	100.3	Good test, sample failed
30	25.4	3.64	6.415	42.45	42.65	7.7896	100.5	Good test, sample failed
28	25.4	3.72	6.415	44.33	44.77	8.0735	101.0	Good, sample failed

B – PRELIMINARY IMPACT TEST RESULTS WITH NASA AND NRC TEST FIXTURES

Table B1: Preliminary impact test results using the NRC and NASA fixtures
with 25.4 mm and 12.7 mm diameter impactors, respectively.

	E_{imp}	E_{abs}	F_{max}	$\% E_{abs}$
	J	J	kN	(%)
NRC Fixture 292.1 mm X 292.1 mm Coupon, 203.2 mm diameter circular opening	11.3	8.0	3.2	70.44
	20.4	13.8	4.7	67.90
	30.8	21.8	6.2	70.66
	41.1	30.6	7.3	74.53
	45.1	34.4	7.7	76.24
	48.7	36.0	7.3	73.87
	48.3	46.0	6.8	95.22
	48.5	48.5	7.1	100.03
	50.0	50.7	7.4	101.45
	51.3	50.7	8.1	98.79
NASA Impact Fixture 254 mm X 177.8 mm Coupon with a 101.6 mm X 101.6 mm Square Opening	2.250	1.219	1.254	54.18
	2.252	1.238	1.262	54.97
	2.265	1.135	1.282	50.11
	3.890	2.180	1.786	56.04
	3.891	2.053	1.843	52.76
	3.918	2.119	1.817	54.08
	5.644	3.425	2.306	60.68
	5.666	3.073	2.313	54.24
	5.679	3.939	2.222	69.36
	5.769	3.260	2.355	56.51
	8.988	5.930	3.021	65.98
	8.898	5.234	3.104	58.82
	9.019	6.139	3.013	68.07
	12.200	7.607	3.763	62.35
	12.250	7.878	3.769	64.31
	12.360	8.146	3.744	65.91
	14.640	9.419	4.229	64.34
	14.800	10.090	4.174	68.18
	14.800	10.030	4.199	67.77
	18.470	12.880	4.740	69.73
	21.790	14.580	5.339	66.91
	26.370	19.580	5.879	74.25
	35.900	26.260	7.120	73.15
	40.180	32.210	7.214	80.16
	40.610	32.340	7.528	79.64
	41.120	32.840	7.601	79.86
	41.330	41.500	7.579	100.41
	41.640	41.750	7.585	100.26
	42.450	42.650	7.790	100.47
	44.330	44.770	8.074	100.99

C – NRC AND NASA IMPACT FIXTURE RESULTS

Figure C1: Absorbed energy for the NASA and NRC impact fixtures.

Figure C2: Maximum force versus impact energy for the NASA and NRC fixtures.

Figure C3: Maximum force versus absorbed energy for the NASA and NRC impact fixtures.

D – FORMAL IMPACT TEST RESULTS

Table D1: Summary of GLARE-3 formal impact test results.

Sample #	E _{nom} J	Impact Velocity		Impact Mass kg	Impact Energy		Energy Absorbed		Maximum Load		Percent Energy Absorbed	Comments
		m/s			J		J		kN			
GL3-11	15	2.25		6.415	16.25		10.27		4.0410		63.2	Good test
GL3-12	25	2.84		6.415	25.97		17.33		5.5146		66.7	Good test
GL3-13	35	3.37		6.415	36.54		26.08		6.7760		71.4	Good test
GL3-14	45	3.82		6.415	46.82		47.25		7.8877		100.9	Punctured completely
GL3-15	55	4.01		6.415	51.59		52.12		8.3143		101.0	Punctured completely
GL3-16	15	2.24		6.415	16.14		9.96		4.1017		61.7	Good test
GL3-17	25	2.83		6.415	25.83		17.31		5.4974		67.0	Good test
GL3-18	35	3.79		6.415	46.18		32.96		7.9488		71.4	Good test
GL3-19	45	3.36		6.415	36.34		25.18		6.8626		69.3	Good test
GL3-20	55	4.21		6.415	56.91		53.33		8.4639		93.7	Good test
GL3-21	15	2.24		6.415	16.13		10.15		4.0713		62.9	Good test
GL3-22	25	2.85		6.415	26.20		17.38		5.5828		66.3	Good test
GL3-23	35	3.37		6.415	36.56		25.27		6.9010		69.1	Good test
GL3-24	45	3.83		6.415	47.26		46.93		8.1362		99.3	Punctured completely
GL3-25	55	4.01		6.415	51.73		53.03		8.0796		102.5	Punctured completely
GL3-26	15	2.23		6.415	16.04		10.34		4.0938		64.5	Good test
GL3-27	25	2.85		6.415	26.05		16.82		5.618		64.6	Good test
GL3-28	35	3.35		6.415	36.12		25		6.8287		69.2	Good test
GL3-29	45	3.81		6.415	46.68		33.78		7.9324		72.4	Good test
GL3-30	55	4.01		6.415	51.75		52.62		8.05		101.7	Punctured completely
GL3-31	15	2.21		6.415	15.73		9.84		3.995		62.6	Good test
GL3-32	25	2.85		6.415	26.19		17.6		5.5515		67.2	Good test
GL3-33	35	3.37		6.415	36.42		25.54		6.7801		70.1	Good test
GL3-34	45	3.81		6.415	46.72		33.84		7.9833		72.4	Good test
GL3-35	55	3.995		6.415	51.33		52.62		8.088		102.5	Punctured completely
GL3-36	15	2.211		6.415	16.43		13.484		4.2584		82.1	Good test
GL3-37	25	2.83		6.415	26.85		20.534		5.5181		76.5	Good test
GL3-38	35	3.34		6.415	37.38		27.942		6.9432		74.8	Good test
GL3-39	45	3.79		6.415	48.23		48.622		8.1121		100.8	Punctured completely
GL3-40	55	3.9		6.415	50.98		38.913		8.345		76.3	Poorly clamped
GL3-41	Control											
GL3-42	25	2.62		6.415	23.13		16.735		5.0946		72.4	Good test
GL3-43	75	4.86		6.415	76.09		52.8		8.088		69.4	Extreme puncture
GL3-44	55	3.94		6.415	52.06		52.95		7.9456		101.7	Punctured completely
GL3-46	Control											

Table D2: Summary of GLARE-4 formal impact test results.

Sample #	E _{nom}	Impact Velocity		Impact Mass	Impact Energy		Energy Absorbed		Maximum Load	Percent Energy Absorbed	Comments
	J	m/s	kg	J	J	J	kN				
GL4-01	25	2.89	6.717	26.74	17.09	5.8451	63.9	Good test			
GL4-02	35	3.39	6.717	36.88	25.51	7.0741	69.2	Good test			
GL4-03	45	3.83	6.717	47.00	36.83	8.293	78.4	Failed			
GL4-04	55	4.08	6.415	53.35	46.3	8.8661	86.8	Failed			
GL4-05	60	4.36	6.415	61.08	55.15	8.9485	90.3	Failed			
GL4-06	25	2.89	6.415	26.73	17.11	5.394	64.0	Good test			
GL4-07	35	3.41	6.415	37.29	25.19	7.1621	67.6	Good test			
GL4-08	45	3.83	6.415	47.03	40.54	8.2127	86.2	Failed			
GL4-09	55	4.09	6.415	53.67	49.49	8.3052	92.2	Failed			
GL4-10	60	4.36	6.415	61.00	54.255	8.474	88.9	Failed			
GL4-11	25	2.88	6.415	26.59	17.82	5.6918	67.0	Good test			
GL4-12	35	3.39	6.415	36.91	24.69	7.0997	66.9	Good test			
GL4-13	45	3.82	6.415	46.69	32.97	8.1717	70.6	Good test			
GL4-14	55	4.08	6.415	53.52	47.83	8.4454	89.4	Failed			
GL4-15	60	4.36	6.415	60.85	55.62	9.095	91.4	Failed			
GL4-16	25	2.87	6.415	26.44	17.11	5.7429	64.7	Good test			
GL4-17	35	3.39	6.415	36.90	24.71	7.1212	67.0	Good test			
GL4-18	45	3.83	6.415	46.94	40.725	8.1008	86.8	Failed			
GL4-19	55	4.09	6.415	53.27	47.31	8.9034	88.8	Failed			
GL4-20	60	4.37	6.415	61.16	55.91	8.7573	91.4	Failed			
GL4-21	25	2.86	6.415	26.29	18.06	5.7206	68.7	Good test			
GL4-22	35	3.36	6.415	36.27	24.55	7.0041	67.7	Good test			
GL4-23	45	3.85	6.415	47.59	40.21	8.2917	84.5	Failed			
GL4-24	55	4.02	6.415	51.71	45.89	8.7242	88.7	Failed			
GL4-25	60	4.39	6.415	61.89	55.94	8.7122	90.4	Failed			
GL4-26	25	2.87	6.415	26.35	17	5.7349	64.5	Good test			
GL4-27	35	3.39	6.415	36.88	24.61	7.0956	66.7	Good test			
GL4-28	45	3.81	6.415	46.50	32.48	8.2159	69.8	Good test			
GL4-29	55	4.08	6.415	53.35	48.21	8.5845	90.4	Failed			
GL4-30	60	4.38	6.415	61.51	56.44	8.9448	91.8	Failed			
GL4-31	Control										
GL4-32	Control										

Table D3: Summary of GLARE-5 formal impact test results.

Sample #	E _{nom} J	Impact Velocity m/s	Impact Mass kg	Impact Energy J	Energy Absorbed J	Maximum Load kN	Percent Energy Abs	Comments
GL5-01	25	2.84	6.717	25.72	16.84	5.8228	65.5	Good test
GL5-02	35	3.37	6.717	36.41	25.16	7.2110	69.1	Good test
GL5-03	45	3.82	6.717	47.02	33.59	8.4912	71.4	Good test
GL5-04	55	4.24	6.415	57.74	42.02	9.7107	72.8	Good test
GL5-05	65	4.56	6.415	66.88	66.94	10.5379	100.1	Complete puncture
GL5-06	25	2.83	6.415	26.14	17.11	5.8657	65.5	Good test
GL5-07	35	3.36	6.415	36.23	24.99	7.1665	69.0	Good test
GL5-08	45	3.8	6.415	46.49	33.38	8.4057	71.8	Good test
GL5-09	55	4.21	6.415	57.11	41.59	9.6052	72.8	Good test
GL5-10	65	4.56	6.415	66.90	67.77	10.4544	101.3	Complete puncture
GL5-11	25	2.85	6.415	25.82	16.55	5.8659	64.1	Good test
GL5-12	35	3.37	6.415	36.47	24.96	7.2468	68.4	Good test
GL5-13	45	3.83	6.415	47.12	33.48	8.4722	71.1	Good test
GL5-14	55	4.24	6.415	57.94	53.8	9.6115	92.9	Partial puncture
GL5-15	65	4.5	6.415	65.19	65.9	10.2545	101.1	Complete puncture
GL5-16	25	2.83	6.415	25.85	17.08	5.7767	66.1	Good test
GL5-17	35	3.22	6.415	33.28	21.38	6.8994	64.2	Good test
GL5-18	45	3.64	6.415	42.42	28.32	8.0525	66.8	Good test
GL5-19	55	3.98	6.415	50.82	34.7	9.0805	68.3	Good test
GL5-20	65	4.19	6.415	56.29	39.871	9.5511	70.8	Good test
GL5-21	25	2.58	6.415	21.39	12.42	5.2309	58.1	Good test
GL5-22	35	3.08	6.415	30.48	19.02	6.53	62.4	Good test
GL5-23	45	3.67	6.415	43.13	28.87	8.1166	66.9	Good test
GL5-24	55	3.64	6.415	42.39	28.49	8.0295	67.2	Good test
GL5-25	65	4.16	6.415	55.62	38.73	9.4906	69.6	Good test
GL5-26	25	2.66	6.415	23.74	16.724	5.4418	70.4	Good test
GL5-27	35	3.13	6.415	31.50	17.982	6.8243	57.1	Good test
GL5-28	45	3.87	6.415	48.13	32.37	8.7694	67.3	Good test
GL5-29	55	4.25	6.415	57.90	40.89	9.7291	70.6	Good test
GL5-30	65	4.67	6.415	69.84	51.27	10.9672	73.4	Dimpled
GL5-31				Control				
GL5-32				Control				
GL5-33				Control				

Table D4: Summary of aluminum formal impact test results.

Sample #	E _{nom} J	Impact Velocity		Impactor Mass kg	Impact Energy		Energy Absorbed J	Maximum Load		Percent Energy Absorbed	Comments
		m/s			J				kN		
24T3-01	25	2.79		6.717	26.14		19.69		5.632	75.3	Good test
24T3-02	35	3.28		6.717	36.13		27.52		6.920	76.2	Good test
24T3-03	45	3.71		6.717	46.23		36.27		7.989	78.5	Good test
24T3-04	55	4.24		6.415	57.67		46.62		9.114	80.8	Good test
24T3-05	60	4.4		6.415	62.34		51.01		9.494	81.8	Good test
24T3-06	25	2.83		6.415	25.76		18.4		5.753	71.4	Good test
24T3-07	35	3.36		6.415	36.23		27.24		7.030	75.2	Good test
24T3-08	45	3.79		6.415	46.12		38.51		8.073	83.5	Good test
24T3-09	55	4.24		6.415	57.68		46.57		9.018	80.7	Good test
24T3-10	60	4.41		6.415	62.47		51.17		9.456	81.9	Good test
24T3-11	25	2.83		6.415	25.74		18.98		5.707	73.7	Good test
24T3-12	35	3.34		6.415	35.95		26.92		6.954	74.9	Good test

E – IMPACT DAMAGE PHOTOS

Figure E1: Sample GLARE-3-31 (15.73 J) (a) x-ray and (b) cross-section.

Figure E2: GLARE-3-31 (15.73 J) fibre delamination.

Figure E3: GLARE-3-32 (26.19 J) (a) x-ray and (b) cross-section.

Figure E4: GLARE-3-33 (36.42 J) (a) x-ray and (b) cross-section.

Figure E5: Matrix cracking in 3-33 in the 90° layer.

Figure E6: GLARE-3-34 (46.72 J) (a) x-ray and (b) cross-section.

Matrix cracking

Figure E7: Intralaminar cracking in GLARE-3-34 (46.72 J).

Figure E8: (a) Delamination in GLARE-4-16 and (b) unaffected region.

Figure E9: GLARE-4-08 (47.03 J) cross-section.

Figure E10: Through-crack in GLARE-4-08 (47.03 J).

Figure E11: Matrix cracking in GLARE-4-08 (47.03 J)

Figure E12: Matrix cracking in GLARE-4-27 (36.88 J).

Figure E13: GLARE-4-19 (53.27 J) cross-section.

Figure E14: GLARE-4-19 (53.27 J) detail of cracked area.

Figure E15: Matrix cracking and intralaminar failures in GLARE-4-19.

Figure E16: Delaminated regions in prepreg layers of GLARE-4-19.

Figure E17: GLARE-5-04 (55 J) (a) x-ray and (b) cross-section.

Figure E18: Delamination and fibre-bridging in GLARE-5-04 on (a) left and (b) right of dent.

Figure E19: Matrix cracking in GLARE-5-04.

Figure E20: GLARE-5-11 (25 J) (a) x-ray and (b) cross-section.

Figure E21: GLARE-5-11 showing delamination damage.

Figure E22: GLARE-5-12 (35 J) (a) x-ray and (b) cross-section.

Figure E23: GLARE-5-12 matrix cracking and top 90° layer adhesive failure.

F – ALUMINUM IMPACT SIMULATION RESULTS

Table F1: Summary of peak force results for preliminary aluminum simulations.

		Velocity	Mass	E_{imp}	$(F_{max})_{exp}$	$(F_{max})_{sim}$	Percent Difference
		m/s	kg	J	kN	kN	
Rough Mesh		2.88	6.45	26.8	5.70	6.25	9.71
		3.38	6.45	36.8	6.97	7.4	6.20
		3.81	6.45	46.7	8.03	8.65	7.71
		4.23	6.45	57.8	9.07	10.2	12.51
		4.40	6.45	62.4	9.47	10.4	9.76
Low Density		2.88	6.45	26.8	5.70	6	5.32
		3.38	6.45	36.8	6.97	7.5	7.64
		3.81	6.45	46.7	8.03	8.8	9.57
		4.23	6.45	57.8	9.07	10	10.30
		4.40	6.45	62.4	9.47	10.4	9.76
Medium Density	Simple	2.88	6.45	26.8	5.70	6.2	8.83
		3.38	6.45	36.8	6.97	8	14.81
		3.81	6.45	46.7	8.03	9.1	13.31
		4.23	6.45	57.8	9.07	10.2	12.51
		4.40	6.45	62.4	9.47	10.6	11.87
	Clamped	2.88	6.45	26.8	5.70	6.6	15.85
		3.38	6.45	36.8	6.97	7.9	13.38
		3.81	6.45	46.7	8.03	8.6	7.08
		4.23	6.45	57.8	9.07	10.1	11.41
		4.40	6.45	62.4	9.47	10.4	9.76
High Density		2.88	6.45	26.8	5.70	6.25	9.71
		3.38	6.45	36.8	6.97	7.4	6.20
		3.81	6.45	46.7	8.03	8	-0.39
		4.23	6.45	57.8	9.07	9.6	5.89
		4.40	6.45	62.4	9.47	9.75	2.90

Table F2: Dent depth results for simulated impact events for preliminary aluminum impact simulations.

		Event Parameters			Dent Depth Results		
		Velocity	Mass	Impact Energy	Experimental	Simulation	Percent Difference
					Permanent	Permanent	
		m/s	kg	J	mm	mm	
Rough Mesh		2.88	6.45	26.8	6.50	7.00	7.75
		3.38	6.45	36.8	7.67	8.30	8.17
		3.81	6.45	46.7	8.94	9.80	9.68
		4.23	6.45	57.8	9.43	11.30	19.89
		4.40	6.45	62.4	10.09	11.60	15.02
Low Density		2.88	6.45	26.8	6.50	6.20	-4.57
		3.38	6.45	36.8	7.67	7.60	-0.96
		3.81	6.45	46.7	8.94	9.10	1.85
		4.23	6.45	57.8	9.43	10.30	9.28
		4.40	6.45	62.4	10.09	10.80	7.09
Medium Density	Simple	2.88	6.45	26.8	6.50	6.60	1.59
		3.38	6.45	36.8	7.67	8.00	4.26
		3.81	6.45	46.7	8.94	9.70	8.56
		4.23	6.45	57.8	9.43	10.80	14.59
		4.40	6.45	62.4	10.09	11.20	11.06
	Clamped	2.88	6.45	26.8	6.50	6.40	-1.49
		3.38	6.45	36.8	7.67	7.80	1.65
		3.81	6.45	46.7	8.94	9.30	4.09
		4.23	6.45	57.8	9.43	10.70	13.53
		4.40	6.45	62.4	10.09	11.00	9.07
High Density		2.88	6.45	26.8	6.50	6.60	1.59
		3.38	6.45	36.8	7.67	7.80	1.65
		3.81	6.45	46.7	8.94	9.20	2.97
		4.23	6.45	57.8	9.43	10.40	10.34
		4.40	6.45	62.4	10.09	11.00	9.07

Table F3: Impact energy results for simulated events for preliminary aluminum impact simulations.

		Experimental			Simulation			Percent Difference in E_{abs}
		E_{imp}	E_{abs}	Percent Absorbed	E_{imp}	E_{abs}	Percent Absorbed	
		J	J		J	J		
Rough Mesh		26.8	19.7	73.5	26.1	15.3	58.6	20.3
		36.8	27.4	74.5	36.6	23.3	63.7	14.5
		46.7	36.0	77.1	46.7	32.1	68.7	10.8
		57.8	45.6	78.9	56.9	41.8	73.5	6.9
		62.4	49.9	80.0	61.9	45.3	73.2	8.5
Low Density		26.8	19.7	73.5	26.1	15.50	59.4	19.2
		36.8	27.4	74.5	36.6	23.75	64.9	12.8
		46.7	36.0	77.1	46.7	32.40	69.4	10.0
		57.8	45.6	78.9	56.9	42.00	73.8	6.4
		62.4	49.9	80.0	61.9	45.00	72.7	9.1
Medium Density	Simple	26.8	19.7	73.5	26.1	15.75	60.3	17.9
		36.8	27.4	74.5	36.6	23.50	64.2	13.8
		46.7	36.0	77.1	46.7	32.00	68.5	11.1
		57.8	45.6	78.9	56.9	41.90	73.6	6.7
		62.4	49.9	80.0	61.9	45.00	72.7	9.1
	Clamped	26.8	19.7	73.5	26.1	14.25	54.6	25.7
		36.8	27.4	74.5	36.6	23.00	62.8	15.6
		46.7	36.0	77.1	46.7	31.50	67.5	12.5
		57.8	45.6	78.9	56.9	41.00	72.1	8.7
		62.4	49.9	80.0	61.9	44.80	72.4	9.5
High Density		26.8	19.7	73.5	26.1	15.50	59.4	19.2
		36.8	27.4	74.5	36.6	23.80	65.0	12.7
		46.7	36.0	77.1	46.7	32.40	69.4	10.0
		57.8	45.6	78.9	56.9	42.00	73.8	6.4
		62.4	49.9	80.0	61.9	45.10	72.9	8.9

Table F4: Results of the final impact simulations on aluminum 2024-T3.

(E _{imp}) _{num} (J)	(E _{imp}) _{exp} (J)	E _{abs} (J)				d _{perm} (mm)				F _{max} (kN)			
		Thin	Thick	Solid	Exp.	Thin	Thick	Solid	Exp.	Thin	Thick	Solid	Exp.
25.1	26.8	15	14.3	14.1	19.7	6.8	6.25	5.08	6.5	6.4	6.3	6.67	5.7
35.1	36.8	22	22.2	21.9	27.4	7.5	7.75	6.07	7.67	7.9	8.06	7.98	6.97
44.8	46.7	32	30.3	30	36	9.8	9.25	7.47	8.94	8.9	9.24	9.15	8.03
54.8	57.8	40	38.6	38.5	45.6	11	10	8.4	9.43	10	10.2	10.3	9.07

G – GLARE IMPACT SIMULATION RESULTS

Table G1: Summary of GLARE simulation absorbed energy results.

Material	$(E_{imp})_{exp}$	Average $(E_{abs})_{exp}$	$(E_{imp})_{sim}$	$(E_{abs})_{sim}$ (J)			
	(J)	(J)	(J)	Tied	Simple	Corrected	UMAT
GLARE-3- 2/1	16.12	10.67	N/A	N/A	N/A	N/A	N/A
	25.75	17.67	25	10.4	8.4	9.40	13.4
	36.56	25.84	35	16.1	13.1	14.80	21
	46.98	33.53	45	22.4	18.0	19.6	28
	51.57	52.67	55	28.5	22.8	41	36
GLARE-4- 2/1	26.52	17.36	25	9.46	N/A	N/A	N/A
	36.86	24.88	35	14.81	12.4	N/A	19.26
	46.96	37.29	45	20.38	16.8	N/A	27
	53.15	47.50	55	26.1	21.8	N/A	39.8
	61.33	55.51	65	N/A	N/A	N/A	N/A
GLARE-5- 2/1	24.78	16.72	N/A	N/A	N/A	N/A	N/A
	34.06	22.89	35	12.31	N/A	N/A	18.3
	44.76	31.03	45	17.1	N/A	N/A	24.7
	54.29	38.21	55	22.1	N/A	N/A	30.9
	65.35	61.14	65	27.2	N/A	N/A	39.29

Table G2: Summary of GLARE simulation maximum force results.

Material	$(E_{imp})_{exp}$	Average $(F_{max})_{exp}$	$(E_{imp})_{sim}$	$(F_{max})_{sim}$ (kN)			
	(J)	(kN)	(J)	Tied	Simple	Corrected	UMAT
GLARE-3- 2/1	16.12	4.09	N/A	N/A	N/A	N/A	N/A
	25.75	5.48	25	7.40	7.45	7.50	6.79
	36.56	6.85	35	9.30	9.63	9.64	7.90
	46.98	7.95	45	10.6	10.95	10.79	8.83
	51.57	8.13	55	12.31	12.83	11.75	10.70
GLARE-4- 2/1	26.52	5.69	25	7.61	N/A	N/A	N/A
	36.86	7.09	35	9.26	9.60	N/A	7.23
	46.96	8.21	45	11.16	11.37	N/A	9.31
	53.15	8.64	55	12.91	12.81	N/A	9.78
	61.33	8.77	65	N/A	N/A	N/A	N/A
GLARE-5- 2/1	24.78	5.67	N/A	N/A	N/A	N/A	N/A
	34.06	6.98	35	8.75	N/A	N/A	8.84
	44.76	8.26	45	10.16	N/A	N/A	10.2
	54.29	9.37	55	10.95	N/A	N/A	11.6
	65.35	10.37	65	12.63	N/A	N/A	13.3

Table G3: Summary of GLARE permanent indentation results.

Material	$(E_{imp})_{exp}$	Average $(d_{perm})_{exp}$	$(E_{imp})_{sim}$	$(d_{perm})_{sim}$ (mm)			
	(J)	(mm)	(J)	Tied	Simple	Corrected	UMAT
GLARE-3- 2/1	16.12	5.28	N/A	N/A	N/A	N/A	N/A
	25.75	7.03	25	5.4	2.78	5.24	7.00
	36.56	8.08	35	7.00	3.75	5.89	8.23
	46.98	8.98	45	8.00	4.75	6.68	9.63
	51.57	15.23	55	8.43	5.00	7.75	11.00
GLARE-4- 2/1	26.52	6.40	25	4.87	N/A	N/A	N/A
	36.86	7.49	35	6.33	4.24	N/A	7.87
	46.96	9.35	45	6.62	4.42	N/A	8.9
	53.15	10.27	55	8.00	4.66	N/A	11.2
	61.33	11.27	65	N/A	N/A	N/A	N/A
GLARE-5- 2/1	24.78	5.39	N/A	N/A	N/A	N/A	N/A
	34.06	6.57	35	5.1	N/A	N/A	7.15
	44.76	7.89	45	5.76	N/A	N/A	8.03
	54.29	8.64	55	6.7	N/A	N/A	8.79
	65.35	13.46	65	7.32	N/A	N/A	9.87

H – PARTIAL LIST OF PUBLICATIONS RESULTING FROM THE FML PROJECT

Refereed Journal Publications:

Laliberté, J., Poon, C., Straznicky, P.V. and Fahr, A. “Applications of Fibre-Metal Laminates in Airframe Structures,” *Polymer Composites*, v. 21, n. 4, August 2000.

Fahr, A., Chapman, C. E., **Laliberté, J.** and Poon, C., “Non-Destructive Evaluation Methods for Damage Assessment in Fiber-Metal Laminates,” *Polymer Composites*, v. 21, n. 4, August 2000.

Laliberté, J., Poon, C., Straznicky, P.V. and Fahr, A. “Post-Impact Fatigue in Fiber-Metal Laminates,” *International Journal of Fatigue*, 2002, v. 24, pp. 249-256.

Conferences and Workshops:

Laliberté, J., Straznicky, P.V. and Poon, C. “Numerical Modelling of Low-Velocity Impact Damage in Fibre-Metal Laminates,” *2002 ICAS Congress*, Toronto, Canada, September 2002.

Laliberté, J., Straznicky, P.V. and Poon, C. “Mode-I Delamination in Cross-Ply Fibre-Metal Laminates,” *ICF-10*, Honolulu, USA, December 2001.

Laliberté, J., Straznicky, P.V. and Poon, C. “Post Impact Shear Behaviour in GLARE,” *3rd Canadian-International Composites Conference*, Montreal, Canada, August 2001.

Laliberté, J., Poon, C. Straznicky, P.V., Fawaz. Z. and McCuaig, K. “Post-Impact Biaxial Fatigue Crack Growth in GLARE,” *21st Symposium of the International Committee on Aeronautical Fatigue*, Toulouse, France, June 2001.

Fahr, A., Chapman, C.E., **Laliberté, J.**, and Poon, C., “Non-Destructive Evaluation of Fiber-Metal Laminates” *7th International Conference on Composites Engineering*, Denver, USA, July 2-8 2000.

Poon, C., **Laliberté, J.** and Straznicky, P.V. “Review of Fibre-Metal-Laminate Technology,” *3rd Joint Canada-Japan Workshop on Composites*, Kyoto, Japan, 2000.

Laliberté, J., Poon, C., Straznicky, P.V. and Fahr, A. “Post-Impact Fatigue in Fiber-Metal Laminates,” *2nd International Conference on the Fatigue of Composites*, Williamsburg, USA, June 2000.

Laliberté, J., Poon, C. and Straznicky, P.V. “Application of Fibre-Metal Laminates in Airframe Structures,” *SPE – Polymer Composites '99*, Quebec City, Canada, October 1999.

Fahr, A., Chapman, C.E., Forsyth, D.F., **Laliberté, J.** and Poon, C., “Non-Destructive Evaluation Methods for Damage Assessment in Fibre-Metal Laminates,” *SPE- Polymer Composites '99*, Quebec City, Canada, October 1999.

Laliberté, J., Poon, C. and Straznicky, P.V. “Low-Velocity Impact Damage in GLARE Fibre-Metal Laminates,” *ICCM-12*, Paris, France, July 1999.

Laliberté, J. and Poon, C. “Durability of Hybrid Fibre Metal Laminates,” contribution to the *International Committee on Aeronautical Fatigue 1999 Summary*.

Laliberté, J., Poon, C. and Straznicky, P.V., “Impact Damage Tolerance in Composites and Fibre Metal Laminates,” *Canada-Taiwan Symposium on Aircraft Materials Testing, Maintenance and Repair*, Ottawa, Canada, April 1999.

Internal NRC Reports

Laliberté, J., Poon, C. and Straznicky, P.V. “Durability Testing of Fibre-Metal Laminates,” IAR-SMPL, Laboratory Report LTR-ST-2231, October 1999.

McCuaig, K., **Laliberté, J.**, Poon, C. and Straznicky, P.V. “Post-Impact, Off-Axis Fatigue of Glare-4 3/2 Variant and Aluminum,” IAR-SMPL, Laboratory Memorandum, November 2001 [under review].

Price, A., **Laliberté, J.** and Poon, C. “Tensile Testing of GLARE-3,” IAR-SMPL Laboratory Memorandum, March 2002 [under review].